SATELLITE ATTITUDE CONTROL IN THE PRESENCE OF REACTION WHEEL FAILURES

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ABSTRACT

Reaction wheel failures are a current concern in the spaceflight field as many missions have been affected by such failures in the past years. In this project, the impact in the controllability of a reaction wheel failure in a 150 kg satellite place in a low Earth orbit will be analysed. As expected, it has been shown that the failure of a reaction wheel will make the satellite to be unstable and uncontrolled in the axis where the failure appeared. However, it has also been shown that both magnetic torquers and thrusters can be a good replacement when a reaction wheel failure happens. In the case of magnetic torquers, it was demonstrated that by using them it is possible to regain control of the satellite in the uncontrolled axis. Moreover, it is even feasible for the satellite to be controlled with the use of only 1 reaction wheel and 2 magnetic torquers. The only drawback is that magnetic torquers present a lower torque capability than reaction wheels resulting in a higher time to stabilise the satellite. On the other hand, thrusters present the same torque capability than reaction wheels and therefore they take the same amount of time to stabilise the satellite. However, it is not feasible to only equip a satellite with thrusters because in case of a reaction wheel failure, the fuel would only last for around a year. The main conclusion is that magnetic torquers should be the main option when there is a reaction wheel failure and thrusters should only be used in emergency situations since fuel in small satellites is limited.

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NOTATION

*Note: all the variables in bold means they represent vectors and not single values.

ϕ	Roll angle [rad]
θ	Pitch angle [rad]
ψ	Yaw angle [rad]
$\dot{\phi}$	Rate of change of roll angle [rad/s]
$\dot{\theta}$	Rate of change of pitch angle [rad/s]
$\dot{\psi}$	Rate of change of yaw angle [rad/s]
$\boldsymbol{q} = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix}$	Quaternions vector []
$\dot{\boldsymbol{q}}$	Time derivative of quaternions vector []
$[\boldsymbol{\Omega}']$	Matrix needed for the solution of the differential equation of the quaternion vector in the inertial frame [rad/s]
$[\boldsymbol{\Omega}'_{\mathbf{orb}}]$	Matrix needed for the solution of the differential equation of the quaternion vector in the orbit frame [rad/s]
$\boldsymbol{w} = \begin{bmatrix} w_x \\ w_y \\ w_z \end{bmatrix}$	Angular velocities of the satellite in the inertial reference frame [rad/s]
$[\dot{\boldsymbol{w}}]$	Time derivative of satellite angular velocities in the inertial frame [rad/s^2]
$\boldsymbol{w_{orb}} = \begin{bmatrix} w_{orb_x} \\ w_{orb_y} \\ w_{orb_z} \end{bmatrix}$	Angular velocities in the orbit reference frame [rad/s]
$\dot{\boldsymbol{w}}_{orb}$	Time derivative of satellite angular velocities in the orbit frame [rad/s^2]
R_{quat}	Quaternions matrix rotation from inertial reference frame to body reference frame []
A_{312}	Euler angles matrix rotation from inertial reference frame to body reference frame in a 3-1-2 type conversion []
$\boldsymbol{M} = \begin{bmatrix} M_x \\ M_y \\ M_z \end{bmatrix}$	External moment applied to the satellite [N·m]
$\boldsymbol{h}$	Angular momentum vector of satellite [kg·m^2 / s]
$\dot{\boldsymbol{h}}_{\boldsymbol{I}}$	Time derivative of angular momentum vector of satellite in the inertial reference frame [kg·m^2 / s^2]

$\dot{\boldsymbol{h}}_{\boldsymbol{B}}$	Time derivative of angular momentum vector of satellite in the inertial reference frame [kg·m^2 / s^2]
$I = \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix}$	Matrix moment of inertia of satellite [kg·m^2]
$-\dot{\boldsymbol{h}} = \boldsymbol{N} = \begin{bmatrix} N_x \\ N_y \\ N_z \end{bmatrix}$	Torque induced by reaction wheels & thrusters [N·m]
k_d	Quaternion vector control gain (reaction wheels & thrusters control) []
k_p	Angular velocities control gain (reaction wheels & thrusters control) []
$\boldsymbol{N_m}$	Torque induced by magnetic torquers [N·m]
$\boldsymbol{M}$	Magnetic control moment [N·m/T]
$\boldsymbol{B}$	Earth magnetic field dipole model [T]
$\boldsymbol{e}$	Error vector [N·m]
k_m	Control gain moment of magnetic torquers []
$k_{d_{mag}}$	Quaternion vector control gain (magnetic torquers control) []
$k_{p_{mag}}$	Angular velocities control gain (magnetic torquers control) []
P	Period of oscillation of the satellite around Earth [s]
w_0	Angular velocity of the satellite orbiting around Earth [rad/s]
l_{arm}	Distance from thruster to the centre of gravity of the satellite [m]
m_i	Initial mass of satellite [kg]
m_f	Final mass of satellite [kg]
m_{fuel}	Mass of fuel [kg]
$\dot{m}_{fuel}$	Rate of fuel consumption [kg/s]
ΔV	Total change in velocity that the thruster can induce to the satellite [m/s]
t	Time fuel would last if used continually
$n_{manouvers}$	Number of manoeuvres that may be performed until fuel expires
t_{real}	Times fuel lasts if performing 1 manoeuvre of 3 minutes of duration per day [min]

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I. INTRODUCTION

The need to rely on its onboard instrumentation has made space technology field to achieve high level of complexity and sophistication. On the other hand, in the aircraft technology field where there is a human person physically controlling the aircraft, such need is slightly lower. (Sidi 1997)

Regarding the satellites that orbit around Earth, they can be very different as they go from tonnes of weight to just 20 kg in some cases. Moreover, in terms of orbits there are also big differences since they may be placed in orbits 40,000km above Earth or just 200km. However, there is one main thing they have in common – the dynamics and kinematics that model the attitude of a spacecraft will always be the same. (Sidi 1997)

This project is focused on small satellites, of no more than 200 kg, placed in low-Earth orbits. Generally, these types of satellites are designed for Earth observation focusing on weather forecast. Additionally, they are widely used for communication applications such as digital TV broadcast satellites.

As of the small satellites' hardware equipment, they are provided with sensors that allow the engineers to be aware of their attitude and position at all times. Furthermore, they are also equipped with actuators whose main function is correcting the satellite's position when necessary and a big antenna pointing towards Earth for allowing the communication with ground stations.

The reason why in these types of satellites the attitude determination and control is so important is that a bad orientation may result in an imperfect data recording and data transfer to Earth. For example, a bad attitude of the satellite may lead to the taking of pictures of the wrong Earth area. Additionally, a wrong orientation could also result in the antenna's satellite not pointing towards Earth that would result in the ground stations not receiving the data that the satellite is sending. In both scenarios, the mission would not be a success, as the goals would not be accomplished.

It is important to state that engineers are always trying to think ahead of any possible failure in the satellite to have an alternative backup plan prepared, in case the failure actually happens.

That is the reason why all the spacecraft present redundancy in all its hardware components – from the sensors to the actuators and antennas. As said, in case something fails in the satellite, we need to make sure that it will still be functional for completing the objectives of the mission.

Once gone through the main characteristics of small satellites, it is also important to overlook the historical background of actuators failures in the spaceflight field.

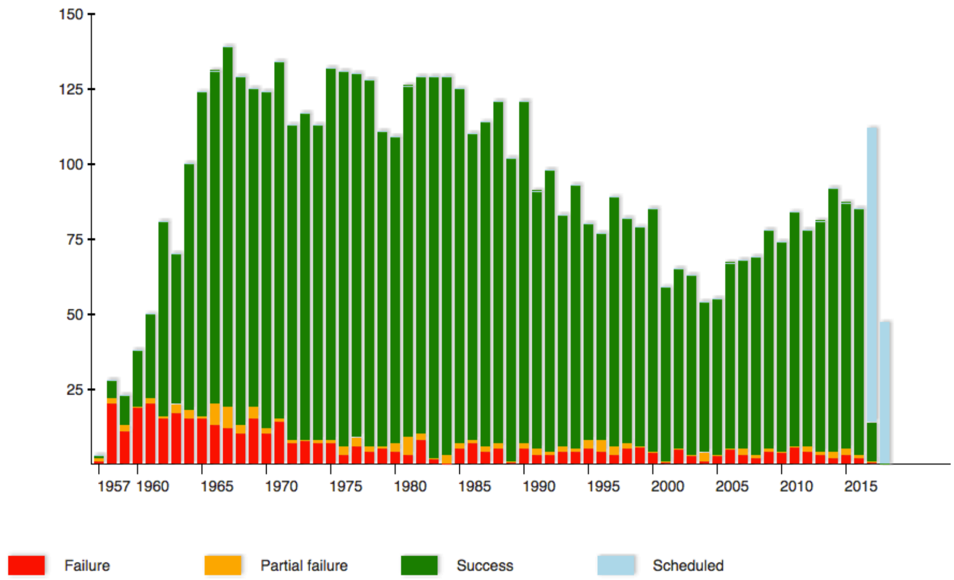

Figure 1. Evolution of the number of spacecraft launchs from 1957 to 2015 (Wikipedia Timeline of Spaceflight 2017)

Regarding the number of failures, there has been a considerably high improvement in the number of malfunctions affecting actuators as may be seen in figure 1. It has gone from around 25 failures in the year 1957 to no more than 7-8 from the year 2000 onwards.

The needing of reducing the number of actuators failures is based on the fact that in the spaceflight field, the work does not end when placing a satellite in orbit. Every satellite has a specific mission to accomplish and it can only do so if its actuators are working properly providing a good orbit control and the correct orientation at all times.

Therefore, it is of great importance that the satellite is able to be functional for as long as the mission is not complete. The only way of making that possible is by assuring that there will be no failures or that the satellite will yet be controllable in case there is one.

The huge technologic advancement available in the field nowadays in comparison with previous years has definitely helped the cause. Additionally, there are more experts working on the field who can use previous experiences of failures as a lesson for knowing what has gone wrong in the past for not making the same mistake again.

Regarding the mass of the satellites, it is patent from figure 2 how the mass of the spacecraft launched has been very irregular lately. However, it is possible to spot a trend downwards from the year 2011 onwards, placing the average mass of all satellites launched in 2014 in around 1,000 kg.

The increasing number of small mass satellites launched to low Earth orbits in the recent times may have powered such situation. For instance, satellites of no more than 200-300 kg have been launching increasingly in the past 5 years, which could have brought down the average satellite mass to the values shown in figure 2.

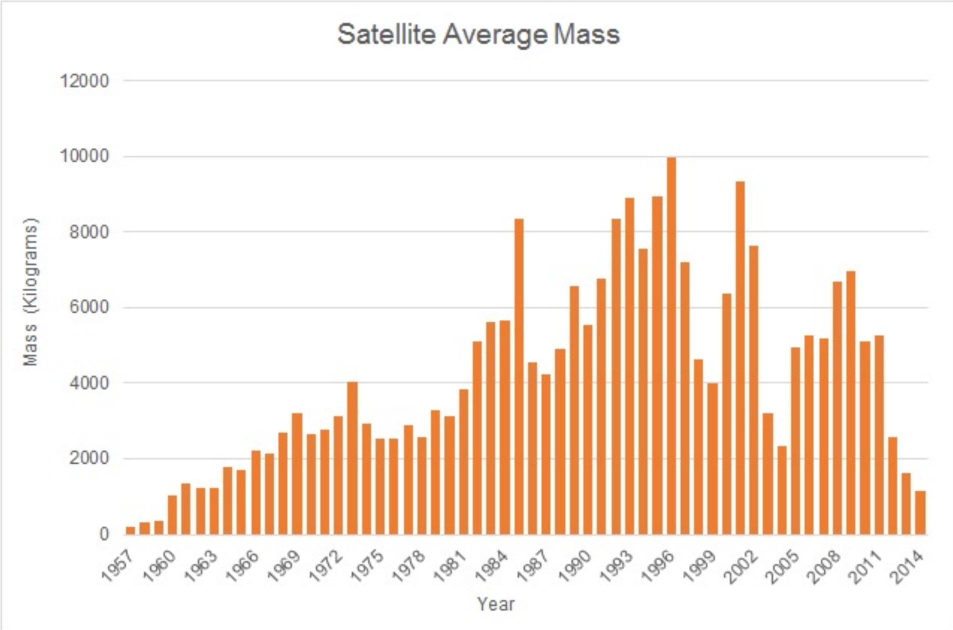

Figure 2. Evolution of satellite average mass (Seradata 2014)

Therefore, it is important to observe how the small satellites are in some way gradually taking over the spaceflight field as in this project we will focus on them.

On one hand, regarding the equations that model the kinematics and dynamics they are the same for all spacecrafts, no difference if they are heavier or lighter. On the other hand, working with a very small satellite will influence the modelling of the actuators control especially the control gains that they present.

Focusing in reaction wheels, the failures that have experienced these actuators are a current concern for the industry. The reason is that a big percentage of the satellites that were equipped with reaction wheels have recorded at least a failure some time along the course of the mission.

For instance, NASA missions “Dawn”, “Kepler”, “Mars Odissey” and “FUSE” have all been recorded with at least one reaction wheel failure. (NASA Spaceflight Forum 2015)

However, such failures have not only happened in government funded missions, as many other missions performed by public institutions such as UoSAT-12 by Surrey University have also suffered the consequences of this kind a failure.

Therefore, it is safe to state that the problem is present in all the levels of the spaceflight field. Furthermore, a reaction wheel failure presents severe consequences for the mission as it leads to the non-controllabilty of the spacecraft.

Hence, in this project such problem will be addressed and it will be demonstrated up to what point is the spacecraft controllable in the presence of a reaction failure. Moreover, we will also work with other actuators that are also equipped in the satellite such as magnetic torquers and thrusters for evaluating if it is possible to effectively control the spacecraft with their help.

II. LITERATURE REVIEW

II.I. Small satellites and type of Earth orbits

In the past 10 years, the volume of Earth observation missions involving small satellites has increased as a result of higher cooperation of space agencies such as ESA and CNES. (Sandau 2010)

However not even a simple matter like when to consider a satellite to be small, there is an absolute truth. Furthermore, there a lot of different classifications that state their own view on what are the requirements for being a small satellite, as it may be seen in figure 3. Even though, the ESA standards categorise a 150kg satellite to be mini, for the purpose of this project we will refer to it as "small satellite".

ESA:	Small 350–700 kg Mini 80–350 kg Micro 50–80 kg
EADS Astrium:	miniXL 1000–1300kg Mini 400–700 kg Micro 100–200 kg
CNES:	Mini 500 kg+P/L (Proteus) Micro 120 kg+P/L (Myriade)

Figure 3. Different satellite characterizations (Sandau 2010)

As shown in figure 4, the main advantages that small satellites present are essentially its low-cost and their short time of response. The combination of good quality with a low price and a short time of response make small satellites to be a very accessible way to the space field for new investors and private companies.

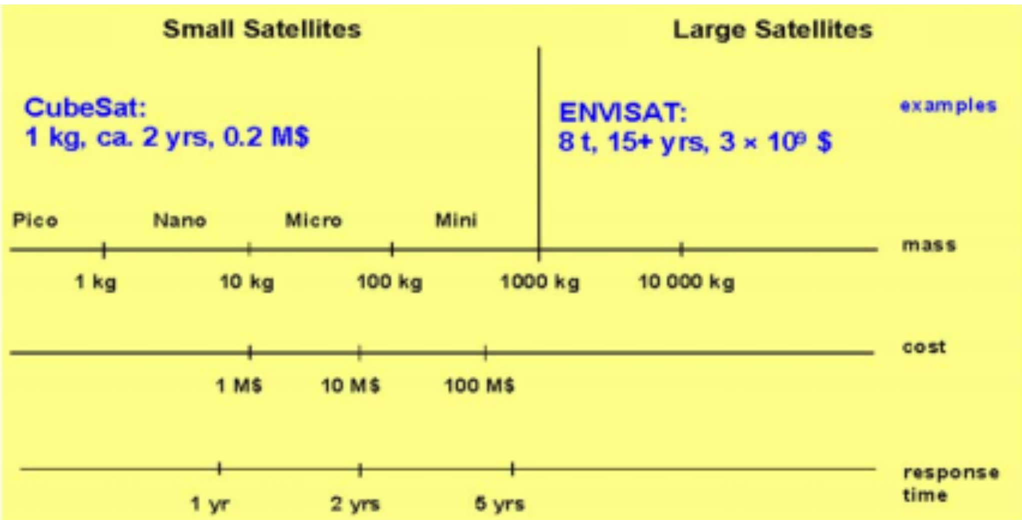

Figure 4. Small and large satellites main characteristics (Sandau 2010)

Due to the low-cost that small satellites present and the fact that they are often designed for Earth-observation missions, small satellites are usually placed on low Earth orbits. (NASA 2016)

Starting with the basics, there are three types of different orbits for satellites orbiting our planet. The closest one to Earth are called low Earth orbits; the furthest away are called high Earth or geosynchronous orbit and finally, the orbits in the middle of them, are called mid Earth orbits. It is worth to mention that the majority of the scientific satellites, including the International Space Station, are placed in low Earth orbits (NASA 2009)

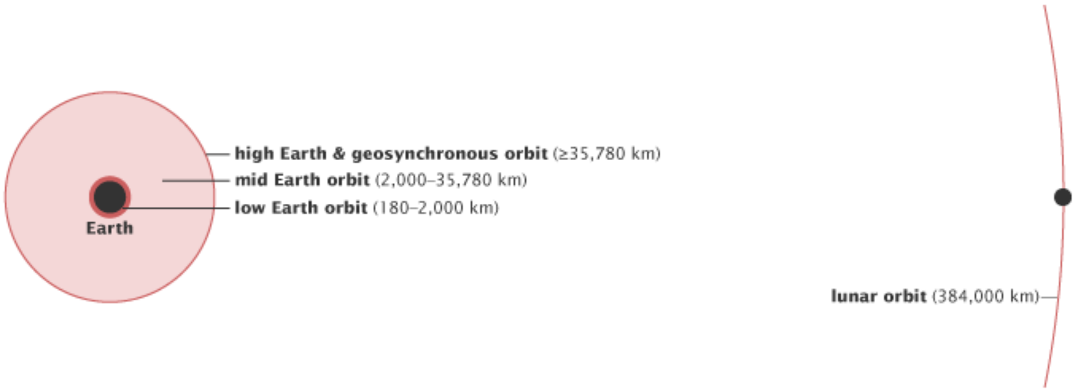

Figure 5. Types of orbits around Earth (NASA 2009)

Regarding the altitude of low Earth orbits, we may observe from figure 5 that the range goes from 180 to 2,000 km. In terms of period, it is obvious that the closer the orbit is to Earth the faster the satellite will orbit as the effect of the gravity from Earth is stronger. Moreover, if assuming completely circular orbits, it is possible to give the approximation of a low Earth orbit period that would probably lie between 88 and 127 minutes.

Additionally, there are two particular types of orbits that are very much worth of mention, one of them is a low Earth orbit and the other one is a high Earth orbit. The names of such particular orbits are sun-synchronous and geosynchronous orbits respectively.

On one hand, the polar-rotating satellites are the ones that, as its name indicates, go through both the North and South pole of Earth. All the satellites in such trajectory are placed in a sun synchronous orbit. The main characteristic of this type of orbit is that any satellite orbiting in it, would cross the equator always at the same ground local solar time. (NASA 2009)

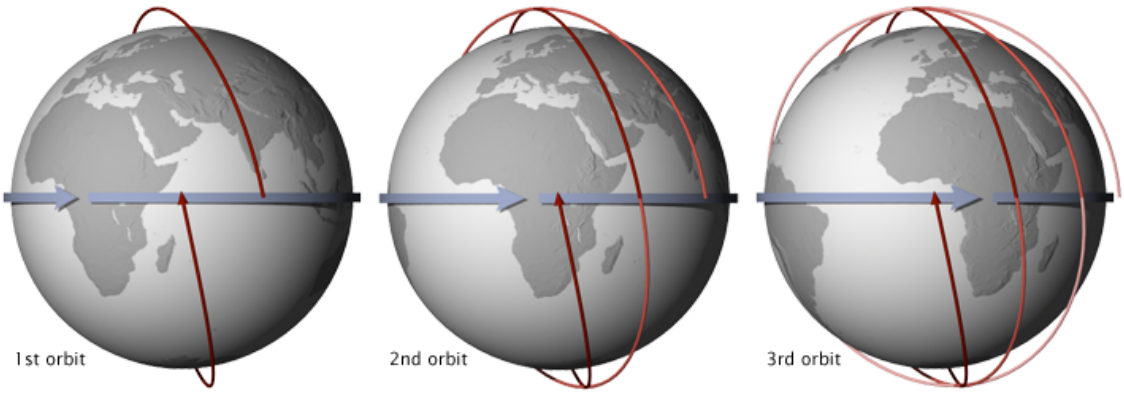

Figure 6. Sun-synchronous orbit (NASA 2009)

On the other hand, the main characteristic of the geosynchronous orbit is that a satellite placed in it, rotates at the same rate than Earth – making its period to be of nearly 24 hours. Moreover, satellites in this type of orbit are often used for weather forecast since they provide with a constant view of the same region of the planet. For instance, the weather monitoring satellites are called GOES (Geostationary Operational Environment Satellite) and every few minutes they send images of their corresponding area of Earth. (NASA 2009)

II.II. Attitude Determination and Control System

ADCS stands for "Attitude Determination and Control System" and is the system in charge of determining the attitude position of the spacecraft and of correcting such position in case it is not the one desired.

As stated above, ADCS has two main functions that are performed by two different types of components. On one hand, the part of the hardware related with the attitude determination are the sensors, where sun sensors, star trackers, horizon sensors, magnetometers and GPS are the ones most widely used. On the other hand, the hardware components in charge of correcting the attitude position of the satellite are the actuators. Among all of the actuators, the ones that will be object of study in this project are reaction wheels, thrusters and magnetic torquers.

Before going into detail of the main actuators and sensors that form part of this system, it is important to mention the main attitude control modes where ADCS may operate. First, "Orbit insertion" refers to the phase and after-phase of the boost that controls the correct placing of the satellite in the final orbit. Second, "Acquisition" refers to the first attitude determination and control – leading to its stabilization – of the satellite when it has already been allocated in the final orbit. Third, "Normal – On Station" is the most used mode used in any mission since it is the one in charge of the attitude determination and control in the standard conditions –conditions it was designed for. Finally, "Contingency or Safe" is the mode used in special circumstances when the Normal mode is not functional (Ketsdever 2017)

II.I.I Sensors

Even though sensors will not be part of our simulation analysis, a brief description of the main ones that are equipped in the vast majority of satellites, may be helpful for the global comprehension of the project.

To start with, star trackers sensors (figure 7) consist of a digital camera that uses the position of many stars to determine the attitude of the spacecraft. The focal plane may present either CCD or CMOS pixels – the first one has lower noise but the second is more resistant to radiation. Moreover, this sensor presents two different modes of operation: tracking mode and initial attitude determination. The main difference between them is that the first mode of operation is relative since it needs beforehand the coordinates of several stars for determination the attitude; however, the initial attitude determination measures the position of at least three bright stars and from such data, it extrapolates the attitude of the satellite. (Markley, Crassidis 2014)

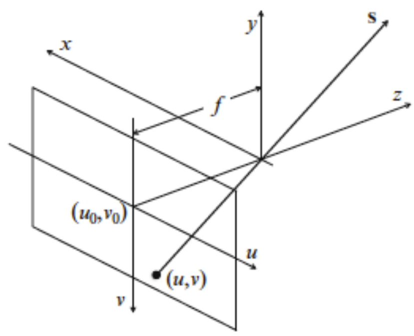

Figure 7. Star tracker configuration (Markley, Crassidis 2014)

Regarding sun sensors, they consist of a photocell or a series of photocells that determine the attitude of the spacecraft from the position where they are receiving the light and its intensity. They normally have a great accuracy but they could be tricked by light reflected on Earth – not coming directly from the Sun – leading to a maximum of 20º attitude measurements errors. The two existing types are coarse Sun sensors (CCSs) and digital Sun sensors (DSSs), being CCSs the more widely used. (Markley, Crassidis 2014)

Additionally, horizon sensors use the position of points in the Earth's horizon to determine its attitude. These sensors are, indeed, equipped in Earth-pointing satellites and present two variants. On one hand, we have the static horizon sensors that can normally be found in small-range satellites with low roll and yaw angles and a limited altitude as they just stick to the reference of one horizon point. On the other hand, the scanning horizon detectors are not fixed and use a FOV detector for locating the position of several (around four) points in the Earth's horizon. (Markley, Crassidis 2014)

Finally, as it is stated in their name, magnetometers uses the sym of the Earth's magnetic field and any local magnetic field created by the spacecraft, for determining the satellite's attitude position. From the numerous advantages they present, its light weight, small size and price and the fact that they can do not necessarily need a clear view of Sun light (unlike sun or horizon sensors) are its main characteristics. (Markley, Crassidis 2014)

II.I.II Actuators

Regarding actuators, reaction wheels are considered the main attitude controls on most satellites. Moreover, if reaction wheels are finally determined to be the equipped in a spacecraft, we have to bear in mind that, in principle, a minimum number of three reaction wheels is required for providing the engineers with a full three axes control of the spacecraft. However, it is rarely common to build a satellite with only three reaction wheels, as redundancy is almost a requirement. (Markley, Crassidis 2014)

For instance, the majority of the satellites are equipped with four reaction wheels as the spacecraft can still be controllable if there is a wheel failure and because they provide greater torque and momentum storage capability. (Markley, Crassidis 2014)

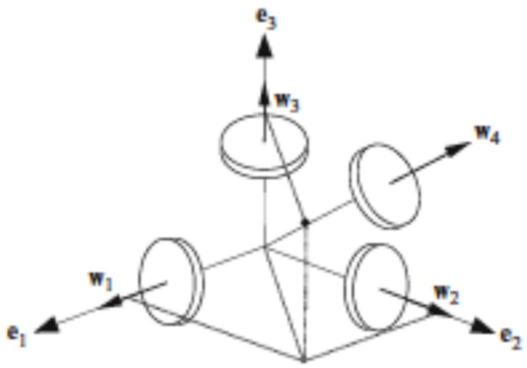

Figure 8. NASA standard four-wheel configuration (Markley, Crassidis 2014)

As of its main components, a reaction wheel is formed by a rotating flywheel, a DC internal motor and the required electronics. Even though it seems to have a quite simple structure, reaction wheels present a main disadvantage that is the disturbance forces (radial and axial) that they create. Furthermore, the axial and radial torques created by the wheels are applied on the centre of mass of the body. Such situation could be mitigated by allocating the wheels as close as possible to the centre of mass of the spacecraft –the close the distance, the lower the torque– but never completely deleted. (Markley, Crassidis 2014)

Regarding thrusters, they can be used for both trajectory and attitude control since they produce forces and torques. Moreover, the advantage they present is that they do not rely on any magnetic or gravitational field, therefore they be functional in any type of orbit. Their main disadvantage its that they are dependant on the propellant supply which makes them not have an infinite life of use and a lower efficiency. (Markley, Crassidis 2014)

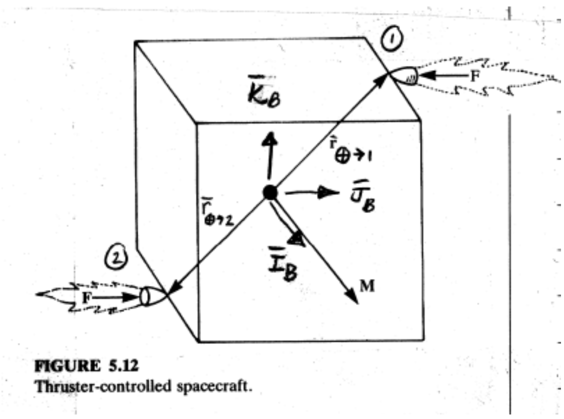

Figure 9. Sketch of moment and forces created by thrusters (Costello 2016)

Furthermore, even though there are three different types of thrusters, the most widely used are the cold gas ones. They normally use gases stored at very high-pressures to induce the boost. Nitrogen and helium very commonly used in these thrusters – especially in the case of helium, since its density is lower and helps reducing the mass of the spacecraft. Just to mention them, the other two thrusters types are the monopropellant and the electric fields thrusters. (Markley, Crassidis 2014).

Finally, regarding the magnetic torques, they work by creating a magnetic dipole moment that induces a torque in the satellite and corrects its attitude position. They normally consist of N turns of wire wrapped up in an area A, where a current I passes through that wire. Moreover, magnetic torques may be used for directly controlling the attitude dynamics of the satellite or for correcting the rotation disturbance that the reaction wheels produce. Finally, it is not usual to find spacecraft with a redundant number of magnetic torquers actuators as, in case they fail, they are already equipped with an extra number of turns of wire. (Markley, Crassidis 2014)

II.III. Spinning, zero momentum and bias momentum satellites

Even though, the spaceflight history is not more than 70 years long, in that short amount of time there has been so much development in both the technology and the design of the satellites. For the better comprehension of this project, we will do a review of the different satellites that have been placed in orbit in terms of its stabilization.

Therefore, we will focus in the study of three types of satellites: spinning satellites, zero momentum or dual spin satellites and bias momentum satellites.

Regarding the spinning satellites, as their name indicates, the constant spin provides them with inertial orientation and stability. The spinning generally occurs around the axis with the greater moment of inertia, in other words, around the axis that takes more effort to spin around and normally present a cylindrical geometry. (Mortari 2005 and Ketsdever 2017)

Their main advantages are its simplicity and long lifetime. Moreover, the spinning, if continuously reoriented, is useful for sky-observation missions. However, the poor power efficiency and the low manoeuvrability due to its high angular momentum are certainly its main drawbacks (Mortari 2005). Finally, one way to stabilize the spinning of the satellite is by using the yo-yo despin technique. (Costello 2016)

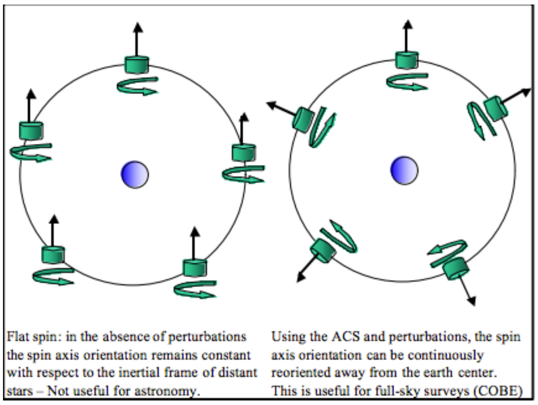

Figure 10. Different orientations of a spinning satellite (Mortari 2005)

As of the zero momentum or dual spin satellites, they are formed by two main components, the platform and the rotor. Both parts are not attached to each other but share a common axis that allows them to rotate in a totally independent way. Generally, it is the rotor the part that spins at a high rate of rotation meanwhile the platform has a much lower rate of rotation, sometimes even zero to provide gyroscopic stability. (Ketsdever 2017)

Amongst its main advantages, we may see that the despun allows a good pointing towards Earth, which results in a really good accuracy. Moreover, it is a reliable system and it can spin along its minimum moment of inertia. The disadvantages are its poor power efficiency, low manoeuvrability and its facility for being disturbed by external torques (Mortari 2005 and Ketsdever 2017)

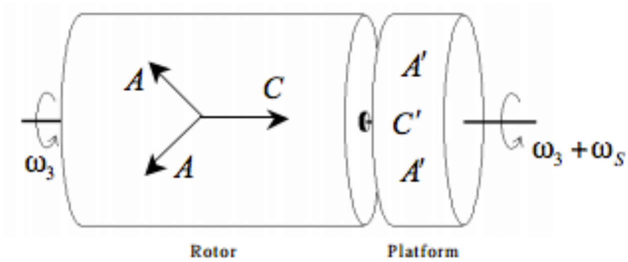

Figure 11. Sketch of a dual-spin satellite (Mortari 2005)

Finally, in the bias momentum satellites, the use of a reaction wheels provide the satellite with the necessary inertial stiffness and the 3-axes control. These type of satellites are especially important in nadir pointing satellites as they can maintain the Z axis of the spacecraft pointing towards Earth at all times. (Brown 2002)

These satellites generally present a long lifetime and their high robustness makes them less likely to be disturbed by external torques. However, it is not that good regarding pointing accuracy as zero momentum satellites. As in the other two other types of satellites, manoeuvrability is really low as well but normally roll, pitch of yaw rotations of more than 90º in an Earth observation satellite are not required. (Brown 2002)

II.IV. Reaction wheel failure in real-life space missions

In the spaceflight field, there are a countless number of missions where an actuator failure occurred, especially if a reaction wheel is involved. In this section, we will go through 2 missions as they represent different approaches when dealing with a reaction wheel failure.

II.V.I. FUSE Mission

The Far Ultraviolet Spectroscopic Explorer (FUSE) mission was launched in June 24 1999 as part of the NASA's programme on space exploration in the far-ultraviolet region. (FUSE 2014)

Even though NASA categorizes FUSE mission as a success since it helped to increase the understanding of our galaxy, there were a lot of hardware problems along its lifetime that are worth of mention. (NASA 2007)

To start with, FUSE satellite was equipped with a redundant number of four reaction wheels. Out of the total four, three of them were placed specifically for the control of each of the three axes of the satellite (X, Y, Z), meanwhile the fourth one presented a skewed angle that would allow it to take control of any of the 3 axes in case of a failure. (Blair and Calvani 2005)

What happened is that around two years after its launch, December 2001, two of the main reaction wheels (axes X and Y) in charge of the attitude control of the spacecraft stopped working. As a result, the spacecraft was held two months in safe mode as the engineers came up a new control using only two reaction wheels. (Blair and Calvani 2005)

They did manage to control the satellite with only 2 functional reaction wheels and the help of magnetic torquers that mitigated the loss of the other wheels. Furthermore it was in 2004 when the mission was again suspended as the reaction wheel responsible for the Z axis failed which led the control of the satellite reliable only in the skewed-angle reaction wheel (Blair and Calvani 2005). The mission was finally terminated by NASA on the 18th October 2007 (NASA 2007).

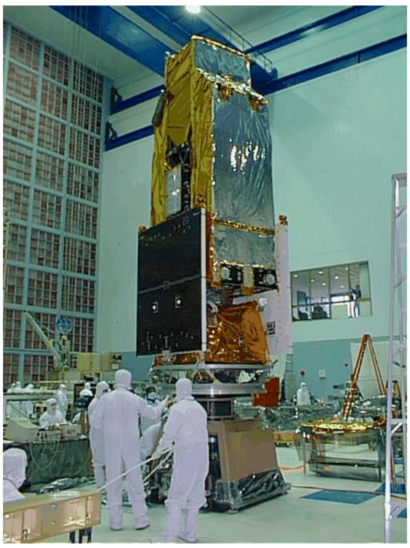

Figure 12. FUSE mission satellite (NASA 2017)

II.IV.II. UoSAT-12 mission

Another example of a reaction wheel failure is the UoSAT 12 mission. Developed by Surrey University and built by SSTL in the UK, UoSAT 12 is –as its name states– the number 12 satellite in a series of missions that was launched on the 21st April 1999. (Surrey Satellite n.d.)

UoSAT 12 was an experimental mission that had the testing of innovative technologies as its main goals. For instance, imaging cameras and a high speed S-band downlink. (Surrey Satellite n.d.)

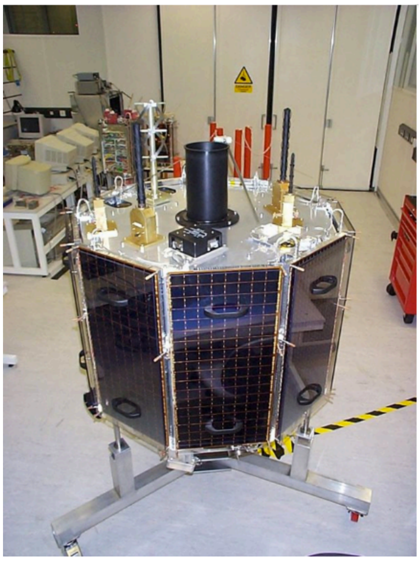

Figure 13. UoSAT-12 flight ready (Surrey Satellite Technology 2017)

The mission lasted for 4 years as it concluded in 2003 and even though it is considered a success – it even imagined a lunar eclipse – there were some problems along the way.

As it had already happened in a countless number of missions, one of the three reaction wheels in charge of the control of the aircraft failed. In contrast to what it meant to the evolution of the other missions – a total disaster – UoSAT 12 mission was still functional after the failure. (Horri & Hodgart n.d.)

In this case, it was the Z-axis the one that stopped working which left the engineers with a couple paths to follow. First, the use of magnetometers was proposed but not finally chosen, as the controllability they offer is very limited. Moreover, the use of thrusters had also been taken into account but was finally dismissed as it involves fuel consumption. (Horri & Hodgart n.d.)

Finally, in this particular mission, the full control of the 3 axes of the spacecraft was achieved with only 2 reaction wheels arranged in standard orthogonal 3-wheel configuration. (Horri & Hodgart n.d.)

II.V. Reference frames & attitude representations

In the spaceflight field, there are some different reference frames that are generally used depending the scenario. In this chapter we will go through the main four different reference frames as in this type of projects a good understanding of this matter is essential.

Among the four, the body and inertial reference frame are the most widely known ones. The first one has normally its origin located in the centre of mass of the spacecraft and presents 3 Cartesian axes and rotates with it. Moreover, it is normally used to check the alignment of the different body parts of the satellite during its assembly. (Markley, Crassidis 2014).

As of the inertial reference frame, it is really useful to work in this reference frame as the Newton Laws may be applied. Its origin is generally located in the Earth centre and even though it is not completely true that the Earth centre is inertial – Earth orbits around Sun –, it can be assumed. (Markley, Crassidis 2014).

The other two widely used reference frames are the Earth-Centered and Local-Horizontal frames that will be explained in more detail below.

II.V.I. Earth-Centered / Earth-fixed reference frame

The ECEF frame is denoted by $\{\epsilon_1, \epsilon_2, \epsilon_3\}$ as shown in figure 14. As can be seen, the Z axis is aligned with the Z axis of the inertial frame and extends through the true North. However in this case X and Y axis are not aligned with the ones in the inertial frame since X axis intersects the Earth sphere at 0º latitude and 0º longitude and the Y axis completes the right-hand rule.

Unlike in the inertial reference frame, ECEF frame rotates along with Earth and the rotation angle is called Greenwich Mean Sideral Time (GMST) angle and can be found as θ_{GMST} in figure 14.

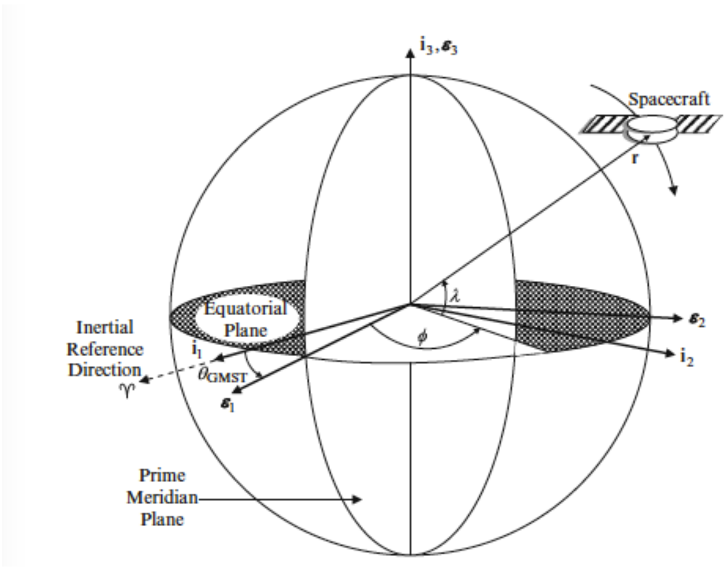

Figure 14. Representation of ECEF reference frame (Markley, Crassidis 2014)

II.V.II. Local-Vertical / Local-Horizontal Reference frame

Different from the other reference frames, LVLH reference frame is referenced to the spacecraft's orbit. As shown in Figure 17, the Z axis will be pointing towards nadir – the centre of the Earth. Moreover, its Y axis will be pointing downwards along the orbit normal and the X axis completes the right hand rule. (Markley, Crassidis 2014)

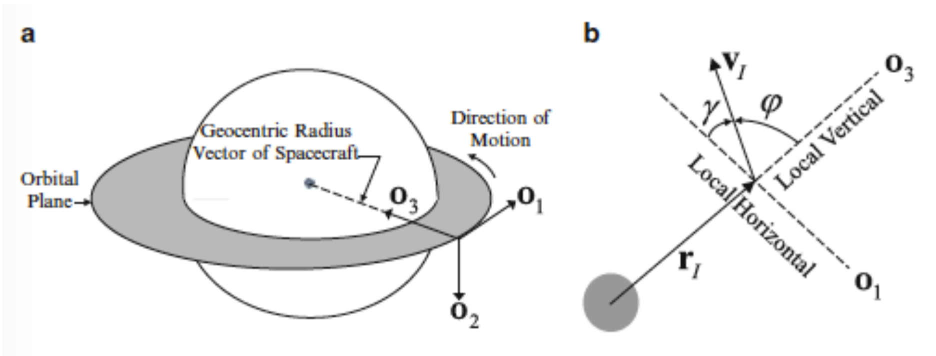

Figure 15. Representation of LVLH reference frame (Markley, Crassidis 2014)

As shown above in figure 15 as well, the angle between the X axis of the reference frame and the velocity vector of the spacecraft in the orbit is called flight path angle. Such angle is of great value in the determination field and transfer orbit manoeuvres.

II.V.III. Euler angles & quaternions

Regarding the attitude representations in the space field, Euler angles and quaternions are definitely the two main ones. First, Euler angles are generally used because they are a very intuitive way of expressing the rotation of a spacecraft and quaternions, even though they seem to be more difficult to understand and abstract at first sight, they are computationally more convenient. (De Weck 2001)

Euler angles representation is defined by three different rotations along the all three axes of a spacecraft, going from an initial reference frame I to a final reference frame F. There are intermediate reference frames in between the different angle rotations but the important thing is to know clearly from which initial reference to which final reference frame the rotation is intended to be.

The principal symbols associated with the angle rotations are the ones as follows: ϕ, θ, ψ. Moreover, the typical axis representation is ZYX, which means that Z is the first axis of rotation, Y the second and X the third one. However, the direction that we give to any of them is totally arbitrary as long as it is consistent with the right-hand rule.

In Euler angles, the only requisite is that no more than one rotation can be done along the same axis in a row. As there are 3 axis of rotation, that gives us 12 different possible rotations, 6 of them will be symmetric sets and the other 6 will be asymmetric. For simplifying the expressions, the axes of the spacecraft will be represented as 1,2,3 representing the rotations along the X,Y,Z axes respectively.

- Symmetric Euler angles set of rotations: 1-2-1, 1-3-1, 2-1-2, 2-3-2, 3-1-3, 3-2-3
- Asymmetric Euler angles set of rotations: 1-2-3, 1-3-2, 2-1-3, 2-3-1, 3-1-2, 3-2-1

For instance, a rotation matrix $A_{312}(\phi, \theta, \psi)$ would be mean that the first rotation is along Z axis of a ψ angle. Then, a rotation along X axis would be performed by an angle θ. To complete the rotation, a rotation of ϕ would be performed along axis Y.

On the other hand, quaternions are much more abstract than Euler angles. They are composed by 4 different values that are divided into a vector part and a scalar one. There are many ways to represented quaternions, but as it will be explained in the paragraph below, the way quaternions will be represented in this project is with vector part containing the values from 1 to 3 and the scalar one containing the 4th one.

Therefore, the whole quaternions vector will be represented by $\boldsymbol{q} = [q_{1:3}, q_4]^T$, where $q_{1:3} = [q_1, q_2, q_3]^T$ is the vector part and q_4 the scalar one. (Markley, Crassidis 2014).

Quaternions may be best explained by the application of Euler's Theorem as found in (de Weck 2001: 12): "The orientation of a body is uniquely specified by a vector giving the direction of a body axis and a scalar specifying the angle about the axis"

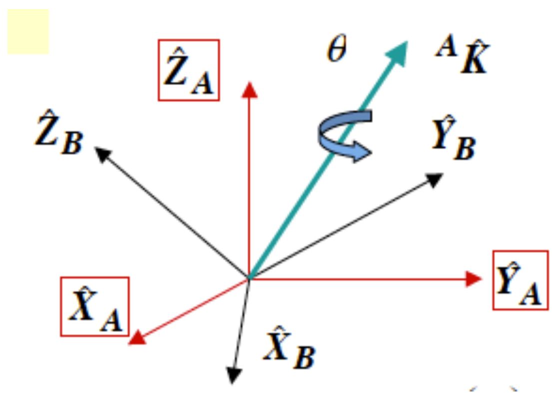

Figure 16. Quaternions representation (De Weck 2001)

II.VI. Attitude control objectives

As stated at the beginning of the project, when dealing with small satellites located in low Earth orbits, Earth observation is very often the main objective of the mission. Therefore, an accurate and precise pointing towards Earth is required for the success of the observation.

II.VI.I. Nadir pointing

In terms of observation techniques, one of the most widely used one is named nadir pointing. As its names states, the satellite's payload cameras will be located directly down towards Earth surface as seen in figure 17. Therefore, the satellites presenting the nadir pointing technique will not be focused on a specific point on Earth's surface but they will provide views of areas all around the planet (Institut für Raumfahrtsysteme 2017)

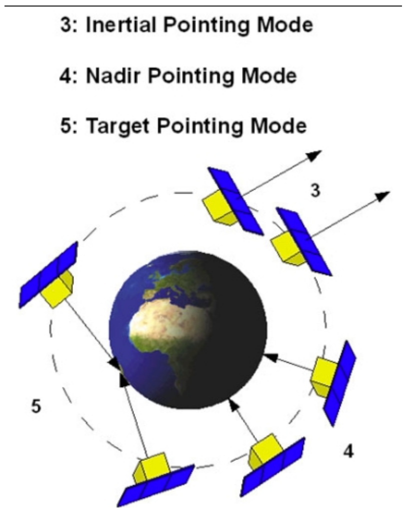

Figure 17. Nadir pointing (Institut für Raumfahrtsysteme 2017)

As of the requirements regarding the accuracy of the pointing, in figure 18 we may observe that such requirements vary depending attitude determination and control. These values were extracted from (Ketsdever 2017) and reflect the regular standards for nadir pointing Earth observation satellites.

TABLE 11-3. Typical Attitude Determination and Control Performance Requirements. Requirements need to be specified for each mode. The following lists the areas of performance frequently specified.

Area	Definition*	Examples/Comments
	DETERMINATION	
Accuracy	How well a vehicle's orientation with respect to an absolute reference is known	0.25 deg, 3 σ, all axes; may be real-time or post-processed on the ground
Range	Range of angular motion over which accuracy must be met	Any attitude within 30 deg of nadir
	CONTROL	
Accuracy	How well the vehicle attitude can be controlled with respect to a commanded direction	0.25deg,3 σ;includesdeterminationand controlerrors,maybetakenwithrespect to an inertial or Earth-fixed reference
Range	Range of angular motion over which control performance must be met	All attitudes, within 50 deg of nadir, within 20 deg of Sun
Jitter	A specified angle bound or angular rate limit on short-term, high-frequency motion	0.1 deg over 1 min, 1 deg/s,1 to 20 Hz; usually specified to keep spacecraft motion from blurring sensor data
Drift	A limit on slow, low-frequency vehicle motion. Usually expressed as angle/time.	1 deg/hr, 5 deg max. Used when vehicle may drift off target with infrequent resets (especially if actual direction is known)
Settling Time	Specifies allowed time to recover from maneuvers or upsets.	2 deg max motion, decaying to < 0.1 deg in 1 min; may be used to limit overshoot, ringing, or nutation

* Definitions vary with procuring and designing agencies, especially in details (e.g., 1 or 3 σ, amount of averaging or filtering allowed). It is always best to define exactly what is required.

Figure 18. Requirements for attitude determination and control in nadir-pointing satellites (Ketsdever 2017)

The conclusion from figure 18 is that in terms of attitude determination the standards are quite high as the accuracy is established at 0.25 degrees. Moreover, the range for the determination is quite high as well (30 degrees) which gives us the idea that nadir pointing is a very accurate and long-range system. Moreover, in the control part the standards seem to follow the same standards as the accuracy is of around 0.25 degrees and the range where standards performance have to be met is at all attitudes, within 50 degrees of nadir. (Ketsdever 2017)

III. METHODOLOGY

III.I. Spacecraft attitude kinematics

The main goal of the modelling of the attitude kinematics of a spacecraft is calculating the attitude position of the satellite at any specific instant in time. In this project, we will stick with numerically solving the differential equations that model such kinematics for obtaining both the Euler angles and quaternions along time. Therefore, if we are able to know how a spacecraft is rotated and its previous evolution in time, we are much closer to being able to stabilize and control it.

III.I.I. Modelling with Euler angles

As shown in (Sidi 1997), the kinematics equations of a non-spinning satellite depend on the type of rotation applied in each case. Therefore, such equations will not be equal if making a Euler angle rotation of 3-2-1 or a Euler rotation of 2-3-1 – for each particular rotation, the equations are unique.

As stated in (Costello 2016), the most common rotation in the spaceflight field for going from the body reference frame to the inertial is to perform a 3-1-2 rotation. Therefore, the rate of Euler angles that result from such rotation are the ones as follows as shown in (Costello 2016).

$$\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} c_\theta & 0 & s_\theta \\ s_\theta t_\phi & 1 & -c_\theta t_\phi \\ -s_\theta / c_\phi & 0 & c_\theta / c_\phi \end{bmatrix} \begin{bmatrix} w_x \\ w_y \\ w_z \end{bmatrix} \tag{1}$$

Only by choosing some initial conditions for the Euler angles, these three non-linear differential equations allows us to calculate the values of the three Euler angles along time.

III.I.II. Modelling with quaternions

In the case of working with quaternions as our attitude representation, there are two main ways to obtain them. First, it is possible to use the modelling with Euler angles and then calculate the quaternions from the values of the angles obtained in previous equation (1).

However, there are some differential equations that, if solved, can directly provide us with the values of the quaternions and that simplifies considerably the process as no conversion from one attitude representation to another is required – the probability of making an error is also reduced.

As stated before, the differential equation (2) provides us with the values of the quaternions along time. In case of working with these equations, as long as some initial conditions **q**(0) are chosen as inputs, the values of the quaternions can be numerically calculated. (Sidi 1997)

$$\frac{d}{dt}\boldsymbol{q} = \frac{1}{2}[\boldsymbol{\Omega}']\boldsymbol{q} \tag{2}$$

$$[\boldsymbol{\Omega}'] = \begin{bmatrix} 0 & w_z & -w_y & w_x \\ -w_z & 0 & w_x & w_y \\ w_y & -w_x & 0 & w_z \\ -w_x & -w_y & -w_z & 0 \end{bmatrix} \tag{3}$$

In case we are working with an inertial pointing satellite, it will be the omega matrix shown in equation (3) the one used. However, in the case we are working with a nadir pointing satellite, in other words, a satellite that is always pointing towards the centre of the Earth, some modifications need to be included in such matrix.

$$[\boldsymbol{\Omega}'_{\boldsymbol{orb}}] = \begin{bmatrix} 0 & w_{orb_z} & -w_{orb_y} & w_{orb_x} \\ -w_{orb_z} & 0 & w_{orb_x} & w_{orb_y} \\ w_{orb_y} & -w_{orb_x} & 0 & w_{orb_z} \\ -w_{orb_x} & -w_{orb_y} & -w_{orb_z} & 0 \end{bmatrix} \tag{4}$$

As seen in equation (4), the only difference with respect to the former matrix is that the angular velocities in it are not the inertial ones anymore. The new matrix is formed by the orbital angular velocities of the satellite that can be obtained as in equation (5). (Sidi 1997)

$$\boldsymbol{w_{orb}} = \boldsymbol{w_I} + R_{quat}\begin{bmatrix} 0 \\ -w_0 \\ 0 \end{bmatrix} \tag{5}$$

As we will see later, "R_q" stands for a rotation matrix and "w_0" represents the angular rate around Earth of the satellite as shown in equation (6). Note that we are considering the period of our small satellite to be of 6,000 seconds – around 100 minutes. That value lies in the range of 88-127 minutes stated in the Literature Review section of what the period of a satellite placed in a low Earth orbit should be, therefore it is consistent with our research.

$$w_0 = \frac{2\pi}{P} = \frac{2\pi}{6000} \approx 0.001047\, rad/s \tag{6}$$

As stated in (Sidi 1997), the vast majority of controllers that work in the orbit frame also work in the inertial one. Therefore, all of the equations will be expressed in terms of the inertial frame as they can always be extrapolated to the orbit frame (e.g. nadir pointing satellites).

III.I.III. Conversion from quaternions to Euler angles

As stated before, quaternions are a way of representing the attitude of a spacecraft that is abstract and not very comprehensive. For such reason, the importance of being able to convert from quaternions to Euler angles is high.

Moreover, the difficulty for doing such conversion is not high since the only thing to do is comparing two different rotation matrices, one relative to the Euler angles rotation and the other corresponding to the quaternion rotation.

Therefore, if rotating from the inertial reference frame to the body reference frame, we can use the following quaternion rotation matrix found in (Wikipedia 2017)

$$R_{quat} = \begin{bmatrix} 1-2q_2^2-2q_3^2 & 2(q_1q_2-q_3q_4) & 2(q_1q_3+q_2q_4) \\ 2(q_1q_2+q_3q_4) & 1-2q_1^2-2q_3^2 & 2(q_2q_3-q_1q_4) \\ 2(q_1q_3-q_2q_4) & 2(q_2q_3+q_1q_4) & 1-2q_1^2-2q_2^2 \end{bmatrix} \tag{7}$$

Therefore, the only thing that we will need to do is first to substitute the corresponding values of q1, q2, q3 and q4 in the quaternion matrix rotation and then compare it with our 3-1-2 Euler angle rotation.

In the 3-1-2 case, the Euler angle rotation matrix from the inertial reference frame to the body reference one will be as stated in (Costello 2016).

$$\begin{bmatrix} I_B \\ J_B \\ K_B \end{bmatrix} = \begin{bmatrix} c_\theta c_\psi - s_\psi s_\theta s_\phi & c_\theta s_\psi + s_\theta s_\phi c_\psi & -s_\theta c_\phi \\ -c_\phi s_\psi & c_\phi c_\psi & s_\phi \\ s_\theta c_\psi + c_\theta s_\phi s_\psi & s_\theta s_\psi - c_\theta s_\phi c_\psi & c_\phi c_\theta \end{bmatrix} \begin{bmatrix} I_I \\ J_I \\ K_I \end{bmatrix} \tag{8}$$

Therefore, comparing both rotation matrices, we will obtain nine different equalities that allow the conversion from attitude representation to another. However, as we only need three of them, we will pick the less complex ones to keep the calculations simple.

$$s_\phi = 2(q_2 q_3 - q_1 q_4) \rightarrow \ \phi = \mathrm{asin}\big(2(q_2 q_3 - q_1 q_4)\big) \tag{9}$$

$$-c_\phi s_\psi = 2(q_1 q_2 + q_3 q_4) \rightarrow \ s_\psi = \frac{-2(q_1 q_2 + q_3 q_4)}{c_\phi} \rightarrow \ \psi = \mathrm{asin}\left(\frac{-2(q_1 q_2 + q_3 q_4)}{c_\phi}\right) \tag{10}$$

$$-s_\theta c_\phi = 2(q_1 q_3 + q_2 q_4) \rightarrow s_\theta = -\frac{2(q_1 q_3 + q_2 q_4)}{c_\phi} \rightarrow \ \theta = \mathrm{asin}\left(-\frac{2(q_1 q_3 - q_2 q_4)}{c_\phi}\right) \tag{11}$$

From such expressions, it is fairly simple to obtain the values of the Euler angles of the spacecraft at any time. This will result to be very helpful when trying to understand and analyse the data obtained in the numerical analysis.

III.II. Spacecraft attitude dynamics

Regarding the attitude dynamics of a spacecraft in the torque-free motion scenario, the main objective is to calculate its angular velocities along time.

As shown in (Sidi 1997), the general equation that will model such attitude dynamics is the one following.

$$\boldsymbol{M} = \dot{\boldsymbol{h}}_{\boldsymbol{I}} = \dot{\boldsymbol{h}}_{\boldsymbol{B}} + \boldsymbol{w} \times \boldsymbol{h} \tag{12}$$

What this expression implies is that the moment on the spacecraft will equal the time derivative of its angular moment in the inertial frame. However, the inertia matrix is always expressed with respect to the body frame. Therefore, there are two choices: either we apply Steiner and transport the inertia matrix to the inertial frame; or as in (Sidi 1997), we apply the derivative transport theorem that allow us to make the time derivative in the body reference frame plus the addition of the vectorial product of angular velocity between inertial and body frames and of the angular moment.

Generally the path followed in (Sidi 1997) is widely used as the derivative transport theorem is one of the most powerful theorems when it comes to simplifying equations.

Taking some assumptions that will be explained below, the result of equation (12) will be the Euler moment equations that will provide us with the evolution in time of the angular velocities of the body. The following equations have been obtained from (Sidi 1997) and (Kaplan 1976).

$$\begin{aligned} M_x &= I_x \dot{w}_x + w_y w_z (I_z - I_y) \\ M_y &= I_y \dot{w}_y + w_x w_z (I_x - I_z) \\ M_z &= I_z \dot{w}_z + w_x w_y (I_y - I_x) \end{aligned} \tag{13}$$

The main assumption we are taking in these equations is that the only non-zero term of the inertia matrix of the spacecraft are the principal moments of inertia. Therefore we will be working with an inertia matrix like the following one.

$$[I] = \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix} \tag{14}$$

In the case where no external moments are applied to the spacecraft ($M_x = M_y = M_z = 0$), equation (14) will remain as shown below in equation (15).

$$\begin{aligned} 0 &= I_x \dot{w}_x + w_y w_x (I_z - I_y) \\ 0 &= I_y \dot{w}_y + w_x w_x (I_x - I_z) \\ 0 &= I_z \dot{w}_z + w_x w_y (I_y - I_x) \end{aligned} \tag{15}$$

III.III. Principal attitude control theory with reaction wheels

As shown before, the equation that model the attitude dynamics of the spacecraft is the Euler moment equation that, if assuming no external moments ($\boldsymbol{M} = 0$) and developing the terms in it, might be expressed as in equation (16).

$$I\,\dot{\boldsymbol{w}} = -\boldsymbol{w} \times (I\,\boldsymbol{w}) \tag{16}$$

For the control of the spacecraft with reaction wheels, two terms will need to be included into equation (16). The most important one, located in the right hand side of the equation, represents the control torque applied by the reaction wheels in order to stabilise the satellite. The integral of such control torque is also included into the second term of the vectorial product, as shown in equation (17).

$$I\,\dot{\boldsymbol{w}} = -\boldsymbol{w} \times (I\,\boldsymbol{w} + \boldsymbol{h}) - \dot{\boldsymbol{h}} \tag{17}$$

For showing how to obtain the final expression of the Euler moment equations with the control of reaction wheels, it is first required to express the final value of that complex vectorial product.

$$-\boldsymbol{w} \times (I\,\boldsymbol{w} + \boldsymbol{h}) = \begin{bmatrix} w_y(I_z w_z + h_z) - w_z(I_y w_y + h_y) \\ w_z(I_x w_x + h_x) - w_x(I_z w_z + h_z) \\ w_x(I_y w_y + h_y) - w_y(I_x w_x + h_x) \end{bmatrix} \tag{18}$$

Therefore, deriving equation (17) for the angular accelerations of the spacecraft and substituting equation (18) in it, it is possible to obtain the expressions for the quaternion feedback control with the help of reaction wheels.

$$\begin{aligned} \dot{w_x} &= \left[w_y(I_z w_z + h_z) - w_z(I_y w_y + h_y) - \dot{h_x}\right]/I_x \\ \dot{w_y} &= \left[w_z(I_x w_x + h_x) - w_x(I_z w_z + h_z) - \dot{h_y}\right]/I_y \\ \dot{w_z} &= \left[w_x(I_y w_y + h_y) - w_y(I_x w_x + h_x) - \dot{h_z}\right]/I_z \end{aligned} \tag{19}$$

At this point, the key is to correctly define the quaternion feedback control term that, as its name states, will be the one in charge of the control of the spacecraft.

As mentioned in (Sidi 1997), the term ($-\dot{h}$) can be modelled as follows with k_d & k_p representing the control gains. Note that with the method followed in this project, it is mandatory to work with quaternions as our attitude representation. However, as explained in previous sections, they can always be converted into Euler angles for a more comprehensive representation of the orientation of the satellite.

$$-\dot{\boldsymbol{h}} = \boldsymbol{N} = -k_p\,I\,\boldsymbol{q_{1:3}} - k_d\,I\,\boldsymbol{w} \tag{20}$$

As may be seen from equation (20) above, the quaternion in the controller will only contain its vector part (q1, q2, q3) to make the index of the vector equation consistent. The rest of the variables have already been mentioned as "I" stands for the matrix inertia of the spacecraft and "$\boldsymbol{w}$" represents the angular velocities of the spacecraft in the inertial reference frame.

Therefore, if expanding equation (20) and substituting it into equation (19), it is possible to obtain the Euler moment equations that will be used for the Matlab simulations. Then:

$$\begin{aligned} \dot{w_x} &= \left[w_y(I_z w_z + h_z) - w_z\left(I_y w_y + h_x\right) - k_p\, I_x\, q_1 - k_d\, I_x\, w_x\right]/I_x \\ \dot{w_y} &= \left[w_z(I_x w_x + h_x) - w_x\left(I_z w_z + h_y\right) - k_p\, I_y\, q_2 - k_d\, I_y\, w_y\right]/I_y \\ \dot{w_z} &= \left[w_x\left(I_y w_y + h_y\right) - w_y(I_x w_x + h_z) - k_p\, I_z\, q_3 - k_d\, I_z\, w_z\right]/I_z \end{aligned} \tag{21}$$

Finally, regarding the values of the control gains, it has to be stated that the higher the gain the better the control will be as the angular velocities of the satellite will converge to 0 in a lower amount of time. However, there are values for the maximum torque that the reaction wheels can induce which makes the gain not to be infinite. As in (Sidi 1997), the maximum torque considered for thrusters and reaction wheels is of around $N_{MAX} \approx 0.02\ N \cdot m$. Therefore, the values for the gains that will be used in the numerical simulation will be of $k_p = 0.002$ & $k_d = 0.036$.

Finally, in the case it is needed to work in the orbit frame, equation (28) shown below would need to be employed.

$$\begin{aligned} \dot{w_x} &= \left[w_y(I_z w_z + h_z) - w_z\left(I_y w_y + h_x\right) - k_p\, I_x\, q_1 - k_d\, I_x\, w_{orb_x}\right]/I_x \\ \dot{w_y} &= \left[w_z(I_x w_x + h_x) - w_x\left(I_z w_z + h_y\right) - k_p\, I_y\, q_2 - k_d\, I_y\, w_{\_orb_y}\right]/I_y \\ \dot{w_z} &= \left[w_x\left(I_y w_y + h_y\right) - w_y(I_x w_x + h_z) - k_p\, I_z\, q_3 - k_d\, I_z\, w_{orb_z}\right]/I_z \end{aligned} \tag{22}$$

III.IV. Mitigation of a reaction wheel failure with magnetic torquers

Even though magnetic torquers are considered as having limited torque capability (Horri n.d.), the use of magnetic torquers as a backup plan when there is an actuator failure is widely used.

In this section, the equations that model the control of the satellite with the help of magnetic torquers will be presented.

As stated before, in the scenario of a reaction wheel failure, the torque it induces becomes 0. For instance, let's assume that it is the wheel in charge of the control of the Z axis the one that fails. In such case: $N_{wheel_z} = 0$.

Therefore, as stated in (Chen n.d.), the control torque induced by a magnetic torquer may be expressed as in equation (23).

$$\boldsymbol{N_m} = \boldsymbol{M} \times \boldsymbol{B} \tag{23}$$

From such equation, two main variables may be identified – the momentum of the torquer and the Earth magnetic field.

As of the Earth magnetic field, we will use the expression found also in (Chen n.d.) that corresponds to the dipole model.

$$B = \begin{bmatrix} 23\cos(wt) \\ 2.4 \\ 46\sin(wt) \end{bmatrix} \tag{24}$$

Moreover, the control for the momentum of the torquer will be modelled as presented in (Chen n.d.). In the equation below, vector "e" stands for the error vector of the satellite and "k_m" for the control gain of the magnetic torquer control.

$$\boldsymbol{M} = -k_m \frac{\boldsymbol{e} \times \boldsymbol{B}}{\|\boldsymbol{B}\|} \tag{25}$$

It is important to note that the magnetic control gain has been given a value of $k_m = 10^4$ that provide sensible control torques within the specified ranges: $N_m < 0.005\ N \cdot m$.

Finally, the error vector may be expressed similar as in equation (20), which it basically represents the control torque required to apply by the magnetic torquer to the satellite in order to stabilise it. Therefore, it is modelled as shown below where $k_{p_{mag}}$ & $k_{d_{mag}}$ represent the quaternion and angular velocities control gains respectively for the magnetic torquer. These gains will present the same values as the ones used in the reaction wheel controlled case.

$$\boldsymbol{e} = -k_{p_{mag}}\, I\, \boldsymbol{q_{1:3}} - k_{d_{mag}}\, I\, \boldsymbol{w} \tag{26}$$

III.V. Mitigation of a reaction wheel failure with thrusters

Apart from magnetic torquers, thrusters are actuators that are also employed as a replacement when a reaction wheel stops working. The equations that model the control of a spacecraft with the help of thrusters will be reviewed in this section.

First, it is important to state that the control theory that involves the use of thrusters presents very similar equations to the ones used for the control modelling with reaction wheels.

Therefore, the main equation for the control with thrusters is represented below, which is a modification of the original Euler moment equation.

$$I\,\dot{\boldsymbol{w}} = -\boldsymbol{w} \times (I\,\boldsymbol{w}) - \dot{\boldsymbol{h}} \tag{27}$$

From such equation, it may be seen that the only modification presented is the addition (right hand side of the equation) of the control torque that the thruster applies to the satellite in order to stabilise it.

As all the terms in equation (27) have been already explained, the next step would be to express such equation for the particular case of our satellite. After doing so, equation (28) results.

$$\begin{bmatrix} I_x \dot{w}_x \\ I_y \dot{w}_y \\ I_z \dot{w}_z \end{bmatrix} = -\begin{bmatrix} w_x \\ w_y \\ w_z \end{bmatrix} \times \left(\begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix} \begin{bmatrix} w_x \\ w_y \\ w_z \end{bmatrix} \right) - k_p \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix} \begin{bmatrix} q_1 \\ q_2 \\ q_3 \end{bmatrix} - k_d \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix} \begin{bmatrix} w_x \\ w_y \\ w_z \end{bmatrix} \tag{28}$$

Furthermore, if deriving the rate of angular velocities from the equation above, it is possible to obtain the equations that would model the dynamics of the satellite when a thrusters controller has been added.

$$\begin{aligned} \dot{w}_x &= \left[w_y w_z \left(I_y - I_z\right) - k_p\, I_x\, q_1 - k_d\, I_x\, w_x\right]/I_x \\ \dot{w}_y &= \left[w_x w_z (I_z - I_x) - k_p\, I_y\, q_2 - k_d\, I_y\, w_y\right]/I_y \\ \dot{w}_z &= \left[w_x w_y \left(I_x - I_y\right) - k_p\, I_z\, q_3 - k_d\, I_z\, w_z\right]/I_z \end{aligned} \tag{29}$$

Finally, as stated in (Sidi 1997), both thrusters and magnetic torquers present very similar torque capability. In other words, the maximum that both actuators can produce is practically the same. For such reason, the values for the control gains used in the thrusters controls were the same as the ones presented in the reaction wheels control section. Consequently, $k_p = 0.002$ & $k_d = 0.036$.

III.VI. Introduction to MATLAB

In this project, the analytical software Matlab will be a pivotal part, as we will rely on it for the modelling of the attitude kinematics and dynamics of the satellite. Moreover, it is also responsible for the numerical simulations that will be performed for the analysis of the control of the spacecraft in different scenarios.

Matlab will provide us with the evolution in time of some variables of great interest for the course of the project. For instance, the angular velocities of the satellite, its Euler angles and quaternions or the control torques induced by reaction wheels, magnetic torquers and thrusters are some of the variables that will be simulated with the software. The way to do so for Matlab will be by numerically solving a set of non-linear differential equations.

Even though such equations have been analytically solved by scientists along the history (elliptic integrals), they do not provide representative results in case there are disturbances. Therefore, it is just simpler and more accurate to numerically solve the non-linear differential equations with Matlab.

When working with the software, 2 different scripts will be needed. Therefore, we will encounter with the main script and a function script. On one hand, as of the main script, it will contain the values of all the variables that need to be analysed. Moreover, it also contains the plots of such variables against time for analysing their evolution. Additionally, the conversion from quaternions to Euler angles is also performed in the main script.

On the other hand, the use of the function 'ode45' in the main script makes it mandatory for us to build an extra function containing all the non-linear differential equations that model the dynamics of the spacecraft. As said, such function will contain the modelling of the dynamics of the satellite, the quaternions and the modelling of the control with thrusters, reaction wheels or magnetic torquers.

It should also be mentioned that the numerical analysis could have been also performed with Simulink that is another powerful software.

IV. Numerical simulation results & analysis

In this section of the project, we will present all the different numerical simulation results obtained with Matlab. Generally for every different case, such results will consist of two different graphs.

The first one will contain 6 subplots. Such subplots will correspond to the evolution in time of the 3 angular velocities of the satellite and its three Euler angles – roll, pitch and yaw. Therefore, we will able to both study the dynamics and kinematics of the satellite and assess its stabilization and control.

On the other hand, the second graph will show the control torques applied by the different actuators in all three axes for stabilizing the satellite.

Furthermore, as we are working with non-linear differential equations, the initial conditions of the variables will need to be chosen. For every different case, such initial conditions will be stated for a better comprehension of the simulation results.

Finally, the principal moments of inertia of the satellite we are working with are as follows. These values are applied for all the different study cases.

$$I_x = 19\ kg \cdot m^2; I_y = 19.5\ kg \cdot m^2; I_z = 12\ kg \cdot m^2 \tag{30}$$

IV.I. Torque-free motion

In this initial case scenario, a satellite in free torque-free motion will be simulated. In this type of study, no control of the spacecraft is carried out which will make the satellite to not be stabilized.

Regarding the initial conditions of the angular velocities, we made non-zero in order to see the behaviour of the satellite – if they were 0, the angular velocities would remain as 0. Moreover, the values of the quaternions from 1 to 3 were given a value of 0.1 and then the last one was assigned the corresponding value for keeping the module of the quaternions vector with a value of 1.

$$w_x(0) = w_y(0) = w_z(0) = 0.1\ rad/s \tag{31}$$

$$q_1(0) = q_2(0) = q_3(0) = 0.1\ ; q_4(0) = \sqrt{(1 - 0.03)} = \sqrt{0.97} \tag{32}$$

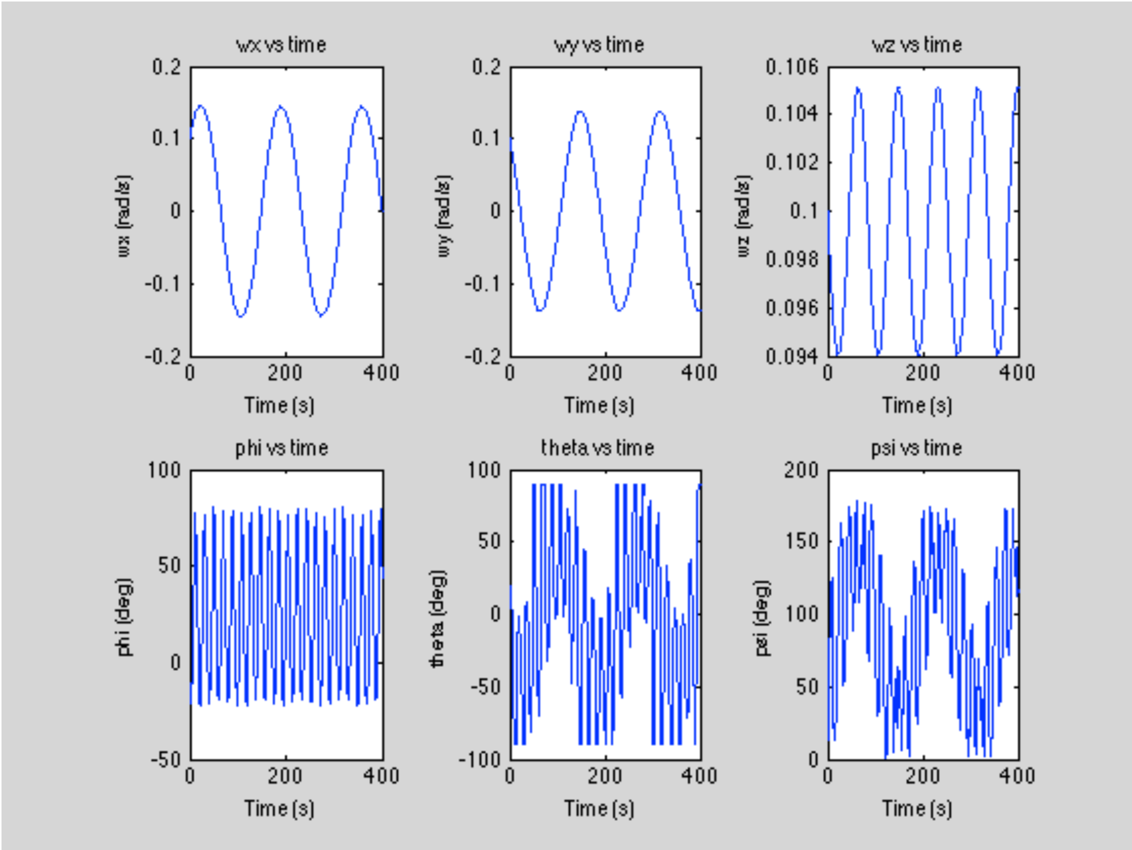

Figure 19. Free-torque satellite numerical simulation results

The main conclusion that may be obtained from the simulation results is that the satellite is neither stabilized nor controlled – as expected. We may see how the angular velocities along the three different axes of rotation do not converge to 0, instead they oscillate with constant amplitude. Because of that fact, we can conclude that both the angular momentum and energy are conserved.

Moreover, we may also see that both the maximum amplitude of the angular velocity in the X and Y axes is really similar as their principal moments of inertia in those axes present almost the same value.

Furthermore, we may see that the Euler angles also oscillate in time as a result of the non stabilization of the spacecraft. Obviously, if the satellite is always turning (as it is in this case), its pitch yaw and roll angles will not converge to any value.

IV.II. Motion controlled with reaction wheels

As explained in previous sections of the project, there are quite a few different actuators that may be used for controlling the orientation of the satellite in orbit.

In this project, a study of a 3 axes reaction wheel controlled satellite will be performed first. Then, it will be analysed which other actuators best replace a reaction wheel when it stops working properly.

Regarding the pointing of the satellite when in orbit, there are two main ways as shown in figure 17. An analysis of the inertial pointing will be performed first which will lead to the analysis of the nadir pointing case afterwards.

Regarding the initial conditions for the quaternions, they will be same ones as used for the torque-free motion case (equation 32). However, in this scenario the initial angular angular velocities will not be non-zero anymore. These initial conditions shown in equation (33) apply for all the simulation shown below unless stated otherwise.

$$w_x(0) = w_y(0) = w_z(0) = 0\, rad/s \tag{33}$$

IV.II.I. Inertial pointing case

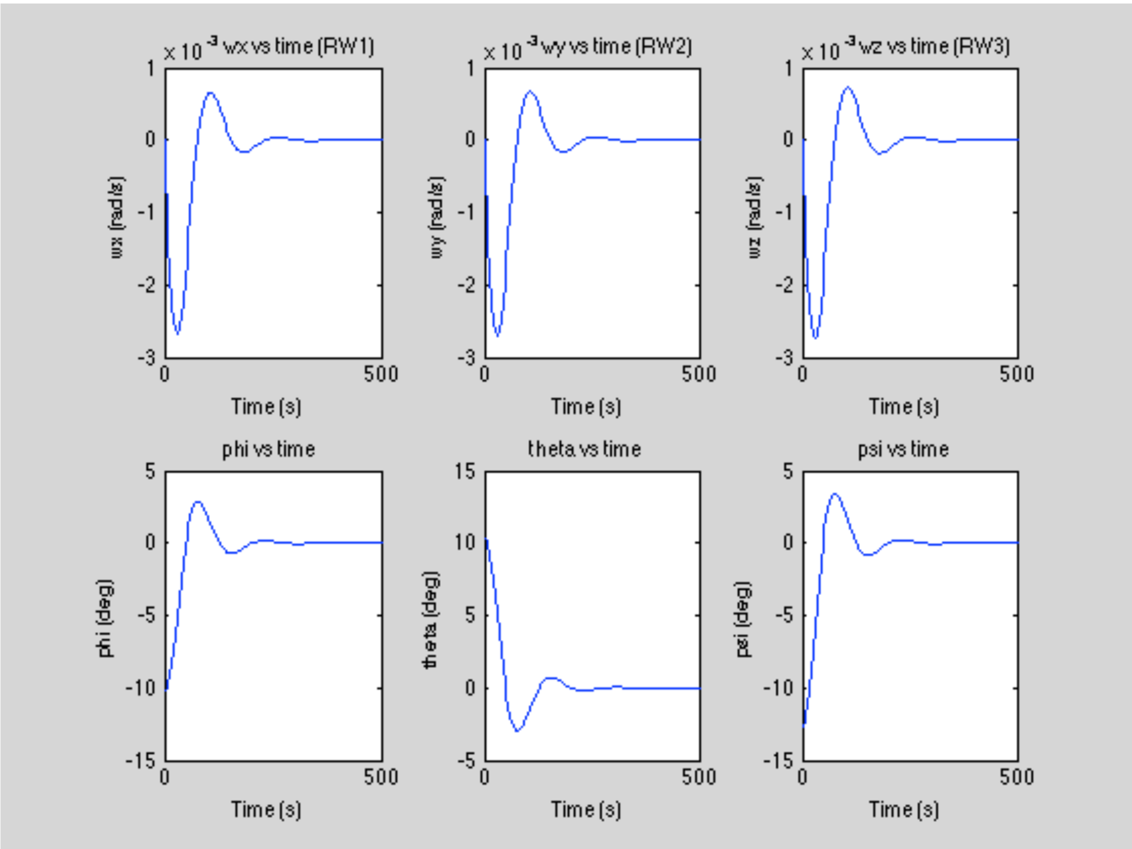

Figure 20. Angular velocities & Euler angles inertial pointing reaction wheel controlled satellite

Unlike in the torque-free motion, we may see here how stabilized and controlled the satellite is now. First, all the angular velocities converge to zero –which means no rotation and therefore stabilization – and so do the Euler angles. The Euler angles also converging to zero means that satellite returns to the initial attitude position, or in other words that it tends to the required rotated position.

The time of convergence of the angular velocities depends on the values of the control gains. In this case, the convergence time is of around 300 seconds, which represent 5 minutes and a not very high value.

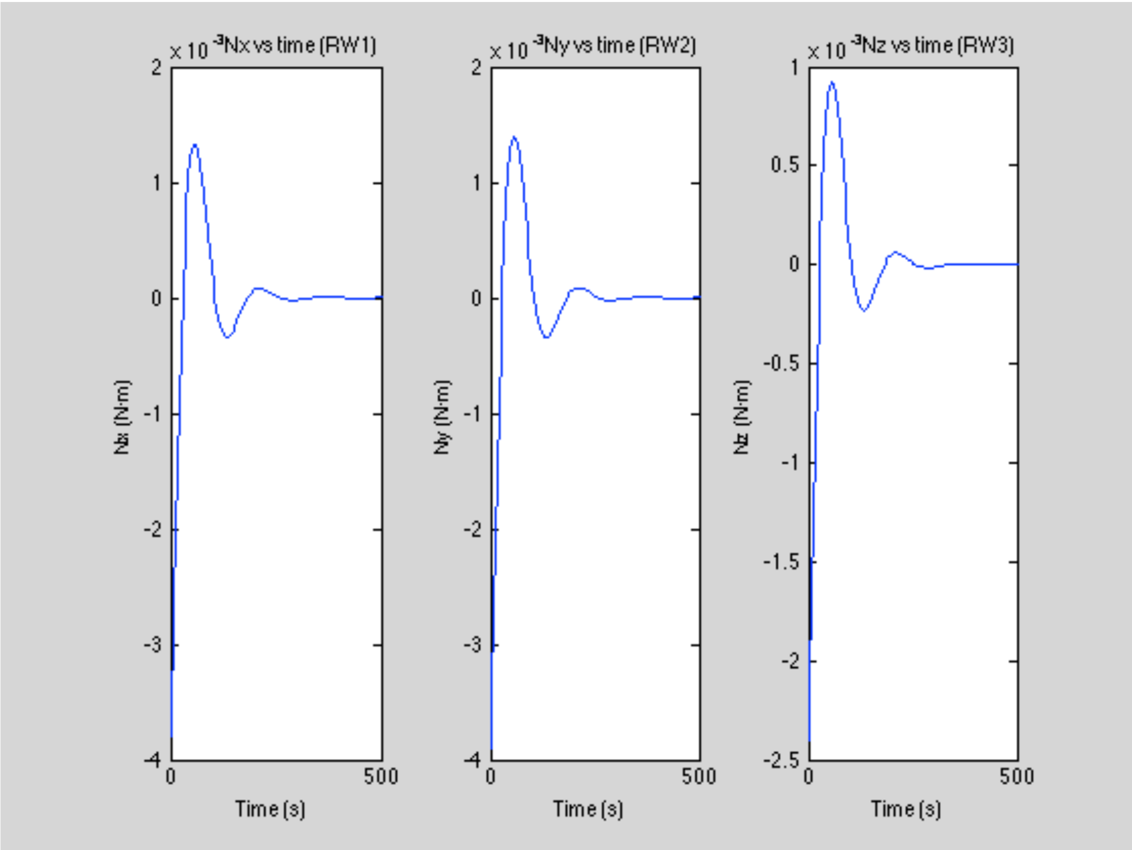

Figure 21. Control torque applied by reaction wheels in inertial pointing satellite

Moreover, figure 21 represents the different control torques applied from the reaction wheels to the satellite. First, it is important to note how the tendency of the control torques is equal in all 3 axes which shows that all 3 reaction wheels have been modelled equally and as expected.

Furthermore, it is also important to observe that the values of the control torques applied by the reaction wheels are similar in the X and Y axes but smaller in the Z axis. Such results represent what it could have been expected as the moment of inertia of the satellite in the Z axis is lower than in the other two. Then, that makes the control torque needed to gain control of such axis to be smaller since less "effort" is required to stabilize it.

IV.II.II. Nadir pointing case

Regarding the nadir pointing, it is important to mention that almost all the initial conditions stated for the inertial pointing will be replicated in order to be able to make a fair analysis between both situations. The only initial condition that will vary will be the angular velocity in the Y axis as now it is not required to converge to 0 but to such initial value. The initial condition for the angular velocity is shown in equation (6) that represents the rate at which our small satellite rotates around Earth. Therefore:

$$w_y = w_0 = \frac{2\pi}{P} = \frac{2\pi}{6{,}000} = 0.001047\ rad/s$$

In such case, the results obtained are shown in figures 22 and 23 below.

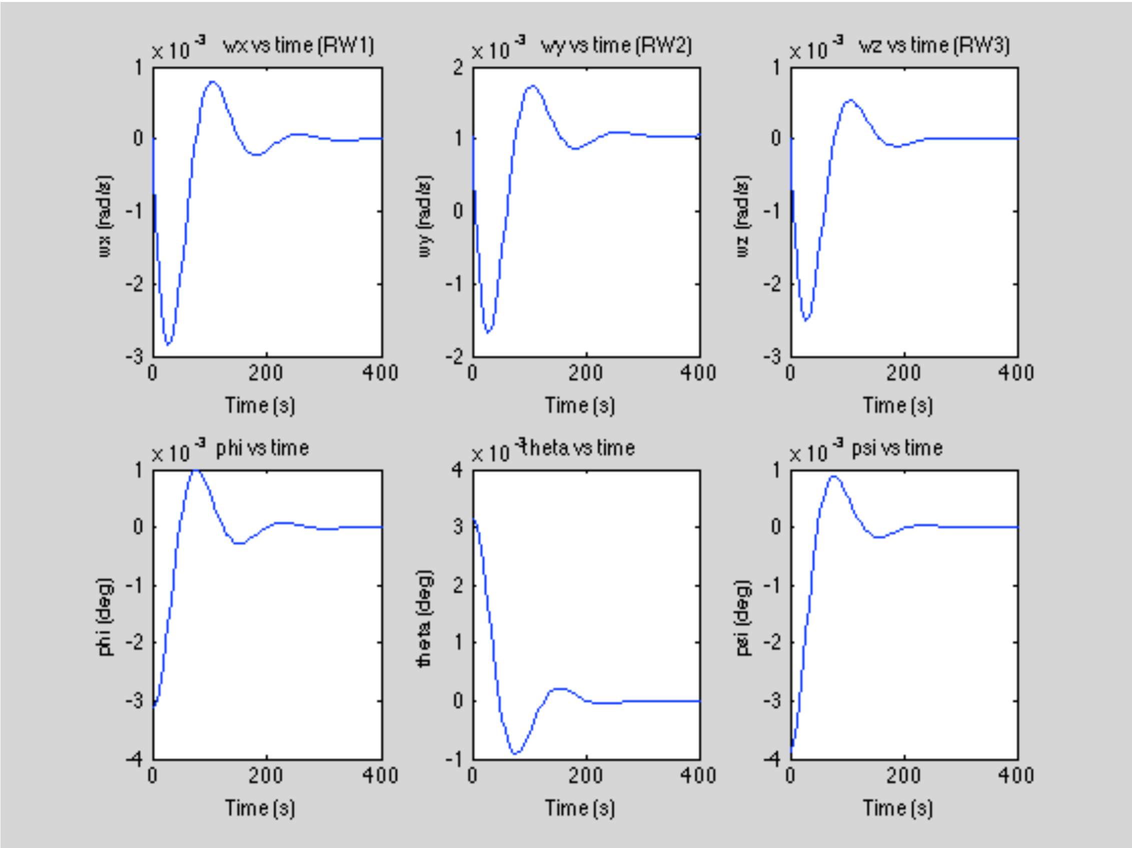

Figure 22. Angular velocities & Euler angles nadir pointing reaction wheel controlled satellite

From figure 22 it can be observed that, as expected, the angular velocity in the Y axis does converge to its new initial value, making the satellite to be pointing towards the centre of Earth at all times as shown in figure 17.

However, regarding the rest of variables it can be stated that nothing has changed with respect to the inertial pointing scenario. For instance, the angular velocities in the X and Z axes still converge to 0. The angular velocity in the Z axis presents lower amplitude due to the lower moment of inertia of the satellite in that axis.

Moreover, the converging time still remains at around 300 seconds; the Euler angles also converge to 0 as in the inertial pointing case.

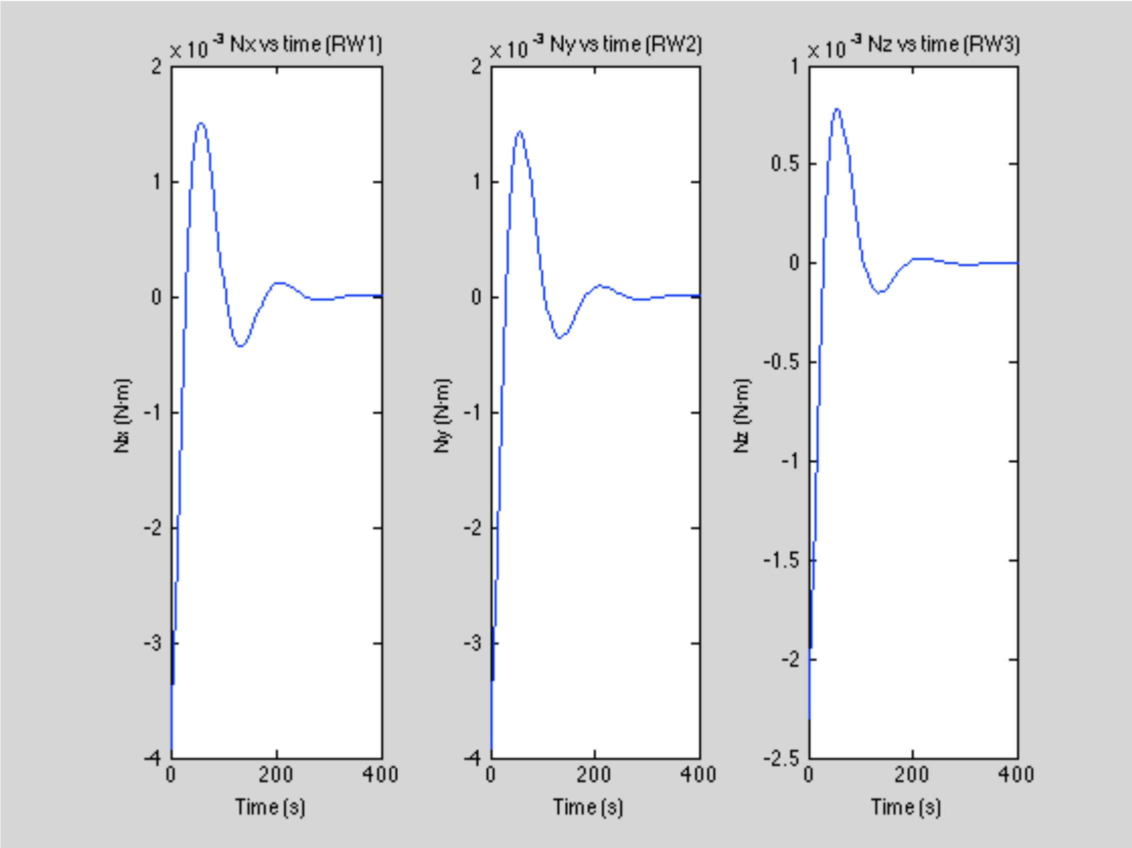

Figure 23. Control torque applied by reaction wheels in nadir pointing satellite

The control torques in all 3 axes present a similar converging time and they tend to zero. These results are what it could have been expected and are the same as the ones obtained in the inertial pointing case.

IV.III. Effect Failure Analysis

As it has been mentioned in the project already, the probability of experimenting a reaction wheel failure is higher than desired if taking into account past missions' experiences.

Therefore, it would advisable to simulate a reaction wheel failure in one of the axes of the satellite to estimate the severity of such failure.

For the purpose of this project, a failure in one axis was only needed but in case it applies, multiple failures may be simulated. As will be seen below, the failure will be simulated in the reaction wheel located in the Z-axis.

It is also important to note that for the reaction wheel simulation, the satellite presents an inertial pointing again and that the initial conditions will be as shown in equations (32) and (33).

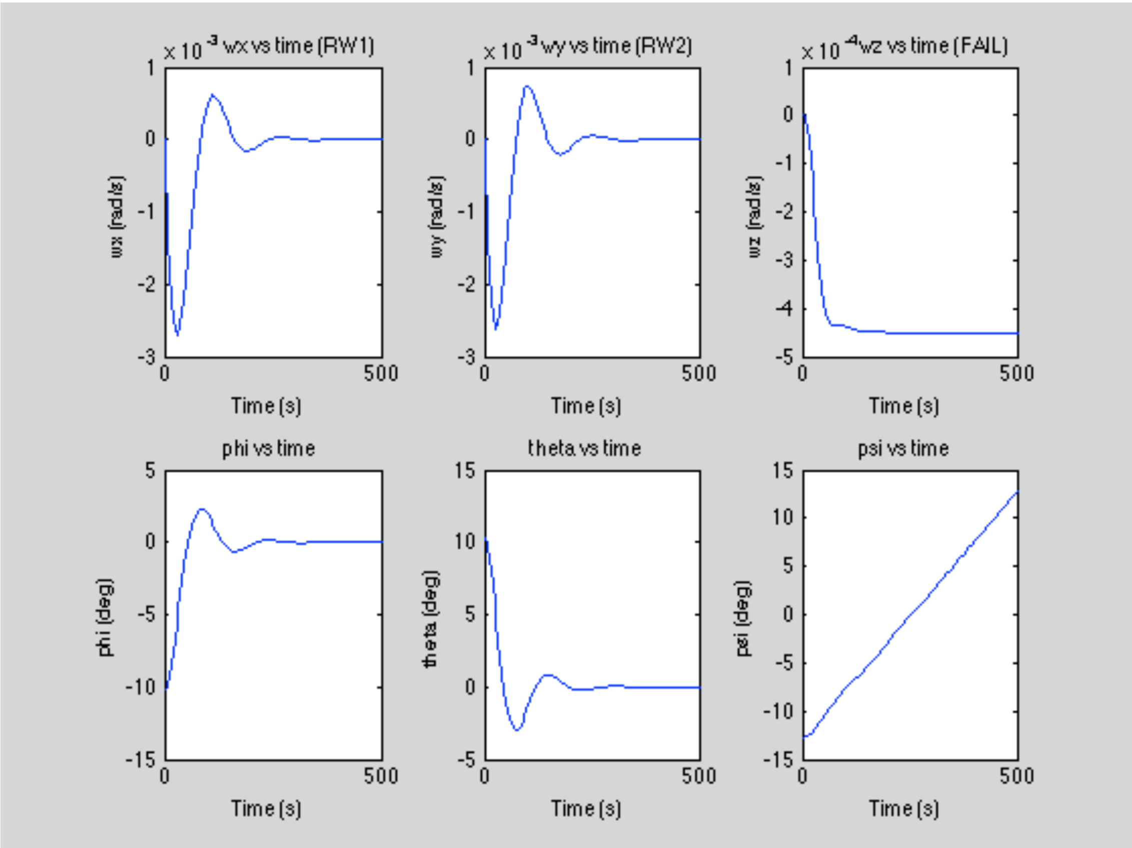

Figure 24. Angular velocities & Euler angles failure in reaction wheel Z axis

From the graph, it might be concluded that a reaction wheel failure definitely affects the stability and control of any spacecraft.

The simulated failure was meant to affect axis Z of the satellite and looking at the results, it is clear that the satellite is out of control in such Z axis. Both X and Y axes present numerical results almost equal (also the convergence time is really similar) to the ones obtained in the previous sections and have not been affected by the failure.

We may observe how the angular velocity in the Z axis does not converge to 0 at any time but instead it stabilizes at a rate of $-4 \cdot 10^{-4}\ rad/s$, which means that the satellite is not controlled. Moreover, yaw angle is also not converging to 0 and keeps increasing along time.

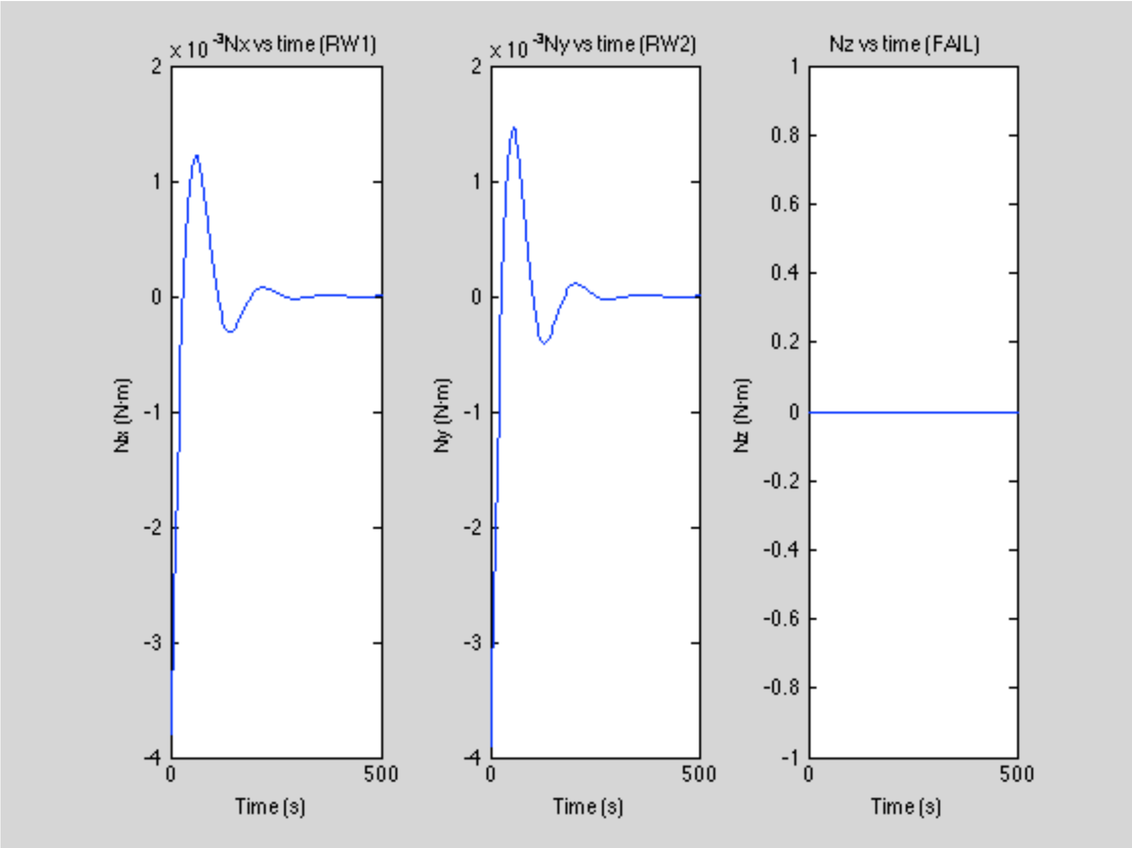

Figure 25. Control torque applied by reaction wheels (failure in RW3)

Additionally, as may be observed from figure 25, it is patent how the control torques applied by the reaction wheels in both the X and Y axis work properly inducing the correct control torque at all times to ultimately return the spacecraft to null angular velocities in those 2 axes.

However, in the Z axis since there has been a failure in the reaction wheel, there will be no control torque applied to the satellite. Therefore, such situation will lead to the non-stabilization of the satellite in that axis, as explained above.

Then, it is safe to state that a reaction wheel failure does affect the dynamics and kinematics of the satellite in the specific axis where the failure takes place and that the use of any other actuator such as thrusters or magnetic torquers is required in this situation to regain control of the spacecraft.

IV.IV. Possible actuators combinations in the case of a reaction wheel failure

When any spacecraft experiences a reaction wheel failure, it jeopardises the entire mission. Therefore, it is vital to have provided the satellite with a backup actuator in case a failure does happen.

As explained in the literature review section, many of the satellites launched to space are provided with a redundant number of actuators just in case a failure takes place. However, not always redundancy is the best option but to provide the spacecraft with a different kind of actuator that does the work as well as a reaction wheel.

In this case, the use of magnetic torquers as the backup plan when there is a reaction wheel failure will be analysed. First, the replacement of a reaction wheel by a magnetic torquer in 1 axis will be simulated. Then, it will be simulated such replacement in 2 axis.

Finally, it is also important to note that in both the magnetic torquers and thrusters controlled cases, the satellite presents an inertial pointing. This is due to the fact that the vast majority of controls that work for inertial pointing satellites, will also work for nadir pointing satellites. (Sidi 1997)

IV.IV.I. Control with 2 reaction wheels and the help of 1 magnetic torquer

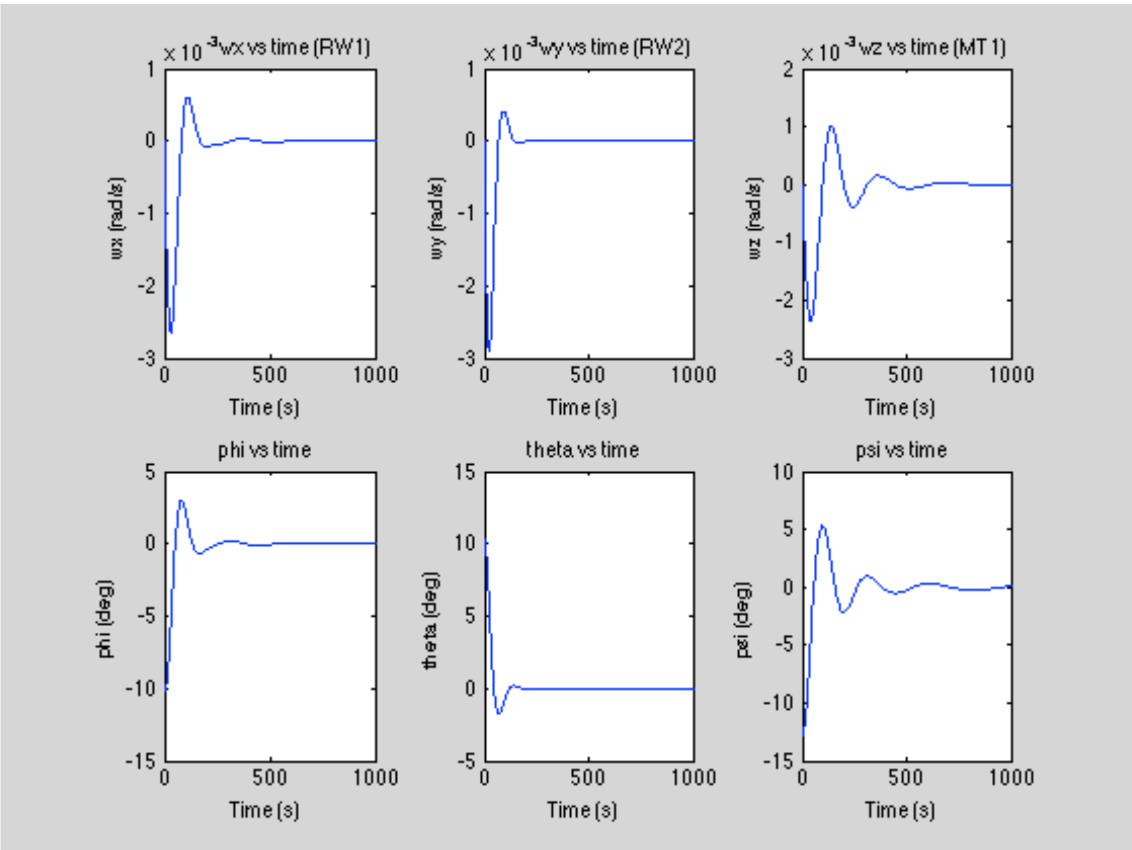

Figure 26. Angular velocities & Euler angles replacement of RW by magnetic torquer in Z axis

As may be seen from figure 26, when added the magnetic torquer to regain control of the Z axis, the satellite is stabilized again. Both the angular velocity and yaw angle converge to 0 as prior to the reaction wheel failure.

Even though the control of the Z axis has been regained, it is important to note how the converging time for the magnetic torquer control is higher than in the reaction wheel case. As may be seen from the figure 27, the converging time for the angular velocity in the Z axis will be of around 800 seconds which a much higher value than the one presented for the angular velocities in the X and Y axes.

Angular velocities and Euler angles in the X and Y axes show a behaviour as expected since they are controlled by reaction wheels.

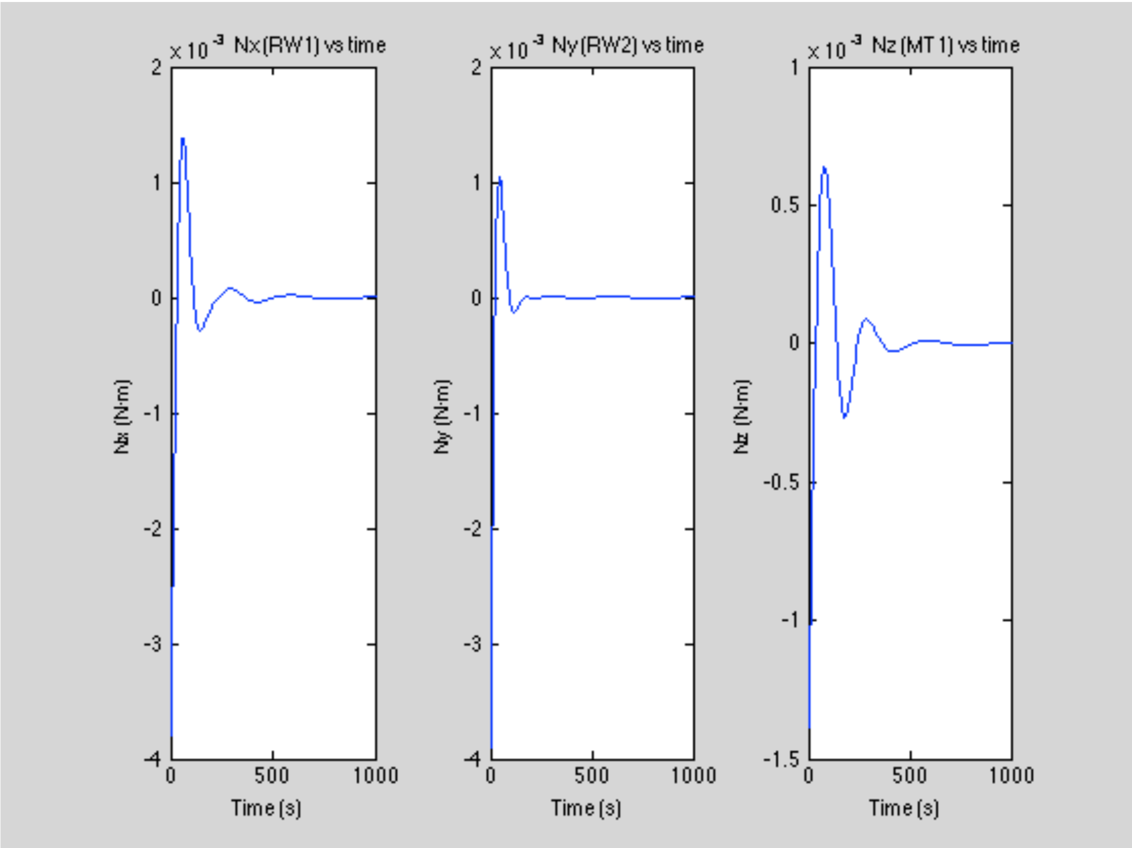

Figure 27. Control torque applied by reaction wheels (X, Y) and magnetic torquer (Z)

Regarding the control torque applied by the magnetic torquer located in the Z axis, it is important to bear in mind that as it is the result of a vectorial product, the control torque will have components in all 3 axes. However, the components in the X and Y axes are negligible as the values are very small. The components of the control torques in the Z axis will be the ones that are important for this analysis.

Moreover, as may be seen from figure 27, the control torque needed by the magnetic torquer is, at all times, lower than the control torques required by the reaction wheels in the X and Y axes to stabilize the spacecraft. Therefore, it means that it requires less energy to do the same job.

In order to obtain an even clearer picture of the main differences between the use of a reaction wheel and a magnetic torquer, in figure 28 a comparison of the angular velocity and control torque applied along the Z axis is shown.

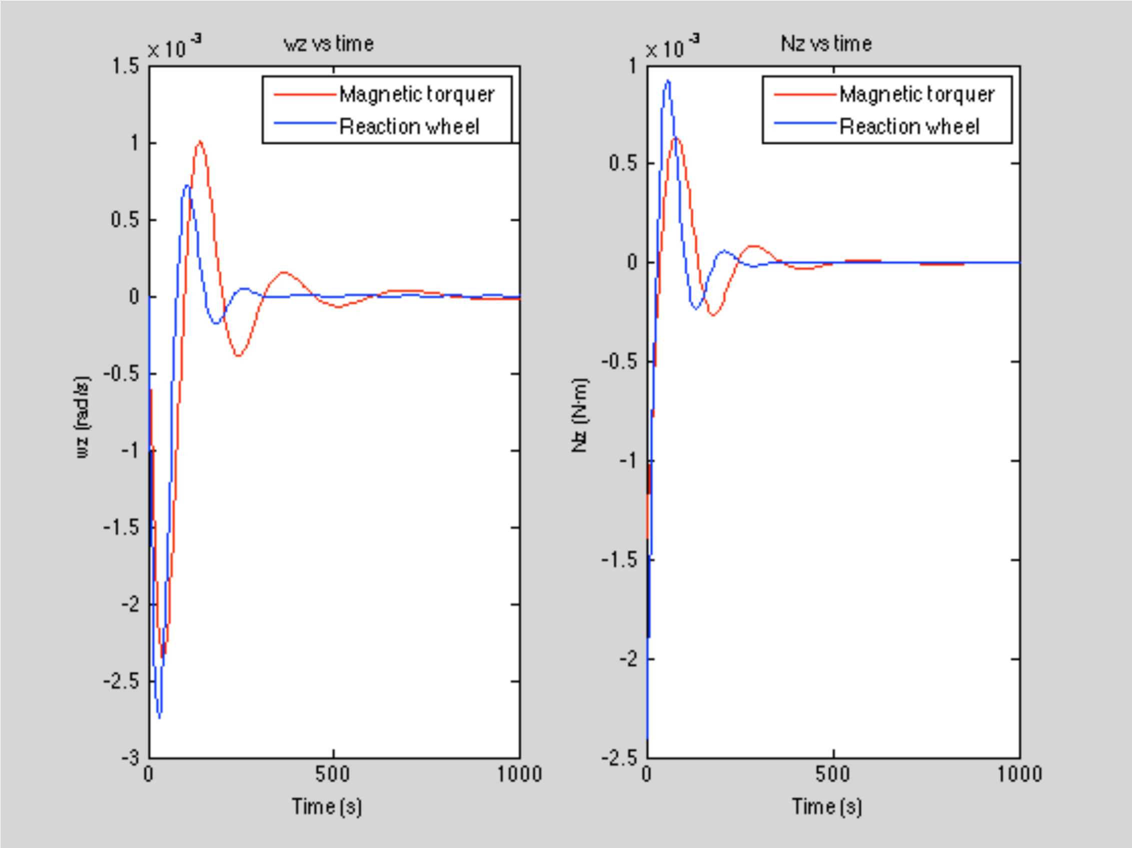

Figure 28. Comparison between reaction wheel and magnetic torquer in Z axis

Regarding the graph containing the angular velocities in Z axis, it is safe to confirm that replacing the reaction wheel by a magnetic torquer will make its stabilization to take a longer amount of time. The amplitude of motion is higher and the converging time to the initial condition is also higher in the case of presenting a magnetic torquer.

However, from the graph containing the control torques applied by both actuators, it is sensible to state that the control torque applied by the magnetic torquer to the satellite is lower than in the case of the reaction wheel. The reason supporting such fact is that the reaction wheels have a higher torque capability than magnetic torquers. Consequently, magnetic torquers use less energy to stabilise the spacecraft than reaction wheels.

IV.IV.II. Control with 1 reaction wheel and the help of 2 magnetic torquers

To finalise the study of the implementation of magnetic torquers, two reaction wheel failures will be simulated (X and Y axes), and control of those 2 axes will be regained with the help of magnetic torquers.

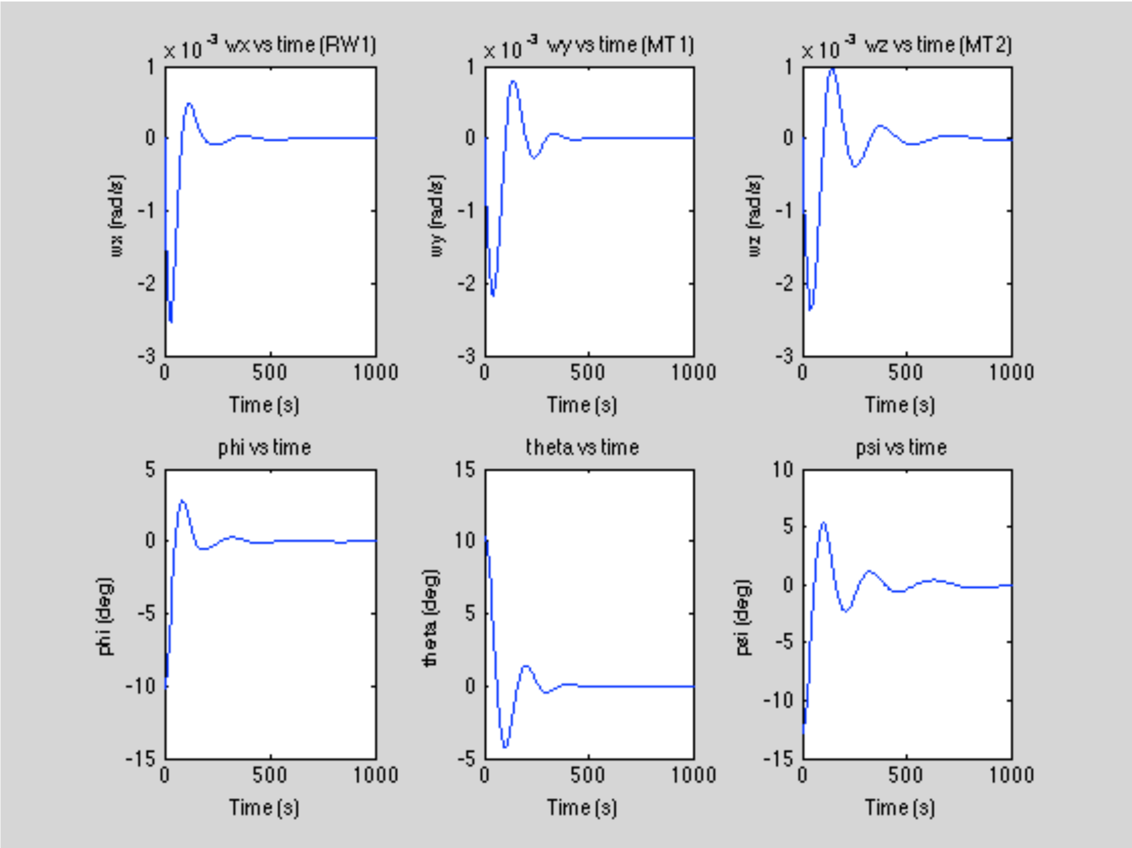

Figure 29. Angular velocities & Euler angle replacement of RW by magnetic torquer in Y and Z axes

As may be observed from figure 29, with the help of 2 magnetic torquers the satellite would also be controllable and stabilized. Both the angular velocities and Euler angles converge to 0. Therefore, the main conclusion from this graph is that even though there 2 reaction wheel failures, control of the satellite could even be regained with the help of magnetic torquers.

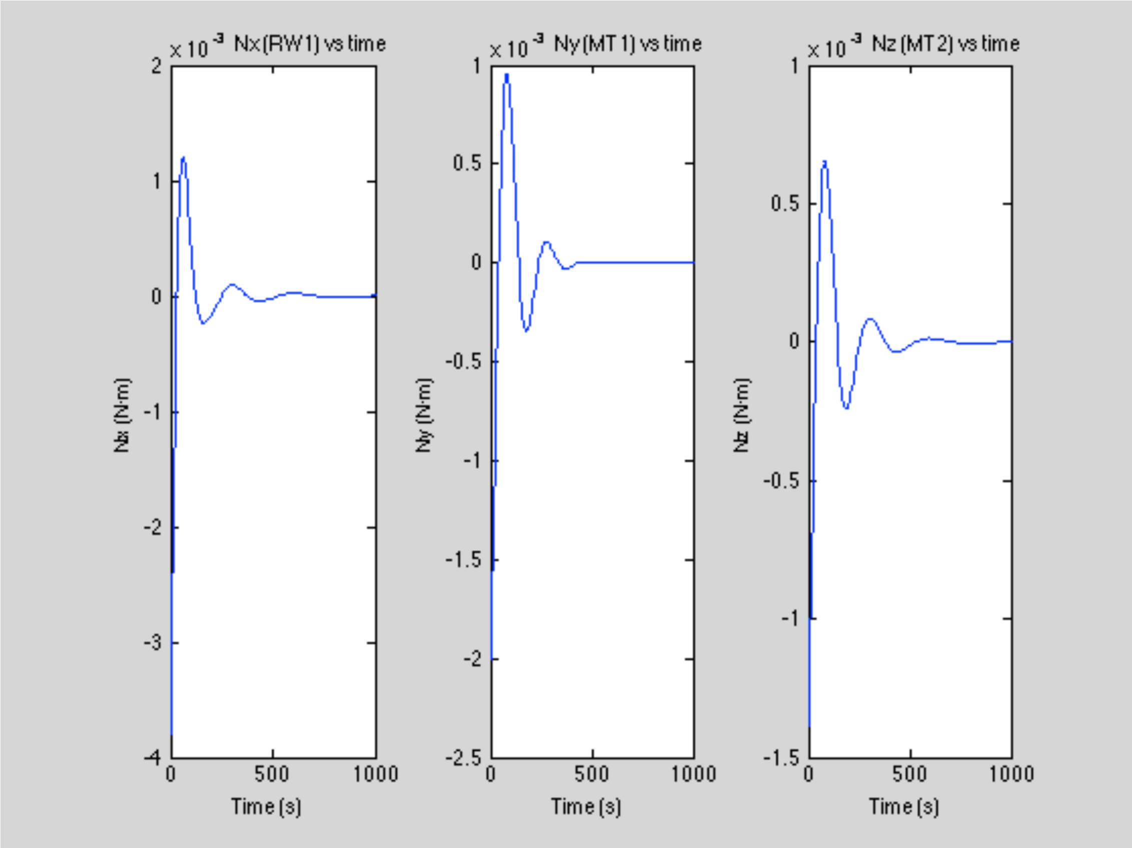

Figure 30. Control torque applied by reaction wheel (X) and magnetic torquers (Y, Z)

Finally, regarding the control torques, it may be observed that for the magnetic torquers, the lower control torque is required in the Z axis as the satellite presents a lower moment of inertia than in the Y axis. This result it could have been expected before running the simulations and prove the results to be sensible.

As a summary, an underactuated satellite may be controlled by the use of two different magnetic torquers in 2 of its axes.

IV.IV.III. Control with 2 reaction wheels and the help of 1 thruster

Another possible backup plan for when a reaction wheel failure happens is to replace it with a thruster. In this subsection of the project, an analysis of the use of a thruster as the main actuator in charge of the attitude control of the satellite in 1 axis will be performed.

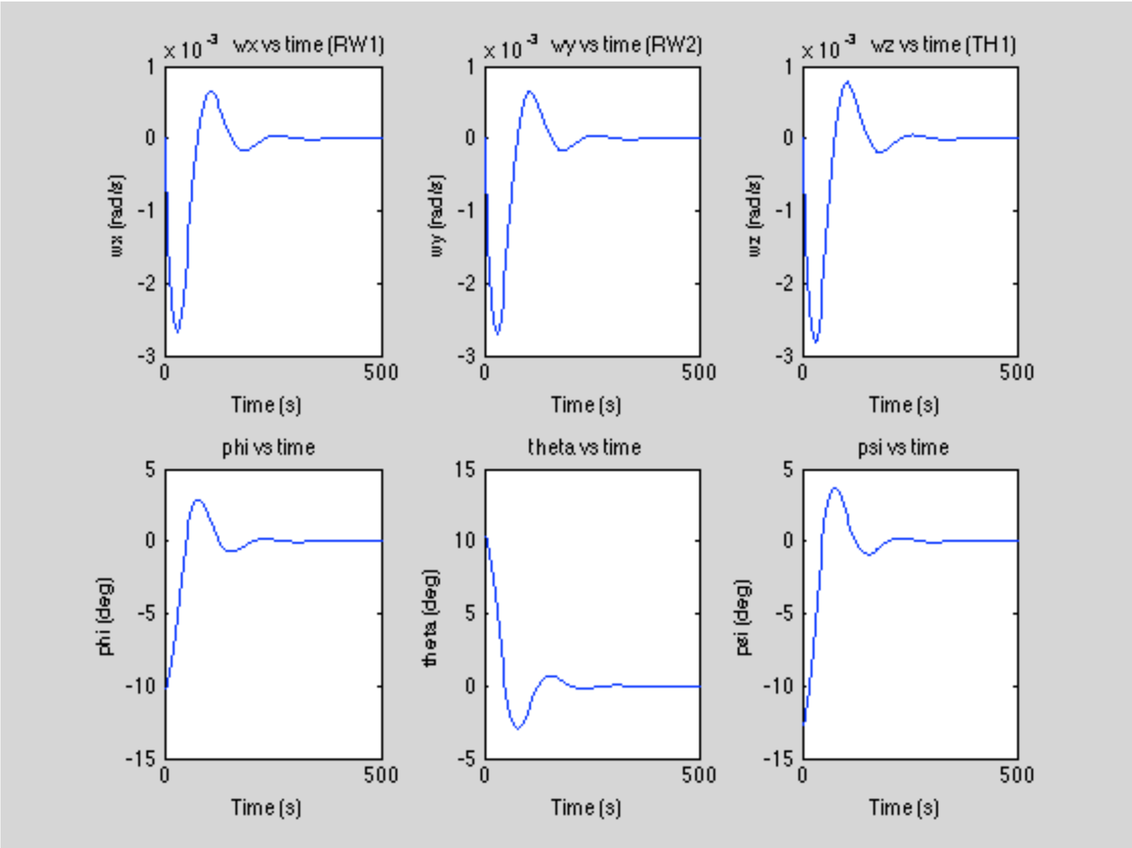

Figure 31. Angular velocities & Euler angles replacement of RW by thruster in Z axis

First, as it may be observed from figure 31, it is patent how by using a thruster control of the Z axis of the satellite is regained. Both angular velocities and Euler angles tend to 0 in a short amount of time (around 300 seconds) and the period of oscillation is also very similar to the one shown in figure 20 where all reaction wheels were working properly.

Therefore, in terms of converging time and period of oscillation, it is safe to state that thrusters provide very similar controls to reaction wheels.

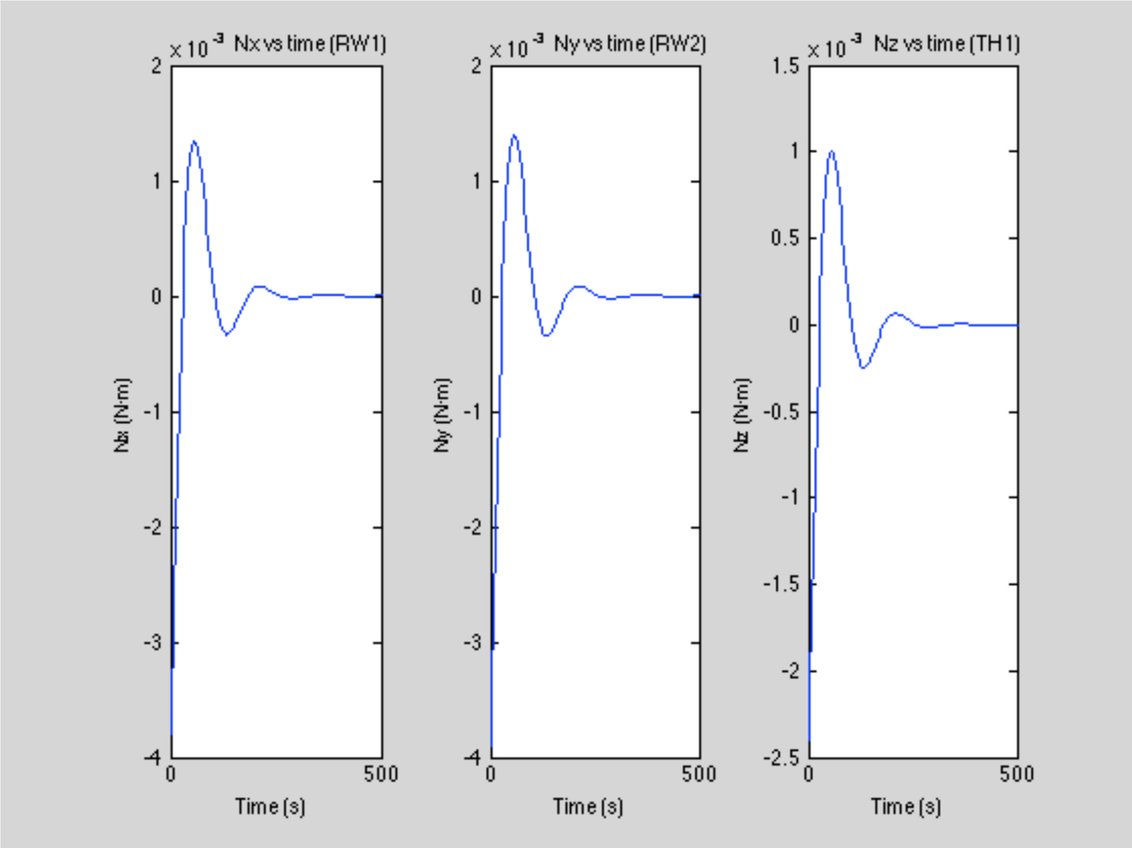

Figure 32. Control torque applied by reaction wheels (X,Y) and thruster (Z)

Moreover, as it could have been expected, the control torque applied by the thruster in the Z axis is lower than the ones applied by reaction wheels as the moment of inertia of the satellite in the Z axis is lower than in the other 2 ones.

As it was done for the magnetic torquer, in figure 33 is displayed a comparison of the angular velocity and control torque applied in both the cases of using a thruster and a reaction wheel. Such graph is intended to display in a very simple and clear way the main differences and similarities of both actuators.

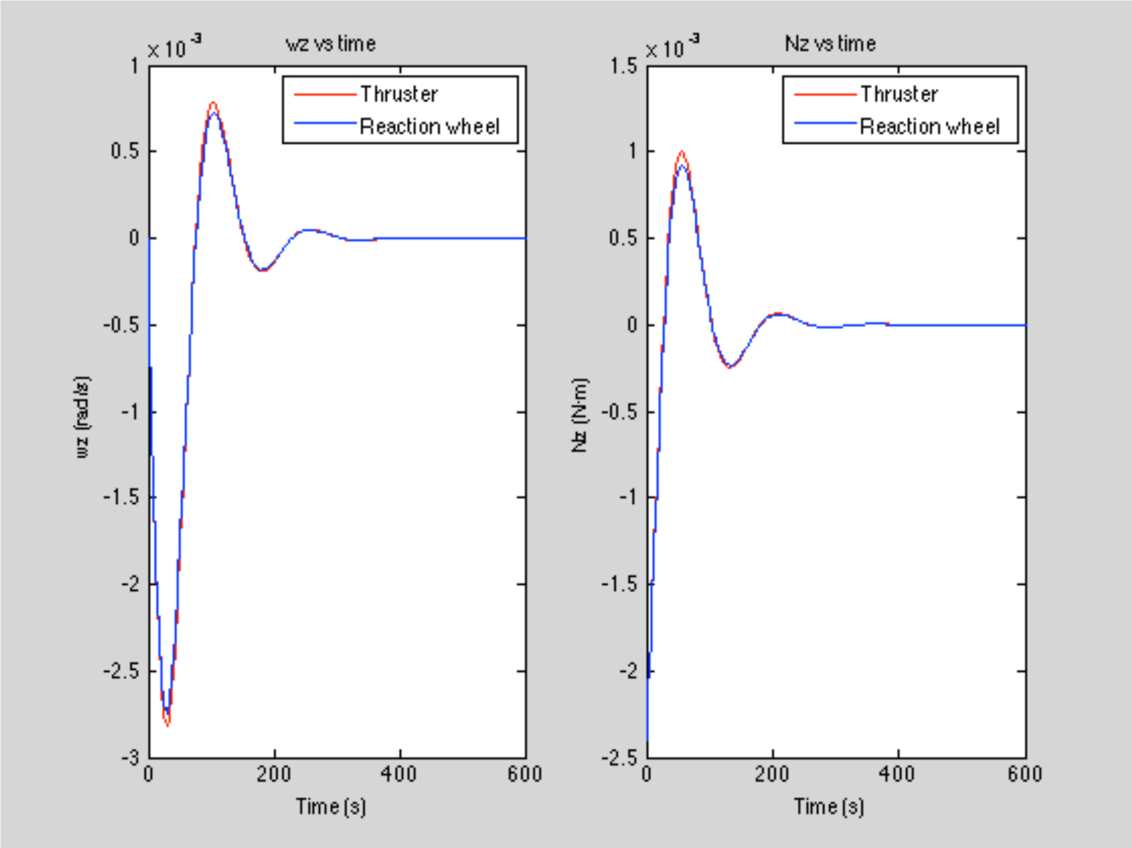

Figure 33. Comparison between reaction wheel and thruster in Z axis

As may be seen from figure 33, there are no differences between using a reaction wheel or a thruster. In fact, in both the angular velocity and control torque applied graphs both lines can be considered as one.

Figure 33 shows that the converging time, period of oscillation and torque capability for both thrusters and reaction wheels are practically identical. Moreover, the energy that both actuators require to stabilise the satellite will also be very similar.

Therefore, it can be concluded that reaction wheels and thrusters provide the same results in terms of the attitude control of a small satellite.

IV.IV.IV. Feasibility of the use of thrusters in small satellites

Unlike reaction wheels and magnetic torquers, thrusters rely on fuel for working. Such dependency is one of the main drawbacks of the use of thrusters in the spaceflight field. The other one is also related to fuel as the need to carry it along the mission, always increases the weight of the satellite.

In this section of the project, it will be performed an approximate analysis of how long the fuel would last in case a thruster was used as the replacement for a reaction wheel that failed. Therefore, it is important to note that in this analysis only one thruster is assumed to be working.

The data used for the calculations is also approximate but it is sensible and has been chosen based on the data provided online for similar missions.

$$\begin{aligned} l_{arm} &= 0.4\,m \\ I_{sp} &= 200\,s \\ m_i &= 150\,kg \\ \Delta V &= 50\,m/s \end{aligned} \tag{34}$$

First, the rate of fuel consumption will be obtained which is calculated as shown in equation (35).

$$\dot{m}_{fuel} = \frac{N_z}{I_{sp}\,g\,l_{arm}} = \frac{0.0552}{200 * 9.81 * 0.4} = 7.033 * 10^{-5}\,kg/s \tag{35}$$

The next step would be to calculate the mass of fuel necessary to be able to produce a total ΔV of 50 m/s. First, the final fuel mass will be computed and from there the mass fuel may be calculated. Then:

$$\begin{aligned} m_f &= \frac{m_i}{e^{\frac{\Delta V}{g * I_{sp}}}} = \frac{150}{e^{\frac{50}{9.81*200}}} = 146.23\,kg \\ m_{fuel} &= m_i - m_f = 150 - 146.23 = 3.77\,kg \end{aligned} \tag{36}$$

Therefore, it may be observed that for a small satellite of around 150 kg, the mass fuel it can carry cannot generally exceed 5 kg, which at the end can be a severe limitation for the use of thrusters in these types of satellites.

Continually, the time such fuel may last will now be computed as shown in equation (37).

$$t = \frac{m_{fuel}}{\dot{m}_{fuel} * 60} = 894.4\,min \tag{37}$$

Finally, if doing an approximation of the number of manoeuvres that are required per day and how long do those manoeuvres last, it is possible to estimate how long will the fuel last for.

Then, if performing 1 manoeuvre per day that lasts for 3 minutes, the fuel will be available for the following number of days:

$$n_{manouvers} = \frac{t}{t_{manouver}} = \frac{894.4}{3} = 298.14 \tag{38}$$

As only 1 manoeuvre per day is performed, the number of days that the fuel will last is shown in equation (39).

$$t_{real} = n_{manouvers} = 298.14\ days \tag{39}$$

Therefore, it is safe to state that in such conditions, the thruster will only be functional for no more than a year, in the best case scenario and also assuming no fuel is destined for orbit trajectory control.

Consequently, if the mission of a small satellite normally lasts for 5 years and the satellite is only equipped with for the case of a reaction wheel failure, the failure would need to happen in the last year of the mission.

That is a very small range of time and shows the main weakness of the use of thrusters as actuators in the spaceflight field: fuel is limited and once is gone, there has to be another way of controlling the spacecraft.

V. CONCLUSIONS

Once all the data has been processed and all the information analysed and presented, it is time to state the main conclusions that can be obtained from this project.

First, reaction wheels have proven to be the best option when controlling the satellite attitude of a satellite. However, they have also been proven to be not very reliable in terms of life of use since many missions have been often recorded with a reaction wheel failure.

Therefore, lacking of a good service life –as they generally fail– has shown to be the biggest weakness of this actuator. Regarding other aspects, reaction wheels are not limited by external factors like thrusters and magnetic torquers are. Then, as long as they do not experience failures, reaction wheels always provide the same good torque capability providing a good controllability of the satellite.

Moreover, giving answer to the title of the project, it is safe to conclude that a satellite can be controlled even though it has experienced a reaction wheel failure in one of its axes. There are quite a few ways to do so as it has been shown previous sections of the project.

In terms of good controllability of the satellite, thrusters are the actuators that present an equal performance as reaction wheels. Therefore, regarding the time of convergence and amplitude of oscillation, thrusters have shown to give results almost identical to reaction wheels. Moreover, in terms of torque capability, thrusters have proven to be really similar to reaction wheels as well.

Therefore, at first sight, thrusters seem to be the best option to replace reaction wheels in case they suffer a failure. However, they present a main drawback that is their reliance on fuel that is limited. Regarding the analysis shown in this project, only by replacing a reaction wheel by a thruster would make the satellite to run out of fuel in around a year. Taking into account that small satellites generally are engaged in 5-year missions, if the failure happened in between the second or third year, just a thruster would not be enough to control it until the mission ends.

Then, thrusters present a great performance but a limited time life that makes us think of thrusters like a perfect actuator for exceptional emergency situations and for orbit control manoeuvres.

On the other hand, magnetic torquers have also shown to be a good option for replacing reaction wheels. The main positive characteristic they present is that, unlike thrusters, have an unlimited time life since they do not depend on fuel.

However, they do depend on the magnetic field of the Earth that makes their torque capability to not be as high as for reaction wheels and thrusters. Especially in terms of converging time, when employing a magnetic torquer it generally takes twice as much time as if it a reaction wheel or thrusters were being used.

Even though the magnetic field is variable which makes magnetic torquers to not provide the same performance in terms of controlling the satellite at all times, magnetic torquers always manage to stabilise the orientation of the spacecraft.

Then, even though thrusters seem to provide a better performance in the short-run, missions can last for many years (even more than previously planned), and if thinking in the long-run, the use of magnetic torquers has proven to be more reliable.

Finally, it is also important to note that equipping the satellite with a great number of redundant reaction wheels in a case a failure happens does not seem to be the best answer to the problem. If a reaction wheel has failed, it is not guaranteed that the one replacing it will not fail as well.

Therefore, I believe that assessing the viability of using any other actuators for replacing reaction wheels when they fail is a more efficient way to solve the problem.

VI. FUTURE RESEARCH DIRECTIONS

The topic covered in this project may be approach from another angle that seems to be very challenging.

In fact, a research direction that could open new ways of controlling an underactuated satellite is to try to model a controller with no more than 2 reaction wheels.

As it has been proved along the project, when a reaction failure happens the control of the orientation of an axis of the satellite is supposed to be lost.

However, it would be priceless if the control in such axis could be regained with the other 2 remaining reaction wheels; no need to use magnetic torquers or even thrusters.

The way to do so is the key that needs to be discovered but concepts such as orientation and saturation of the 2 working reaction wheels seems to be crucial to the cause.

Further study on this topic should be of great help to the spaceflight industry since it may change the way actuators are equipped in the satellites. For instance, redundancy of reaction wheels would no longer that important resulting in a lower the weight and more room in the satellite, both vital aspects.

Even though other actuators will still be equipped in most satellites as the backup option in case reaction wheels fail, assessing the controllability of a satellite with no more than 2 working reaction wheels seems to be the next following step forward in the field.

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APPENDIX I. Project Proposal

Final Year Undergraduate Project Proposal Form

Student Name	Alvaro Guindal Martinez
Course	Project (320EKM) 1617JANMAY
Email	guindala@coventry.ac.uk
Project Module	320EKM

Supervisor	Dr. Nadjim Horri

Project Title:
A proposed title for the project (Should be meaningful, relevant and concise)

Satellite attitude control in the presence of reaction wheel failures

Synopsis:
Explain the background to the project, and provide an overview of what you intend to do (approximately 500 words)

Nowadays there are a countless number of satellites that are launched from Earth every year. Amongst all of those satellites, some of them are satellites that are designed for Earth observation. In such type of satellites, it is really important to be able to control its rotation due to the fact that a bad rotational position may lead into the satellite getting pictures of the wrong spot on Earth or not well communicating with the ground radars receiving all the data.

In other words, the attitude dynamics of the satellite play a key role in this type of missions. Moreover, being able to control the satellite's rotational dynamics and kinematics is one of the most challenging processes when designing the satellite and planning its mission.

Furthermore, placing a satellite in orbit is really expensive and we need to have to sufficient knowledge to foresee any type of problems that may take place.

A satellite is such a complex machine that there are a lot of things that may not go as planned and we should be able to overcome them in the most efficient way.

In economics terms, it is much more profitable to invest more money on research that would allow us to control a satellite even when there is any type of failure (including actuator failure, which the most common one) rather than building a brand new one and place it in orbit again.

Regarding the actuators, they are the parts of the satellite that allows us to control its attitude dynamics and kinematics. Generally, there is one actuator per axis, which means

that we can control the rotation of the satellite in each of its 3 axes of rotation.
Even though there are different types of actuators, in this project we will focus on the study of the ones called reaction wheels.
Therefore, the main objective of this project is to study the controllability of a satellite in a low-earth-orbit (LEO) when there is a reaction wheel failure. We will analyse up to which level the satellite is controllable and how stable and efficient it is in such special circumstances.
However, this may be one of the final parts of the project and for getting there, we will need to understand first the dynamics and kinematics of the satellite when placed in orbit. Once we have a model of such dynamics in MATLAB and SIMULINK, we will then proceed to vary such model for reproducing the actuator failure and examine up to which point will its dynamics and kinematics change.
Finally, as stated before, we will examine whether will the satellite be controllable and how stable and efficient will it be.

Client:
Provide a description of your client (if any), and contact details.

Not applicable

Objectives (provide from 5 to 8): *List the overall objectives of the project. These should be measurable, and will be used to assess the level of achievement of the project.*

- Review the Attitude and Determination Control System (ADCS) for Earth observation missions.
- Understand the fundamentals of theory of spacecraft dynamics, reference frames and attitude representations.
- Model the attitude dynamics and kinematics of a satellite orbiting around Earth.
- Predict with the help of numerical simulation analysis softwares the behaviour of the satellite when there is a reaction wheel failure. Learn how to mitigate it with the help of thrusters and magnetic torquers.
- Evaluate performance in terms of rapidity & energy consumption and demonstrate stability of the satellite by analysing the data in a sensible way. Then, convert such data into information for presenting the results.

Project Deliverables (provide from 5 to 8):
Provide a list of key deliverables of the project (which may be one for each of the above objectives). These can be studies, reports, recommendations, etc.

- Final report
- 2 MATLAB scripts (with & without reaction wheel failure) containing all the programming code with the attitude dynamics and kinematics of the satellite
- 2 SIMULINK models (with & without reaction wheel failure) containing all the controls of the satellite
- MATLAB script for examining the stability and efficiency (energy consumption) of the satellite when there is the actuator failure.
- Simulation results obtained from the numerical analysis

Why are you interested in the project?
Provide a reason for your interest, and describe what greater general interest it serves. Who else could benefit from it?

First, one of the main reasons why I am really interested in this project is because I can work in the field of orbital mechanics that is something I always wanted. Since I was a kid, I have been really interested in Astronomy and I found really interesting anything related to planets, galaxies and space in general. Therefore, finally getting to work in a space-related project is a joy for me.
Moreover, the human kind has always been trying to expand its boundaries along our Solar system for better understanding where we come from and what is around us. A countless number of missions have been launched over the years but not all of them have been successful. One of the reasons for these unsuccesful missions is the actuators failure that happens more often than we may think. Therefore, it is important to invest in research about controllability of the satellite when there is any type of actuator failure rather than adding more actuators to the satellite since it is redundant and does not fix the actual problem.
Furthermore, research on the controllability of a satellite when there is an actuator failure is one of the areas where the space agencies have started to invest more and more money due to its increasing importance. Therefore, I believe that my research is helping in a way to the understanding of this topic.
Finally, any person or private company willing to build any type of satellite and place it in orbit could also benefit from this research.

What are the key questions the project attempts to answer (provide from 1-3)?

- What are the dynamics and kinematics of a satellite in a LEO (low-Earth orbit)?
- What is the behaviour of such satellite when the actuator in any of the 3 axis of rotation of the satellite stops working? In other words, how much will its dynamics and kinematics will change?
- Will we be able to control the satellite when such actuator failure occurs?

How will you judge whether your project has been a success?

Predicting the behaviour of the satellite with and without failures and modelling a good control that let us control it in a stable way, I believe it would be success.
Moreover, being able to reliably evaluate the performance of the satellite when the reaction wheel fails would also be a success.
However, if the satellite turns out to not be controllable when the actuator is not working I will not consider it as failure, as long as my work leads to better understanding of controllability issues.

What research methods do you intend to use?

A deductive approach will be followed in this project. Therefore, it will be based on obtaining data through numerical analysis of the attitude dynamics of the satellite when the actuator fails. Such data will then be processed and some quantitative information will be extracted from it. Thus, with such information we will make a comparative analysis of the dynamics of the satellite when the actuator is working properly and when it is not.

Moreover, we will also make a comparative analysis with existing results and state if the results obtained are sensible. Finally, with SIMULINK we will model the control of the rotational dynamics of the satellite for when the failure in the reaction wheel occurs.

What primary and/or secondary data sources do you intend to use?

In this project, we will be working with secondary data extracted from published research papers and textbooks. Especially, all the equations that model the attitude dynamics and kinematics of the satellite are extracted from secondary data. Moreover, simulations will be performed to obtain primary data using models.

Estimate the number of hours you expect to spend on each of the major project tasks:
(The tasks below are only examples. You will need to edit the table to suit your own project).

Introduction	20
Objectives	30
Literature Review	50
Case Studies	30
Research	80
Final report preparation	80
Total number of hours	290

Signature: ALVARO GUINDAL MARTINEZ

Date: 13th February 2017

APPENDIX II. CU Ethics Approval

Certificate that shows the project has passed the Coventry University Ethic Approval process. The project was confirmed and approved as Low Risk.

Certificate of Ethical Approval

Applicant:

Alvaro Guindal Martinez

Project Title:

Satellite attitude control in the presence of reaction wheel failures

This is to certify that the above named applicant has completed the Coventry University Ethical Approval process and their project has been confirmed and approved as Low Risk

Date of approval:

20 February 2017

Project Reference Number:

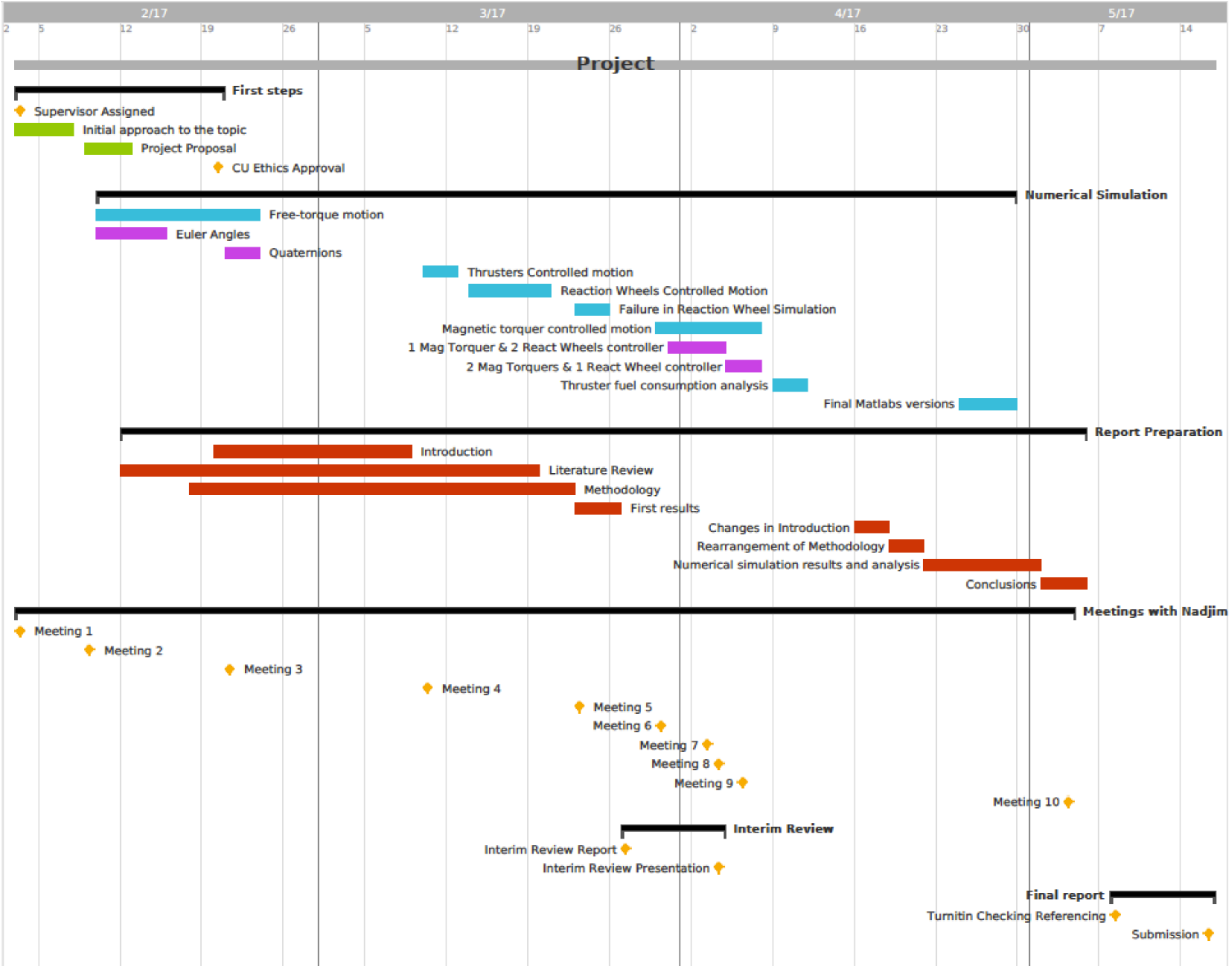
P51645

APPENDIX III. Project Management

The purpose of this section is to show how the time has been managed in order to accomplish all deadlines and complete the work on time. For doing so, a gantt chart and a logbook have been performed as will be presented in the following pages. They are both updated until the day of the final submission of the project.

It is important to note that the number of hours of work stated in the logbook have been estimated and may slightly differ with reality.

I. GANT CHARTT



II. LOGBOOK

Date	No. Hours dedicated	Brief description of activities performed
3/2/17	3	This day I finally had a supervisor assigned. The meeting went very well as Dr. Horri works on the research field of the spacecraft dynamics and control which is the area where I wanted to do my project on. We chatted about the different possibilities the project offered and we came up with a project title. Moreover, Dr. Horri assigned me to review some literature review as an initial approach to the spacecraft attitude dynamics area of research.
4/2/17	3	I found some time to review some of the topics that Dr. Horri advised me to go through. Especially, I focused on the research about ADCS (Attitude Determination and Control System) giving special attention to the study of its hardware – actuators such as reaction wheels, thrusters and magnetic torquers.
5/2/17	4	I kept on with my initial approach to the spacecraft attitude dynamics. On that particular day, I narrowed the literature review to the study of concepts such as references frames describing attitude and attitude kinematic models. From the first one, I focused on the study of the ECI (Inertial frame) and Orbit reference frame (LVLH). From the second one, I was able to find the Euler's Moment equations and the Euler angles' kinematic equations. Moreover, I did a little research on the concept of quaternions and the reason to use them but the concept was still not trivial for me at that time.
6/2/17	3	On this day, I did some research on the historical background of the satellites. The study went from the early spinning satellites to zero-momentum and bias momentum satellites.
7/2/17	3	I studied about the feedback control on 3 axes using 3 reaction wheels. It was focused on PD controllers and quaternion feedback control. However, it seemed of pointless at the time as I realized that I needed to do much further study on the dynamics and kinematics of satellite attitude before being able to jump to the control part of the project.
8/2/17	0	Rest day before meeting with my supervisor
9/2/17	3	First real meeting with my supervisor where we discussed about the literature review research I had done on the weekend. Moreover, he went into detail for explaining to me the differential equations that model the attitude dynamics of the spacecraft. He suggested I could start modelling a MATLAB or SIMULINK program where I could play a bit with the input variables and examine how the output variables changed. Furthermore, I started working on my project proposal.
10/2/17	3	As Dr. Horri suggested me, I started programming both the MATLAB code and SIMULINK model. However, since I was struggling with the SIMULINK model, I decided to focus on the MATLAB program. After coding my function with all the differential Euler moment equations, I coded a MATLAB script for obtaining some graphs where I could study the behaviour of the output (spacecraft angular velocities in the X, Y and Z axes) variables with respect to time. Finally, I continued working on my project proposal.
11/2/17	0	Rest day

12/2/17	4	On this day, I kept on with the literature review on the topics that Dr. Horri had suggested me but instead of using websites as references like I had done in the past, I looked for books in the library and technical papers that could help with my research. Furthermore, I finished the writing of the project proposal and I did the CU Ethics forms.
13/2/17	5	With the help of books like (Sidi 1997) and (Mark, Crossidis 2014) I started the writing of the literature review section of the project. I focused on the ADCS subsection regarding all the actuators and sensors that form part of this system. I also worked on the reference frames and attitude representations section of the project.
14/2/17	4	Tried again with SIMULINK but decided to go on with MATLAB. Free torque/Euler angles As of the literature review, I started reviewing the type of satellites and orbits section and did some writing on it. Moreover, regarding the numerical calculation, I gave SIMULINK a final try but as I could not get to the point I would have liked, I chose Matlab as the software for the project. Therefore, I coded both the main script and main function of the free-torque case and analyse the results obtained. IMPORTANT: in such code I was working with Euler angles not quaternions.
15/2/17	3	I realized that the Matlab I had done could only be applied for axis symmetric body satellites ($I_x = I_y$). I changed that part of the code as such case is very specific and our aim is to study a general free-torque case. Furthermore, I analised the results obtained in order to check if they were sensible and what we could have expected.
16/2/17	2	I kept on with the literature section especially regarding requirements of a nadir pointing satellite. I did a lot of research on this topic, as it was very difficult to find information about it.
17/2/17	0	Rest Day – My birthday
18/2/17	2	On this day, I started the writing of the Methodology section of the project. I focused on writing of the sections about the dynamics and kinematics of a satellite in free-torque motion.
19/2/17	3	I kept on with the writing of the literature review part.
20/2/17	3	I started the Introduction section on this day. I focused on the review of both books and websites where I found graphs that would support my ideas and I started the writing of a draft.
21/2/17	3	Meeting! – Nadjim saw my MATLAB results and conclude they were sensible and consistent. He encouraged me to work now with quaternions as my attitude representation as they present more advantages that Euler angles do. Moreover, he told me to keep on working hard as he was satisfied with my progress so far.
22/2/17	4	I did the re-writing of the main script and function of the torque-free motion case but in this case working with quaternions and not Euler angles. Moreover, I did a lot of research on literature review part regarding the conversion from quaternions to Euler angles. After a couple hours of research, I could finally found both the matrices that have to be compared for performing the conversion. (R_{quat} & A_{312})
23/2/17	3	I finished the Methodology torque-free motion case.
24/2/17	3	In the Methodology section, the writing of the kinematics and dynamics equations that model the behaviour of the spacecraft in free-torque motion was performed on that day.
25/2/17	3	I kept on working on the Introduction part and managed to do more than half of the writing of it.

26/2/17	2	The writing of the "Introduction to Matlab" subsection of the Methodology section was started and finished on that day.
27/2/17	0	Rest day
28/2/17	3	I resumed the writing of the Literature review, especially regarding the reference frames and attitude representations section.
1/3/17	4	I started to do research on the equations that model the control of the spacecraft especially with the help of thrusters and reaction wheels. Such research was performed with the help of books like (Sidi 1997).
2/3/17	3	I did some reviewing of the writing done so far in both the Methodology and Literature Review section.
3/3/17	4	Research on the historical background of the project for the Introduction section. Such historical background was mainly based on the study of the evolution of satellites along the past 60 years.
4/3/17	2	I continued to go through to the Control theory for both thrusters and reaction wheels.
5/3/17	3	Went to the library where I spent some time doing some research on textbooks about spacecraft dynamics & control. Moreover, I tried to implement in Matlab the control with thrusters but I could not get any results, as there were some errors in the coding.
6/3/17	2	Literature review on the dynamics and kinematics of spacecrafts. Moreover, I did literature review on Euler angles kinematics and quaternions and how to convert from one to another.
7/3/17	3	Finished the writing of the ADCS, types of satellites and orbits & reference frames and attitude representation subsections of the Literature Review part.
8/3/17	2	I finished the Introduction section of the project.
9/3/17	5	Cancelled meeting with supervisor. Therefore, I kept on working on my project – I finished all the chapters in the Literature Review section of the project but the one related to "Real-life missions that experienced reaction wheel failures".
10/3/17	5	Meeting with my supervisor for around an hour. We went through the MATLAB script and functions. Moreover, he took a look at my drafts of introduction and literature review and we discussed the approach I should follow for doing the Methodology section. Finally, he told me to start working on the thrusters and reaction wheel control cases and encouraged me to have it ready for our next meeting.
11/3/17	2	Flight to Dubai – lost almost all the day. Due to jet lag, I could work a couple hours at night on the MATLAB functions introducing the controlling with thrusters. Finally, did manage to obtain some results that I will include in the interim review report.
12/3/17	3	I did some reading about the missions that had experienced an actuator failure that Dr. Horri recommended me to include in the literature review part.
13/3/17	2	I started reviewing the writing of the introduction section (grammar, structure, content…)
14/3/17	3	I continued on working with the MATLAB functions as I tried to include the equations for controlling the spacecraft with reaction wheels. However, this case is not as straightforward as the controlling with thrusters was and therefore I could not come up with a good code (there were some errors and I decided to leave for other day)

15/3/17	2	Started the writing of the “Failure missions” section in the literature review but did not make all the progress I would have expected.
16/3/17	0	Road trip to Abu Dhabi – no time for working on the project
17/3/17	3	Finished the writing of the subsection regarding the missions that had experienced a reaction wheel failure in real life.
18/3/17	3	Flight back from Dubai – I managed to work a couple of hours during the flight on the Methodology part. I rearranged the order of the subsections and I started writing the control theory section with the respective equations we are going to use in MATLAB
19/3/17	5	First, I included in my literature review the section about real-life missions that have experienced a reaction wheel. I chose two of them, one performed by NASA and another one performed by Surrey University. Moreover, I did the coding of the Matlab scripts for the control of the spacecraft with reaction wheels but there were some errors since the results were not making sense.
20/3/17	5	I continued with the reaction wheel controlling modelling. I identified the root of the error and I fixed it and the results were much better. I played a little bit with the initial conditions of the variables to see the different outcomes I got.
21/3/17	5	I started reviewing the literature review section. I was especially looking for grammar mistakes and I ended up rewriting a lot of things in a different way – the content remained the same but the way I explained things changed a bit. Moreover, I also checked the references of the section in order to see if everything was correct.
22/3/17	3	I did the same as the day before but in the Methodology section.
23/3/17	6	Meeting with Nadjim. We discussed a lot of things, especially regarding the simulation part. Moreover, he told the steps to follow in the Methodology and Analysis of results section. Following his advice, I added the modelling of the control with magnetic torquers and I simulated the situation where there is an actual failure in a reaction wheel – I added the results to my report as well.
24/3/17	6	I started the analysis of the results I had so far, especially focusing in the main differences and similarities of the different controls we had used. Moreover, I finished the revision of my literature review and Methodology part.
25/3/17	4	I finished the analysis of the results I had so far – I explained the graphs that are presented in the Analysis section. I started and finished the “Notation” section.
26/3/17	5	On this day, I named and numbered all of the figures in the project and included them all in the table of figures. Moreover, I numbered all equations appearing so far in the project. Additionally, I included the cover page of the project and a table of contents. Finally, I thought it would be nice to include a header and footer in the document as well.
27/3/17	3	I worked on the appendices of the report.
28/3/17	0	Rest day
29/3/17	4	I practised the presentation that I had on Friday.
30/3/17	2	I did a final checking of the content of the presentation and I practise it a final time.

31/3/17	3	My presentation was supposed to be held today but it got cancelled as nobody could join Nadjim to attend it. However, the day was productive as Nadjim and I met for an hour where he explained me the further steps to follow in the project. He introduced to the magnetic torquer controller area and gave me good references on the topic.
01/4/17	4	I started working on the modelling with the magnetic torquer controlling but I was having many problems.
02/4/17	3	I kept on with the modelling with magnetic torquers but I was not able to regain control of the Z axis using the magnetic torquer.
03/4/17	4	I held a long meeting with Nadjim where he went through the Matlab code in order to identify the error cause. During the meeting, I improved the simulation results but I was not good enough yet. Once I got home, I fixed all the errors and I was prepared to show Nadjim the results the day after.
04/4/17	4	I finally held the presentation on this day. Nadjim and Nerea attended it and it went very well. After the presentation, I had a quick meeting with Nadjim to show him the Matlab code and discuss the further steps in the project.
05/4/17	3	After the modelling of 2 reaction wheels and 1 magnetic torquer, today I started the modelling of the control with 2 magnetic torquers and only 1 reaction wheel.
06/4/17	3	Final meeting with Nadjim before Easter break, where we discussed the possibility of including the control with only 2 reaction wheels to the project. I also showed him my progress with the magnetic torquers control modelling.
07/4/17	4	I finished the modelling with 2 magnetic torquers and 1 reaction wheel and the simulation results were showing were we would have expected.
08/4/17	3	I started working on the thruster fuel consumption section.
09/4/17	3	I started getting results for the “Feasibility for the use of thrusters” but they still were not good enough for what I wanted to explain.
10/4/17	4	Finally, I finished the modelling of the “Feasibility for the use of thrusters”. The simulation results were sensible and ready to be included in the report.
11/4/17	0	Holidays
12/4/17	0	Holidays
13/4/17	0	Holidays
14/4/17	0	Holidays
15/4/17	0	Holidays
16/4/17	5	After some days off that were needed, I came back with a lot of energy. I realised that my introduction was not good enough so I decided to make some changes to it.
17/4/17	3	I kept on reading and searching for new information to include in the Introduction.
18/4/17	4	I did the re-writing of the Introduction section of the project.
19/4/17	3	I also realised that the Methodology section was also in need for a change. In this case was more of a rearrangement rather

		than a real change. For instance, the control with thrusters section was moved towards the end of the Modelling section.
20/4/17	6	The control with magnetic torquers section was included to the Modelling section. The control with thrusters was rewritten and the fuel consumption equations were added to the section as well.
21/4/17	3	The methodology section changes were finished on this day.
22/4/17	4	On this day, I started to prepare the Numerical simulation and results section. What I did is to generate all the graphs that I wanted to include in the report.
23/4/17	4	Moreover, once I had all the information needed, the analysis of such information began. Moreover, it also took me some time to decide the structure of the section.
24/4/17	3	I kept with the analysis of the simulation results.
25/4/17	6	On this day, I also started to modify my Matlab scripts and functions to make them more presentable. To make their reading easier, I focused on well structure them and include the name of every variable and their units.
26/4/17	4	I kept on with the analysis of the simulation results and also with the modification of the Matlab scripts.
27/4/17	3	I included in the report, the project proposal form and CU Ethics certificate. I also kept working on the report.
28/4/17	3	The final versions of the Matlab scripts and functions were ready.
29/4/17	4	I realised on this day that it would be useful to include a couple of graphs including a comparison of the control with reaction wheels and thrusters and magnetic torquers. In such graphs, it would be very intuitive to identify the strengths and weaknesses of every actuator.
30/4/17	3	I kept on working on the analysis of results.
31/4/17	6	As last day, I was working on the analysis of the results I had obtained. It was close to being finished.
01/5/17	3	I finished the section of "Numerical simulation results and analysis".
02/5/17	4	As I had finished the section of analysis of results, I started working on my conclusions.
03/5/17	5	I kept on working on my conclusions, and I also I did a draft of the abstract of the project.
04/5/17	4	Final meeting with Nadjim. I printed the analysis and conclusions sections in order to get some feedback from Nadjim. He identified some errors and encouraged me to include a final section about "Future research directions"
05/5/17	6	I corrected all the errors that Nadjim had pointed out and updated my report.
06/5/17	4	I wrote the final version of the abstract and the acknowledgements. Moreover, I completed the Notation of the report.
07/5/17	5	I did the writing of the "Future research directions" section of the report. Additionally, I updated the Gantt Chart.
08/5/17	2	Turnitin Reference Checking. Originality signature

*Note that the dates highlighted in yellow represent a day where I held a meeting with my supervisor.