POLITECNICO DI TORINO

Master's Degree in Aerospace Engineering

Master's Degree Thesis

Satellite Miniaturization Study and Experimental Integration of Reaction Wheels in a PoketQube Satellite

Supervisors

Prof. Marcello ROMANO

Prof. Chantal CAPPELLETTI

Candidate

Armando Vittorio ATZORI

April 2023

"Innovation is the unrelenting drive to break the status quo and develop anew where few have dared to go."

- Steven Jeffes.

Summary

The miniaturization of satellite components has revolutionized the space industry by allowing for the development of smaller, cheaper, and more efficient satellites. This trend has been fueled by advancements in technology, such as the development of microelectronics and nanotechnology, which have enabled the creation of smaller and more powerful components. As the demand for smaller and more affordable satellites grows, the need for smaller and more efficient subsystems becomes more pressing. This thesis will explore the State of the Art of miniaturized technologies for each subsystem typically present in a small spacecraft and the perspectives of their development. After that, the Attitude Determination and Control System will be taken more into consideration focusing on the physical limit of one of its main components, the Reaction Wheels, with both analytical and statistical analysis. At the end, to validate the theory, a model of PocketQube Satellite following the main standard specification has been developed trying to integrate three reaction wheels in it and running some tests to see what are actually their capabilities with a limited mass, volume and power budgets.

Table of Contents

List of Tables

List of Figures

Chapter 1

Introduction

In this chapter will be given an overview of the satellite miniaturization history, advantages and challenges that this trend could lead to, following a detailed research on high level articles and documentation [1] [2] [3] [4].

1.1 Satellite Miniaturization

Satellites have played a crucial role in various applications, ranging from telecommunications to remote sensing, navigation, weather forecasting, and military surveillance. Over the years, advancements in technology have led to a significant reduction in satellite size and weight while improving their capabilities, which has created new opportunities for various stakeholders, including private companies, research institutions, and governments. The miniaturization of satellites has been a key driving force in the space industry, enabling the development of small satellites with reduced costs and quicker launch times than traditional satellites. The miniaturization of satellites has also allowed for the democratization of space exploration, enabling more people to access and leverage space-based data for various applications. This chapter aims to provide an overview of satellite miniaturization research and development, focusing on its history, key drivers, challenges, and future prospects. The paper begins by defining satellite miniaturization and exploring its evolution over the years. It then discusses the various drivers of satellite miniaturization, including technological advancements, cost reductions, and the growing demand for small satellites. It also examines the challenges associated with miniaturization, including limited power, reduced capabilities, and reliability issues. Finally, explores the future prospects of satellite miniaturization, including emerging trends, such as the use of Artificial Intelligence and machine learning in satellite technology.

1.2 What is Satellite Miniaturization?

Satellite miniaturization refers to the process of designing and developing small satellites with reduced size, weight, and cost compared to traditional satellites. Miniaturized satellites are typically classified based on their weight, with most small satellites weighing between 1-500 kg. The miniaturization of satellites is a recent trend, with the first small satellites launched in the early 2000s. Prior to this, most satellites were large, complex, and expensive, making them accessible only to a few governments and large corporations.

1.3 The history of Satellite Miniaturization

The miniaturization of satellites began in the 1990s when advancements in technology, such as the development of microelectronics and miniaturized sensors, made it possible to design and develop smaller satellites. In 1998, the first small satellite, a CubeSat, was developed by researchers at Stanford University. CubeSats are small, cube-shaped satellites measuring 10 cm x 10 cm x 10 cm and weighing 1.33 kg. The development of CubeSats marked a significant milestone in satellite miniaturization, as it demonstrated that small satellites could be designed and developed at a lower cost and launched quickly [5].

Since the development of the first CubeSat, there has been a significant increase in the number of small satellites launched into orbit, with most of them being CubeSats or similar microsatellites. The miniaturization of satellites has enabled various stakeholders, including private companies, research institutions, and governments, to access space-based data and services, leading also to a democratization of space exploration.

1.4 Key Drivers of Satellite Miniaturization

The miniaturization of satellites has been driven by various factors, including technological advancements, cost reductions, and the growing demand for small satellites. Below are listed some of what can be considered the key drivers of satellite miniaturization:

- Technological Advancements: Advancements in technology, particularly in the field of microelectronics and miniaturized sensors, have been a critical driver of satellite miniaturization. The development of microelectronics has enabled the design and production of smaller and more powerful computers, which can perform complex tasks, such as data processing and communication, with minimal power consumption. Additionally, the miniaturization of sensors, such as cameras, accelerometers, and magnetometers, has made it possible to collect data from space with a high degree of accuracy, without the need for large, heavy, and expensive instruments.

- Cost Reductions:

  Satellite miniaturization has also been driven by the need to reduce the cost of developing and launching satellites. Traditional satellites are typically large, complex, and expensive, requiring extensive ground infrastructure, sophisticated manufacturing processes, and advanced launch vehicles to get them into orbit. Miniaturized satellites, on the other hand, can be designed and developed at a lower cost, using off-the-shelf components and simplified manufacturing processes. Additionally, the reduced size and weight of small satellites allow for multiple satellites to be launched simultaneously, reducing the cost of launch and increasing access to space.

- Growing Demand for Small Satellites:

  The growing demand for small satellites has been another significant driver of satellite miniaturization. Small satellites are well-suited for various applications, including earth observation, telecommunications, and scientific research. They are also ideal for emerging applications, such as internet of things (IoT) and global connectivity, which require a large number of satellites in low earth orbit (LEO). The increasing demand for small satellites has created new opportunities for private companies, research institutions, and governments to access and leverage space-based data and services.

1.5 Challenges of Satellite Miniaturization

Despite the numerous benefits of satellite miniaturization, there are also several challenges associated with the development and operation of small satellites and below are listed some of the key challenges of satellite miniaturization.

1.5.1 Size Limitations

The most obvious limitation when it comes to miniaturizing satellites is size. The smaller the satellite is, the less room there is for components such as antennas, solar panels and batteries. This can limit the performance of a satellite in terms of both communication range and power generation capabilities. Additionally, small satellites may not have enough structural strength or thermal protection to survive in space for an extended period of time. For these reasons it is important that any miniaturization efforts take into account these limitations in order to ensure that the satellite will be able to perform its intended mission effectively.

1.5.2 Weight Limitations

Weight is also an important factor when it comes to miniaturizing satellites. Smaller satellites are often lighter than their larger counterparts due to their reduced component count and overall size. However this can also lead to instability during launch or operation in orbit due to their reduced mass relative to other objects in space such as debris or other spacecrafts. Additionally, lighter satellites may require additional support structures or thrusters in order to maintain their position in orbit over an extended period of time which can add weight and complexity back into the system design process making them less desirable from a cost perspective.

1.5.3 Power Requirements

Power requirements are another major limitation when it comes to miniaturizing satellites as they often require more power than larger versions due to their increased component count and decreased mass which reduces efficiency gains from traditional solar panel designs . Additionally , small form factor batteries may not have enough capacity or lifespan for longer duration missions making them unsuitable for certain applications . To combat these issues , designers often turn towards advanced energy storage solutions such as fuel cells , flywheels , or even nuclear power sources . These solutions tend to be more expensive however they allow designers greater flexibility when it comes designing smaller , more efficient systems .

1.5.4 Mission Requirements

The final limitation when it comes designing miniature satellites lies within its mission requirements . Depending on what type of mission a satellite will be performing , certain components may need more attention than others in order for it operate successfully . For example , if a satellite needs high precision pointing capabilities then additional components such as reaction wheels or reaction control systems may need added onto its design which can add complexity and cost back into the system design process . Additionally , if a satellite needs communication capabilities then large antennae may need added onto its design further increasing its size and weight making it less desirable from a cost perspective . As such , any miniature design should take into account all mission requirements before attempting any sort of miniaturization efforts .

1.5.5 Reliability Issues

The reliability of small satellites is a significant concern, given their reduced size and weight. The launch and operation of small satellites require high levels of precision, and any error or malfunction can result in mission failure or loss of the satellite. Additionally, small satellites are more susceptible to space weather and radiation, which can affect their performance and reliability over time.

1.6 Future Prospects of Satellite Miniaturization

The miniaturization of satellites is expected to continue in the future, driven by advancements in technology, increased demand for small satellites, and the growing importance of space-based data and services. For that reason some key prospect aspects of this development can be indentified.

1.6.1 Distributed Satellite Networks

Distributed satellite networks, also known as satellite constellations, have gained significant attention and investment in recent years due to their potential to provide global coverage and enable new applications in areas such as global internet connectivity, earth observation, and weather forecasting [6]. A satellite constellation consists of multiple small satellites that are designed to work together to provide a specific service or coverage area. Unlike traditional large satellites that are launched individually and operated independently, satellite constellations are typically designed to work as a coordinated network, with each satellite communicating with its neighboring satellites to share data and maintain a specific formation. Satellite constellations offer several advantages over traditional large satellites. First, they can provide much greater coverage than a single large satellite, as they can be designed to cover specific regions of the globe or provide continuous global coverage. Second, they can offer redundancy and resiliency, as the loss of one satellite in the network does not necessarily impact the overall functionality of the constellation. Finally, they can enable new applications that require low-latency, high-bandwidth communication or frequent and high-resolution imaging of the Earth's surface. Several companies are currently developing or operating satellite constellations. One of the most well-known examples is SpaceX's Starlink, which aims to provide global internet connectivity through a network of thousands of small satellites in low Earth orbit. Other companies such as OneWeb, Telesat, and Amazon's Project Kuiper are also developing similar networks. However, the development and operation of satellite constellations present several technical and regulatory challenges. First, maintaining a specific formation of satellites requires sophisticated communication and control systems, as well as precise orbital maneuvers. Second, the growing number of satellites in orbit has raised concerns about orbital debris and collisions, prompting regulators to impose stricter requirements for satellite design and operation. Finally, the large-scale deployment of satellite constellations has raised concerns about the impact on astronomy and the night sky. Despite these challenges, the potential benefits of satellite constellations have driven significant investment and research efforts in this area. The ongoing development of advanced technologies such as inter-satellite communication, artificial intelligence, and advanced propulsion systems is expected to enable new capabilities and applications for satellite constellations in the future.

1.6.2 Inter-satellite Communication

Inter-satellite communication is an emerging technology that can enable small satellites to communicate with each other, rather than relying on ground-based infrastructure. This technology can improve the efficiency and speed of data transmission, as well as enhance the resilience of space-based communication networks [7]. Traditionally, communication between satellites and the ground has been accomplished using radio-frequency (RF) communication, with data being transmitted between the satellite and ground stations using microwave frequencies. This method has several limitations, including limited bandwidth, long latency, and susceptibility to interference. Inter-satellite communication, on the other hand, allows small satellites to communicate directly with each other using optical, RF, or other forms of wireless communication. This technology can enable the transfer of data between satellites in a constellation or enable direct communication between satellites in different orbits. There are several potential benefits of inter-satellite communication. First, it can enable high-speed, low-latency communication between satellites, which can be critical for applications such as earth observation, remote sensing, and real-time data transmission. Second, it can reduce the need for ground-based communication infrastructure, which can be costly and vulnerable to disruptions. Finally, it can enhance the resilience of space-based communication networks, as satellites can continue to communicate with each other even if ground-based infrastructure is disrupted. Several companies and research organizations are currently developing and testing inter-satellite communication systems. For example, the European Space Agency (ESA) is developing a laser-based communication system called the European Data Relay System (EDRS), which enables high-speed communication between low-Earth orbit satellites and ground stations. In addition, companies such as Mynaric and BridgeSat are developing laser-based communication systems that can enable high-speed communication between satellites in low-Earth orbit. Despite its potential benefits, inter-satellite communication also presents several technical and regulatory challenges. First, designing and implementing an inter-satellite communication system requires sophisticated communication and control systems, as well as precise orbital maneuvers. Second, regulatory agencies such as the International Telecommunication Union (ITU) must ensure that the use of inter-satellite communication does not interfere with other communication systems or cause harmful interference. Finally, the use of laser-based communication systems for inter-satellite communication can raise concerns about the potential impact on the environment and the safety of other spacecraft in orbit. Despite these challenges, the ongoing development of advanced inter-satellite communication technologies is expected to continue to drive innovation in the space industry, enabling new capabilities and applications for small satellites.

1.6.3 Artificial Intelligence and Machine Learning

Artificial intelligence (AI) and machine learning (ML) are rapidly transforming the way small satellites are designed, operated, and utilized. These technologies can enable small satellites to perform complex tasks such as autonomous navigation, data analysis, and decision-making, which were previously only possible with larger, more expensive satellites [8]. AI refers to the ability of machines to perform tasks that would normally require human intelligence, such as perception, reasoning, and decision-making. ML is a subset of AI that focuses on algorithms that can learn from data and improve their performance over time. Together, AI and ML can enable small satellites to operate more efficiently, reduce costs, and enable new applications. One of the most significant applications of AI and ML in small satellite development is in autonomous navigation. Traditional satellite navigation systems rely on ground-based systems or expensive on-board sensors to determine a satellite's position and orientation. However, with the development of AI and ML algorithms, small satellites can use their on-board sensors and cameras to autonomously navigate in space. AI and ML can also be used to analyze data collected by small satellites. For example, small satellites equipped with cameras can collect large amounts of data on the Earth's surface, which can be used for applications such as earth observation, agriculture, and disaster response. With the use of AI and ML algorithms, small satellites can automatically process and analyze this data, identifying patterns and anomalies that would be difficult or time-consuming for humans to detect. In addition, AI and ML can enable small satellites to make decisions in real-time based on data collected during a mission. For example, small satellites equipped with sensors can detect changes in environmental conditions, such as changes in temperature or atmospheric pressure. With the use of AI and ML algorithms, small satellites can make decisions on the best course of action based on these environmental changes, such as adjusting their orbit or activating specific sensors to collect additional data. Several companies and research organizations are currently developing and testing AI and ML algorithms for small satellites. For example, NASA's Jet Propulsion Laboratory is developing autonomous navigation algorithms for small satellites, while companies such as BlackSky and Capella Space are using AI and ML to analyze earth observation data collected by their small satellite constellations. Despite the potential benefits of AI and ML for small satellite development, there are also several challenges that must be addressed. One of the main challenges is the limited computing power and storage capacity of small satellites, which can make it difficult to implement complex AI and ML algorithms. In addition, the use of AI and ML in small satellites raises concerns about the potential for errors or malfunctions in autonomous systems, which could have significant consequences. Despite these challenges, the ongoing development of AI and ML technologies is expected to continue to drive innovation in the small satellite industry, enabling new capabilities and applications for small satellites.

1.6.4 3D Printing and Additive Manufacturing

3D printing and additive manufacturing (AM) are technologies that can revolutionize the design and production of small satellites. These technologies can enable the production of complex parts and structures with high precision and accuracy, while also reducing costs and lead times. 3D printing refers to the process of creating a three-dimensional object by layering material, such as plastic or metal, according to a digital design. AM is a broader term that encompasses a range of technologies that use additive processes to create objects, including 3D printing. One of the main advantages of 3D printing and AM for small satellite development is their ability to produce complex geometries that would be difficult or impossible to produce using traditional manufacturing techniques. For example, 3D printing can enable the production of lightweight, high-strength parts with intricate internal structures, which can improve the performance and efficiency of small satellites. In addition, 3D printing and AM can reduce the lead time and costs associated with traditional manufacturing techniques. With traditional manufacturing, producing a prototype or a small batch of parts can be expensive and time-consuming, as it requires the creation of specialized tooling and molds. 3D printing and AM can eliminate the need for these tools, enabling rapid prototyping and the production of small batches of parts at a lower cost. Several companies and research organizations are currently developing and testing 3D printing and AM technologies for small satellite development. For example, NASA's Marshall Space Flight Center is using 3D printing to produce rocket engine components, while companies such as Made In Space and Relativity Space are using 3D printing to produce rocket components and entire rockets. Despite the potential benefits of 3D printing and AM for small satellite development, there are also several challenges that must be addressed. One of the main challenges is the limited range of materials that can be used in 3D printing and AM, particularly for applications that require high strength and durability. In addition, the precision and accuracy of 3D printing and AM can be affected by factors such as temperature and humidity, which can make it difficult to produce parts with consistent quality. Also, the ongoing development of 3D printing and AM technologies is expected to continue to drive innovation in the small satellite industry, enabling new capabilities and applications for small satellites.

1.6.5 Advances in Propulsion Systems

Advances in propulsion systems, such as electric propulsion and micro-thrusters, can enhance the maneuverability and operational lifespan of small satellites. These systems can enable small satellites to maintain their orbit, avoid orbital debris, and perform complex maneuvers, such as formation flying and orbital rendezvous.Advances in propulsion systems, such as electric propulsion and micro-thrusters, can enhance the maneuverability and operational lifespan of small satellites. These systems can enable small satellites to maintain their orbit, avoid orbital debris, and perform complex maneuvers, such as formation flying and orbital rendezvous.

1.7 Sustainability

The 2030 Agenda for Sustainable Development, adopted by all United Nations Member States in 2015, provides a shared blueprint for peace and prosperity for people and the planet, now and into the future [9]. At its heart are the 17 Sustainable Development Goals (SDGs), which are an urgent call for action by all countries, developed and developing, in a global partnership. They recognize that ending poverty and other deprivations must go hand-in-hand with strategies that improve health and education, reduce inequality, and spur economic growth while tackling climate change and working to preserve our oceans and forests. The space

Figure 1.1: Sustainable Development Goals

industry and all the research programs are day by day more and more compliant on achieve at least some of them in their contribute to achieves those honorable goals. The research and implementation of miniaturized technologies in the space environment could provide achive many of them:

- 1) No Poverty.
  The miniaturization of satelites could lead to cheaper solutions to provide communications even in the most remote places on the earth allowing the population living in those areas to grow and become more and more sofisticated in the future.

- 4) Quality education.
  A cheaper and miniaturized satellite could provide a very powerful educational tool for students to interface with being able to actively interact with a scaled

technologies and better learn how it works and even suggest what can be improved about them.

- 8) Decent Work and Economic Growth
  Again the space industry and economy is becoming year by year bigger and bigger and making space technologies cheaper and more powerful is certanly a strong help in this direction for it involving more people directly and providing more solutions to today problems.

- 9) Industry Innovation and Infrastructure.
  As its a quite new field of intensive research the miniaturization of space technologies is clearly a great help for the all the space industries continusly providing innovations and different solutions.

- 10) Reduced Inequalities. As cited before the implementation of miniaturized and cheaper solution in the space industry coul provide much more services on Earth and accessible to more and more pople around the world.

Chapter 2

Satellites Miniaturized Technologies

2.1 Introduction

In this chapter will be given a general overview of some of the most promising technologies in the miniaturized satellite industry for some of the generally present subsystem on a spacecraft. There will be shown many state of the art technologies that could be miniaturized even more in the future following the path described in the previous chapter which states the main drivers that will lead the research on this particular field.

2.2 Payload Subsystem

In recent years, there have been significant developments in the miniaturization of payloads, which has enabled the creation of more capable miniaturized satellites. For example, there have been advancements in the development of miniaturized spectrometers, which can be used for remote sensing and scientific research missions. In addition, miniaturized attitude control systems are being developed, which will enable the creation of more maneuverable and capable S/Cs.

Looking ahead, the trend in the development of small satellites payloads is towards further miniaturization and increased capability. As technology continues to advance, it is likely that more sophisticated sensors and instruments will be developed that can be integrated into the small form factor of satellites. In addition, there is a growing trend towards the development of miniaturized satellites for specific applications, such as Earth observation and space exploration missions. These satellites will carry payloads that are tailored to their specific mission objectives.

Overall, while the limited size poses challenges for the integration of payloads, advancements in miniaturization are enabling the creation of more capable and versatile satellites. As technology continues to advance, it is likely that the capabilities of S/Cs payloads will continue to improve, opening up new opportunities for scientific research and commercial applications.

As technology continues to advance, it is likely that more sophisticated sensors and instruments will be developed that can be integrated into the small form factor satellites and some of them are listed below:

- Hyperspectral Imagers: Hyperspectral imagers can be used to capture images of the Earth or other planetary surfaces in hundreds of spectral bands, which can provide valuable information about the composition of those surfaces. While some hyperspectral imagers can be large and complex, there are also miniaturized versions that could potentially fit within a 5 cm cube [10]. For example, a miniaturized hyperspectral imager is the Micro-Hyperspec developed by Headwall Photonic, which has a volume of approximately 6 cm x 6 cm x 6 cm and weighs around 400 grams. It's worth noting that miniaturized hyperspectral imagers may have trade-offs in terms of spatial or spectral resolution compared to larger systems, and they may require additional components such as a pointing system or data processing capabilities to function properly. But overall, miniaturized hyperspectral imagers are a promising area of research and development for small spacecraft applications. Those could be integrated into into a small form factor satellite for remote sensing and scientific research missions.

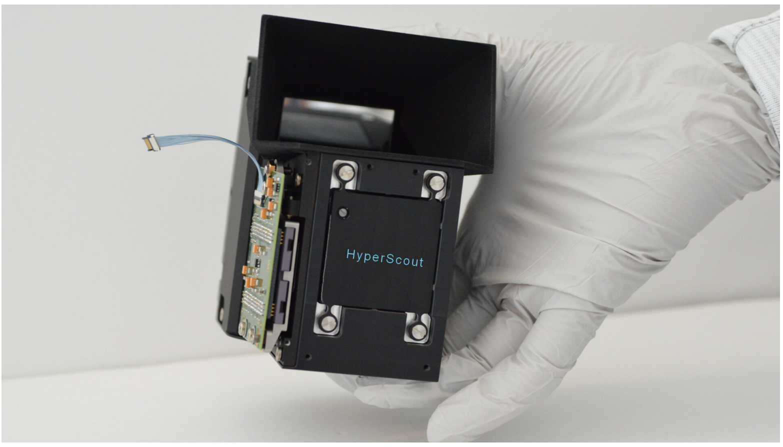


Figure 2.1: Hand-sized hyperspectral camera (ESA)

- Miniature X-ray/Telescope Monitor: Miniature X-ray/Telescope Monitors (XRTMs) are designed to detect and monitor X-rays and gamma rays from space sources, such as solar flares and gamma-ray bursts [11]. They typically consist of a small detector, electronics, and sometimes a collimator to focus the incoming radiation. A small telescope has been theorized to fit a standard pico-satellite weighting around 100 grams and with a FOV of around 2 deg x 2 deg. This type of payload could be suitable in a constellation of satellites. A constellation of XRTMs could provide coverage over a larger area and enable more complete monitoring of the X-ray and gamma-ray sky. However, there are some technical and logistical challenges to consider, such as coordinating the orbits and ensuring that the data from each satellite is transmitted and processed in a timely manner. Thus, the specific requirements and trade-offs will depend on the scientific objectives and technical constraints of the mission.

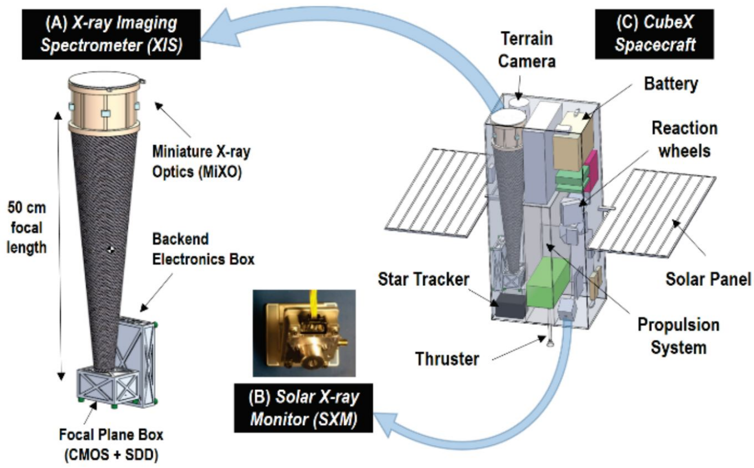


Figure 2.2: Compact X-ray/Telescope Monitor (CubeX)

- Visible/IR phototransistors: Visible/IR phototransistors are electronic devices that can detect light in the visible and near-infrared regions of the electromagnetic spectrum. They are typically small and low-power, which makes them well-suited for use in small spacecraft. They can be used for a variety of applications, such as star trackers, Earth observation, and imaging. For example, the NASA Star Tracker System (STS) uses a visible-light phototransistor to determine the attitude and orientation of a spacecraft relative to the stars. Other small spacecraft missions, such as CubeSats, have also used phototransistors for Earth observation and imaging. One advantage of phototransistors is that they are relatively simple and low-cost compared to other imaging technologies, such as cameras. They can also operate in low-light conditions and have high sensitivity to specific wavelengths of light. However, they may have limitations in terms of spatial resolution and image quality compared to other imaging technologies. Overall, visible/IR phototransistors are a viable option for a small spacecraft payload, and they have been used successfully in a variety of applications.

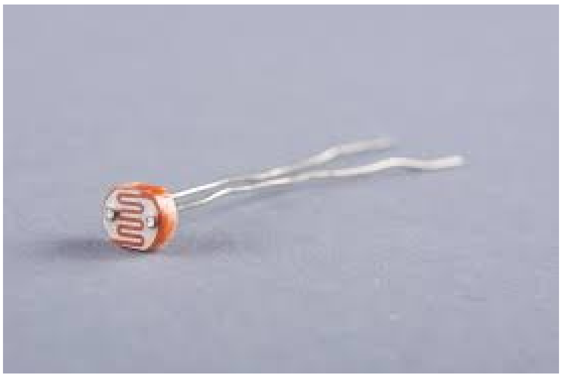


Figure 2.3: IR phototransistors

- Magnetospheric Plasma Analyzers: Magnetospheric Plasma Analyzers (MPAs) are instruments that can measure the properties of plasma in the Earth's magnetosphere. They are typically composed of a set of electrostatic analyzers that can measure the energy and angular distribution of ions and electrons in the plasma [12]. MPAs have been used in a variety of scientific missions to study the Earth's magnetosphere, including the Magnetospheric Multiscale (MMS) mission launched by NASA in 2015. The MMS mission used four identical spacecraft equipped with MPAs to study the fundamental processes of magnetic reconnection, which can release large amounts of energy and affect space weather. Overall they can be suitable for small spacecraft applications, but there are some challenges to consider. MPAs require complex electronics and high-voltage power supplies to operate, which can increase their mass and power requirements. They also require careful calibration and data processing to ensure accurate measurements of plasma properties.

2.3 Guidance Navigation and Control System

Small satellites are becoming increasingly popular for their lower cost and faster development timelines. They are also being used for various applications such as remote sensing, Earth observation, communication, and scientific research. One of the critical components of small satellites is the Guidance Navigation and Control (GNC) system, which is responsible for ensuring that the satellite is correctly oriented and positioned in space. The GNC system typically consists of Attitude Determination and Control System (ADCS) and Global Positioning System (GPS) receiver.

This chapter provides an overview of the GNC systems in small satellites and discusses the main functions of the ADCS and GPS subsystems. It also explores the different technologies applicable to small form factor satellites, including integrated units, reaction wheels, magnetic torquers, star trackers, magnetometers, sun sensors, horizon sensors, atomic clocks, and GPS receivers. Finally, it discusses future development technologies that are driving the miniaturization of space technologies.

The Attitude Determination and Control System (ADCS) is responsible for controlling the attitude and orientation of the satellite in space. The primary functions of ADCS include:

- Attitude Determination.
  This function determines the satellite's orientation relative to the Earth and the celestial sphere. Attitude determination is accomplished by using sensors such as sun sensors, magnetometers, and star trackers.

- Attitude Control.
  This function controls the satellite's orientation by applying external forces. Attitude control is achieved by using actuators such as reaction wheels, magnetic torquers, or even thrusters.

- Pointing Accuracy.
  This function ensures that the satellite points accurately towards the target. It is crucial for the success of many satellite missions such as remote sensing and Earth observation.

Small satellites have revolutionized the space industry, providing an accessible and affordable platform for various scientific, commercial, and military applications. With the miniaturization of space technologies, the Guidance, Navigation, and Control (GNC) system has become an essential component of small satellites, enabling them to perform their missions accurately and efficiently. This chapter will focus on the main functions of the GNC system in small satellites, specifically the Attitude Determination and Control System (ADCS) and the Global Positioning

System (GPS). It will also provide a comprehensive overview of the main technologies applicable to small form-factor satellites, such as integrated units, reaction wheels, magnetic torquers, star trackers, magnetometers, sun sensors, horizon sensors, atomic clocks, and GPS receivers. Lastly, this chapter will discuss future developments in GNC systems, focusing on the trend towards miniaturization and the integration of advanced technologies.

2.3.1 Attitude Determination and Control System (ADCS)

The ADCS is a critical component of the GNC system that enables a small satellite to maintain its desired attitude, i.e., orientation, in space. The ADCS is responsible for determining the satellite's attitude, maintaining it, and correcting any deviations from the desired attitude. The main components of the ADCS are the sensors, actuators, and control algorithms [13].

In particular, control algorithms are used to determine the required actuator torques to maintain the satellite at the desired attitude. Control algorithms are responsible for processing the sensor data and determining the required actuator torques to maintain the satellite's attitude. There are several types of control algorithms that are commonly used in ADCS systems for small satellites, including:

- Proportional-Integral-Derivative (PID) Control.
  PID control is a very common control algorithm that is used to maintain the satellite's attitude at the desired setpoint. The algorithm calculates the error between the measured attitude and the desired setpoint, and applies proportional, integral, and derivative terms to determine the required actuator torques.

- Non-Linear Control - Quaternion Feedback Algorithm.
  Quaternion feedback is a popular non-linear control algorithm used for attitude control in satellites. It is based on the concept of quaternion rotation, which is a mathematical tool used to represent the rotation of an object in three-dimensional space. The algorithm works by sensing a rotation and using the quaternion rotation to calculate the required torque to correct the attitude of the satellite.

- Model Predictive Control
  MPC is an advanced control algorithm that uses a mathematical model of the system to predict the future behavior of the system and determine the required actuator torques to maintain the desired attitude.

- Kalman Filter
  The Kalman filter is an algorithm that is used to estimate the satellite's attitude and rate of change using sensor data. It combines the sensor measurements

with a mathematical model of the system to produce an optimal estimate of the state of the system.

2.3.2 Technologies Applicable to Small Form-Factor Satellites

Small satellites have unique design requirements and constraints, which require the use of specialized technologies for the GNC system and the following rows will provide an overview of the main technologies applicable to small form-factor satellites looking at the state of the art of today technology.

- Integrated units.
  Integrated units are devices that combine multiple components of a GNC system into a single package. They typically include sensors for attitude determination (such as magnetometers and sun sensors), as well as actuators for attitude control (such as reaction wheels and magnetic torquers). Examples of integrated units for small satellites include the Attitude Determination and Control System (ADCS) developed by CubeSpace, and the integrated GNC system developed by Blue Canyon Technologies.

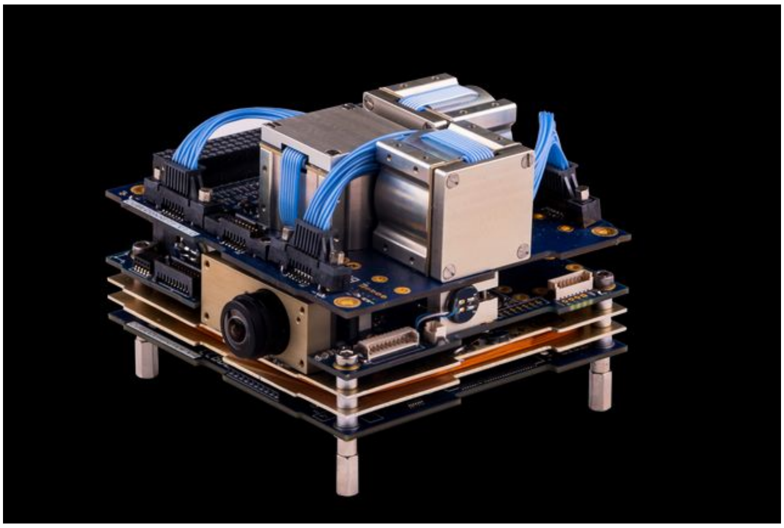


Figure 2.4: Miniaturized Integrated Unit for CubeSats (CubeSpace)

- Reaction wheels.
  Reaction wheels are devices that use the principle of conservation of angular momentum to control the attitude of a satellite. They consist of a spinning wheel that can be accelerated or decelerated to change the satellite's orientation. By controlling the speed of the reaction wheels, the satellite can adjust its

pitch, yaw, and roll. Reaction wheels are a popular choice for small satellites due to their low power consumption and high accuracy. Examples of reaction wheel products for small satellites include technologies from Blue Canyon Technologies, GomSpace and NanoAvionics.

Figure 2.5: Miniaturized 4 Reaction-Wheel Cluster (NanoAvionics)

- Magnetic Torquers. Magnetic Torquers are devices that use electromagnetic coils to interact with the Earth's magnetic field and generate a torque on the satellite. By controlling the current in the coils, the satellite can adjust its orientation. Magnetic torquers are particularly useful for small satellites that do not have large reaction wheels or thrusters, as they require very little power and can provide precise control. Examples of magnetic torquer products for small satellites include the MTQ30 from AAC Clyde Space and the Miniature Magnetic Torquer from Blue Canyon Technologies.

Figure 2.6: Scalable Magnetic Torquers for Small Satellite (AAC Clyde Space)

- Star trackers.
  Star trackers are devices that use a camera and sophisticated algorithms to determine the orientation of a satellite based on the positions of stars in the sky. By comparing the observed positions of stars to a catalog of known stars, a star tracker can determine the satellite's attitude with high accuracy. Star trackers are a popular choice for high-precision missions, such as Earth observation and remote sensing. Examples of star tracker products for small satellites include the Fine Sun Sensor and Star Tracker from AAC Clyde Space and the Hydrazine Star Tracker from GomSpace.

Figure 2.7: PicoStar Star Tracker from Space Micro

- Magnetometers.
  Magnetometers are devices that measure the strength and direction of the Earth's magnetic field. By measuring changes in the magnetic field, a magnetometer can detect changes in the satellite's orientation. Magnetometers are a common component of integrated GNC systems, as they provide a low-cost, low-power method for attitude determination. Examples of magnetometer products for small satellites include the Miniature Magnetometer from Blue Canyon Technologies and the Tri-axial Magnetometer from GomSpace.

- Sun sensors.
  Sun sensors are devices that measure the position of the sun relative to the satellite. By measuring changes in the sun's position, a sun sensor can detect changes in the satellite's orientation. Sun sensors are a popular choice for low-cost missions, as they require very little power and provide a simple method for attitude determination. Examples of sun sensor products for small satellites include the CubeSense from Pumpkin Space Systems and the Sun Sensor from Redwire Space.

- Horizon sensors.
  Horizon sensors are devices that measure the position of the Earth's horizon relative to the satellite. By measuring changes in the horizon's position, a

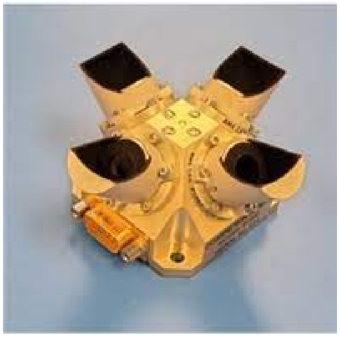


Figure 2.8: Miniaturized Sun Sensor from Redwire Space

horizon sensor can detect changes in the satellite's orientation. Horizon sensors are commonly used in conjunction with other sensors, such as sun sensors and magnetometers, to provide redundant attitude determination.

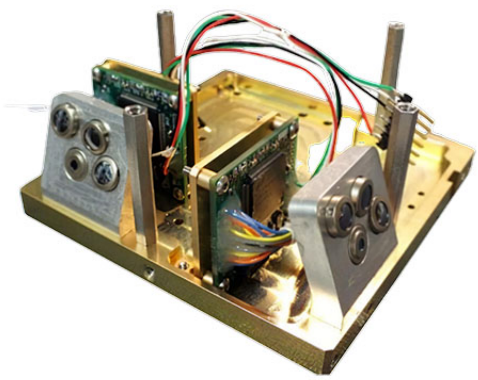


Figure 2.9: Miniaturized Horizon Sensor from Adcole Maryland Aerospace

- Atomic clocks.
  Atomic clocks are devices that measure time based on the vibrations of atoms. They are used in GNC systems to provide precise timing information for satellite operations, such as scheduling maneuvers or coordinating data transfers [14]. Atomic clocks are particularly important for scientific missions, where accurate timing is critical. Examples of atomic clock products for small satellites include the Miniature Atomic Clock from Spectratime and the CSAC Clock from Microchip Technology.

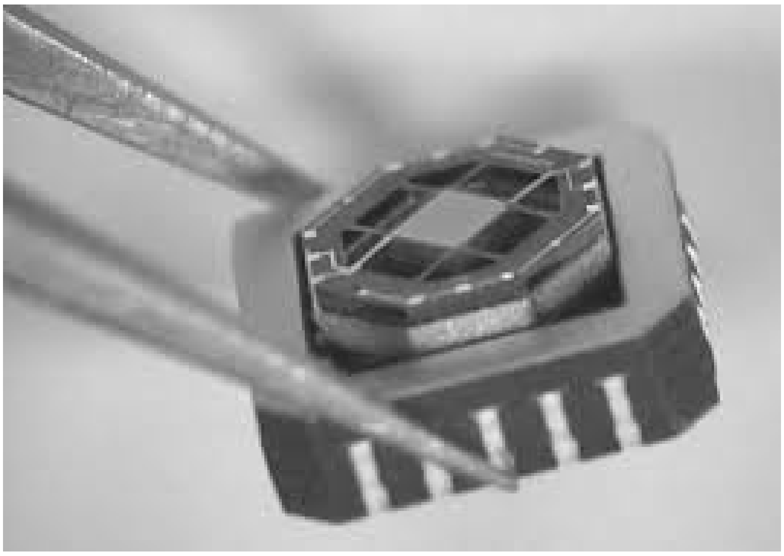


Figure 2.10: Chip-Size Atomic Clock (Charles Stark Draper Laboratory)

The trend towards miniaturization of space technologies is expected to continue in the future, driven by the need for more cost-effective and accessible space missions. This trend is expected to result in the development of new technologies and improvements in existing technologies that are more suitable for small form-factor satellites. This chapter will provide an overview of some of the future development technologies that are expected to have an impact on small satellites' GNC systems.

- Miniature Inertial Measurement Units (IMUs).
  Miniature IMUs are expected to be developed in the future, providing accurate and reliable attitude determination in a small form factor. These devices will likely use micro-electromechanical systems (MEMS) technology and will be integrated into small form-factor packages.

- Reaction Wheels. Even smaller reaction wheels are expected to be developed in the future, providing more precise attitude control in a small form factor. These devices will likely use advanced materials and manufacturing techniques to achieve high precision and low weight.

- Miniature Magnetometers.
  Miniature magnetometers are expected to be developed in the future, providing accurate attitude determination in a small form factor. These devices will likely use advanced materials and manufacturing techniques to achieve high precision and low weight [15].

Figure 2.11: MEMS IMU from Honeywell Aerospace

Figure 2.12: Miniaturized Reacrion Wheel (Astrofein)

- Optical Navigation Sensors.
  Optical navigation sensors are expected to be developed in the future, providing accurate and reliable navigation information in a small form factor. These devices will likely use advanced imaging technology and will be integrated into small form-factor packages.

- Quantum Sensors.
  Quantum sensors are a new type of sensor that use quantum mechanics to measure physical quantities with high precision. These sensors are currently being developed for a variety of applications, including navigation and timing [16]. In the future, quantum sensors may be integrated into small form-factor packages, providing highly accurate and reliable navigation and timing information for small satellites.

2.3.3 Summary

Guidance, Navigation, and Control (GNC) systems are critical components of small satellite missions, enabling precise control of the satellite's position and

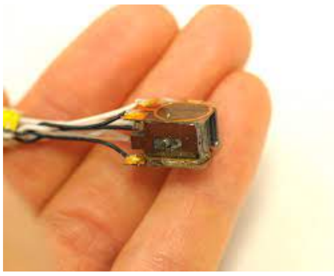


Figure 2.13: Miniaturized atomic vapor cell magnetometer (National Institute of Standards and Technology)

Figure 2.14: Miniaturized Quantum Sensor (LI-COR)

orientation. The miniaturization of space technologies has led to the development of a wide range of GNC technologies that are well-suited for small form-factor satellites. These include integrated units, reaction wheels, magnetic torquers, star trackers, magnetometers, sun sensors, horizon sensors, atomic clocks, and GPS receivers. As the field of small satellite missions continues to evolve and mature, GNC systems will play an increasingly important role in enabling complex and innovative missions. With the continued development of new technologies and the growing trend towards miniaturization, the possibilities for small satellite missions are limited only by our imagination.

2.4 Communication Subsystem

One of the critical subsystems in a satellite is the communication subsystem. This subsystem enables the satellite to communicate with the ground station and other satellites in the network. The communication subsystem in a miniaturized satellite is different from that in a traditional satellite, mainly due to the space and power constraints. This chapter provides an overview of the communication subsystem in a satellite and the technologies adapted in small form factor ones. It also discusses the various technologies that can be implemented considering the volume, mass, and power budget [17]. The primary purpose of the communication subsystem in a PocketQube satellite is to enable communication with the ground station and other satellites in the network. The communication subsystem consists of two primary functions, i.e., uplink and downlink. Uplink refers to the transmission of data from the ground station to the satellite. The data transmitted could include commands, software updates, or scientific data to be analyzed onboard the satellite. Downlink, on the other hand, refers to the transmission of data from the satellite to the ground station. The data transmitted could include telemetry data, scientific data, or images captured by the onboard cameras. Apart from uplink and downlink, the communication subsystem in a small form factor satellite also supports inter-satellite communication. Inter-satellite communication enables communication between multiple satellites in a network, enabling them to work together to achieve a common objective. For example, a constellation of miniaturized satellites could be used for remote sensing or earth observation applications. There are several frequency bands that can be used for communication in space, depending on the application and mission requirements. Here are some of the most common frequency bands used in space communication:

- VHF (Very High Frequency): VHF frequencies range from 30 MHz to 300 MHz and are often used for short-range communication and telemetry in low-Earth orbit (LEO) missions.
- UHF (Ultra High Frequency): UHF frequencies range from 300 MHz to 3 GHz and are commonly used for satellite-to-satellite and satellite-to-ground communication in LEO missions.
- L-Band: L-Band frequencies range from 1 GHz to 2 GHz and are used for a variety of satellite communication applications, including mobile satellite services, satellite navigation, and remote sensing.
- S-Band: S-Band frequencies range from 2 GHz to 4 GHz and are commonly used for telemetry, command, and data transmission in satellite missions.
- C-Band: C-Band frequencies range from 4 GHz to 8 GHz and are often used for

satellite communication, including television and radio broadcasting, mobile satellite services, and remote sensing.

- X-Band: X-Band frequencies range from 8 GHz to 12 GHz and are commonly used for high-data-rate communication, including satellite-to-ground and satellite-to-satellite communication.
- Ku-Band: Ku-Band frequencies range from 12 GHz to 18 GHz and are often used for high-data-rate satellite communication, including direct broadcast television and satellite internet.
- Ka-Band: Ka-Band frequencies range from 26.5 GHz to 40 GHz and are used for high-data-rate satellite communication, including broadband satellite internet and remote sensing.

The choice of frequency band depends on several factors, including the required data rate, the distance between the satellite and the ground station, the power consumption of the communication subsystem, and the regulatory environment. Mission designers must carefully evaluate the available frequency bands and choose the most suitable one based on their specific mission requirements.

2.4.1 Antennas

Antennas are typically used to tranmsmit data trough space using electromagnetic waves and are generally sized for their respective frequencies. For missions that dont require a high data rate a simple patch omnidirectional antenna could provide enough performances even when the satellite is thumbling or its pointing accuracy is not accurate enough. There are two main categories of antennas:

- Fixed Antennas. those remain stationary and do not require any mechanisms which can guarantee a much reliable mechanical system. Some examples can be patch antennas, monopole antennas, or array antennas.

- Deployable Antennas. They usually require power or some sort of elastic mechanism to deploy and reach their final position which could lead to a potential failure of the system but they can also lead to a more compact design saving some much needed volume. Some examples of those mechanisms are represented by a quadrifilar helical antenna or the EnduroSat quadruple UHF Antenna that can deploy from each side of the panel.

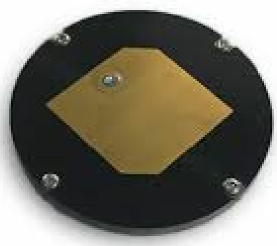


Figure 2.15: Miniaturized Omnidirectional Patch Antenna from ISIS Space

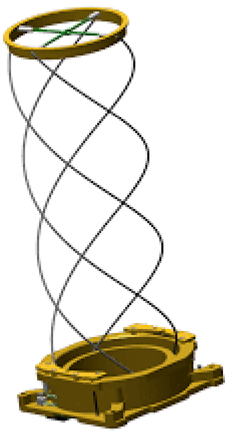


Figure 2.16: Quadrifilar Helical Deployable Antenna (HCT)

2.4.2 Inter-Satellite Communication

Some of the technologies that can be used for satellite-to-satellite communication include inter-satellite optical links which use PIC-based lasers that offer reduced cost, size, weight, and power compared to conventional systems. Satellite Internet of Things (IoT) is another major trend enabling unprecedented connectivity across industries and empowering 5G and upcoming 6G capabilities. Satellites manufacturers and operators are bringing technological innovations to ground stations as well as orbital service that however are not discussed in this thesis since the focus is more on the technologies on board of the spacecraft.

2.4.3 Optical Communication

One of the main promising technology applicable to this subsystem are the Laser receivers which implements the use of electromagnetic radiation to transmit data between terminals [18]. This technology is quite new and the most reliable proof of concept has been mad into laboratories. This relatively new technology could offer better overall performances than the general RF systems due to its larger bandwidth and sub-millimeters wavelengths. In fact Laser communications terminals in space use narrower beam widths than radio frequency systems, providing smaller "footprints" that can minimize interference or improve security by drastically reducing the geographic area where someone could intercept a communications link. However, a laser communications telescope pointing to a ground station must be exact when broadcasting from hundreds of kilometers away. A deviation of even

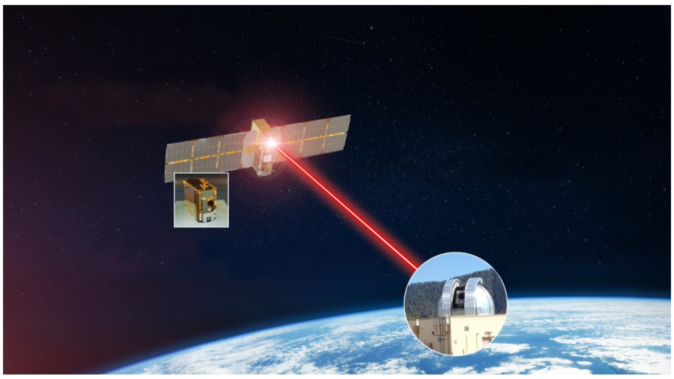


Figure 2.17: Optical Communication Concept for Small Satellites (MIT Lincoln Lab)

a fraction of a degree can result in the laser missing its target entirely. Also they are very difficult to intercept and are less subject to external disturbance providing a more reliable and stiff system. One of the other main advantages brought by this technology is also its lower power consumption providing an overall better effinceny of the system which is crucial point for the direction of the trend.

2.5 Propulsion Subsystem

Small satellites have been gaining significant attention in recent years due to their low-cost and increased accessibility for various space missions. The propulsion system is an essential component of small satellites that enables them to perform orbital maneuvers, adjust their position and altitude, and deorbit at the end of their mission life. The propulsion system's primary function is to provide thrust to overcome the gravitational force and maintain the desired orbit. The propulsion system's performance is crucial to the mission's success and directly impacts the satellite's lifetime and capability. This chapter will discuss the main propulsion technologies applicable to small form factor satellites, their technical descriptions, and key integration and operational considerations towards the miniaturization of each one.

2.5.1 Propulsion Technologies applicable to Small Form Factor Satellites

In this subsection will be listed the more promising technologies that can be implemented in the future to have a reliable propulsion system in small satellites that could help achieve many more mission goals not only in the Low Earth Orbit but even further.

- Alternative Monopropellants and Bipropellants.
  Alternative monopropellants and bipropellants are the most commonly used propulsion systems in small satellites due to their high specific impulse and thrust capabilities. Monopropellants are liquids that decompose into gases and generate thrust, while bipropellants require a combination of two liquids to produce thrust. These propulsion systems offer high efficiency and are easy to handle, making them popular choices for small satellites. Alternative monopropellants, such as hydrogen peroxide, offer higher performance and safety compared to conventional monopropellants like hydrazine. Hydrogen peroxide is a powerful oxidizer that decomposes into steam and oxygen when it comes in contact with a catalyst [19]. This reaction generates thrust, which can be controlled by varying the catalyst's concentration. On the other hand, bipropellants like nitrogen tetroxide and hydrazine provide high thrust and specific impulse but require careful handling due to their toxicity and corrosiveness. The use of alternative monopropellants and bipropellants in small satellites requires careful storage, handling, and safety measures to avoid contamination and corrosion. The propulsion system's components, such as tanks, valves, and filters, must be designed to withstand the corrosive nature of the propellant. Additionally, the integration of the propulsion system with

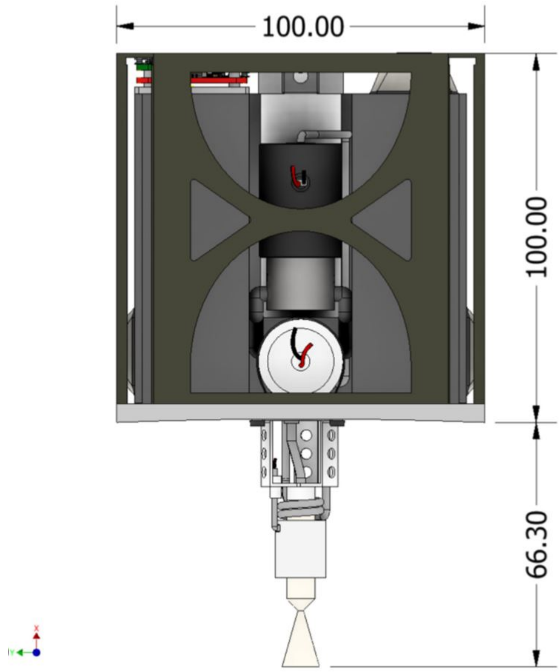


Figure 2.18: Green Propellant Micro-Thruster Design from Delft TU

the satellite's power and control systems must be carefully planned to ensure smooth operation and optimal performance.

- Cold Gas/Warm Gas.
  Cold gas and warm gas propulsion systems are relatively simple and easy to operate, making them popular choices for small satellites. These propulsion systems use compressed gas to generate thrust, with cold gas systems using unheated gas and warm gas systems heating the gas before expansion. Cold gas propulsion systems use compressed gas, such as nitrogen or helium, to generate thrust [20]. The gas is stored in high-pressure tanks and is released through a nozzle to generate thrust. Warm gas propulsion systems, on the other hand, heat the gas before expansion to increase its specific impulse. The gas is heated using resistive heaters or by passing it through a catalytic bed. The heated gas is then expanded through a nozzle to generate thrust. Cold gas and warm gas propulsion systems require relatively simple integration and operational considerations compared to other propulsion systems. However, careful design and planning are necessary to ensure optimal performance and reliability. The propulsion system's tanks, valves, and filters must be carefully designed to withstand the high pressures and temperatures involved in the operation. Additionally, the integration with the satellite's power and control

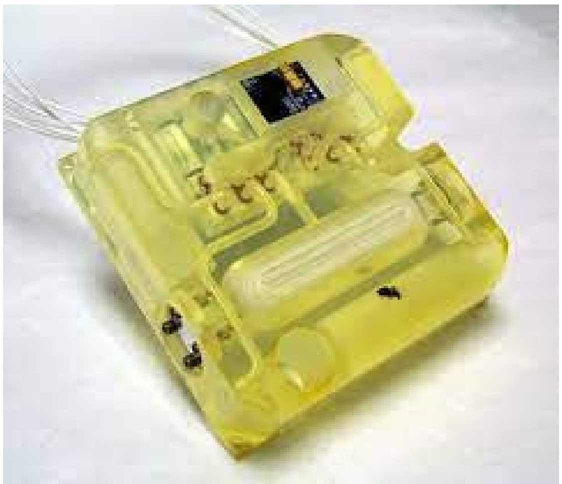


Figure 2.19: MEMS Propulsion of the MEPSI Mission

systems must be carefully planned to ensure smooth operation and optimal performance.

- Electrothermal Propulsion.
  Electrothermal propulsion systems use electric energy to heat a propellant and generate thrust, providing high specific impulse and low thrust capabilities. These propulsion systems offer good performance and efficiency for small satellites, making them popular choices for low-thrust maneuvers [21]. Electrothermal propulsion systems use electric energy to heat a propellant, such as a noble Electrothermal propulsion is another technology used in small satellites, where electrical energy is converted into thermal energy to vaporize a propellant and produce thrust. In this method, an electrical resistor or an arc discharge is used to heat the propellant, which is then expelled out of a nozzle to produce thrust. Electrothermal propulsion systems have a high specific impulse compared to chemical propulsion systems, making them more efficient. Additionally, they are more controllable than cold gas thrusters and simpler than ion thrusters. However, the specific impulse is lower than that of ion thrusters, which limits their use in deep space missions. For what concerns integration and operational considerations for electrothermal propulsion systems it has to be included the need for a power supply, which is a key consideration in small satellites. Efficient power conversion and management systems are required to provide sufficient electrical power to the propulsion system while maintaining the overall power budget of the satellite. The heating elements and propellant tanks must also be designed to withstand the high temperatures generated during operation.

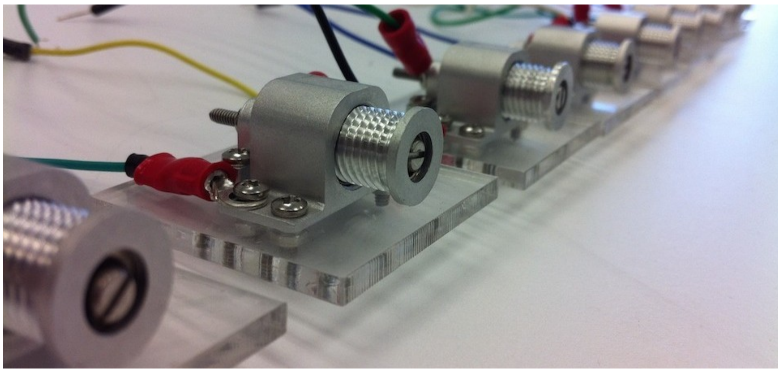


Figure 2.20: Miniaturized Concept of a Resistojet Propulsion from the George Washington University

- Electrospray Propulsion.
  Electrospray propulsion is a type of electric propulsion that utilizes electrospray thrusters to generate thrust by expelling ionized propellant. The system consists of a liquid propellant tank, an electrospray emitter, and a power supply [22]. The propellant is ionized by applying a high voltage to the electrospray emitter, and the ions are expelled through a nozzle to generate thrust. Electrospray propulsion offers a high specific impulse and low propellant consumption, making it ideal for small satellite applications. However, it has a low thrust-to-weight ratio, limiting its use for missions that require high thrust. The integration and operational considerations for electrospray propulsion

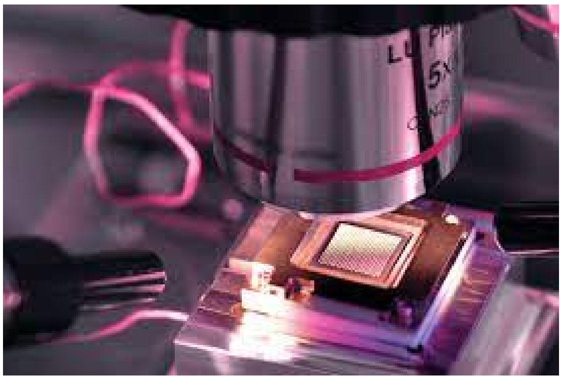


Figure 2.21: Miniaturized Concept of ElecroSpray Propulsion from NASA Laboratories

  systems include the need for a high voltage power supply and efficient power conversion and management systems. The emitter and propellant tank must also be designed to withstand the high voltages generated during operation.

Furthermore, the system must be designed to minimize the risk of electric discharge and arcing, which can damage the satellite and affect its operation.

- Gridded Ion Propulsion.
Gridded ion propulsion is another type of electric propulsion that uses a grid of electrodes to ionize a propellant and generate thrust. The system consists of a propellant tank, an ionization chamber, and an acceleration grid. The propellant is ionized by applying a voltage to the ionization chamber, and the ions are accelerated through the acceleration grid to generate thrust. Gridded ion propulsion systems offer a high specific impulse and low propellant

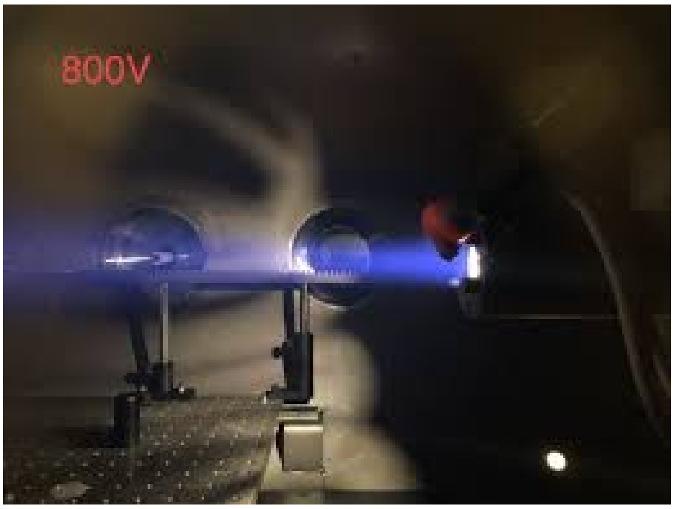


Figure 2.22: Miniaturized Concept of Gridded Ion Propulsion from Perdue University

consumption, making them ideal for deep space missions. However, they require complex and expensive components, making them less suitable for small satellite applications. Integration and operational considerations for gridded ion propulsion systems include the need for a high voltage power supply and efficient power conversion and management systems. The ionization chamber and acceleration grid must also be designed to withstand the high voltages generated during operation. Furthermore, the system must be designed to minimize the risk of arcing and discharge, which can damage the satellite and affect its operation.

2.5.2 Propellant-less Solutions

In this section will be discussed also, for completion, some of the most accredited propellant-less solution that can be applied to achieve different goals and be compliant with some mission requirements. Propellantless solutions, such as solar sails, electrodynamic tethers, and aerodynamic drag, offer an alternative to traditional propulsion systems that rely on the use of propellants. Solar sails use the pressure of sunlight to generate thrust, while electrodynamic tethers utilize the Earth's magnetic field to generate an electric current that can be used to produce thrust. Aerodynamic drag can also be used to slow down or reposition a satellite. Integration and operational considerations for propellantless solutions include the need for efficient power management systems to operate the systems. The materials used in solar sails must be strong, lightweight, and capable of withstanding the harsh conditions of space. Electrodynamic tethers must be designed to minimize the risk of breakage or failure, which can affect the operation of the satellite. Aerodynamic drag solutions must be designed to minimize the risk of damage to the satellite.

2.5.3 Summary

In conclusion, small satellite propulsion technologies have advanced significantly in recent years, with a wide range of options available for mission planners and spacecraft designers. Alternative monopropellants and bipropellants, cold gas and warm gas propulsion, electrothermal propulsion, gridded ion propulsion, and propellantless solutions such as solar sails, electrodynamic tethers, and aerodynamic drag can all be used to achieve the desired propulsion characteristics for a given mission. Integration and operational considerations for each technology must be carefully addressed to ensure successful mission outcomes, taking into account the specific requirements of the spacecraft and the mission objectives.

2.6 Electrical Power System

In the last few decades, the development of small form factor satellites has revolutionized the space industry. These small satellites have opened up new avenues for scientific research, commercial applications, and communication networks. However, as small satellites have limited space and power budgets, their electrical power system (EPS) is a critical component that must be designed with careful consideration.

The primary function of the EPS is to generate, store, and distribute electrical power to the spacecraft's various subsystems, instruments, and payloads. In this chapter, we will discuss the main components of the EPS and the different technologies that can be employed for power generation and storage in small satellites.

2.6.1 Electrical Power Generation

Electrical power generation in small satellites typically relies on solar cells or batteries. Solar cells are the most common means of generating power in small satellites. They convert sunlight into electrical power through the photovoltaic effect. The amount of power generated by solar cells is dependent on the surface area of the solar panels, the efficiency of the cells, and the angle at which the panels are oriented towards the sun.

Batteries, on the other hand, are used to store electrical energy generated by the solar cells. They are also used to provide power to the spacecraft when it is in eclipse or when the solar panels are not facing the sun. Batteries used in small satellites are typically rechargeable and can be made of various chemistries, including lithium-ion, nickel-cadmium, and nickel-hydrogen.

In recent years, researchers have been exploring new ways of generating electrical power in small satellites. One such technology is the use of radioisotope thermoelectric generators (RTGs). RTGs are powered by the heat generated by the radioactive decay of isotopes such as plutonium-238. They can provide a constant source of power for decades, making them ideal for deep space missions or missions that require continuous power generation.

Another technology being explored for power generation in small satellites is the use of fuel cells. Fuel cells generate electrical power by reacting hydrogen and oxygen to produce water and electricity. This technology has the potential to provide higher power densities than traditional battery systems, making it ideal for high-power applications.

2.6.2 Electrical Power Storage

As mentioned earlier, batteries are commonly used to store electrical power in small satellites. Lithium-ion batteries are the most common type of battery used in small satellites. They have a high energy density and can be made in small form factors, making them ideal for use in small satellites. Nickel-cadmium and nickel-hydrogen batteries are also commonly used in small satellites, although they have lower energy densities than lithium-ion batteries.

Another technology being explored for electrical power storage in small satellites is the use of supercapacitors. Supercapacitors can store electrical energy and release it quickly, making them ideal for high-power applications. They have a lower energy density than batteries but can be recharged quickly, making them ideal for applications that require rapid charging and discharging.

In recent years, researchers have been exploring the use of flywheel energy storage systems for small satellites. Flywheels store kinetic energy in a rotating mass and can release the energy quickly when needed. They have a higher energy density than batteries and can provide high-power output, making them ideal for high-power applications.

2.6.3 Power Management and Distribution

Power management and distribution is a critical component of the EPS in small satellites. The EPS must be designed to manage the power generated by the solar cells or other power generation systems and distribute it to the various subsystems and instruments in the spacecraft.

Power management and distribution in small satellites typically involve the use of power conditioning units (PCUs) and power distribution units (PDUs). PCUs are used to regulate and condition the electrical power generated by the solar cells or other power generation systems to ensure that it meets the requirements of the spacecraft's subsystems and instruments. PDUs are used to distribute the electrical power to the various subsystems and instruments in the spacecraft.

PCUs typically include voltage regulators, power converters, and other conditioning components that are used to regulate the voltage and current of the electrical power generated by the solar cells or other power generation systems. PDUs, on the other hand, typically include switches, fuses, and other components that are used to distribute the electrical power to the various subsystems and instruments in the spacecraft.

Power management and distribution in small satellites must also take into consideration the power budget of the spacecraft. The power budget is the total amount of electrical power that the spacecraft can consume over a given period. The power budget is typically determined by the available power generation and storage systems, as well as the power requirements of the spacecraft's subsystems and instruments.

To optimize the power budget, power management and distribution in small satellites must be designed to minimize power losses and maximize the efficiency of the power generation and storage systems. This can be achieved through the use of efficient power conditioning and distribution components, as well as the implementation of power-saving strategies such as power cycling and power prioritization.

2.6.4 Miniaturization of EPS Technologies

The miniaturization of EPS technologies is a critical component of the development of small satellites. As small satellites have limited space and power budgets, the EPS must be designed to be as compact and efficient as possible.

The miniaturization of solar cells has been a major focus of research in recent years. Researchers have been exploring new materials and manufacturing techniques that can be used to produce solar cells that are smaller, lighter, and more efficient. One promising technology being explored is the use of thin-film solar cells. Thin-film solar cells are made of flexible, lightweight materials and can be produced in a variety of shapes and sizes.

The miniaturization of batteries is also an important area of research. Lithium-ion batteries are currently the most commonly used battery technology in small satellites, but researchers are exploring new chemistries and manufacturing techniques that can be used to produce batteries that are smaller, lighter, and more efficient. One promising technology being explored is the use of solid-state batteries. Solid-state batteries have the potential to provide higher energy densities than traditional batteries, making them ideal for use in small satellites.

The miniaturization of power management and distribution components is also an important area of research. Researchers have been exploring new designs and manufacturing techniques that can be used to produce power conditioning and distribution components that are smaller, lighter, and more efficient. One promising technology being explored is the use of microelectromechanical systems (MEMS) for power management and distribution. MEMS are tiny devices that can be used to control and manipulate electrical signals with high precision, making them ideal for use in small satellites.

2.6.5 Summmary

In conclusion, the EPS is a critical component of small satellites that must be designed with careful consideration. The EPS must be designed to generate, store, and distribute electrical power to the spacecraft's various subsystems, instruments, and payloads. Solar cells and batteries are the most common means of generating and storing electrical power in small satellites, although researchers are exploring new technologies such as RTGs, fuel cells, supercapacitors, and flywheel energy storage systems.

Power management and distribution is a critical component of the EPS in small satellites. The EPS must be designed to manage the power generated by the solar cells or other power generation systems and distribute it to the various subsystems and instruments in the spacecraft. Power management and distribution in small satellites must also take into consideration the power budget of the spacecraft.

The miniaturization of EPS technologies is a critical component of the development of small satellites. Researchers are exploring new materials, designs, and manufacturing techniques to produce smaller, lighter, and more efficient solar cells, batteries, and power management and distribution components. MEMS and other microtechnologies show great promise in creating highly efficient and precise power management and distribution systems.

The continued development and improvement of EPS technologies will play a crucial role in the success and advancement of small satellites. As the demand for small satellites grows, EPS technologies must be designed to meet the unique requirements and challenges of small spacecraft, including limited space and power budgets.

In addition, the EPS must also be designed to withstand the harsh space environment and operate reliably over long periods of time. The reliability and durability of EPS components are critical to the success of small satellite missions and must be thoroughly tested and validated before deployment.

Overall, the EPS is a fundamental aspect of small satellite design and must be given careful consideration during the design and development process. The continued improvement and advancement of EPS technologies will enable the development of increasingly capable and sophisticated small satellites, opening up new opportunities for scientific exploration, commercial applications, and space-based services.

2.7 Thermal Control Subsystem

One of the critical subsystems of a satellite is the thermal control subsystem. The thermal control subsystem is responsible for regulating the temperature of the satellite's components and ensuring that they operate within their temperature limits. Miniaturization of components of the thermal control subsystem is essential for the development of small satellites. Small satellites, such as PocketQube Satellites, have restricted volumes and limited mass budgets and as described before the power density of the electrical components is going to become higher and higher. Therefore, the thermal control subsystem's components must be miniaturized to fit within the satellite's limited space and mass constraints. This section will discuss the technologies that can be used for the miniaturization of components of the thermal control subsystem of a satellite and here are listed some of the more promising technologies that could play an important role in the miniaturization of this specific subsystem that are considered as SoA.

2.7.1 Passive Systems

Passive thermal control subsystems are a critical component of small spacecraft design that helps to regulate the temperature inside the spacecraft. These systems rely on the inherent thermal properties of materials to dissipate or retain heat to maintain an optimal temperature range.

Passive thermal control subsystems can take many forms, but the most common approach is to use specially designed materials, coatings, and insulation to regulate the temperature. These materials are chosen for their ability to absorb, reflect, or emit thermal energy, and they are strategically placed on the spacecraft to achieve the desired thermal balance. For example, highly reflective materials can be used to reflect solar radiation and prevent it from heating up the spacecraft, while insulating materials can be used to prevent heat from escaping the spacecraft at night or in cold environments. Passive thermal control systems are advantageous in small spacecrafts as they require minimal power, are lightweight and can operate for extended periods without intervention. They are also less complex and less expensive than active thermal control systems that rely on heaters or radiators to regulate temperature. Additionally, materials with high thermal conductivity can be used to transfer heat away from sensitive components, such as electronics or batteries. Also In addition to passive thermal control technology, structural and electrical design methods also contribute to managing the thermal environment, passively. These design methods include:

- Materials selection;
- Mission Design;

- Spacecraft Design;
- Electronics Design.

Hereafter are listed some of the main technologies commonly used in small form factor spacecrafts that could also be miniaturized to achieve their purposes.

- Multi Layer Insulation (MLI).
  Multi-layer insulation (MLI) is a common thermal control solution used in small satellites to maintain stable temperatures and protect sensitive equipment from extreme temperature variations. MLI consists of multiple layers of low-conductivity materials, such as Mylar or Kapton, interleaved with layers of high-conductivity materials, such as aluminum foil or metalized plastic films. The state of the art of MLI in small satellites has advanced significantly in recent years, driven by the increasing demand for high-performance thermal control solutions in the rapidly growing small satellite market.Researchers are exploring new materials for MLI, such as aerogels and nanostructured materials, which have lower thermal conductivity and higher thermal stability than traditional MLI materials. These materials can provide better thermal insulation and help to reduce the overall mass of the satellite.Also, Advancements in automated manufacturing technologies, such as robotic cutting and folding machines, have enabled the efficient and accurate production of MLI blankets with complex shapes and geometries. This reduces the time and cost of manufacturing MLI and allows for more customized solutions for individual small satellite missions. Some researchers are exploring novel MLI designs that incorporate shape memory alloys or phase-change materials, which can adjust their thermal properties in response to changes in temperature. These designs can provide more precise thermal control and reduce the need for additional thermal control systems.In summary, the state of the art of MLI in small satellites is continuously evolving, with new materials, manufacturing techniques, and design approaches being developed to improve the performance and efficiency of thermal control systems. These advancements will continue to play a crucial role in the development of small satellite technology, enabling new missions and applications in space.

- Thermal Straps
  Thermal straps are another common thermal control solution used in small satellites. A thermal strap is a flexible, high-conductivity material that is used to transfer heat between two points in a spacecraft. Thermal straps can be made from a variety of materials, including Copper,due to its high thermal conductivity and low mass, Aluminum or even composite materials like Carbon Fiber Reinforced Plastic which could offer a good balance of thermal conductivity, strength, and lightweight. The choice of material for a

Figure 2.23: SoA Multi Layer Insulation (Aerospacefab)

Figure 2.24: Close up of Multi Layer Insulation

thermal strap depends on the specific requirements of the mission, such as the desired thermal conductivity, mass, and flexibility.

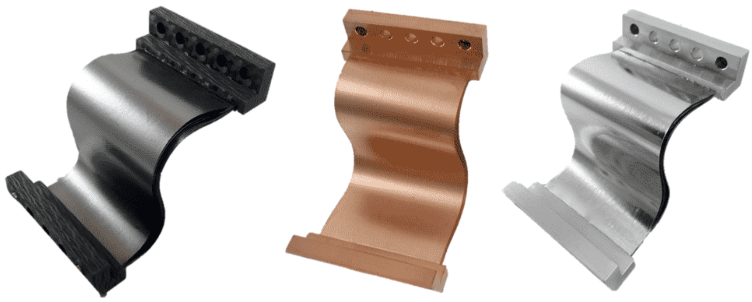


Figure 2.25: Examples of different materials Thermal Straps (Thermal Space)

- Thermal Interface Materials. Thermal interface materials (TIMs) are another important component of thermal control systems in small satellites. They are used to enhance the thermal conductivity and transfer of heat between two surfaces by filling any gaps or voids between them. TIMs are typically used to improve the thermal performance of electronic components, such as power amplifiers, memory modules, and processors, by improving the heat dissipation from these components. There are several types of solution used as TIMs, including highly viscous Thermal Greases, Thermal Pads, Phase-Change Materials, which provide high thermal conductivity and adaptability to different thermal environments, and Thermal Adhesives that create a stronger contact then Thermal Pads and are usually used in applications where a permanent bond is required. The choice of TIM depends on the specific application and requirements of the small satellite. Factors such as thermal conductivity, thermal resistance, durability, and ease of application must be considered when selecting the appropriate TIM for a particular application.

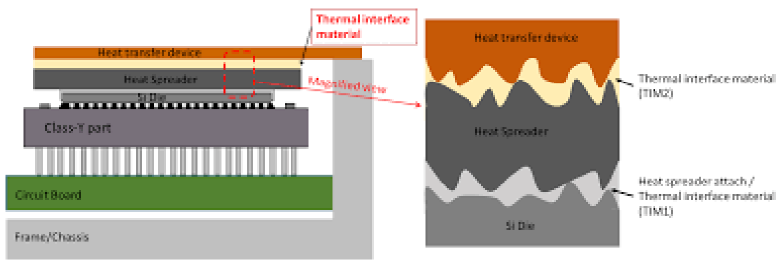


Figure 2.26: Function of Thermal Interface Materials

- Deployable Radiators.
  During the research of thermal control technologies one of the most promising one was for sure the deployable radiators. Deployable radiators are a type of thermal control solution that has been implemented in small satellites to

regulate the temperature of sensitive electronic components. They consist of a set of radiator panels that are folded or stowed during launch and then deployed in space to provide additional surface area for thermal radiation.They can of course be used to dissipate excess heat generated by most sensitive components or to provide additional cooling capacity in environments with high thermal loads. Deployable radiators are typically made of lightweight, high-conductivity materials such as aluminum or graphite, which can efficiently transfer heat away from electronic components. The radiator panels are designed to be highly emissive, allowing them to radiate heat into space and regulate the temperature of the spacecraft. Some deployable radiator designs also incorporate movable louvers or shades that can be used to adjust the radiative properties of the panels and improve the thermal control performance of the system. Deployable radiators offer several advantages over other thermal control solutions. First, they provide a large surface area for thermal radiation, which can effectively dissipate heat generated by electronic components. Second, they can be stowed during launch, reducing the overall mass and volume of the spacecraft. Third, they can be deployed on demand, providing additional cooling capacity when needed.

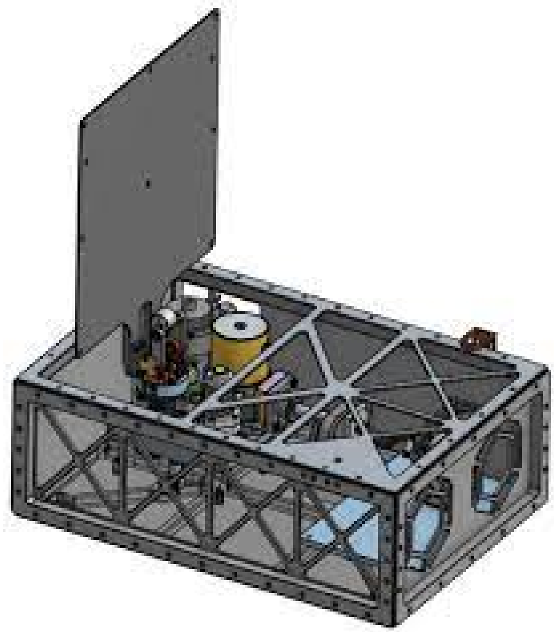


Figure 2.27: Example of Deployable Radiator Application in CubeSats (NASA)

- Heat Pipes.
  A traditional heat pipe is a passive device comprised of a metal container (pipe) that holds a liquid under pressure and has a porous wick-like structure within the container. When heat is applied to one end of the tube, the liquid inside the tube near the hot end vaporizes into a gas that moves through the tube to the cooler end, where it condenses back into a liquid. The wick transports the condensed liquid back to the hot end via capillary action. Heat pipes are an efficient passive thermal transfer technology, where a closed-loop system transports excess heat via temperature gradients, typically from electrical devices to a colder surface, which is often either a radiator itself, or a heat sink that is thermally coupled to a radiator. One of the most advanced example is reported in Fig. 2.28 showing 'FlexCool',which is a bent, flat heat pipe developed as a cross between a heatpipe and a thermal strap that can be customized for higher heat fluxes by increasing the thickness. It has ten times the thermal conductivity of copper, while being even lighter and was used in a 6U CubeSat deployed from the ISS in 2020.

Figure 2.28: FlexCool Micro HeatPipe in a 1U frame

2.7.2 Active Systems

Active thermal control systems are essential for small spacecraft to regulate temperature and prevent overheating or freezing of critical components. The state of the art in active thermal control systems for small spacecraft is evolving rapidly, driven by advancements in technology and the increasing demand for small, low-cost spacecraft.

One popular approach to active thermal control in small spacecraft is using passive radiators combined with heaters and temperature sensors to maintain a stable temperature range. However, this approach has limitations in its ability to handle large temperature variations and requires significant power consumption. Alternatively, some spacecraft use active thermal control systems that incorporate thermal loops, which consist of a heat source, a heat sink, and a fluid that circulates between them. This method is effective for larger spacecraft but can be more complex and difficult to implement in small spacecraft due to size and weight constraints. Another approach is to use thermoelectric coolers, which are small, lightweight devices that use electrical power to generate a temperature difference between two surfaces. These can be integrated into the thermal control system to provide localized cooling or heating as needed. Recent advancements in materials and manufacturing techniques have also led to the development of new types of materials and coatings that can help improve the efficiency of thermal control systems in small spacecraft. For example, thin-film coatings with high thermal emissivity can help radiate heat more effectively, while multi-layer insulation materials can provide enhanced thermal resistance. Again here are listed some of the most promising technologies that can be implemented in a small form factor satellite:

- Heaters.
  Heaters are devices that are used to generate heat and maintain a specific temperature in small spacecrafts. Small spacecrafts, such as CubeSats, are particularly susceptible to temperature variations due to their compact size and limited thermal control systems. This can cause damage to sensitive electronics or affect the performance of propulsion systems.

  Heaters can be used to regulate the temperature of critical components such as batteries, antennas, and sensors. They work by converting electrical energy into heat energy, which is then transferred to the surrounding environment. Heaters can be controlled by a thermal control system that monitors the temperature and adjusts the heat output as needed to maintain the desired temperature range. In addition to regulating temperature, heaters can also be used to prevent the formation of ice or frost on critical components. This is particularly important for spacecraft that operate in cold environments, such as those in orbit around planets or moons. Overall, heaters play a critical role in ensuring the safe and reliable operation of small spacecrafts by regulating temperature and preventing damage due to extreme temperature variations.

Figure 2.29: Flexible Micro Heater for Space Applications (MINCO)

- Cryogenic Cooling.
  Cryocoolers are another type of thermal management system that can be used in small spacecrafts. Unlike heaters, which generate heat, cryocoolers are used to actively remove heat and maintain a specific low temperature range. Cryocoolers work by using a refrigeration cycle to transfer heat from the spacecraft to the environment, similar to how a refrigerator works. They use a combination of compressors, expanders, and heat exchangers to cool down and circulate a refrigerant, which absorbs heat from the spacecraft and transfers it to the external environment. They are commonly used in small spacecrafts for cooling infrared detectors or other scientific instruments that require very low temperatures to operate effectively. They can also be used to cool down other sensitive electronics that generate a lot of heat and require active cooling to prevent overheating. One of the benefits of using cryocoolers is that they can maintain a more precise temperature range than heaters, which may have wider temperature variations. However, they can also be more complex and expensive to design and operate than heaters. Overall, cryocoolers play important roles in the thermal management of small spacecrafts being particularly useful for cooling sensitive scientific instruments or electronics that require very low temperatures.

Figure 2.30: Micro Cryocooler (Lockheed Martin Space)

- Active Thermal Architecture
  One of the most advanced cooling solution found during this survey was for sure the Active Thermal Architecture (ATA). The Active Thermal Architecture (ATA) [23] system is an advanced, active thermal control technology for small

satellites in support of advanced missions in deep space, helio-physics, earth science, and communications. The ATA technology is capable of high-power thermal rejection, and zonal temperature control of satellite busses, payloads, and high-energy density components and the project was developed by the Center for Space Engineering at Utah State University (CSE, USU) and funded by the NASA SST program in partnership with JPL. The most promising

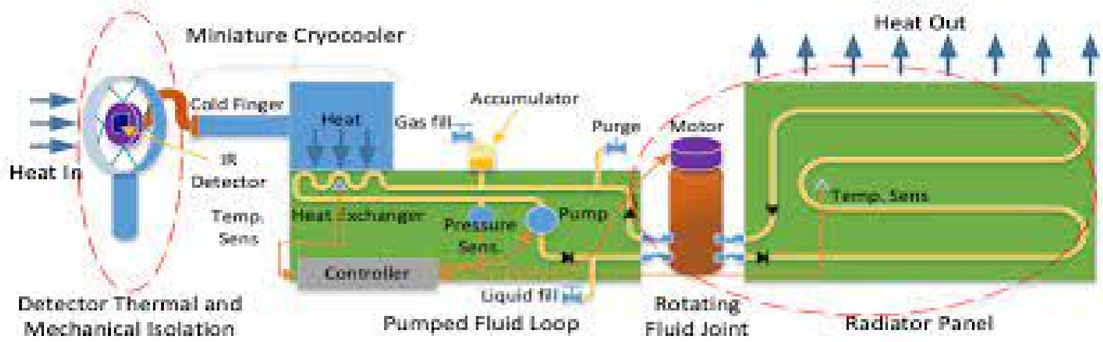


Figure 2.31: ATA Principles of Function (NASA)

fact about it is that Ultrasonic additive manufacturing (UAM) techniques were used to simplify and miniaturize the ATA system by embedding the fluid channels directly on the satellite chassis, and the external radiator, creating integrated multi-function structures.

2.8 Structure and Materials

2.8.1 Primary Structure

he primary structure of a small spacecraft is the backbone that supports the entire mission. It is responsible for carrying the payload and all other components, including the propulsion system, power supply, and communication equipment, as well as providing structural integrity and stability. The primary structure of a small spacecraft is typically composed of lightweight materials such as composites, aluminum, or titanium. These materials are chosen for their strength-to-weight ratio, which is critical for a spacecraft's performance and launch cost. The design of the primary structure also takes into consideration the forces experienced during launch, deployment, and operation in space, including thermal and mechanical stresses, vibration, and shock.In addition to the choice of materials, the design of the primary structure must also take into account the mission requirements and constraints. Small spacecraft are often designed with a modular approach, where individual components are assembled in orbit. This requires the primary structure to be designed with flexibility in mind, allowing for the integration of different modules and subsystems.Furthermore, the primary structure must also be designed with long-term reliability in mind. Spacecraft can remain in orbit for many years, and the primary structure must withstand the harsh conditions of space, including radiation and micrometeoroid impacts. Therefore, the materials and construction methods used for the primary structure must be able to withstand these environmental factors over extended periods. Overall the primary structure of a small spacecraft is a critical component that provides the necessary support, stability, and reliability for the entire mission. It must be designed with a careful consideration of materials, forces, and mission requirements to ensure the success of the mission. In the current State of the Art they can be divided into two main categories, Monocoque Construction and Modular Frame Designs. Monocoque construction is a technique where the skin of the spacecraft provides the primary structural support. This approach is similar to an eggshell, where the thin outer layer supports the entire structure. Monocoque construction offers several benefits, including reduced weight and increased strength. The skin of the spacecraft is designed to distribute loads evenly, resulting in a structure that is strong, stiff, and lightweight. Monocoque construction is also easier to manufacture, as the skin can be formed from a single piece or a few large panels, reducing the number of joints and fasteners required. Modular frame designs, on the other hand, are constructed by assembling individual modules that make up the spacecraft's primary structure. These modules are usually designed to be interchangeable, allowing for greater flexibility in spacecraft design and the ability to swap out or replace individual components. This approach provides several benefits, including the ability to launch larger spacecraft in smaller launch vehicles, as the modules can be packed more efficiently. Modular frame designs also allow for easier maintenance and repair, as individual modules can be replaced if needed. Here are listed more in detailed what can be considered the main advantages and disadvatages the Monocoque Construction:

- Lighter weight.
  Monocoque construction typically results in a lighter spacecraft structure, as there is less material used in building the overall structure.
- Increased stiffness.
  The single-piece structure of a monocoque design results in increased stiffness, which can be beneficial in reducing vibration and providing more stable flight characteristics.
- Better aerodynamics.
  Monocoque structures typically have smoother external surfaces, resulting in better aerodynamic performance.
- Limited access to internal components.
  Due to the nature of monocoque construction, accessing internal components can be challenging, which can make maintenance and repairs more difficult.
- Higher manufacturing costs.
  Building a monocoque structure can be more complex depending on the objective and require more specialized equipment, which can result in higher manufacturing costs.

On the other hand a similar comparison can be done for Modular Frame Designs:

- Flexibility.
  The modular design approach provides more flexibility in terms of adding or removing modules, making upgrades and modifications easier.
- Easier maintenance.
  Modular designs typically provide better access to internal components, which can make maintenance and repairs easier.
- Lower manufacturing costs.
  Building a modular design can be less complex and require less specialized equipment, which can result in lower manufacturing costs.
- Heavier weight.
  Modular designs typically require more material to be used in building the overall structure, resulting in a heavier spacecraft.

- Reduced stiffness.
  The modular design approach can result in a less stiff structure, which can lead to more vibration and less stable flight characteristics.

In summary, both monocoque construction and modular frame designs have their own advantages and disadvantages when it comes to small spacecraft structures. Choosing the best approach will depend on a variety of factors, including the mission requirements, budget, and available manufacturing resources.

2.8.2 Additive Manufacturing

Additive Manufacturing (AM) technology is revolutionizing manufacturing processes across various industries, and the space industry is no exception. AM technology, also known as 3D printing [24], offers unique advantages such as the ability to produce complex geometries, customization, and reduced material waste. In miniaturized space applications, AM technology can offer significant benefits such as reduced manufacturing costs, rapid prototyping, and the ability to produce parts with high precision and quality. This paper will focus on the use of AM technology for miniaturized space applications, with a focus on the Technology Readiness Level (TRL) requirements and a comparison of different types of materials such as photopolymers, PLA, and ABS. As cited beefore the Technology Readiness Level (TRL) is a measure used by the space industry to assess the maturity of a technology. The TRL ranges from 1 to 9, with TRL 1 indicating the lowest level of technology maturity and TRL 9 indicating the highest level of maturity, where the technology has been proven to work in an operational environment. The TRL requirements for miniaturized space applications are typically lower than for larger space applications, as the risks and costs associated with failure are generally lower.

The TRL requirements for AM technology in miniaturized space applications are typically in the range of TRL 3 to TRL 6. At TRL 3, the technology is proven to work in a laboratory environment, and at TRL 6, the technology has been demonstrated in a relevant environment. The key factors that influence the TRL requirements for AM technology in miniaturized space applications include the size and complexity of the parts being produced, the materials being used, and the performance requirements of the parts.

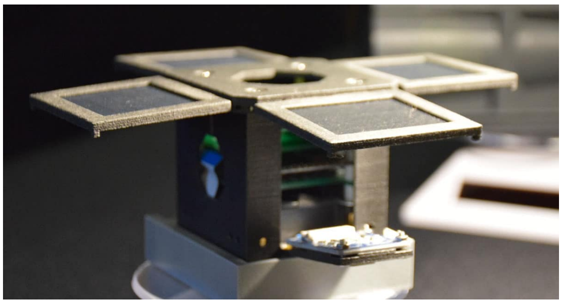


Figure 2.32: 3D Printed PQ Satellite (Mini-Cubes)

The main materials used in this field are listed below with some of their main characteristics:

- Photopolymers.
  Its a lightweight material with high resolution printing capabilities for intricate designs and fast curing time for quick production. It also can be used to produce small-scale parts with high accuracy and its characterized by low shrinkage for accurate and consistent parts.

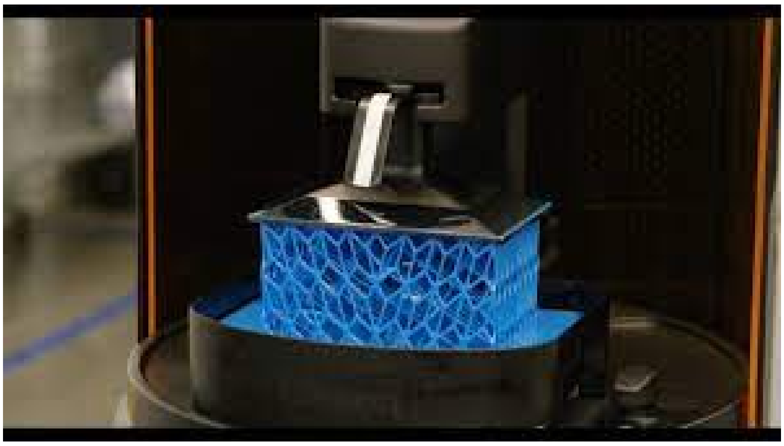


Figure 2.33: Photopolymer 3D Printing

- PLA. Made from renewable resources, makes it an eco-friendly option and being degradable could be a good choice for space debris mitigation. Also it is a lightweight material which reduces the overall weight of spacecraft. Although one of its main characteristics is its low thermal expansion coefficient, making it suitable for use in temperature-sensitive environments.

Figure 2.34: PLA 3D Printing

- ABS.
  High strength and durability, making it suitable for parts that require mechanical strength Good heat resistance, making it suitable for use in high-temperature environments Good chemical resistance, making it suitable for use in harsh chemical environments Suitable for creating complex geometries, making it useful for creating intricate parts.

Figure 2.35: ABS in-space 3D Printing (ESA)

Chapter 3

Reaction-Wheel Miniaturization Limit

In this chapter will be evaluated, based on some relevant statistical data, the miniaturization of one of the main components of the ADCS, the reaction wheels. For this analysis will be considered two main factors as stated in [25]. First, the dominant disturbance torques on small spacecraft in low earth orbit, aerodynamic drag and solar radiation pressure, both become relatively larger as spacecraft size decreases. Second, the effectiveness of spinning rotors reduces as the rotor inertia decreases with the square or the wheel radius. In this study will be introduced more data relatively to more advanced and miniaturized technologies reported in the previous chapter that could provide more data to analyze the limit. Here is reported the initial curve determined in 3.1 for some relevant typologies of satellite and for different scale of length. It is put into ratio the characteristic length of each satellite and the mass ration between the mass of the reaction wheels and the entirety of the satellite. the limit follows the law:

$$\mu_A DCS = a(\lambda)^b. \tag{3.1}$$

The coefficient 'a' and 'b' are calculated interpolating the data reported and the result obtained are:

$$a = 0.06500; \tag{3.2}$$

$$b = 0.6277. \tag{3.3}$$

As shown in the graph is also reported in orange the zero payload line which simply states the physical limit in which all the mass of the satellite would be made all by the mass occupied by the reaction wheels.

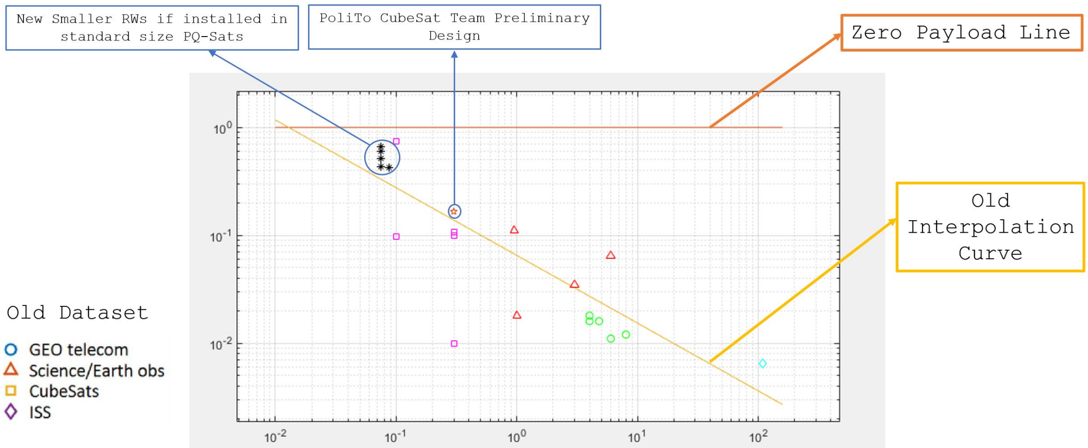


Figure 3.1: Old interpolation Line of reaction Wheels Miniaturization Limit

3.1 Statistical Data

In the previous graph were also presented some data that did not contributed to the interpolation and are those reported with a black star and are the data added using some of the newest technologies for the reaction wheels to this day. Those data take into account the installation of some reaction wheels in some reference size PocketQube satellites, which standard will be discussed later in this thesis. For the generic 1P standard a total mass of 250g is taken into consideration and the characteristic length is calculated based of the total volume of the satellite, and not taking only the maximum length of the satellite. The data reported are the one stated in the following table:

RW Name	Dim [mm]	Mass [g]
Nano-Avionics RWO	[43.5,43.5,24]	137
CubeSpace CUBEWHEEL S+	[33.4,33.4,29.7]	90
CubeSpace CW0057	[35,35,24]	115
CubeSpace CW0017	[28,28,26]	60
Sinclaire Interplanetary RW-0.01	[50,50,30]	120

Table 3.1: State of the Art Reaction Wheels

Given the data reported, after some consideration on the number of reaction wheels present in a model and the size of the model (1P, 2P or 3P) it can be easly calculated their specific length and the mass fraction and can finally be inserted with as shown in the previous chart. In the following MatLab Code reported in Fig. 3.2 are explained the assumptions made:

```
% PQ Data

lamda_1P= 0.05;
m_1P= 0.225; % average mass for a 1P unit
mu_CW0057= 2*0.115/(2*m_1P);
lamda_CW0057= 1.5*lamda_1P;
mu_CW0017= 3*0.065/(2*m_1P);
lamda_CW0017= 1.5*lamda_1P;
mu_CWS_plus= 3*0.09/(2*m_1P);
lamda_CWS_plus= 1.5*lamda_1P;
mu_CW_M= 2*0.150/(2*m_1P);
lamda_CW_M= 1.5*lamda_1P;
mu_RW_001= 2*0.120/(2.5*m_1P);
lamda_RW_001= 1.75*lamda_1P;

mu_PQ=[ mu_CW0057 mu_CW0017 mu_CWS_plus mu_CW_M mu_RW_001];
lamda_PQ= [lamda_CW0057 lamda_CW0017 lamda_CWS_plus lamda_CW_M lamda_RW_001];
```

Figure 3.2: PQ Sat Assumptions

3.2 Data Interpolation

Given those data it can be made a new interpolation following the same law as before but with more empirical data that should provide a different result shown in Fig. 3.3: As shown the new coefficients of the curve are:

$$a = 0.0298; \tag{3.4}$$

$$b = -1.136. \tag{3.5}$$

The results can be much better evaluated by comparing the old and new curve in the following logaritmic graph in Fig. 3.4: As shown in red is reported the old interpolation line while in blue is reported the new one. As it is shown is also clear that the intersection of the curve with the Zero Payload Line happens at much higher values than before going from 12.8 mm to a huge value of 44.4 mm, at the limit of the PQ Standard. This would suggest a reconsideration of the limits of the miniaturization, however this is only a statistical limit based on a very semplistic model and more considerations have to be done following those results.

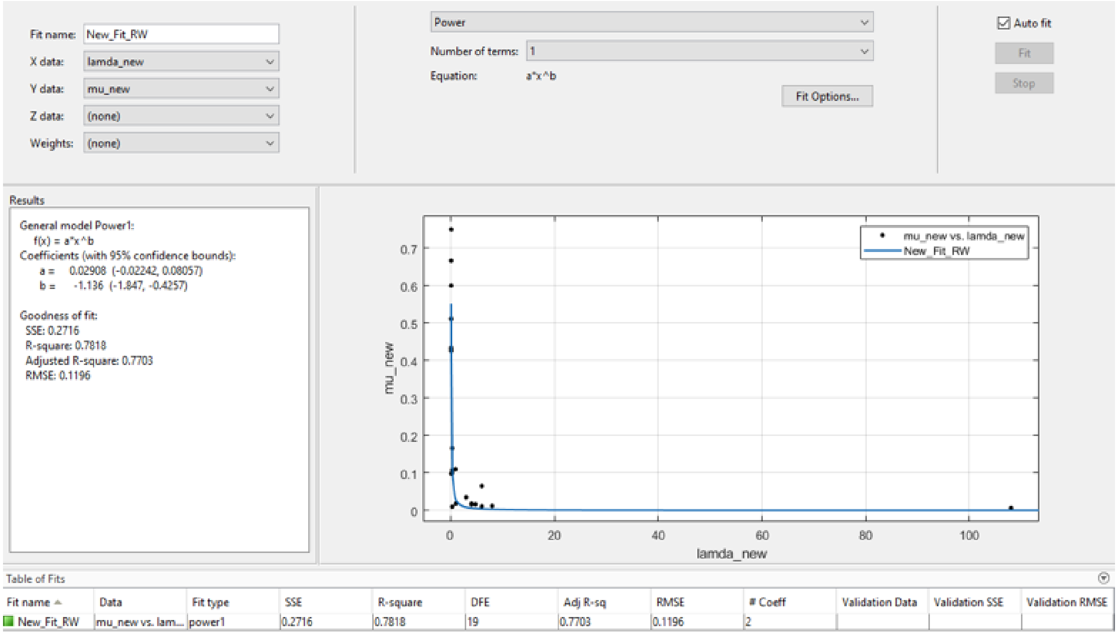


Figure 3.3: New Interpolation of Data

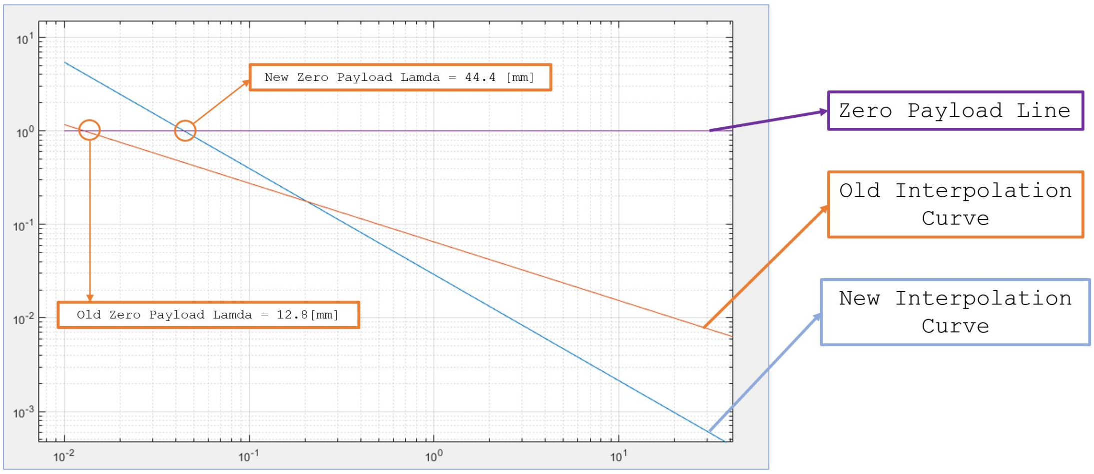


Figure 3.4: Old and New Interpolation Curve Comparison

3.3 Conclusions and Consideration

The new limit is set much higher than the previous one. This could be to the fact that there are many more points above the old curve and could be more effective to properly weight those interpolation points. Also, the configurations added are just a very simple concept of RW configuration and are not validated satellites as the ones reported in the original article, thus, a more deep research could lead to more accurate configurations of PQ-Sat and provide a much accurate curve. As highlighted in the 3° slide the Residual calculated during the iterative interpolation is around 0.8 which is not that bad, but not even that good. To improve this aspect there should be taken into consideration satellites with similar purposes since the size of the AOC System is very sensitive to that reducing the variance of the data taken into consideration. This would allow to have less variancies in the DataSet and provide a much accurate law.

Chapter 4

Design and Integration of Attitude Control System for a PocketQube Satellite based on Reaction Wheels

In this Chapter will be discussed all the proceedings that lead to the complete design and integration of an Attitude Control System Satellite based on Reaction Wheels in a standard PocketQube Satellite following general design rules provided again by high level articles and documentation [26] [27].

4.1 PoketQube Standard

The PocketQube standard is a miniaturized satellite standard developed by Glasgow-based company, Alba Orbital. PocketQubes are very small, with a standardized size of just 5cm cubed and a weight of up to 250 grams. They are much smaller and lighter than the Cubesat standard, which has a size of 10cm x 10cm x 10cm. The PocketQube standard was created to allow small companies, universities and individuals to build and launch their own satellites at a relatively low cost. The small size and low weight of PocketQubes means that they can be launched as secondary payloads on existing rocket launches, thereby reducing the cost of launching them.

In this article, we will provide a detailed introduction to the PocketQube standard. We will begin by providing a brief overview of the history of the PocketQube standard and how it came to be. We will then go on to discuss the technical specifications of the standard, including the size, weight and power requirements of a PocketQube. We will also discuss the various subsystems that are typically included in a PocketQube, such as the communication, power and propulsion systems. In addition, we will discuss the different types of missions that PocketQubes can be used for and provide examples of some of the missions that have been launched so far. Finally, we will discuss the future of the PocketQube standard and the potential impact it could have on the space industry.

4.1.1 History of the PocketQube Standard

The PocketQube concept appeared in 2009, from an idea of professor Robert J. Twiggs, as a result of a collaboration between Morehead State University (MSU) and Kentucky Space which developed some specifications with respect to PocketQubes The first PocketQube was launched using MRFOD (Morehead Rome Fempto Orbital Deployer) installed inside UniSat-5 microsatellite as a result of a cooperation between Morehead State University GAUSS Srl and Kentucky Space.

4.1.2 Technical Specifications of the PocketQube Standard

The PocketQube standard has a standardized size of 5cm x 5cm x 5cm, and a weight of up to 250 grams. This makes it much smaller and lighter than the Cubesat standard, which has a size of 10cm x 10cm x 10cm and a weight of up to 1.33 kilograms. The small size and low weight of the PocketQube standard means that it can be launched as a secondary payload on existing rocket launches, which significantly reduces the cost of launching a PocketQube.

The PocketQube standard also has specific power requirements. The standard requires that a PocketQube must be able to operate on a maximum power budget of 2 watts. This means that the communication, power and propulsion systems of a PocketQube must be designed to operate within this power budget.

PocketQubes typically include a number of subsystems that allow them to function in space. Regarding communication systems, PocketQubes typically use low-power radio transmitters and receivers to communicate with ground stations and other satellites. They are typically designed to be compatible with standard amateur radio frequencies, making it easier for hobbyists and students to communicate with and operate their PocketQube. However, some PocketQubes may use commercial off-the-shelf communication systems, especially if they require higher data rates or greater reliability.

Power systems for PocketQubes typically consist of solar panels and a battery. The solar panels are used to generate power for the satellite, while the battery is used to store excess power and provide power when the satellite is not in sunlight. Due to the limited surface area available on a PocketQube, the solar panels must be highly efficient and may use advanced materials such as multi-junction cells or thin-film solar cells.

In addition to these subsystems, PocketQubes may also include a range of sensors and scientific instruments, depending on the mission requirements. For example, a PocketQube designed for scientific research might include a set of cameras or sensors to collect data on the Earth's atmosphere or magnetic field. A technology demonstration mission might include a payload to test a new material or technology in the space environment. As mentioned earlier, the simplicity and flexibility of the PocketQube standard allows for a wide range of mission types to be carried out. The small size and low cost of these satellites make them an attractive option for universities, research institutions, and even individual hobbyists who want to experiment with space technology. However, the small size of PocketQubes also presents some challenges in terms of subsystem design and mission requirements. Due to the limited space and weight available, subsystems must be highly compact and efficient, and missions must be carefully planned to maximize the scientific or technological return on investment.

PocketQubes can be used for a wide range of space missions, including technology demonstrations, Earth observation, and scientific research. Due to their small size and low weight, PocketQubes are often used to carry out missions that are not possible with larger satellites. Some of the types of missions that PocketQubes can be used for include:

Technology Demonstrations: PocketQubes are ideal for testing new technologies and subsystems in space. They can be used to demonstrate new communication, power, and propulsion systems, as well as new sensors and instruments.

Earth Observation: PocketQubes can be used to take images of the Earth, which can be used for a variety of applications, such as monitoring crop growth, tracking weather patterns, and detecting natural disasters.

Scientific Research: PocketQubes can be used to carry out scientific research in space. They can be used to study phenomena such as the Earth's magnetic field, radiation in space, and the behavior of microorganisms in microgravity.

Educational Outreach: PocketQubes are an excellent tool for educational outreach. They can be used to engage students in space-related activities and provide hands-on experience with satellite technology.

4.1.3 PocketQube Development

As the PocketQube standard has gained popularity in recent years, there has been a growing interest in developing new technologies that are suitable for use in these small, low-cost satellites. Many of these technologies are focused on making the subsystems required for PocketQubes more compact, efficient, and reliable, in order to maximize the capabilities and mission lifetimes of these satellites. In this section, we will discuss some of the key technological developments that are relevant to

PocketQube satellites.

One important area of development for PocketQube technology is in communication systems. As mentioned earlier, PocketQubes typically use low-power radio transmitters and receivers to communicate with ground stations and other satellites. To maximize the range and reliability of these systems, researchers and engineers have been developing new antenna designs that are more compact and efficient than traditional designs. For example, some research has focused on using metamaterials to create antennas that are smaller and more efficient than traditional antennas, while others have explored the use of phased array antennas that can steer the satellite's beam towards a specific target on the ground. Another important area of development is in power systems. As mentioned earlier, PocketQubes typically use solar panels and a battery to generate and store power. To maximize the efficiency of these systems, researchers and engineers have been developing new solar panel designs that are more lightweight and efficient than traditional designs. For example, some research has focused on using multi-junction solar cells that are more efficient at converting sunlight into electricity, while others have explored the use of thin-film solar cells that can be more easily integrated into the surface of the satellite.

To improve the reliability and lifetime of PocketQube satellites, researchers and engineers have also been developing new attitude control systems. The attitude control system is responsible for maintaining the satellite's orientation in space, and typically consists of a set of reaction wheels and/or magnetorquers. To improve the accuracy and responsiveness of these systems, researchers have been exploring the use of new sensors and control algorithms that can better account for the complex interactions between the satellite and its environment.

Another important area of development is in payload technology. Payloads are the scientific or technological instruments that are mounted on the satellite to carry out the mission objectives. To maximize the scientific or technological return on investment, researchers and engineers have been developing new payloads that are more compact, lightweight, and efficient than traditional payloads. For example, some research has focused on developing new imaging sensors that can capture high-resolution images of the Earth's surface using a fraction of the power and weight of traditional sensors, while others have explored the use of micro-electromechanical systems (MEMS) to create new types of scientific instruments that can be more easily integrated into a PocketQube.

In addition to these areas of development, researchers and engineers are also exploring new manufacturing and assembly techniques that can make it easier and more cost-effective to build PocketQubes. For example, some researchers are exploring the use of 3D printing to create satellite components that are more lightweight and efficient than traditional components, while others are exploring the use of nanoscale assembly techniques to build complex systems on a microscopic scale.

Overall, the development of new technologies for PocketQubes is an important area of research that is helping to expand the capabilities and mission lifetimes of these small, low-cost satellites. While there are still many challenges to overcome, the ongoing research and development in this field is helping to pave the way for a new era of low-cost space exploration and research.

4.1.4 Future of the PocketQube Standard

The future of the PocketQube standard is bright, with the technology offering numerous advantages over traditional larger satellites. While the PocketQube standard has already seen significant growth and success, there are several areas of potential for the future.

One area of future development for PocketQubes is the integration of more sophisticated payloads. Payloads are the scientific or technological instruments that are mounted on the satellite to carry out the mission objectives. As technology continues to advance, there will be a greater ability to miniaturize and integrate sophisticated payloads into PocketQubes. This could lead to the development of a new generation of highly capable, low-cost satellites with advanced scientific capabilities.

Another area of potential for the future of PocketQubes is the development of more advanced propulsion systems. Currently, PocketQubes are primarily deployed using a "ride-share" approach, in which they are launched as secondary payloads on larger rockets. This limits the ability of PocketQubes to be placed into specific orbits, which can impact their mission capabilities. The development of advanced propulsion systems, such as micro-thrusters, could allow PocketQubes to be more precisely placed into orbit and could open up new mission opportunities.

Another potential area for future development is the expansion of applications for PocketQubes. Currently, PocketQubes are primarily used for scientific research and technology demonstrations. However, as the technology continues to mature, there may be additional opportunities for commercial and government applications, such as telecommunications or Earth observation.

The development of more efficient and reliable communication systems is also a potential area for future development. While current communication systems for PocketQubes are effective for transmitting small amounts of data over short distances, more advanced systems could expand the capabilities of these satellites. For example, the use of optical communication systems could allow for faster and more reliable communication over longer distances.

Finally, the development of more sustainable and environmentally friendly PocketQubes is also a potential area for future development. While PocketQubes are already a low-cost and environmentally friendly alternative to traditional satellites, there is still potential for further development in this area. For example, the use of alternative energy sources, such as fuel cells or nuclear power, could allow for longer mission lifetimes and reduce the environmental impact of these satellites.

Overall, the future of the PocketQube standard is bright, with many potential areas for development and growth. As technology continues to advance, it is likely that PocketQubes will become even more capable, efficient, and versatile, opening up new opportunities for scientific research, commercial applications, and space exploration. The potential of this technology is truly limitless, and it will be exciting to see what the future holds for PocketQubes.

4.2 PocketQube Model for Testing

In this section will be reported the process of designing and construction of a PQ Satellite model with three reaction wheels inside. This phase of the thesis was performed at the University of Nottingham under the direct supervision of one of the PQ Standard authors, Prof. Chantal Cappelletti whose guide was fundamental for the realization of the project.

4.2.1 Electric Motors Choice and Configuration

The first approach at the problem was of course the correct choice of the components to be integrated in the model, thus since the test has as a primary objective th performance of the reaction wheels its suitable to start from the motors that will be used. A preliminary market analysis was performed to find which motors were the most suitable for the use case setting as the main driver only the size of it since three of them had to fit in such a small volume and, of course, they cannot occupy all of it not leaving space for any other components. Therefore hereafter are reported the main choices as the result of this preliminary market analysis:

- Faulahber S/G.
  The faulhaber S/G is a coreless DC motor from Faulhaber, which is one of the most high end brands in this field. It's small form factor make it suitable for the current application with a power consumption at full regime of around 0.2 W and high rotation speed. It's also made of high quality stainless steel and has high performance magnets inside that allows them to be very responsive and guarantee a precise and fine control of the speed trough its voltage. Hereafter are reported its main dimensions and some of its main performance data.

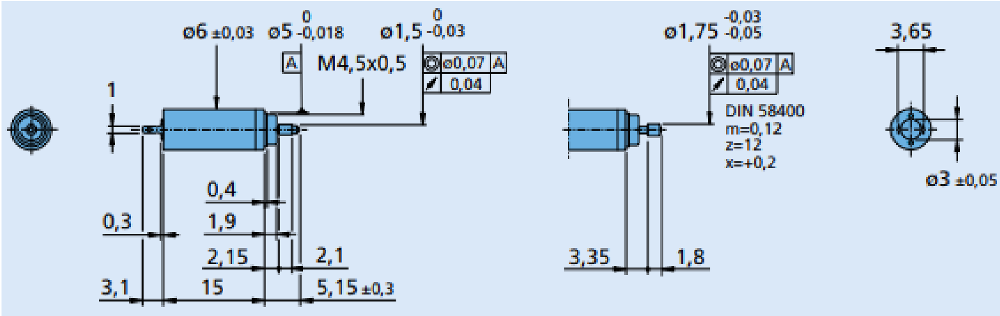


Figure 4.1: Faulhaber S/G Dimensions

Nominal Voltage	1.5-4.5 [V]
Rated Torque	0.17 [mNm]
No-Load speed	20200 [rpm]
Power	0.2 [W] [rpm]

Table 4.1: Faulhaber S/G main performance data

- Faulahber B-Flat. As the previous this motor is from the Faulhaber company but uses a different technology with a 4 pole design and it also differentiate for its extremely flat design with an height of about 9 mm. It is able to do the electronic commutation of the signal using three digital hall sensor allowing a decent and precise control on it. although it also may require a proper driver to control it trough DC voltage. It also has a higher torque then the S/G model at the price of course of a higher power consumption. Again hereafter are reported the main characteristics of the motor:

- Faulahber B-Micro.
  The last motor from Faulhaber is the B-micro which is a coreless brushless DC motor having the same form factor as the fist one while having a more compact design making it on paper a better candidate for the current application. Also its power consumption is much lower with an operating power consumption of around 33 mW at the cost of course of very limited performances in terms of nominal torque. Again this motor would require a proper speed controller to perform as wanted.

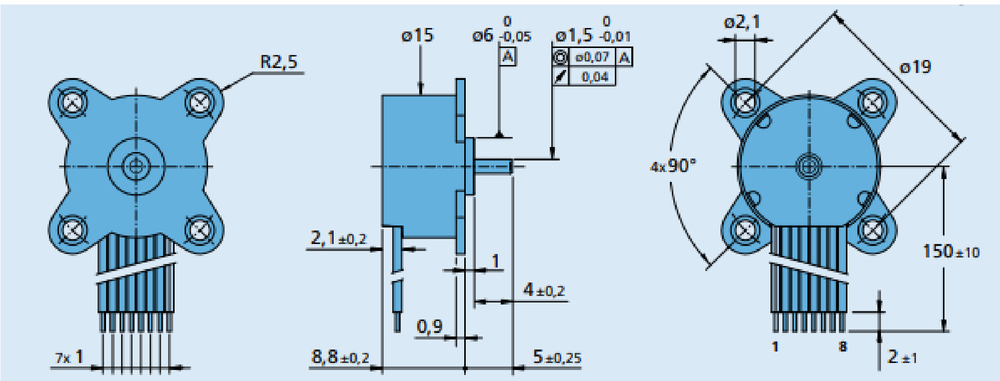


Figure 4.2: Faulhaber B-Flat Dimensions

Nominal Voltage	6-12 [V]
Rated Torque	0.45 [mNm]
No-Load speed	15000 [rpm]
Power	0.6 [W] [rpm]

Table 4.2: Faulhaber B-Flat main performance data

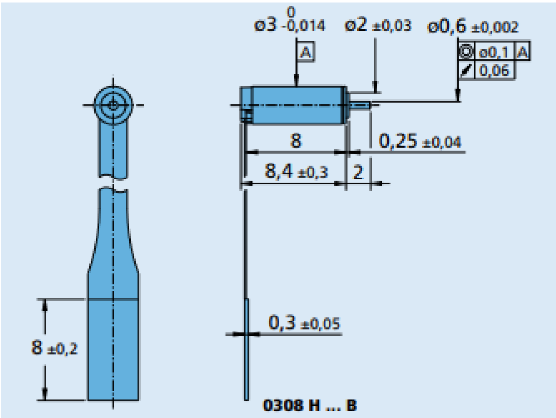


Figure 4.3: Faulhaber B-Micro Dimensions

Nominal Voltage	3 [V]
Rated Torque	0.013 [mNm]
No-Load speed	61000 [rpm]
Power	0.05 [W]

Table 4.3: Faulhaber B-Micro main performance data

- Generic Micro Coreless Motor.
  To find more cost effective solutions also a generic coreless DC motors has been considered and out of all the possibilities the more suitable for this application is the model reported in Fig. 4.4.

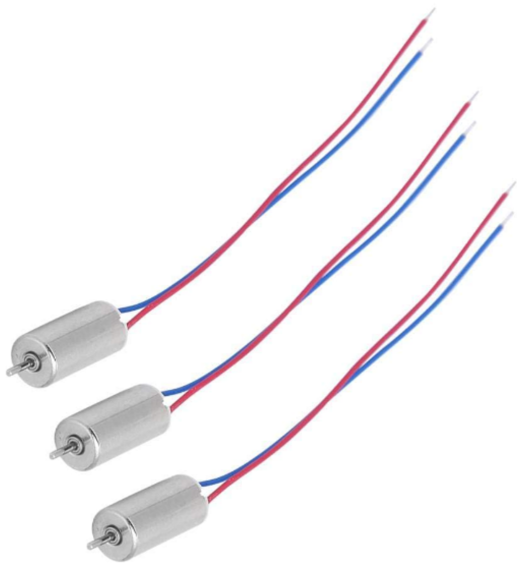


Figure 4.4: Generic Micro Coreless DC Motor

It's size, technology and performances are very similar to Faulhaber S/G but unfortunately since its a generic product the quality and reliability are not as high. Since no data-sheet could been found for this specific motor the dimension are just reported on the site and its diameter is about 6 mm and has an height of 12 mm. As before the main performance data are reported in Tab. 4.4.

Nominal Voltage	1-3.7 [V]
Rated Torque	0.17 [mNm]
No-Load speed	20000 [rpm]
Power	0.3 [W]

Table 4.4: Generic Coreless DC Motor main performance data

- Brushless DC Motor from 30 mm generic fan.
  For completion, it has been taken in consideration to use the Brushless DC motors from a generic 30 mm computer fan, retooling it to be the reaction wheel of our model. It would be a very cost-effective solution, and could be controlled via a simple PWM signal. On the other hand the performance would not be very clear and has to be estimated and the rated power consumption its at the limits of the wanted application.

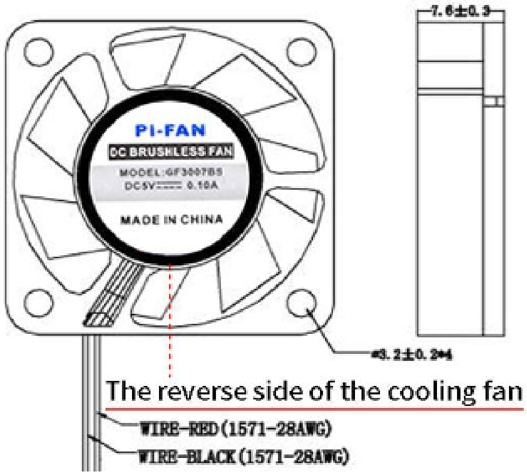


Figure 4.5: Generic 30 mm Brushless Computer Fan

4.2.2 Motors Trade-Off

After the Preliminary market analysis based mainly on the size of the motors other factors has been taken into consideration to make a pondered and good choice of the main component of the model. The following Drivers are taken in consideration for a Trade-Off of the previous cited five motors:

- Cost.
  The cost of the motor is a huge factor in this project since its not been payed through properly allocated research founds and its just a proof of concept. Thus a competitive price for the components is set as one of the main drivers of the project.

- Time.
  Since the project has to be finished in about two months delivery time should be as low as possible and not above 1 or 2 weeks. Again, since time management is concerning this is one of the main drivers for this Trade-Off.

- Operability. Since time management and cost are the main drivers, also it has to be taken in consideration how easily a motor can be controlled since a better one could avoid wastes of times and troubleshooting sessions during development.

- Integration.
  A good, easy and proper integration of the motors in the model would play a very big role during the project since volume and masses are very limited.

- Performances.
  The performances of the motors are again a very important driver since have to be balanced for the use case and it cant risked to have the model with inadequate specifications.

- Power Requirements. Finally the last driver is the respect of the power requirements of the PQ Standard cited in the previous section. Although this is set to be the less important one since those are components not specifically designed for this use case and could be optimized in further iterations of the project.

So, following a simple and basic procedure of Trade-Off a weight from 1 to 3 is attributed to each driver and for each motor a score from 1 to 5 is attributed to each driver. Multiplying the score for its weight and making the sum of each one a final standings of the best motors should be the result narrowing the options to choose from. The scores and totals are reported in Tab 4.5. For a better vistualization of the results they are reported also in a Star Diagram in Fig. 4.6:

	Cost	Time	Operability	Integration	Performances	Power Requirements	Total Score
Driver Weigth	3	3	2	2	2	1	-
Faulhaber S/G	1	1	3	3	4	4	29
Micro Coreless Motor	5	4	4	3	3	3	45
BL Fan's DC Motor	4	4	4	2	2	2	38
Faulhaber B-Micro	2	3	2	4	3	5	36
Faulhaber B-Flat	1	3	2	2	4	2	29

Table 4.5: Motors Trade-Off Table

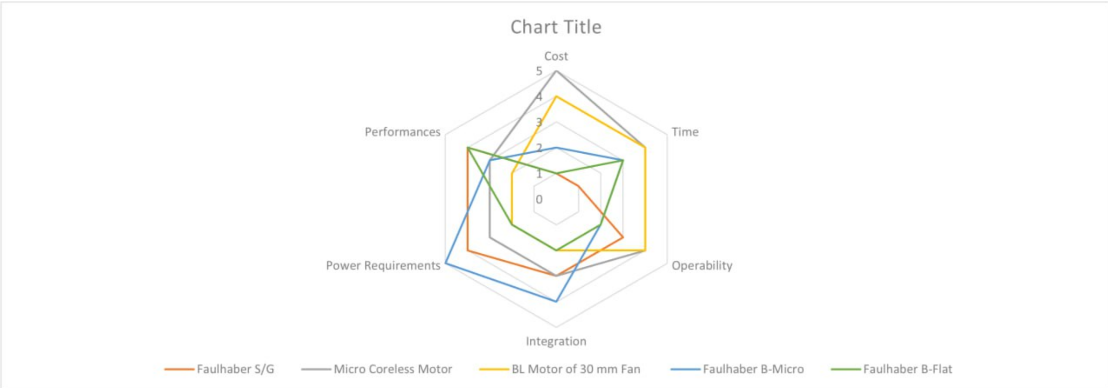


Figure 4.6: Motors Trade-Off Diagram

After a quick evaluation of the results it seems clear that for the use case the generic Coreless Motor is seems the most reasonable chooich that better matches the set drivers of the project providing very contained costs, Fast delivery times up to next day, great controllability and good enough performances in ratio to the power needed. Overall it is a solid choice that fits well the study case. Its important top know that that model in particular is a brushed motors are not suitable for space applications since their technologies provide some issues. First of all there could be some arcing phenomena between brush and commutator wich is not ideal for its operability being thus less efficent and having poor thermal characteristics in vacuum. They can also cause some Electromagnetic interference that should be avoided at all cost for the correct functioning of the overall satellite. On the other hand, Brushless motor provide a higher electronics cost and a much greater motor drive complexity which does not fit well the current state of the project. Also the technology used for the brushed DC motor chosen could be found with similar performances and power specification in its brushless version. In fact the Faulhaber-Micro previously taken into consideration is in fact a clear example of that.

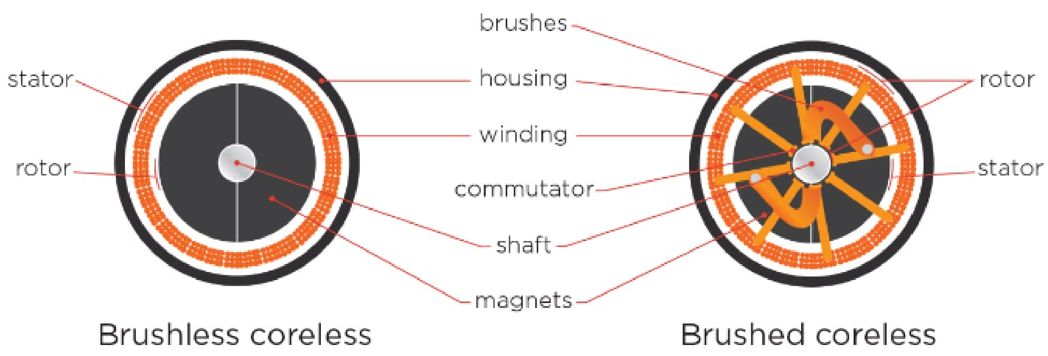


Figure 4.7: Brushless and Brushed Corless Motor Comparison

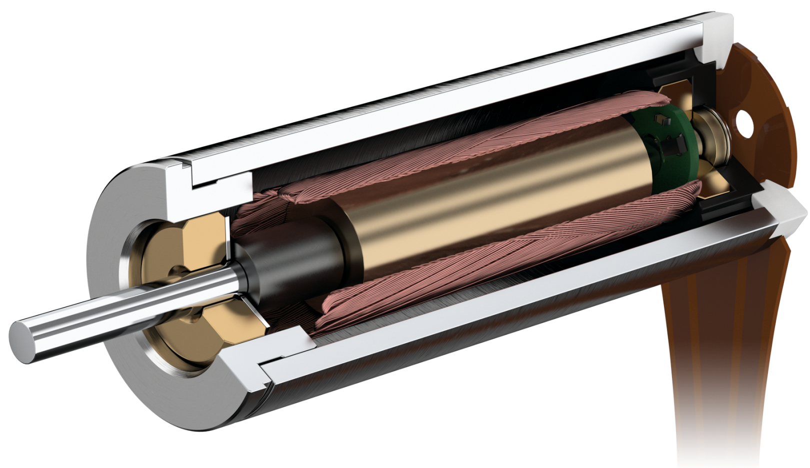


Figure 4.8: Faulhaber B-Micro Teardown

4.2.3 Motors configuration

After the definitive choice of the motors model it's necessary to think about a proper and suitable configuration and since there are only three wheels available the reasonable choices are just two:

- Orthogonal Configuration;
- Tetrahedral Configuration.

Their main differences, also considering the main advantages and disadvantages, can be listed as follows:

- Geometry.
  The main difference between the two configurations is their geometry. The standard orthogonal configuration consists of three reaction wheels arranged orthogonally along the x, y, and z axes, whereas the tetrahedral configuration consists of three reaction wheels arranged in a tetrahedron shape, with one wheel at the center and the other two at the vertices of an equilateral triangle.
- Control.
  The standard orthogonal configuration provides good control over pitch, yaw, and roll. However, it is susceptible to singularities, which can cause in some extreme cases loss of control. The tetrahedral configuration, on the other hand, provides good control over all three axes without singularities giving more robustness to the system.
- Complexity.
  The tetrahedral configuration requires more power to operate than the standard orthogonal configuration due to the additional complex control algorithms needed to maintain stability.
- Volume usage.
  The tetrahedral configuration requires less space than the standard orthogonal configuration since the wheels can be placed closer together. This can be an advantage for smaller spacecraft with limited space.
- Robustness.
  The tetrahedral configuration is more robust than the standard orthogonal configuration in the face of external disturbances such as solar wind or gravitational pull from nearby objects. This is due to the fact that the tetrahedral configuration provides better control over all three axes.

In summary, while the standard orthogonal configuration provides good control over pitch, yaw, and roll, the tetrahedral configuration offers more redundancy, better control over all three axes, and more robustness to external disturbances. The tetrahedral configuration also provides better control over all three axes and is less susceptible to singularities, which can be advantageous in certain situations at the price of a more complex algorithm to control the spacecraft. So, Given the strict volume budget and the great progress that in the last years has been made to create smaller but still really powerful controllers giving up some algorithm complexity in order to achieve better volume management, in particular in this early stage of the design, the tetrahedral configuration seems the most suitable and flexible choice. So a preliminary CAD model of the motors could be created to visualize the configuration and a preliminary volume for the flying wheel can be estimated and reported in Fig 4.9 and Fig. 4.10.

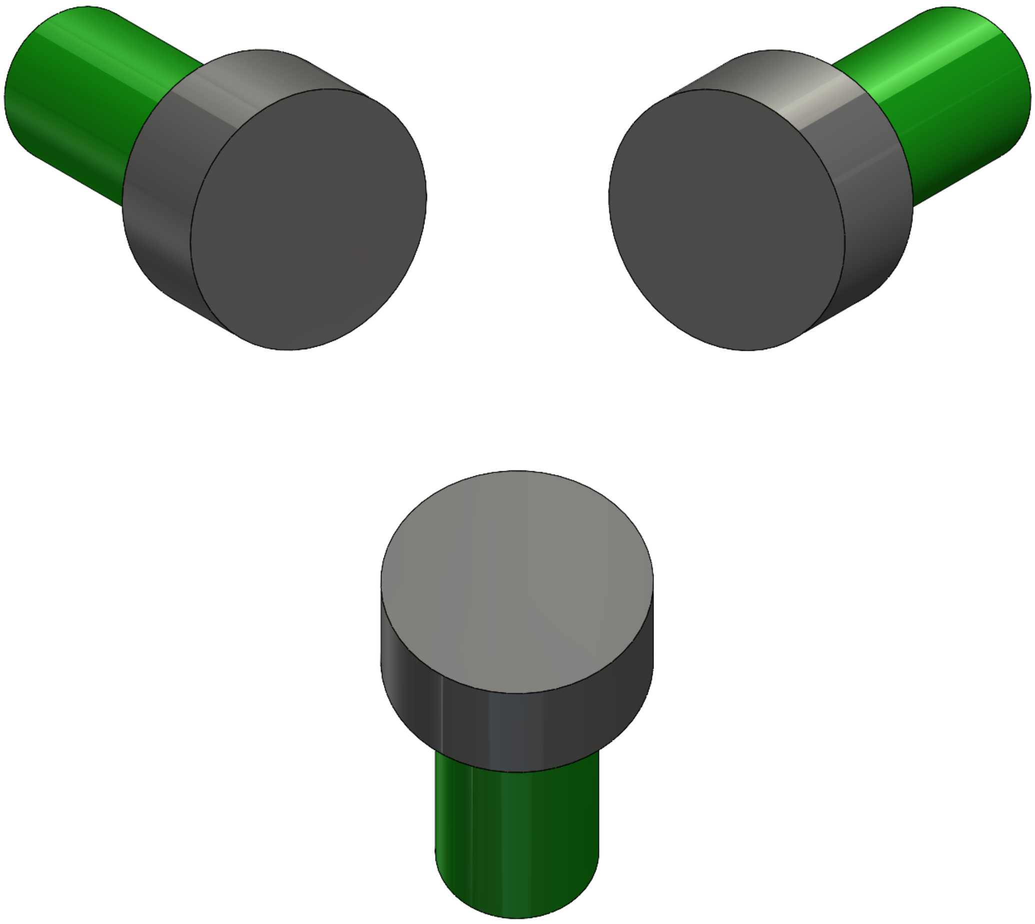


Figure 4.9: fig:Motors Configuration Top View

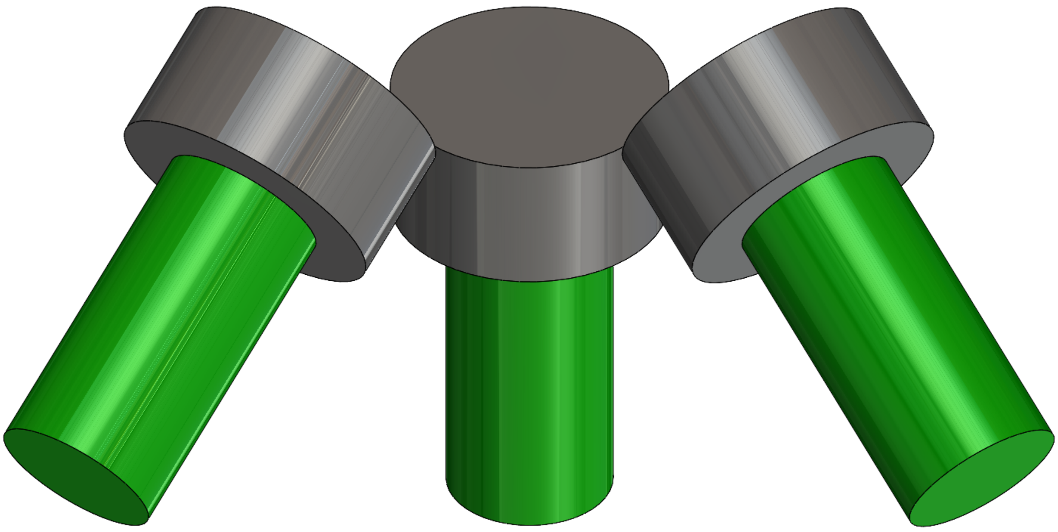


Figure 4.10: Motors Configuration Side View

Finally since the configuration has been fixed it is possible and very important for future steps to calculate the rotational matrix for this particular configuration stating the components of the angular momentum generated by each wheel on what are assumed to be the principal axis of inertia of the overall satellite. To set a generic convention from now on the vertical axe of the configuration is set as the y axis, while the other two forming an orthogonal triplet are the x and z axes. A quick scheme of the configuration of the axis is reported below in Fig. 4.11.

To define then the rotational Matrix for the configuration the angles between the axis of the motors and the orthogonal axis stated have to be defined. Starting for the y axis, all the motors follow the regular tetrahedral shape, thus the angle between tre axis of the rotor and the y axis is equals to:

$$\Delta = 35.26\,\mathrm{deg}\,. \tag{4.1}$$

Concerning the plane angles, the motors are all placed on the vertices of an equilateral triangle with motor [2] lying on the z axis and the others spaced 120 deg from it. Following these assumption the rotational matrix can be calculated as follows using simple trigonometric formulas:

$$Z = \begin{bmatrix} sin(\Delta)cos(\pi/6) & 0 & -sin(\Delta)cos(\pi/6) \\ cos(\Delta) & cos(\Delta) & cos(\Delta) \\ sin(\Delta)sin(\pi/6) & -sin(\Delta) & sin(\Delta)sin(\pi/6) \end{bmatrix}. \tag{4.2}$$

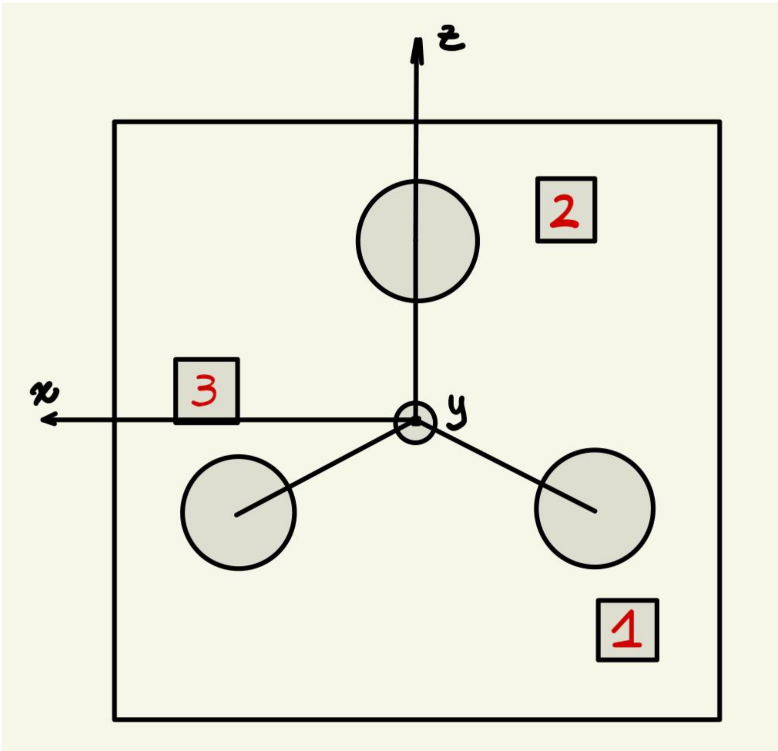


Figure 4.11: Motors Enumeration and Axis definition

4.3 Electronics Components

In this section will be discussed the other components needed to create a proper model of the PQ Satellite including all the Electronics and hardware. Most of them were directly provided by the University of Nottingham which they use for their FlatSat Project. This was possible mainly because all the components provided in this project were characterized by compatible dimensions and requirements for the model and knowing already that they will work together without many issues was a very important factor for the time-effectiveness of the project. On the other hand being stucked with fixed components could fossilize the project a little bit a make it lose some versatilty. However they could be considered less important for the main purpose of the project so they can be taken as viable for it.

4.3.1 Micro Controller-Pimoroni™ Tiny 2040

The first component needed in the model is, of course, the 'brain' of the PQ Satellite. This is the only component that has been changed from the FlatSat project of the University of Nottingham since it originally used the Raspberry Pi™ Pico (RP Pico) which is unfortunately over the size capability of the PQ Satellite standard with its 51 mm of length. The point is that not all the General Purpose Pins (GPIO) that the RP Pico provide are actually necessary for the test and therefore a miniaturized version of it can be used. The choice happened to be the PIMORONI Tiny 2040 which has the exact same specs of the RP Pico but its much smaller. For comparison an image of both micro-copntrollers is reported in Fig. 4.12. In terms of size, the Tiny 2040 measures just 20mm x 40mm, making it one

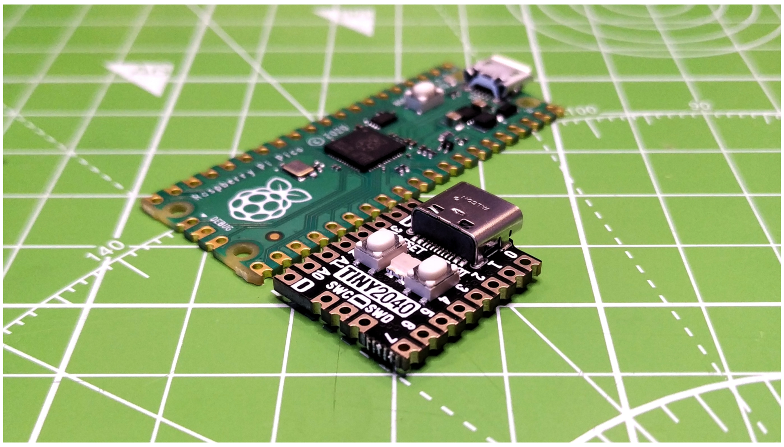


Figure 4.12: RP Pico and PIMORONI™ TINY 2040

of the smallest and most compact microcontroller boards available. This makes it well suited for use in space-constrained applications such as in this project, where size and weight are critical considerations. The Tiny 2040 features a 32-bit Arm Cortex-M0+ processor with 48 MHz clock speed, 256 KB flash memory, 32 KB RAM, and a variety of communication interfaces, including USB, SPI, I2C, and UART making it very versatile and it also includes an on-board LED and a reset button, which could be very important during test phase. In Fig. 4.13 is reported its pin layout and specification used to integrate then all the other components. Overall the PIMORONY TINY 2040 seems the perfect choice for the model since

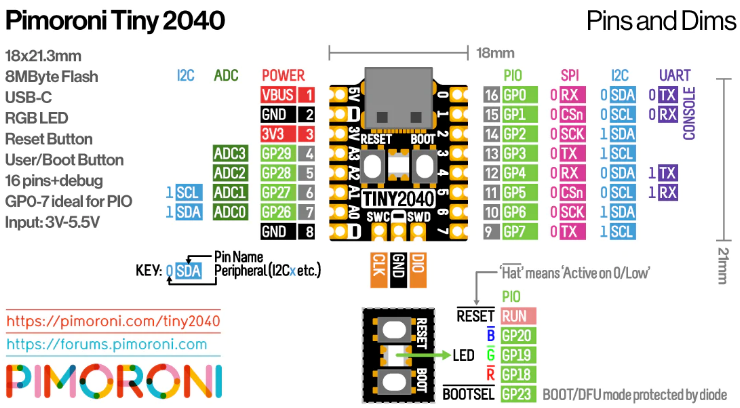


Figure 4.13: PIMORONI$^{\text{TM}}$ TINY 2040 Pin Layout

its processor is capable to do complex calculation in a very restricted time which is fundamental for the control algorithm necessary for the test.

4.3.2 Battery Pack

For what concerns the battery pack the choice was very limited to what was available at the moment in the laboratory of the University of Nottingham. The module used for this project happened to be the only one that fits the dimensions restrictions of the PQ Standard. A pic of it is reported in Fig. 4.14. Overall this standard battery pack provides exactly what is needed with its operating voltage of 3.7 V and its capacity of 500 mAh should provide enough power for all the duration of the test. It is also embadded with a standard JST (Japanese Solderless Terminal) which is very convenient for the integration with other components of the model with much complications.

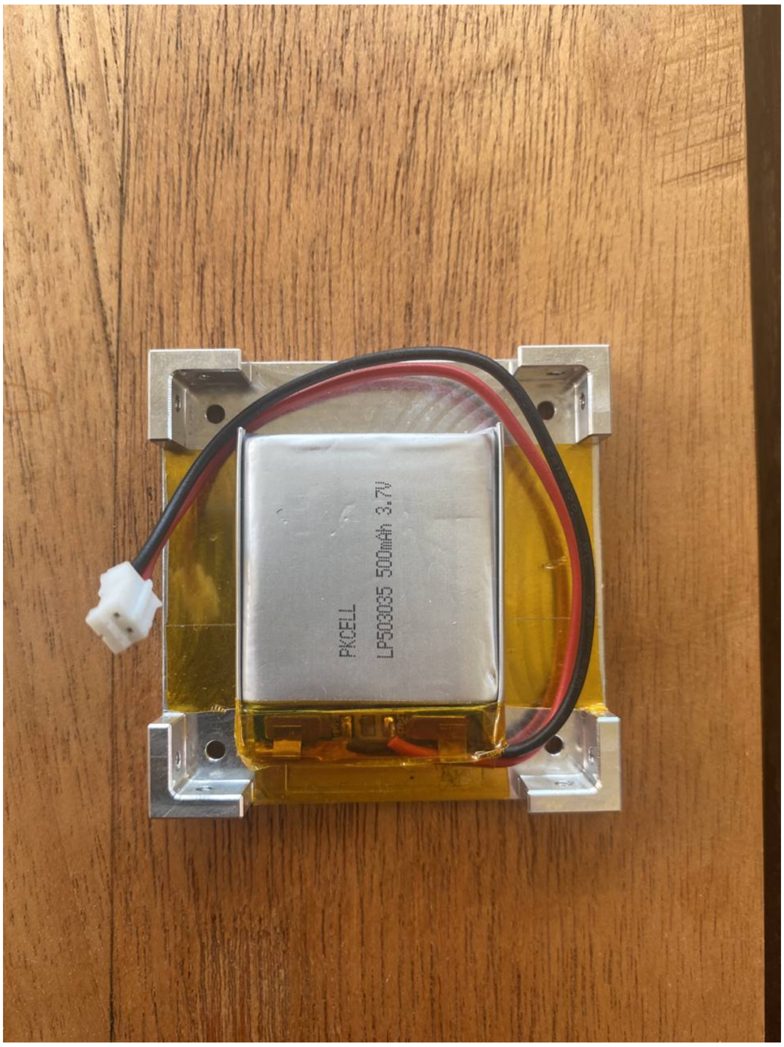


Figure 4.14: Battery Pack

4.3.3 Battery Board - Power Boost 500 Charger

The Adafruit™ Powerboost 500 Charger is a small, portable device that allows you to power and recharge electronic projects with a single lithium-ion battery and here can be listed some of the main features that make it suitable for this application:

- Charging capability.
  The Powerboost 500 Charger is capable of charging a single lithium-ion battery with a capacity of up to 500mAh with the same connector of the battery previously described.

- Voltage boost.
  The device features a voltage boost circuit that can increase the battery voltage to 5V, which is suitable for powering most electronic projects.

- Overload protection.
  Given the experimental nature of the project, the Powerboost 500 Charger has built-in overload protection to prevent damage to the battery or your electronic device in case of a short circuit or other electrical malfunction.

- Compact design.
  The device is small and lightweight, making it ideal for the use case.

- Low battery indicator.
  The device has an LED indicator that signals when the battery is running low. which will come handy when running the tests avoiding interruptions caused by a discharged battery.

For completion an image of the device is reported in Fig. 4.15.

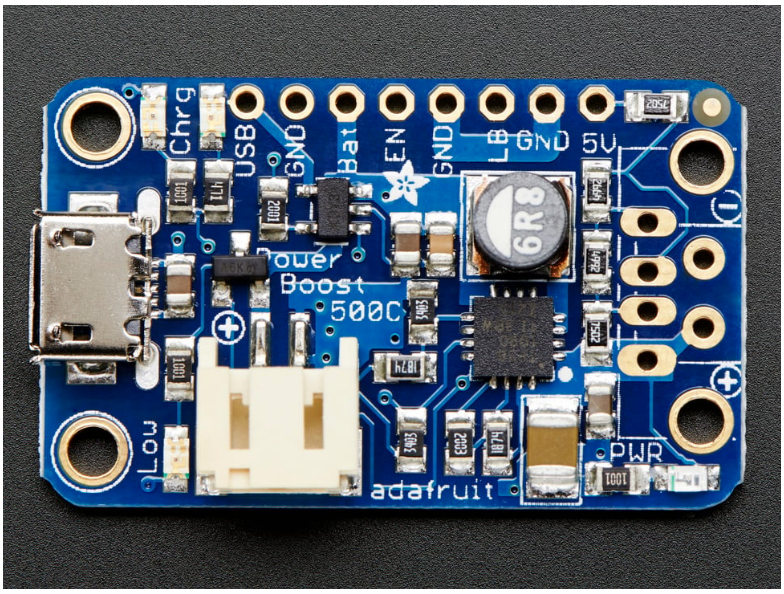


Figure 4.15: Adafruit^{TM} Power Boost 500 Charger

4.3.4 H-Bridge - L9110H

At this point seems evident that the plan is to control the motors using the general purpose pin of the PIMORONI Tiny 2040 that would send a PWM input to the motor and being able to control it through its voltage. Although to do so te motor cannot be connected directly to the microcontroller since its not capable to provide enough power and so an motor driver is needed. The solution to this issue is just to use an h bridge driver that would take the power directly from the battery board and provide given a proper input from the micro-controller a proportional voltage to the motors. It is important to note that an opportune choice has to been made because h-bridges usually are made to output a lot of power and since the motors are rated for just 3.7 V and 300 mA it is important to fulfil those requirements. A good choice happened to be the L9110H from Texas Instrument which can be easily found on online stores and provide the correct output needed. Also its very compact size make it very easy to integrate in the model. A quick schematics of how it works is reported in Fig. 4.16 As its shown, this also allows to control the

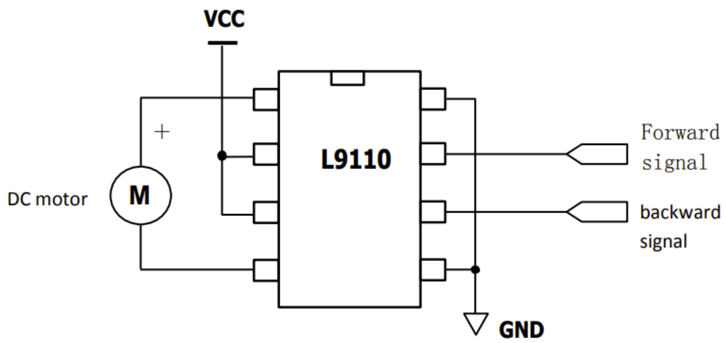


Figure 4.16: L9110H Schematics

motor in both ways which is crucial for this application.

4.3.5 Power Monitor Module - INA 260

To monitor the power consumption from the power module, which would basically rapresent the overall power consumption of the entirety of the model, a power monitor module is necessary. That should provide crucial information about the overall power consumption during the test to compare with the estimated results. The choice happen to be another component from Adafruit™, the INA 260 which the main characteristics are listed below:

- Voltage and Current sensing.
  The INA260 can measure voltage levels up to 36V and current levels up to 15A, with high accuracy.

- Digital interface.

The INA260 communicates using I2C or SMBus digital interface, which allows for easy integration with microcontrollers, processors, and other digital devices.

- Accuracy.
  The INA260 has high accuracy and precision, with a maximum current measurement error of ± 0.0015 and a maximum voltage measurement error of ± 0.001.

- Integrated functions.
  The INA260 includes features like overvoltage and undervoltage detection, overcurrent detection, and overtemperature protection.

- Low power consumption.
  The INA260 has a low power consumption, which makes it suitable for battery-powered applications.

- Wide operating temperature range.
  The device can operate in a wide temperature range from -40°C to 125°C.

Again an image of the component is reported for good measure in Fig. 4.17.

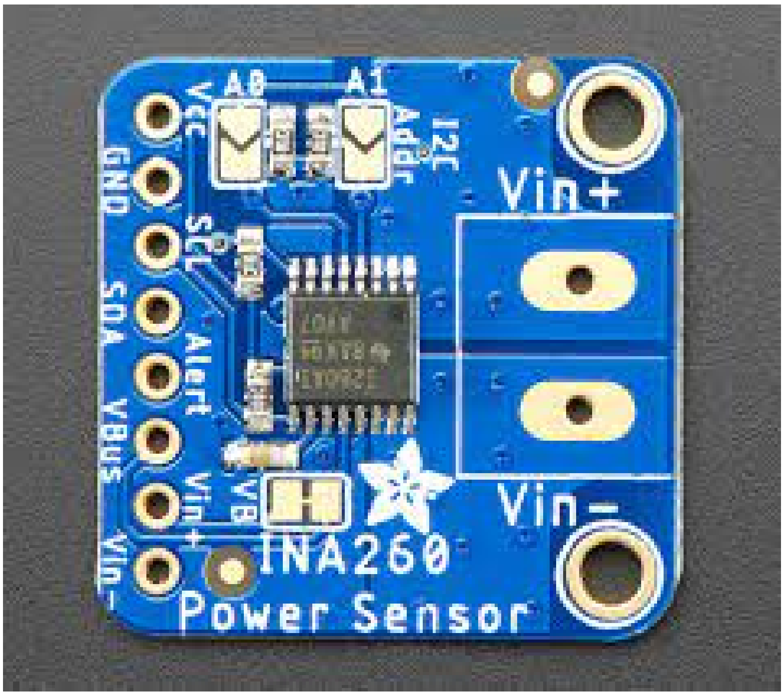


Figure 4.17: Adafruit™ INA 260

4.3.6 Motion Control Sensor - ICM 20948

Since the test would require the change of the attitude of the model one of the main components that needs to be integrated is a motion control module to detect movements and send the microcontroller the data needed to properly activate the electric motors. The Adafruit ICM 20948 is a 9 degree of freedom (DOF) motion sensor module that combines a 3-axis accelerometer, 3-axis gyroscope, and 3-axis magnetometer in a single package. Here again are listed some of its main features that makes it a good candidate to be used for attitude and heading reference systems, as well as for navigation and guidance:

- High precision.
  The ICM 20948 has a resolution of 16-bits for the accelerometer and gyroscope, and 14-bits for the magnetometer, which allows for high precision measurements.

- Low power consumption.
  The module has a low power consumption mode, which is ideal for battery-powered applications.

- Multiple communication protocols.
  The ICM 20948 can communicate with a host processor using I2C or SPI interface protocols.

- Built-in motion processing.
  The module has built-in motion processing algorithms that can be used to detect motion, calculate orientation, and track gestures.

- Wide operating range.
  The ICM 20948 can operate within a temperature range of -40°C to 85°C.

- Compact size.
  The module measures only 2.5mm x 3mm, making it suitable for applications where space is limited.

It is important to note that since the main data that will be required from this module are the angular velocities around the main three axis it will be necessary to refer them to the principal axis of the overall satellite and also convert them from degrees per second to radians per second since all the calculation will be done using the last one as measure. An Image of the motion sensor used can be found at Fig. 4.18.

Figure 4.18: PIMORONI^{TM} ICM-20948

4.3.7 Air Quality Sensor - BME 680

Along the motion sensor a similar form factor module that can be integrated in the model is an air quality sensor that could provide crucial information regarding pressure and temperature on board. The one used for the model is the PIMORONI^{TM} breakdown board of the BME-680 which combines a gas sensor, humidity sensor, pressure sensor, and temperature sensor in one compact package. Again, its high enough accuracy and low power consumption make it very suitable for the use case and its easily integratable on board of the model. It uses the same I2C or SPI interface as the Motion Control Sensor and the Power Monitor Board making it very compatible with all the components previously cited. In terms of size, the BME680 measures just 3mm x 3mm x 1mm, making it one of the smallest and most compact air quality sensors available. This makes it well suited for use in space-constrained applications, such as PocketQube satellites, where size and weight are critical considerations.

Overall, the BME680 is a highly capable and versatile air quality sensor that provides real-time information about indoor air quality in a compact and low-power package and an image of it is reported in Fig. 4.19.

Figure 4.19: PIMORONI^{TM} BMETM-680

4.3.8 433 MHz Antenna Module

To provide some basic communication capabilities with an hypothetical ground station it has been decided to integrate also a basic transceiver module providing Communication on the UHF band at around 433 Mhz that can be controlled using the UART pin on the PIMORONI^{TM} Tiny 2040. The DWM-APC220-433MHz/470MHz Data Link module is a wireless communication module that uses frequency modulation (FM) to transmit and receive data. Here are some of its main features:

- High data rates.
  The module supports data rates up to 19.2kbps, which is suitable for transmitting sensor data, commands, and other types of digital data.
- Multiple channels.
  The module can operate on multiple channels, allowing for multiple data links to operate in close proximity without interference making it very versatile.
- Low power consumption.
  The module uses a low power sleep mode, which is ideal for battery-powered applications such as this one.
- Compact size.
  The module has a small form factor and can be easily integrated into a variety of projects.

A representative picture of the module can be found in Fig. 4.20.

Figure 4.20: 433 MHz Transceiver Module

In the context of a PocketQube satellite, the 433 MHz Antenna Module is likely used to transmit telemetry data from the satellite back to the ground station from an altitude of around 300 kilometers and this one in particular can only transmit and receive data from 1000 meters. Of course this is not representative of a real case scenario but since the test is meant to be only on the motors the availability and low cost of this module make it good enough for the current iteration of the project.

4.3.9 Solar Panels

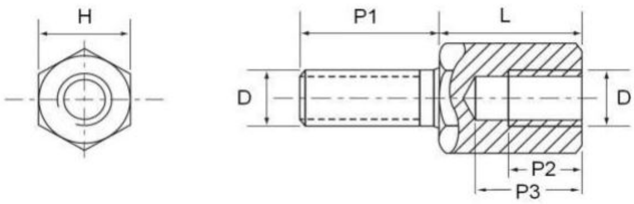
To make the model even more close to an actual satellite it has been decided, since they were available directly in the laboratory of the University of Nottingham, to also integrate three solar panels on the sides of the satellite. The performance of them were not very clear since no data sheet for them was found, but they are still representative for the overall mass of the model and its inertia.

4.4 Structure and Integration

After having all the components needed to make the satellite function properly its time to integrate them following the PocketQube Standard designing ad-hoc structures and baseplates to integrate them all together along with sides panels to mount the solar panels on. Following those assumptions a structure based on different levels its been supposed following also future iteration in which the all the electronics could be integrated in a single PCB whit some proper circuit design.

4.4.1 Stand-Offs

Following those assumptions it was clear that a structure like this in a PQ-Standard Model could only be achieved using Stand-Offs and the only availability for the fitting size and threaded hole dimension of the M2 Standard were the ones from RS Components which states at an height of 10 mm per stand-off. This dimension is way too much for the current application and a half of this size would have been optimal, but unfortunately were not available at the moment so the project and the design of the model all turned around of those measures. Some technical detail of the Stand-Off used are reported in Fig. 4.21 for the Male-Female version and in Fig. 4.22 for the Female-Female version necessary for the coupling of all of them.

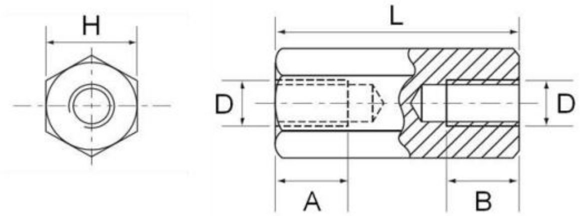


Specifications ±0.1mm

RS No.	Material	Finish	P1+L	H	D	P1	P2	L
2240411			10				3	5
2240412			15					10
2240413			20					15
2240414	Brass	Nickel Plated	25	4	M2	5		20
2240415			30					25
2240416			35					30

Figure 4.21: MF Stand-Offs Technical details

Those Stand-Offs should provide opportune flexibility to the structure configuration and also guarantee a good strength of the overall assembly.



Specifications ±0.1mm

RS No.	Material	Finish	L	H	D	Hole(A/B)	
2240365			5			5	
2240361			10			10	
2240362	Brass	Nickel Plated	15	4	M2	5	5
2240363			20			5	5
2240364			30			5	5

Figure 4.22: FF Stand-Offs Technical details

4.4.2 Main Baseplates

The main baseplates design followed two main drivers:

- Manufacturing Cost.
  Since those baseplates are meant to be manufactured by machining of aluminium 5005, which is quite expensive. The design should have been as simplistic as possible featuring the less possible sharp edges which could not be done properly and some heavy tolerances have to be applied.

- Functionality.
  The main baseplate should provide all the threaded holes to mount all the standoffs and also all the sides panels.

The result of the design is shown in the following CAD model reported in Fig. 4.23.

As it is shown the bottom holes are meant to allocate the stand-offs, the main base can also be accounted to place some components such as the battery after isolating it with a piece of Kapton Adhesive. Also the edges can be easly used as support of a side panel with threaded M2 holes equally spaced.

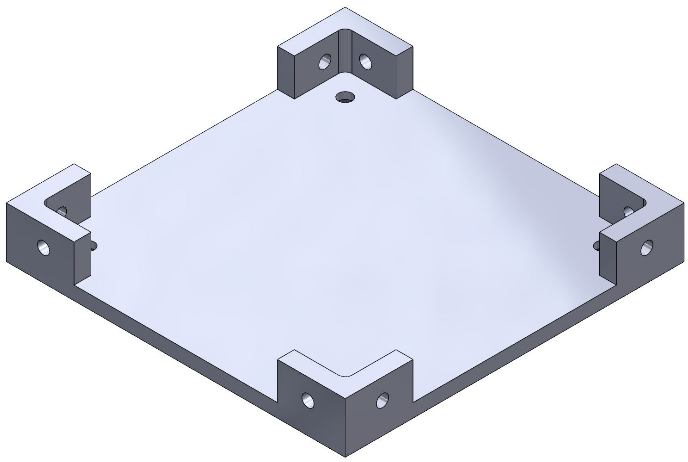


Figure 4.23: Main Baseplate Design

4.4.3 Internal Baseplates

The internal baseplates were meant to be 3D printed, so a more detailed design could be made and more focused on the integration of all the components considering not only how they could fit in a specified surface but also following their wiring references to avoid complications during the soldering process of all the components. On the other hand also the baseplate that would hold the motors has been designed following the sketch and the configuration reported before. Doing the design it as been take into account also the tolerances of the 3D printer that would have been used to manufacture the components of 0.2 mm exception made for the motor baseplate for which a tight fit would have been preferred since they had to be very stable during the test. The shape of the holes has been made following the projection of the cylinder of the motor at an angle of 35.26°. So an image of each baseplate is reported in the following images:

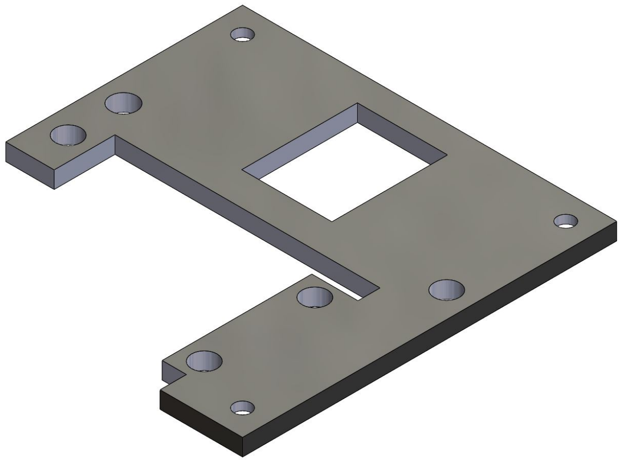


Figure 4.24: L1 Baseplate Design

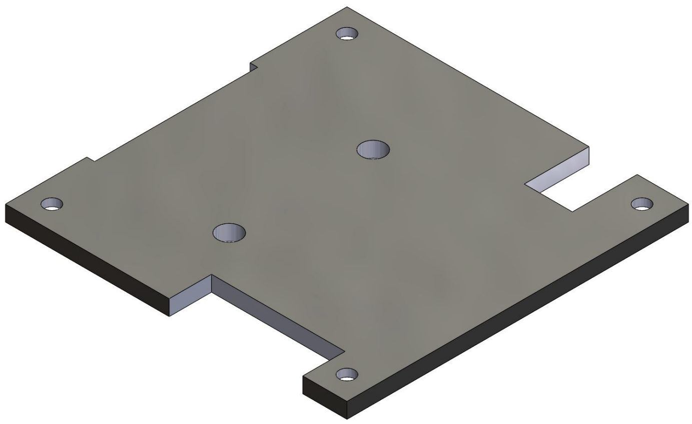


Figure 4.25: L2 Baseplate Design

Figure 4.26: RW Baseplate Design

4.5 Overall CAD Design and Model

At this Point all the elements are ready for an overall cad integration of all the components and the final result after all the work is reported in the following images:

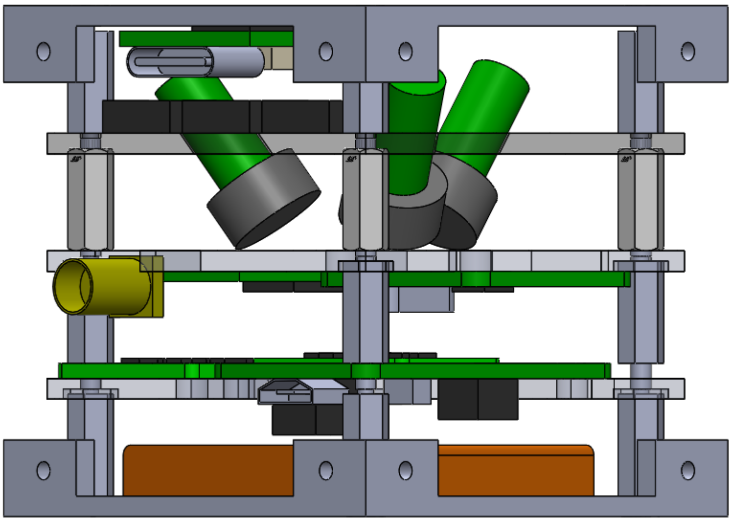


Figure 4.27: Assembly without side panels

As shown its clear that the overall volume of the satellite is not used at its maximum potential due to all the empty space caused by the use of 100 spacers

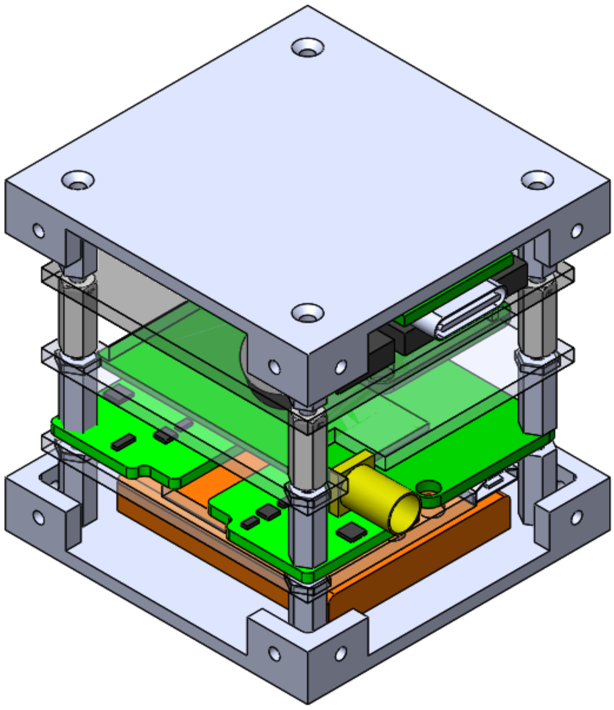


Figure 4.28: Assembly without side panels

from each level of the model. The bottom plate is used to attach the battery that will be plugged to the battery board directly on the following level which comprehend also the motion control sensor and the temperature sensor. On the level up above are integrated the antenna module along with the power monitor module and following that enough space for the electric motors and their flying wheels. At the end on the top Baseplate its shown the main component of the model which is the microcontroller along with three (one for each motor) H-bridges. It is important top note that the panel facing the antenna and the USB-C exit of the controller was properly modelized to fit those exits and facilitate the interface with them during the testing procedure.

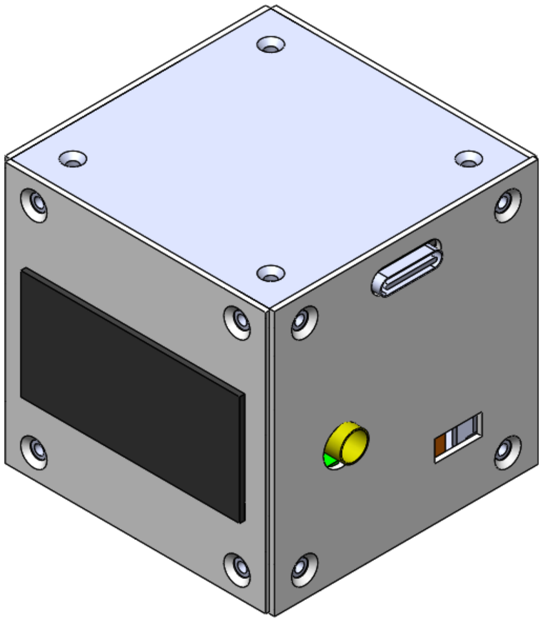


Figure 4.29: Assembly with side panels

4.5.1 Mass Budget and Properties

After having a proper rappresentation of the model all components have been weighted with a poper scale and an overall mass budget can be estimated. Following the logic of the components they have been divided into 3 main categories:

- Structure: 54,93 [g];
- Electronics: 37,48 [g];
- Reaction Wheels: 13,41 [g].

So given the weigth of each main subcategory they can be more easily visualized in the following diagram showing the percentage for each one in Fig 4.30. As its clearly shown the majority of the total mass is represented by the structure and only the 0.13 of the total is occupied by the reaction wheels showing the potential of this application. The total mass of the virtual model is:

$$m_tot = 105.82[g]. \tag{4.3}$$

The current estimation doesn't take into consideration the total mass of the wires necessary to connect all the components together and can be estimated to be about

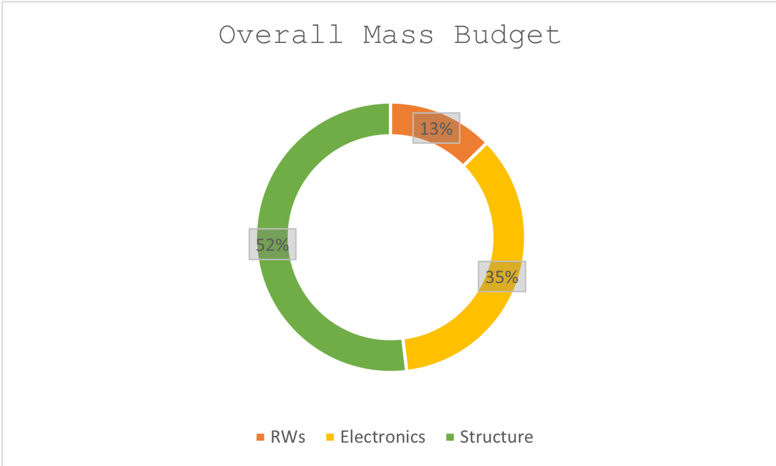


Figure 4.30: Total Mass Budget

20 percent more. From the CAD model is very important for the project to extract the position of the principal axys of inertia and the value of the principal moment of inertia of the model around those. the weight of the satellite and its volumes are pretty much evenly distributed distributed so as the principal axis of inertia can be taken into account the orthogonal triplet originating from the center of mass of the satellite. Solidworks^{TM} can estimate at this point also the Inertia Matrix of all the model and is reported hereafter:

$$I = \begin{bmatrix} 48821.48 & -19.14 & -208.58 \\ -19.14 & 46576.31 & -78.48 \\ -208.58 & -78.48 & 50198.01 \end{bmatrix} [gmm^2]. \tag{4.4}$$

This Matrix is of course fundamental for the control algorithm.

4.5.2 Costs

The same thing can be done from a cost perspective to show how powerful the cost reduction is to implement such a model. In the following table are listed the main costs in GBP (£) for each component.

At around 280 GBP the model proved to be really cost effective with most of the costs reported for the structure since it was necessary for the purpose of the test to be machined from bare Aluminum and this operation happened to absorb most of the budget. Overall the total cost differentiation can be made as before following the same categories and is reported in Fig. 4.31.

Component	Cost [£]
Motor	2
H-Bridge	2
Motion Sensor	14
Temperature Sensor	20
Power Monitor	10
Battery Module	15
PIMORONI™ Tiny 2040	10
Antenna Module	15
Solar Cell	15
Stand-Offs	28.80
Main Baseplates	130
Total	281.80

Table 4.6: Individual Costs of each Component

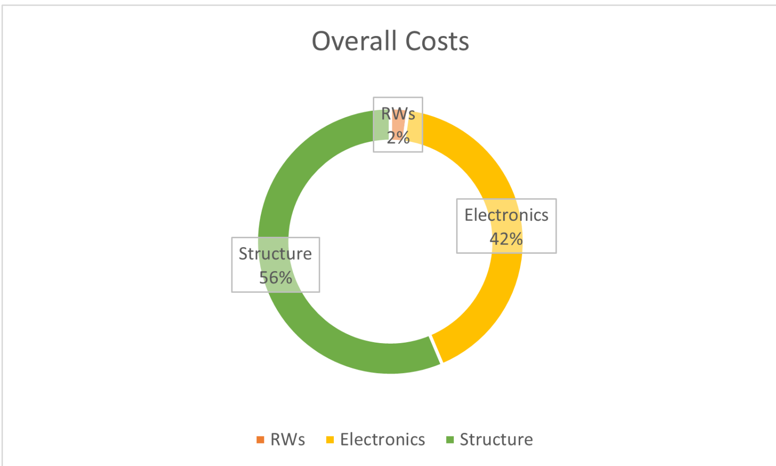


Figure 4.31: Total Cost Division

Chapter 5

Model Simulation and Test

5.1 Control Algorithm - Quaternion Feedback Algorithm

Satellites are important tools for space exploration, earth observation, and communication. As satellites operate in the harsh environment of space, they can experience a variety of disturbances that can cause them to deviate from their intended orbits. One such disturbance is the change in angular velocity, which can occur due to a variety of factors such as atmospheric drag, solar wind, or gravitational perturbations from other celestial bodies. To maintain the desired orbit, it is essential to detect and correct such disturbances. In this context, the quaternion feedback algorithm can be used to sense the change in angular velocity of a satellite and perform a maneuver to restore its initial position. This article describes the use of the quaternion feedback algorithm for satellite attitude control.

Quaternion feedback is a popular algorithm used for attitude control in satellites. It is based on the concept of quaternion rotation, which is a mathematical tool used to represent the rotation of an object in three-dimensional space. A quaternion is a four-dimensional vector that can be used to represent the rotation of an object around an axis. It consists of a scalar component and a three-component vector.

In satellite attitude control, the quaternion feedback algorithm is used to sense the change in angular velocity of the satellite and perform a maneuver to restore its initial position. The algorithm works by measuring the angular velocity of the satellite using a gyroscope and using the quaternion rotation to calculate the required torque to correct the attitude of the satellite. The algorithm involves the following steps:

Sensing the angular velocity of the satellite using a gyroscope.

Calculating the error quaternion, which represents the difference between the desired and actual orientation of the satellite. This is done by multiplying the conjugate of the desired quaternion by the actual quaternion.

Using the error quaternion to calculate the required torque to correct the attitude of the satellite. This is done by multiplying the error quaternion by a gain matrix, which is a matrix that determines the sensitivity of the system to the error.

Applying the required torque to the satellite using reaction wheels to correct its attitude.

The quaternion feedback algorithm is a closed-loop control system, which means that it continuously adjusts the required torque based on the feedback from the gyroscope. This ensures that the satellite maintains the desired orientation even in the presence of disturbances.

Inertia Matrix and Configuration

To perform a maneuver to restore the initial position of a satellite, it is essential to know the inertia matrix of the satellite and its three reaction wheels. The inertia matrix is a mathematical representation of the distribution of mass and rotational inertia of an object. It is used to calculate the torque required to change the angular velocity of the object.

The configuration of the reaction wheels is also important for attitude control. The reaction wheels are used to change the angular momentum of the satellite and maintain its desired orientation. The configuration of the reaction wheels can have a significant impact on the performance of the satellite. For example, a symmetric configuration can lead to a more stable and controllable system.

Maneuver to Restore Initial Position

To perform a maneuver to restore the initial position of a satellite using the quaternion feedback algorithm, the following steps can be followed:

- Sensing the angular velocity of the satellite using a gyroscope.
- Using the sensed angular velocity to calculate the error quaternion.
- Using the error quaternion and the gain matrix to calculate the required torque to correct the attitude of the satellite.
- Applying the required torque to the satellite using the reaction wheels.
- Monitoring the attitude of the satellite and repeating steps 1-4 until the satellite is in its initial position.

The key to a successful maneuver is to apply the required torque in the correct direction and magnitude. This can be achieved by using the gain matrix to adjust the sensitivity of the system to the error. A high gain can result in a more aggressive correction, while a low gain can result in a slower and less effective correction. The gain matrix is typically chosen based on the characteristics of the satellite and the desired performance of the system.

Once the required torque is calculated, it needs to be applied to the satellite using the reaction wheels. The reaction wheels are typically arranged in a triangular or tetrahedral configuration to provide three-axis control. The torque is applied by changing the rotational speed of the reaction wheels, which changes the angular momentum of the satellite.

During the maneuver, it is essential to monitor the attitude of the satellite and repeat the process until it is in its initial position. The monitoring can be done using sensors such as gyroscopes, magnetometers, and sun sensors. The sensors provide feedback to the quaternion feedback algorithm, which adjusts the torque required to correct the attitude of the satellite.

In some cases, it may be necessary to perform a series of maneuvers to restore the initial position of the satellite. For example, if the satellite experiences a large disturbance, it may require multiple maneuvers to correct its attitude.

5.1.1 Challenges and Limitations

The use of the quaternion feedback algorithm for satellite attitude control is not without its challenges and limitations. One of the main challenges is the accuracy of the sensors used to sense the angular velocity and orientation of the satellite. Any errors in the sensor measurements can lead to inaccurate calculations and suboptimal performance.

Another challenge is the complexity of the system. The quaternion feedback algorithm involves multiple components such as the gyroscope, gain matrix, and reaction wheels, which need to be carefully calibrated and configured to achieve optimal performance.

The main limitations of the system include in this case the availability of power. The reaction wheels require a significant amount of power to operate. As such, the system needs to be carefully designed to ensure that it can operate within the power constraints of the satellite.

At the end satellite attitude control is essential for maintaining the desired orbit and achieving the objectives of the mission. The quaternion feedback algorithm provides an effective way to sense the change in angular velocity of a satellite and perform a maneuver to restore its initial position. The use of the quaternion feedback algorithm for satellite attitude control is not without its challenges and limitations, but with careful calibration and configuration, it can provide a proper compromise for this study case.

5.2 Simulation

After a description of the algorithm chosen to implement the test its important to rum some mathematical simulation of the test that the project has as an objective. To do so, a simple MatLab™ implementation of the model and the algorithm should be sufficent. This is also necessary to set the proportional and derivative gains needed by looking at the responsiveness of the system. The main output of the model would be each element of the reference quaternion, the angular velocity of the satellite around its own axis and the output torque of each Reaction wheel of the Motor.

5.2.1 SimuLink™ Model

Fist of all its important to define the mathematical model of the algorithm and this can be easily implemented using the valuable software SimuLink™. in Fig. 5.1 is reported the high level block diagram of the algorithm. As can be noticed

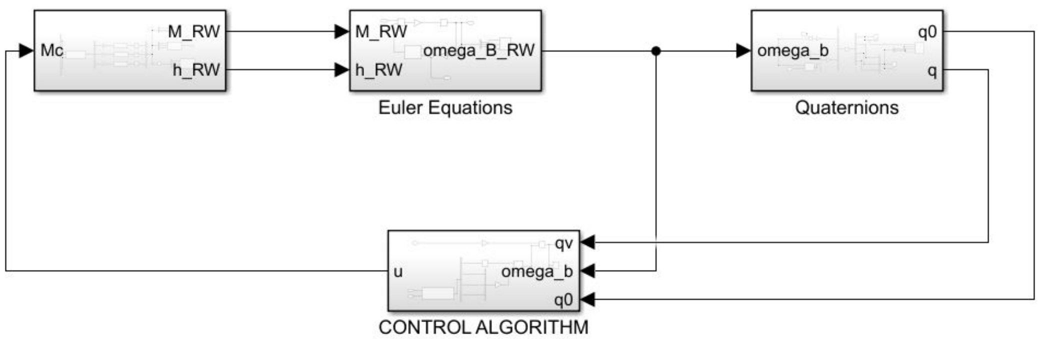


Figure 5.1: High Level QFA Block Diagram

the main block are just 4:

- Euler equation Block.
  This block is responsible for the implmentation of the Euler Equation to the model. As shown in Fig. 5.2 it takes in input the moment applied by the reaction wheels to the satellite and the angular momentum generated to calculate with a simple integrator the angular velocity of the satellite after the application of the control. As stated, the main output of this Block is the angular velocity of the satellite around its principal axis of inertia.

- Quaternion Block.
  This block take as only input the angular velocity of the satellite and determines

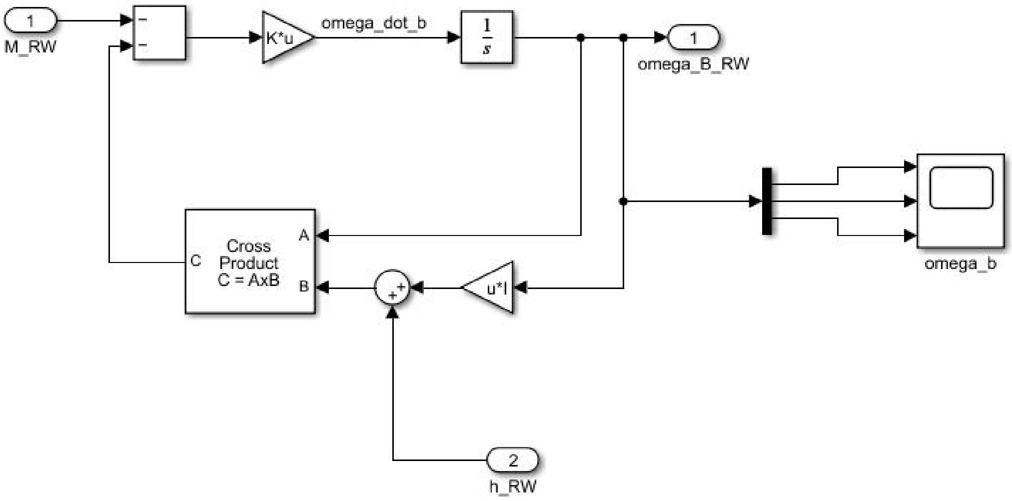


Figure 5.2: Detailed Euler Equation Block

the change of the quaternion following that specific angular velocity. Then Via a simple integrator the new quaternion after the rotation can be calculated and therefore the new asset of the overall satellite. All the operation needed are describe in Fig. 5.3. Again, the main output of this block is the quaternion

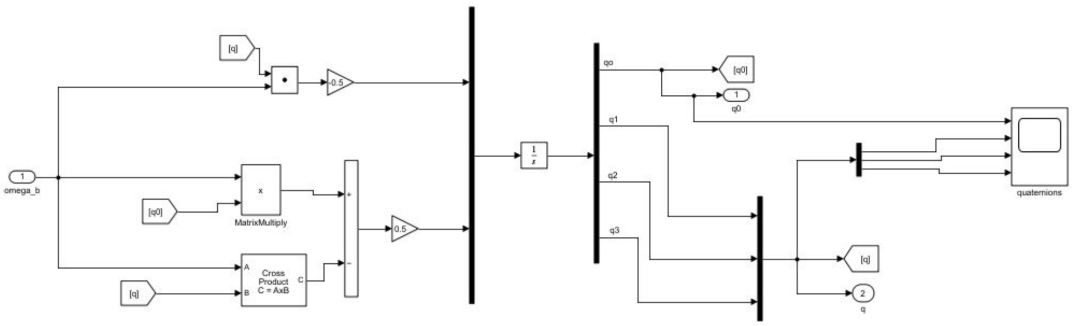


Figure 5.3: Detailed Quaternion Block

after the control is applied.

- Control Algorithm Block.
  In this block enter as input both the angular velocity of the satellite and the current quaternion coming from the previous blocks. The first thing is to calculate the quaternion error between the current quaternion and the

quaternion desired and on that basis calculate the required torque around the principal axis of inertia of the model to perform that rotation. In this block have to be implemented the proportional and derivative gains since the torque needed are calculated following the law:

$$M_c = -K_p q_e - K_d \omega_b; \tag{5.1}$$

A detailed description of this block is reported in Fig. 5.4.

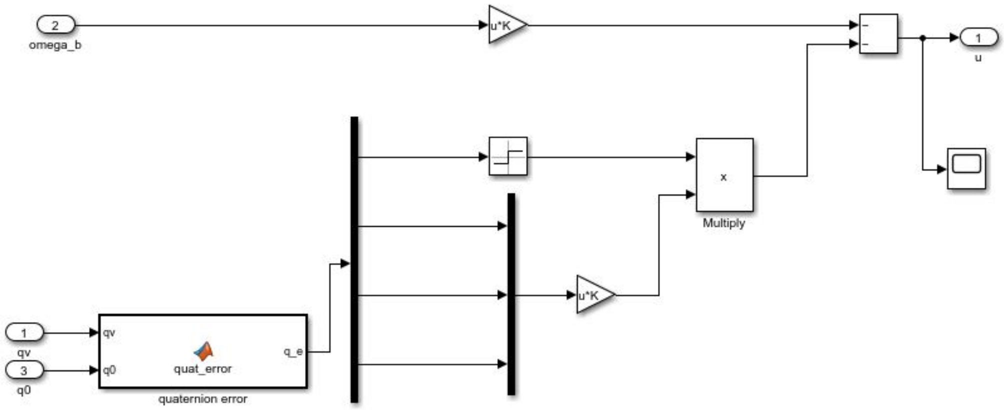


Figure 5.4: Detailed Control Block

- Reaction Wheel Block.
  The last highlighted block is the one that take into consideration the actual actuation of the Reaction Wheels. The first operation performed in this block is the one necessary to translate the control input from the Control Block to the moment that each Reaction Wheel has to output by performing a 'rotation' of the control vector following the configuration of the Reaction Wheels. Its here that the matrix Z calculated in the previous section of the chapter is needed. After that operation it performs some calculation based on the effective responsiveness of the motors to a step signal and saturate the output to the maximum torque of the Reaction Wheels. After those operation not only are given all the Moments required for each Reaction Wheel but the rotation matrix can be applied to return to a standard orthogonal configuration which will be taken as input for the Euler Equation Block in the cycle. A detailed description of this block is reported in Fig. 5.5.

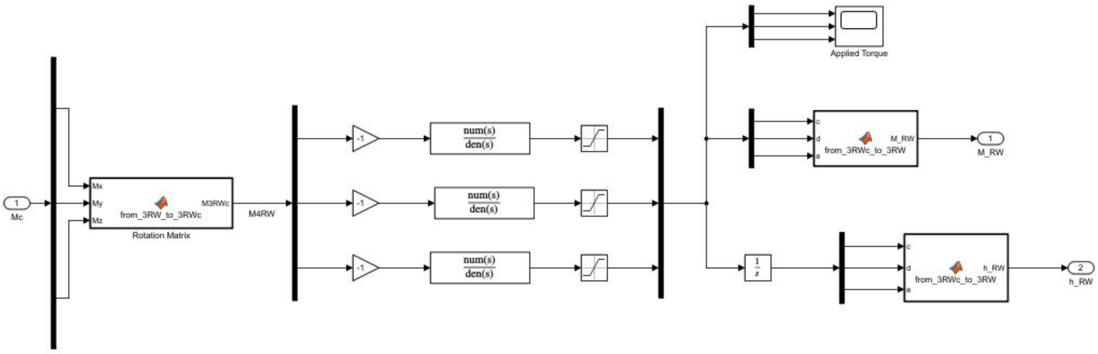


Figure 5.5: Detailed Reaction Wheels Block

5.2.2 Simulation Data

The Data needed for the simulation are of course the Inertia Matrix of the model which was reported before, but is also reported here for completion:

$$I = \begin{bmatrix} 48821.48 & -19.14 & -208.58 \\ -19.14 & 46576.31 & -78.48 \\ -208.58 & -78.48 & 50198.01 \end{bmatrix} [gmm^2]. \tag{5.2}$$

It is important to note that for the simulation all the measures used are compliant with the one used for the SI in order for the automated calculations to be correct. Also in all simulation the quaternion desired is always the fundamental quaternion:

$$q_d = \begin{bmatrix} 1 \\ 0 \\ 0 \\ 0 \end{bmatrix} \tag{5.3}$$

and the desired angular velocity is of course:

$$\omega_d = \begin{bmatrix} 0 \\ 0 \\ 0 \end{bmatrix} [rad/s] \tag{5.4}$$

since its necessary that the satellite will not rotate after performing the maneuver.. Also its important to set the saturation limit for each reaction wheel that could be taken from the datasheet of the motor chosen for the model and set as:

$$U_{max} = 0.17[mNm]. \tag{5.5}$$

Also the technology of the motor provide great responsiveness of those so a time constant of:

$$\tau_{RW} = 0.1 \tag{5.6}$$

is chosen for those simulations.

Its important now to set the use case of the Reaction wheels by defining the start condition of the satellite which will basically states the maneuver that the satellite will make under the Quaternion Feedback Algorithm.

For this analysis case the cases that are taken into consideration are:

- The reaction to an impulse given from an external disturbance that generate a change in the angular velocity of the satellite around its vertical axis. This could be achieved by setting the initial condition of the angular velocity of the satellite at an prefixed value. For this analysis case, and most importantly to set correctly the gains for the model an initial angular velocity of:

$$\omega_{0b} = \begin{bmatrix} 0 \\ 1 \\ 0 \end{bmatrix} [rad/s]. \tag{5.7}$$

  This angular velocity is way above every disturbance that could affect the satellite in Low Earth Orbit. but its opportune to overestimate it for setting the gains correctly and also considering the worst case scenario for the motors.

- A maneuver of a 60° rotation around the Y axis. This maneuver can be simulated by imposing the initial quaternion as:

$$q_0 = \begin{bmatrix} 0.8660 \\ 0 \\ 0.5 \\ 0 \end{bmatrix}. \tag{5.8}$$

  This simulation will provide crucial information on the responsiveness of the system performing a wide enough manouvre. Again the path angle is set to be way higher that any manouver that could be needed but, again it is necessary to analyze if the system would be stable under a worst case scenario.

5.2.3 Simulation Results

Hereafter are reported the results of simulation and the transient for the model to the two cases reported before. This is the completion of an iterative procedure to set the value of the proportional and derivative gains in order to achieve a good responsiveness of the system to the algorithm and stabilize it. At the end the value set for those gains are:

$$K_d = 6.72 * 10^{-6}; \tag{5.9}$$

$$K_p = 0.336 * 10^{-4}. \tag{5.10}$$

- CASE 1.

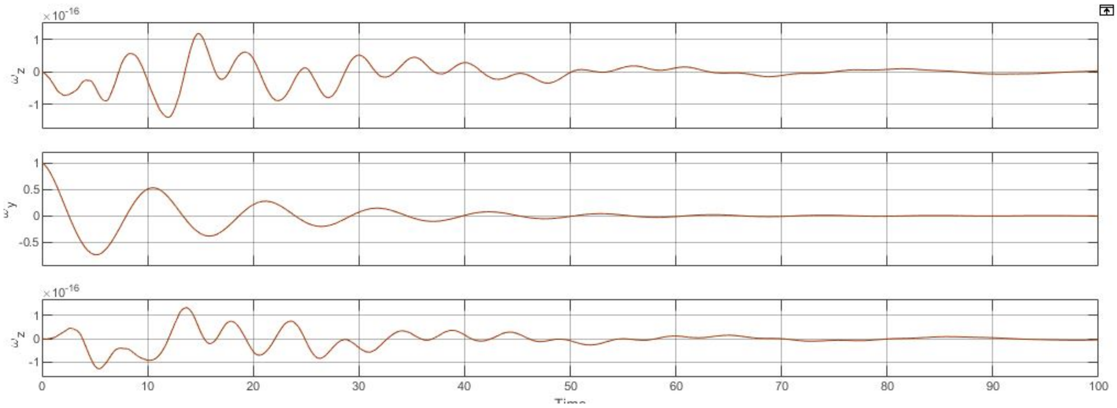


Figure 5.6: Angular Velocity of the model around its Principal axys of Inertia (CASE 1)

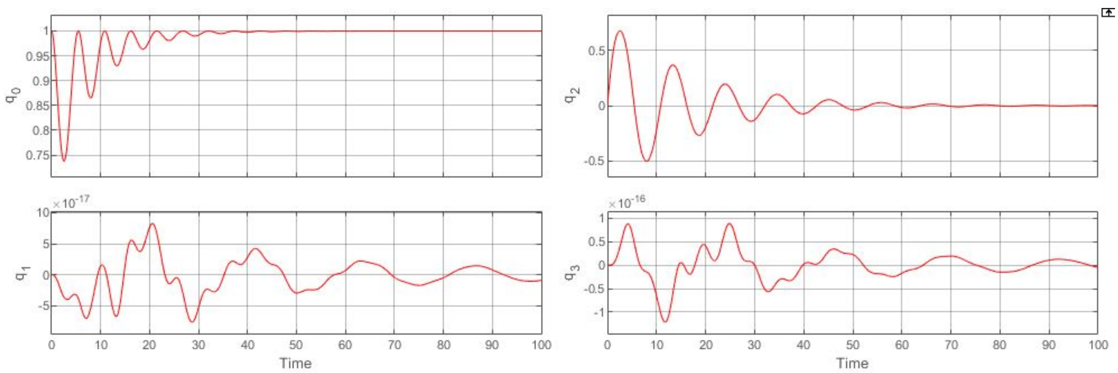


Figure 5.7: Quaternion Transient of the Model (CASE 1)

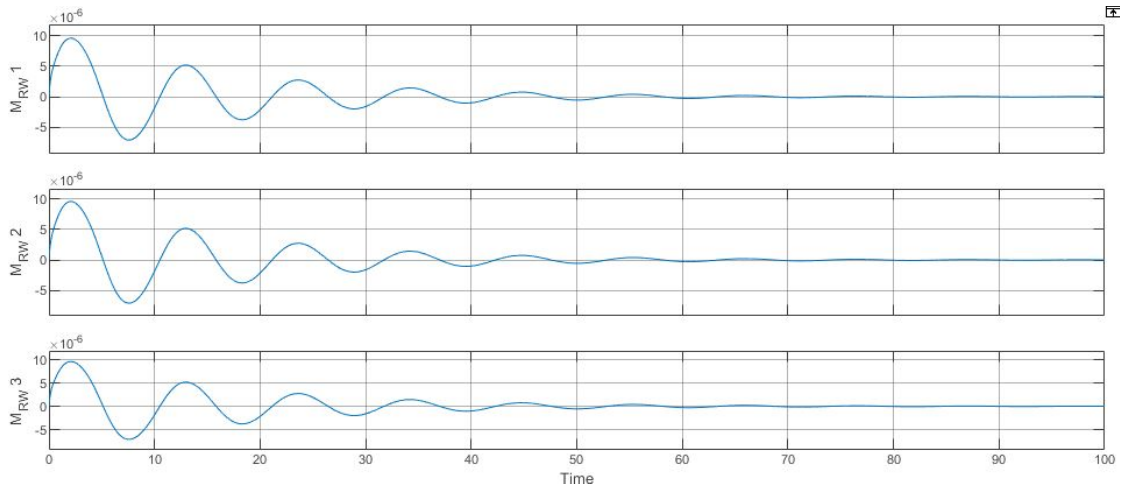


Figure 5.8: Quaternion Transient of the Model (CASE 1)

As shown, those results highlight a great responsiveness of the system that manage to activate the Reaction wheels in contrast of an external disturbance. The transient is not very long and, however the graph around the x and z axis seems very unstable, the order of magnitude of those vibration are much inferior to the overall movement of the satellite, therefore those results are very promising. Also it is important to note that the maximum torque required to the reaction wheels to perform such maneuver is very low and around two order of magnitude inferior to the maximum output torque of each one of them.

- CASE 2:
  As shown in those results the system is very stable and reaches is desired position after only 100 seconds which is very important. Again the transient is not very long and however the graph around the x and z axis seems very unstable the order of magnitude of those vibration are much inferior to the overall movement of the satellite, therfore those results are very promising. Again it is important to note that the maximum torque required to the reaction wheels is again around two order of magnitude inferior to the maximum output torque of each one of them.

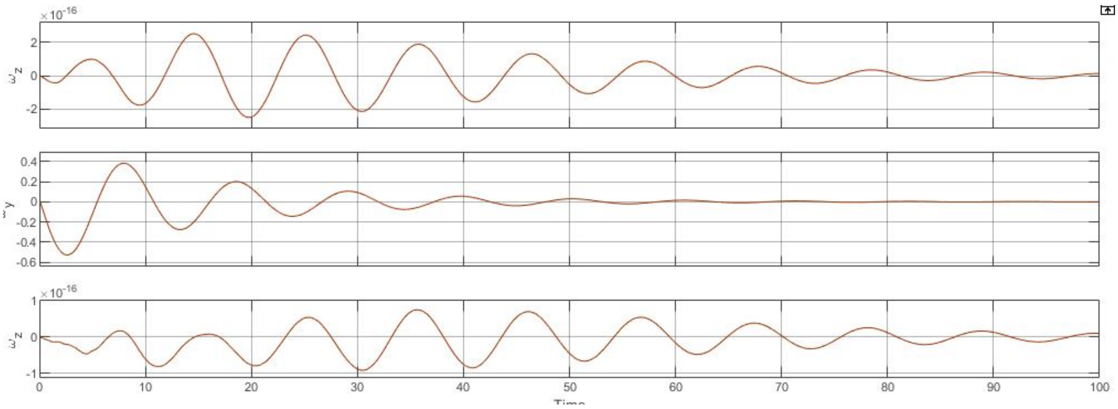


Figure 5.9: Angular Velocity of the model around its Principal axys of Inertia (CASE 2)

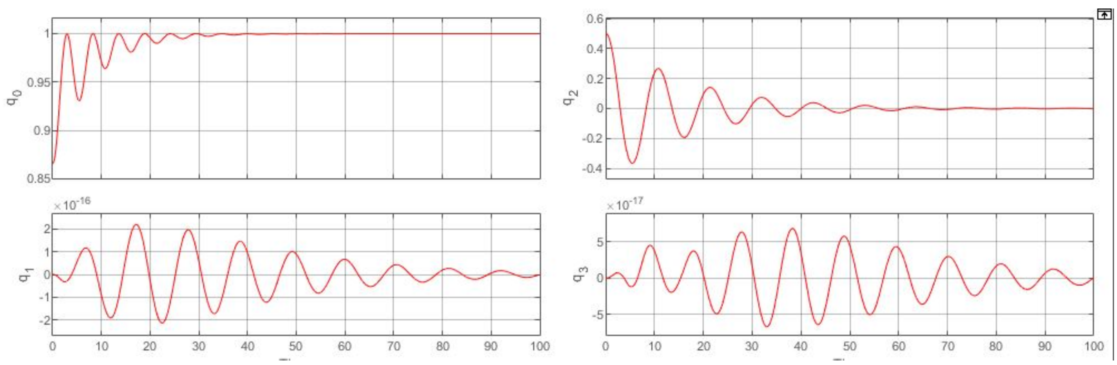


Figure 5.10: Quaternion Transient of the Model (CASE 2)

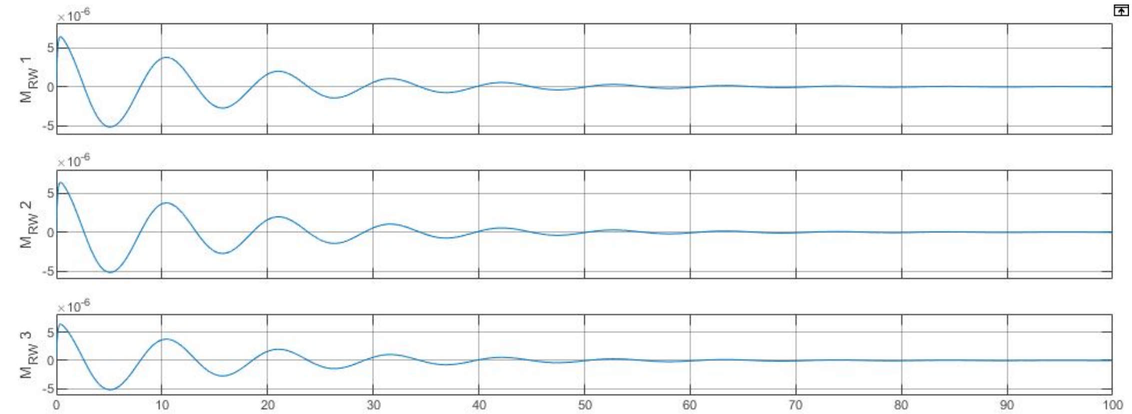


Figure 5.11: Quaternion Transient of the Model (CASE 2)

Those results clearly states that a maneuver and in general the control of the asset of such a small satellite is possible and can be done using miniaturized components.

5.3 Experimental Test

After the promising results set by the SimuLink^{TM} model the project can be put together and all the components are integrated and connected using some thin cables to avoid excessive volume usage. After some test and troubleshooting the final result is reported in Fig. 5.12 and Fig. 5.13.

Figure 5.12: Display of Final Integration

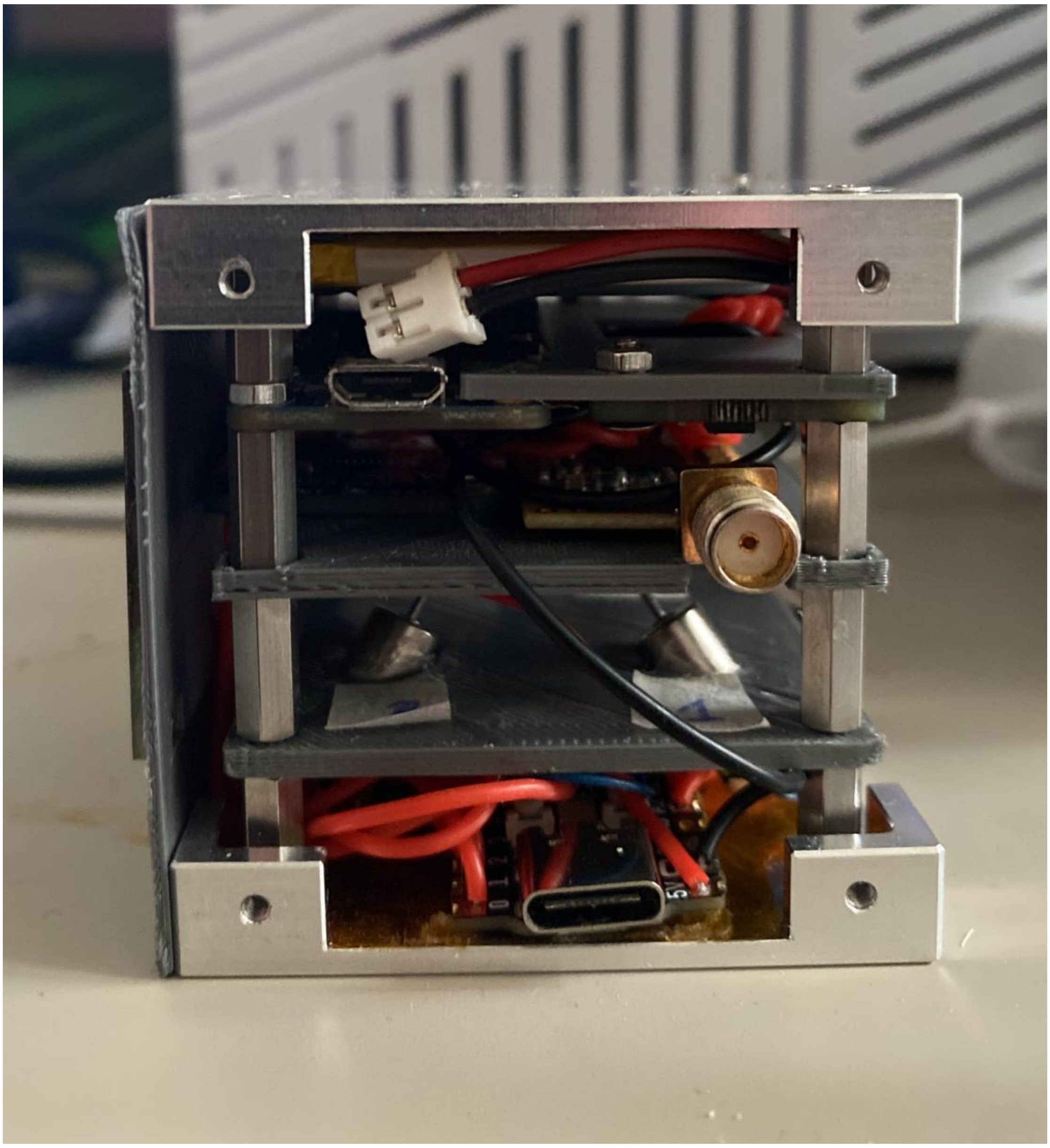


Figure 5.13: Display of Final Integration - User Interface

5.3.1 Python Language

The first issue was that of course the microcontroller installed in the model was not able to read a SimuLink^{TM} model and to be able to connect with other electronic components another programming language shoud be used. The choice happened to be CircuitPython. CircuitPython is a programming language designed to simplify experimenting and learning to code on low-cost microcontroller boards such as the PIMORONI^{TM} Tiny 2040. It's based on Python which is one of the fastest growing programming languages. With CircuitPython, there are no upfront desktop downloads needed. Once you get your board set up, open any text editor, and start editing the main code. It makes getting started easier than ever with no upfront desktop downloads needed.

Advantages:

- CircuitPython is easy to learn and use, especially for beginners.
- It supports modules and packages which means it's easy to reuse your code for other projects.
- CircuitPython makes it easier to get started with microcontroller programming because it doesn't require any upfront desktop downloads.

Disadvantages:

- CircuitPython may not be as performant as other programming languages like C++.
- Some drivers such as display drivers are written in more low-level languages which means they are direct hardware drivers.

Given the nature of the language chosen some proper drivers for each component should be installed to allow the microcontroller to communicate with each one and being able to read the main data to perform the algorithm.

5.3.2 Test Bench

For the very experimental nature of the test the manouver will be tested only on the vertical axis by suspending the test model on a string attached to the top of it via the screw holes designed to make the standoffs attached to the main baseplate. This could lead to some problems but they can be resolved via software. For example one relevant problem was the reading of angular velocities around the x and z axys which would never be compensated. This problem can be resolved by setting the reads of the gyroscope around those axys as 0 simulating a perfectly flat movement and how the satellite would react to it.

5.3.3 Python Code

The next step has been a very hard one since all the procedures implemented in MatLab and SimulLink had to be translated ion Circuitpython that of course doesnt have all the numerical functions previously used (i.e. integrators, calculation of a matrix inverse, etc...) but a library that simulate some of those can be installed and its called ulab. This library allows the processor to perform the calculation needed for the algorithm to work.

The control algorithm is set to activate as soon as a quaternion different from the starting one is calculated and therefore a quaternion error different from the value:

$$q_e = \begin{bmatrix} 1 \\ 0 \\ 0 \\ 0 \end{bmatrix} \tag{5.11}$$

is calculated. This is don by constantly reading the angular velocity from the gyroscope and calculating the resulting quaternion from it. of course the tolerance is set on the norm of the quaternion error to trigger the control algorithm. Also to avoid the build up of the error calculated if the algorithm does not trigger the code is set to set the position as neutral at the start of each angular velocity reading.

The final product can be divided in two main section:

- Device settings and Parameters Definition.

content/QFA_PQ.py

```
import os
from ulab import numpy as np
from ulab import scipy as sp
import time
import board
import adafruit_icm20x # Motion Sensor
import adafruit_bme680 # Air Quality Sensor
import adafruit_ina260 # Power Sensor
import busio
import pwmio
import math

os.mkdir('/gyro_data')

i2c = busio.I2C(board.GP7, board.GP6)
icm = adafruit_icm20x.ICM20649(i2c)
bme680 = adafruit_bme680.Adafruit_BME680_I2C(i2c , address = 0x76
    )
ina260 = adafruit_ina260.INA260(i2c)

```

```
pwm_1 = pwmio.PWMOut(board.GP0)
pwm_2 = pwmio.PWMOut(board.GP1)
pwm_3 = pwmio.PWMOut(board.GP2)
pwm_1r = pwmio.PWMOut(board.GP28)
pwm_2r = pwmio.PWMOut(board.GP27)
pwm_3r = pwmio.PWMOut(board.GP26)


# Quaternion inverse
def quat_inverse(q):
    w, x, y, z = q
    magnitude = np.sqrt(w**2 + x**2 + y**2 + z**2)
    return np.array([w/magnitude, -x/magnitude, -y/magnitude, -z/magnitude])

# Quaternion product function
def quat_product(q1, q2):
    w1, x1, y1, z1 = q1
    w2, x2, y2, z2 = q2
    w = w1 * w2 - x1 * x2 - y1 * y2 - z1 * z2
    x = w1 * x2 + x1 * w2 + y1 * z2 - z1 * y2
    y = w1 * y2 + y1 * w2 + z1 * x2 - x1 * z2
    z = w1 * z2 + z1 * w2 + x1 * y2 - y1 * x2
    return np.array([w, x, y, z])

def transpose(vector):
    # Determine the shape of the input vector
    input_shape = vector.shape

    # Create an output vector with the transposed shape
    output_vector = np.zeros((input_shape[0], 1))

    # Transpose the input vector by assigning its values to the output vector in a transposed order
    for i in range(input_shape[0]):
        output_vector[i, 0] = vector[i]

    return output_vector


def deg_per_sec_to_rpm(deg_per_sec):
    deg_per_min = deg_per_sec * 60  # convert degrees per second to degrees per minute
    rounds_per_min = deg_per_min / 360  # convert degrees per minute to rounds per minute
    return rounds_per_min

# Define the spacecraft parameters

```

```
I = np.array([[ 49205.16/(10^9), 0, 0], [0, 47018.60/(10^9), 0], [0, 0, 50657.2/(10^9)]]) # overall inertia matrix
Z = np.array([[ -0.499945823958681, 0, 0.499945823958681],
              [ 0.816540811885746, 0.816540811885746, 0.816540811885746],
              [-0.288643856042774, 0.577287712085548, -0.288643856042774]]) # reaction wheel configuration
J_rw = 0.0000372335 # reaction wheels inertia (motor)
I_inv = np.linalg.inv(I)
pi_Z = np.linalg.inv(Z)

tau_RW = 0.08
U_max_RW = 0.17 # TBD
w_max = 2000 # rpm TBD
tol = 0.1 # Set the minimum angular velocity necessary to trigger the reèosition manouver
dt=0.1

# Define the initial conditions
q_des = np.array([1, 0, 0, 0]) # initial quaternion and quaternion desired
q = q_des
w = np.array([0, 0, 0]) # initial angular velocity and angular velocity desired
qd = quat_inverse(q)
qe = quat_product(qd, q)

# Define the control gains
kp = np.array([ 0.672 , 0.745 , 0.672]) # proportional gain
kd = np.array([ 3.36 , 3.32 , 3.36]) # derivative gain

# Read Gyro data

gx, gy, gz = icm.gyro
omega_b = np.array([ -deg_per_sec_to_rpm(gy), deg_per_sec_to_rpm(gz), -deg_per_sec_to_rpm(gx)]) # convert from dps to rpm and refer the read to the coordinate system of the satellite
print(omega_b)
t_real = time.time()

with open('/gyro_data/gyro_log.txt', 'w') as f:
```

In this section of the code the main libraries and function needed in the control algorithm are defined along with the channels of communication and output to regulate the voltage of the motors via a PWM signal. Also all the variables of the system calculated in the previous sections are defined and a first read of the data from the gyroscope is done.

- Main Cycle The Main Cycle is set to function as the Block Diagram reported before on SimuLink™ but for the nature of the test has some more implementation in it. First of all the control algorithm is set to start only after a quaternion error is calculated from the reading of the angular velocity and only if it exceed a determined tolerance bis set to start. Therefore a preliminary if is implemented making a check on the quaternion error. The other implementation is the simulation in the control algorithm of the control of the Reaction Wheels since Circuitpython is not able to perform the transient analysis of the response of a system to a step signal. therefore The signal is simplified to be set as as direct step, which is a quite strong assumption but could still provide some valuable results. The Main Cycle code is reported hereafter in Fig. **??**

content/QFA_PQ.py

```

# Main Cycle   q[-3:]

while True:

    if abs(qe[0] - 1) > tol or abs(qe[1]) > tol or abs(qe[2]) > tol or abs(qe[3]) > tol:

        #####################
        # Quaternion Block #
        #####################

        # Calculate quaternion

        pop = -0.5 * np.dot(omega_b , q[-3:])
        trash4 = np.cross(omega_b, q[-3:])
        pop1 = 0.5 * (q[0] * omega_b - trash4)
        trash1 = pop1[0]
        trash2 = pop1[1]
        trash3 = pop1[2]
        q_dot = np.array([pop, trash1, trash2, trash3])


        # Integrate the Quaternion
        q = q + (q_dot * dt)

        # Normalize the Quaternion
        q = q / np.linalg.norm(q)

        # Calculate the Quaternion Error

        qe =  quat_product(qd, q)

```

```
            ##################
            # Control Block #
            ##################

            M_xyz = -(qe[0]/abs(qe[0])) * qe[-3:] * kp - kd * omega_b


            M_RW = -1*np.dot(pi_Z, transpose(M_xyz))

#           # Define the numerator and denominator coefficients for the Transfer Function
#           num = [-1]
#           den = [0.08, 1]
#
#           # Create a transfer function object
#           sys = sp.lti(num, den)
#
#           # Define the time range for the simulation
#           t = np.linspace(t_prev,t_real)
#
#           # Define the step input signal
#           u1 = np.ones(len(t)) * M_RW[1]
#           u3 = np.ones(len(t)) * M_RW[2]
#           u3 = np.ones(len(t)) * M_RW[3]
#
#           # Simulate the response of the system to the step input
#           t_trash, M_RW1_model, x_trash = sp.step(sys, T=t, u1)
#           t_trash, M_RW2_model, x_trash = sp.step(sys, T=t, u2)
#           t_trash, M_RW3_model, x_trash = sp.step(sys, T=t, u3)
#
#           M_RW1_model = M_RW1_model[-1]
#           M_RW2_model = M_RW2_model[-1]
#           M_RW3_model = M_RW3_model[-1]

            #Saturate to maximum/minimum output
            M_RW[0]=np.clip(M_RW[0], -U_max_RW, U_max_RW)
            M_RW[1]=np.clip(M_RW[1], -U_max_RW, U_max_RW)
            M_RW[2]=np.clip(M_RW[2], -U_max_RW, U_max_RW)

            # Integrate to find angular velocity of each RW

            w_RW1 = (M_RW[0]/J_rw)*(dt)
            w_RW2 = (M_RW[1]/J_rw)*(dt)
            w_RW3 = (M_RW[2]/J_rw)*(dt)
            print(w_RW1)
            print(w_RW2)
            print(w_RW3)
```

```

            h_RW = Z * np.array([w_RW1, w_RW2, w_RW3])

            M_xyz = Z * M_RW

            # Set the PWM from microcontroller to perform the RW speed based on set curve
            w_pct1 = w_RW1 / w_max
            w_pct2 = w_RW2 / w_max
            w_pct3 = w_RW3 / w_max
            w_pct1= np.clip(w_pct1, -1, 1)
            w_pct2= np.clip(w_pct2, -1, 1)
            w_pct3= np.clip(w_pct3, -1, 1)
            print(w_pct1[0])
            print(w_pct2[0])
            print(w_pct3[0])

            # Set RW 1

            if w_pct1[0] > 0:

                pwm_1r.duty_cycle = 0
                pwm_1.duty_cycle = math.ceil(w_pct1[0]*65535)

            elif w_pct1[0] < 0:

                pwm_1.duty_cycle = 0
                pwm_1r.duty_cycle = math.ceil(-w_pct1[0]*65535)

            else:

                pwm_1.duty_cycle = 0
                pwm_1r.duty_cycle = 0

            # Set RW 2

            if w_pct2[0] > 0:

                pwm_2r.duty_cycle = 0
                pwm_2.duty_cycle = math.ceil(w_pct2[0]*65535)

            elif w_pct2[0] < 0:

                pwm_2.duty_cycle = 0
                pwm_2r.duty_cycle = math.ceil(-w_pct2[0]*65535)

            else:

                pwm_2.duty_cycle = 0
```

```
                    pwm_2r.duty_cycle = 0

                # Set RW 3

                if w_pct3[0] > 0:

                    pwm_3r.duty_cycle = 0
                    pwm_3.duty_cycle = math.ceil(w_pct3[0]*65535)

                elif w_pct3[0] < 0:

                    pwm_3.duty_cycle = 0
                    pwm_3r.duty_cycle = math.ceil(-w_pct3[0]*65535)

                else:

                    pwm_3.duty_cycle = 0
                    pwm_3r.duty_cycle = 0

                ##########################
                # Euler Equation Block #
                ##########################

                # Integrate Euler equation
                omega_b_anal = (I_inv * (-M_xyz - np.dot(omega_b,I*
    omega_b+h_RW)))

                gx, gy, gz = icm.gyro
                omega_b = np.array([ 0, deg_per_sec_to_rpm(gz), 0]) #
     convert from dps to rpm and refer the read to the coordinate
    system of the satellite
                print(omega_b)
                f.write(f'{omega_b[1]}\n')
                time.sleep(dt)
                # Update time

    #           t_prev = t_real
    #           t_real = time.time()
    #           dt = t_real - t_prev

            else :

            ################################################
            # Continue Reading Angular Velocity from ICM #
            ################################################
                pwm_1.duty_cycle = 0
                pwm_1r.duty_cycle = 0
                pwm_2.duty_cycle = 0
                pwm_2r.duty_cycle = 0
```

```
            pwm_3.duty_cycle = 0
            pwm_3r.duty_cycle = 0
            gx, gy, gz = icm.gyro
            omega_b = np.array([ 0, deg_per_sec_to_rpm(gz), 0]) # convert from dps to rpm and refer the read to the coordinate system of the satellite
            print(omega_b)
            f.write(f'{omega_b[1]}\n')

            ####################
            # Quaternion Block #
            ####################

            # Calculate quaternion

            pop = -0.5 * np.dot(omega_b , q[-3:])
            trash4 = np.cross(omega_b, q[-3:])
            pop1 = 0.5 * (q[0] * omega_b - trash4)
            trash1 = pop1[0]
            trash2 = pop1[1]
            trash3 = pop1[2]
            q_dot = np.array([pop, trash1, trash2, trash3])


            # Integrate the Quaternion
            q = q + (q_dot * dt)

            # Normalize the Quaternion
            q = q / np.linalg.norm(q)

            qe =  quat_product(qd, q)

            time.sleep(dt)
```

5.3.4 Failure of Test

Unfortunately during the last part of the tests just a couple of weeks from the end of the project the model has started to have some problems. First of all, the power line of the motors which was initially run trough the microcontroller trough 3 V stopped working. This has probably happened because that line on the microcontroiller was only rated for a maximum of 300 mA which is also the maximum rated current for the motors from datasheet. The problem occurred probably because all three H-Bridges were connected in series and were all requesting current from the same power lane, that lead to an overdrive of that lane on the microcontroller that stopped working. The fix was quite simple tough since changing the power lane that provides current to the motors to come directly from the battery board which could provide up to 1A should solve the issue. After doing the opportune soldering and separating the power lane from the PIMORONI™ Tiny 2040 also a software fix was needed since the power provided to the motors was at 5 V for which them are not rated for so the pwm signal from the microcontroller is set to be multiplied for a factor of 0.6 accounting for that change on the hardware and providing the same voltage as the previous tests. So a quick test with the motors spinning at a duty cycle of 0.5 is performed to see if everything was fine before closing the satellite and starting the tests on the test bench again. After closing everything up the satellite is put back on the string and the after it has stabilized in a neutral position the microcorntroller is rebooted to initiate the algorithm again. At this moment after a couple of runs in which the motors did not seems to generate enough torque to stop the satellite from its rotation the test was stopped and corrections on the proportional and derivative gains were made. After the satellite stabilized again the controller was again rebooted and a light spin was given to the satellite to simulate an impulsive external disturbance. Unfortunately an error on the code made all three motors start to rump up at their maximum speed and something started smoking from the bottom of the satellite. The test was immediately shut down and the battery disconnected. Once the model was disassembled it could be noticed that the H-Bridges were still really hot and the smell of smoking was coming from them and was clear that the model would have not worked anymore. Again the problem resulted to be the current limit of the components. Each H-Bridge was rated to support a maximum current of 800 mA, but since were again all connected in series all the current requested from all three motors operating at their maxim um rating current of 300 mA may have caused an overdrive of the components and destroying them. It has been tried in the last weeks of the project to buy some different components that could be delivered on time but all has ad some size or hardware incompatibility with the system which was designed to use those specific ones. This bad end of the project does not indeed mean a failure of the overall study and further more implementation can be made to improve it and make it work in the future.

Chapter 6

Conclusions

In this final chapter will be discussed some overall conclusion of all the study some future implementation of the project along with some future implementation of the project.

6.1 Conclusions

The main aim of this thesis is set to give an general overview on what is the current state of the art of miniaturized satellite technologies for each one of the main subsystems typically present on a spaceecraft highlighting what are they main advantages and also the main technological and physical limits. As shown the state of the art is more concentrated on the CubeSat standard but as shown with the software test results run trough this project a miniaturization of some of those is certainly possible. It is important to note that as has been highlighted those miniaturized satellite are meanly meant to operate in large constellation and so they may have not need all the components usually displayed in more standard sized satellite. For example one of the main promising concept of using those satellite is to deploy multiple of them from a single mother-ship, therefore the communication may only be necessary from each satellite to the main spacecraft and the ground communication could be performed by the deployer saving much power while also providing a reliable line of communication. such mission can be represented by the BEESat mission from the TU of Berlin and the Andesite mission from the Boston University and their concept are reported in Fig. 6.1 and Fig. 6.2.

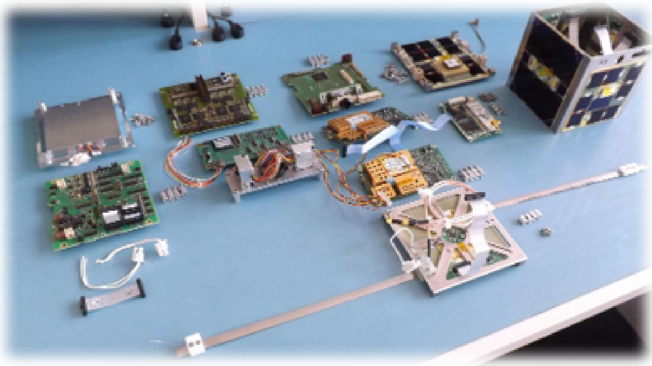


Figure 6.1: BEESat Concept (TU of Berlin)

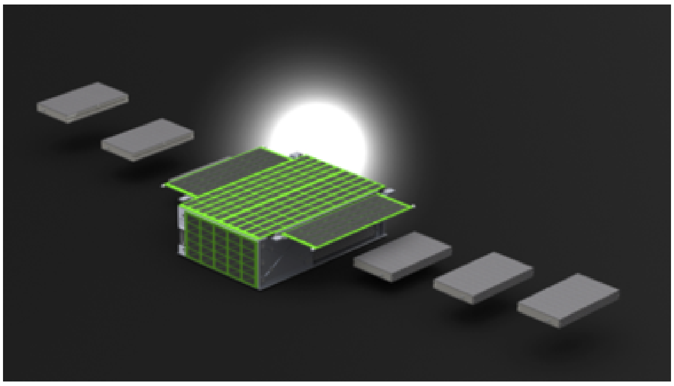


Figure 6.2: Andesite Mission Concept (Boston University)

Aside also from it pure scientific use case this project was a clear example of what a powerful didactical tool those satellites could be providing a real world application of the concepts maturated during the years of study. Their cheap and simple nature is very valuable for student projects with a limited budget while also providing many new skills that can be fruited such as programming in different language, hardware and electronics manipulation and even product design lokking at all the processes that the model has to go trough. Overall, despite the failure of the physical test and the burnout of the model, this study was able to provide some important data to the study of the limits of the miniaturization of satellites and in particular of the Reaction Wheels in hope to gauge more and more interest on this field that its on my advise a crucial aspect for the sustainability of the space economy and industry along with the research and teaching in an Academic surrounding.

6.2 Future Implementations

here ar listed some of the future implementation of the overall project that could provide some valuable data on the overall study and can surely help the research on this field of miniaturized satellites:

- First of all speaking from a software test perspective some more test can be performed on other different levels to achieve more data on the current configuration of the model, such as maneuvers and disturbance contrasts also along the other two axis.

- Another important simulation that for time restriction was not possible to study is a dethumbling maneuver of the satellite using the model designed. To perform that simulation a B-dot algorithm written ad-hoc for the model needs to be implemented. This could provide critical data on the performance that can be obtained from a miniaturized system like this in a fundamental phase of the mission of a deployed satellite.

- Another point brought to the table could be represented by an analysis of a different configuration of the Reaction Wheels that could be more performant and also easier to control. However the volume limitation were very stict in this implementation in the future with a better integration of the components it could be possible.

- In this model all the electronics component were soldered together using live wires and integrated using custom made 3D Printed baseplates. In a future iteration it will be necessary to avoid wiring and short problems, which were very common given the inexperience on the field, to custom design the circuit boards. This will also help saving much volume otherwise occupied by all the wires and baseblates.

- This last implementation could also allow the use of much smaller standoffs saving even much space and allowing the integration of other components such as a camera or other batteries to provide a backup power storage and alternative power lane. that would provide a much reliable an robust model closer to a real life satellite and not representing only a simplified test model.

Bibliography

[1] Halis C Polat, Josep Virgili-Llop, and Marcello Romano. «Survey, statistical analysis and classification of launched CubeSat missions with emphasis on the attitude control method». In: *Journal of small satellites* 5.3 (2016), pp. 513–530 (cit. on p. 1).

[2] Giovanni Minelli et al. «O/OREOS nanosatellite: A multi-payload technology demonstration». In: (2010) (cit. on p. 1).

[3] Maria Antonietta Viscio, Nicole Viola, Sabrina Corpino, Fabrizio Stesina, Silvano Fineschi, Federico Fumenti, and Christian Circi. «Interplanetary CubeSats system for space weather evaluations and technology demonstration». In: *Acta Astronautica* 104.2 (2014). Future of Space Exploration: Towards the Stars, pp. 516–525. ISSN: 0094-5765 (cit. on p. 1).

[4] Oliver Montenbruck, Eberhard Gill, and Fh Lutze. «Satellite orbits: models, methods, and applications». In: *Appl. Mech. Rev.* 55.2 (2002), B27–B28 (cit. on p. 1).

[5] Chantal Cappelletti, Simone Battistini, and Benjamin Malphrus. *Cubesat handbook: From mission design to operations.* Academic Press, 2020 (cit. on p. 2).

[6] Eberhard Gill, Prem Sundaramoorthy, Jasper Bouwmeester, B Zandbergen, and R Reinhard. «Formation flying within a constellation of nano-satellites: The QB50 mission». In: *Acta Astronautica* 82.1 (2013), pp. 110–117 (cit. on p. 5).

[7] W Edmonson, S Gebreyohannes, A Dillion, Radhika Radhakrishnan, Jules Chenou, A Esterline, and Fatemeh Afghah. «Systems engineering of inter-satellite communications for distributed systems of small satellites». In: *2015 Annual IEEE Systems Conference (SysCon) Proceedings.* IEEE. 2015, pp. 705–710 (cit. on p. 6).

[8] Vasileios Leon, George Lentaris, Dimitrios Soudris, Simon Vellas, and Mathieu Bernou. «Towards Employing FPGA and ASIP Acceleration to Enable Onboard AI/ML in Space Applications». In: *2022 IFIP/IEEE 30th International Conference on Very Large Scale Integration (VLSI-SoC)*. IEEE. 2022, pp. 1–4 (cit. on p. 7).

[9] Heinz Stoewer and Marcello Romano. «Business e sostenibilità nel futuro dello Spazio». In: Politecnico di Torino - Biennale Tecnologia, 2022 (cit. on p. 10).

[10] Shen-En Qian. «Hyperspectral satellites, evolution, and development history». In: *IEEE Journal of Selected Topics in Applied Earth Observations and Remote Sensing* 14 (2021), pp. 7032–7056 (cit. on p. 13).

[11] J Hong, S Romaine, B Ramsey, L Nittler, K Gendreau, Daniele SPIGA, M Elvis, and J Grindlay. «Miniature Lightweight X-Ray Optics (MiXO) and CubeSat X-Ray Telescope (CubeX) for Solar System Exploration». In: (2017) (cit. on p. 14).

[12] DJ McComas, SJ Bame, BL Barraclough, JR Donart, RC Elphic, JT Gosling, MB Moldwin, KR Moore, and MF Thomsen. «Magnetospheric plasma analyzer: Initial three-spacecraft observations from geosynchronous orbit». In: *Journal of Geophysical Research: Space Physics* 98.A8 (1993), pp. 13453–13465 (cit. on p. 15).

[13] Loris Franchi, Lorenzo Feruglio, Raffaele Mozzillo, and Sabrina Corpino. «Model predictive and reallocation problem for CubeSat fault recovery and attitude control». In: *Mechanical Systems and Signal Processing* 98 (2018), pp. 1034–1055 (cit. on p. 17).

[14] JF DeNatale, RL Borwick, C Tsai, PA Stupar, Y Lin, RA Newgard, RW Berquist, and M Zhu. «Compact, low-power chip-scale atomic clock». In: *2008 IEEE/ION Position, Location and Navigation Symposium*. IEEE. 2008, pp. 67–70 (cit. on p. 21).

[15] James S Bennett, Brian E Vyhnalek, Hamish Greenall, Elizabeth M Bridge, Fernando Gotardo, Stefan Forstner, Glen I Harris, Félix A Miranda, and Warwick P Bowen. «Precision magnetometers for aerospace applications: A review». In: *Sensors* 21.16 (2021), p. 5568 (cit. on p. 22).

[16] Guglielmo M Tino et al. «Atom interferometers and optical atomic clocks: New quantum sensors for fundamental physics experiments in space». In: *Nuclear Physics B (Proceedings Supplements)* 166 (2007), pp. 159–165 (cit. on p. 23).

[17] Roberto Sorrentino, Luca Pelliccia, and Simone Bastioli. «Recent progress in miniaturized and reconfigurable filters for advanced communications and space applications». In: *2011 MICROWAVES, RADAR AND REMOTE SENSING SYMPOSIUM*. IEEE. 2011, pp. 20–23 (cit. on p. 25).

[18] Ryan W Kingsbury. «Optical communications for small satellites». PhD thesis. Massachusetts Institute of Technology, 2015 (cit. on p. 28).

[19] Kjell Anflo and R Möllerberg. «Flight demonstration of new thruster and green propellant technology on the PRISMA satellite». In: *Acta Astronautica* 65.9-10 (2009), pp. 1238–1249 (cit. on p. 30).

[20] HR Shea. «MEMS for pico-to micro-satellites». In: *Moems And Miniaturized Systems Viii*. Vol. 7208. SPIE. 2009, pp. 208–215 (cit. on p. 31).

[21] George Lewis Teel. «Development and Characterization of the Heated-Anode Cathode Arc Thruster (HA-CAT)». PhD thesis. The George Washington University, 2014 (cit. on p. 32).

[22] Douglas Spence, Eric Ehrbar, Nate Rosenblad, Nate Demmons, Tom Roy, Samuel Hoffman, Dan Williams, Vlad Hruby, and Chris Tocci. «Electrospray propulsion systems for small satellites». In: (2013) (cit. on p. 33).

[23] K Janzer, M Killian, M Langer, D Krejci, and A Reissner. «Thermal Control of Higher Power Applications on CubeSats». In: *Proceedings of the 69th International Astronautical Congress*. 2018 (cit. on p. 47).

[24] Rachael E Enfield, Janam K Pandya, Jiakai Lu, David Julian McClements, and Amanda J Kinchla. «The future of 3D food printing: Opportunities for space applications». In: *Critical Reviews in Food Science and Nutrition* (2022), pp. 1–14 (cit. on p. 51).

[25] Brett Streetman, Joseph Shoer, and Leena Singh. «Limitations of scaling momentum control strategies to small spacecraft». In: *2017 IEEE Aerospace Conference*. IEEE. 2017, pp. 1–7 (cit. on p. 54).

[26] Heinz Stoewer, Ralf Hartmann, and LAJ Baron von Richter. «5.4. 2 An Advanced Methodology for the Design Process of a Satellite». In: *INCOSE International Symposium*. Vol. 10. 1. Wiley Online Library. 2000, pp. 505–512 (cit. on p. 59).

[27] Simone Chesi, Octavio Perez, and Marcello Romano. «A dynamic, hardware-in-the-loop, three-axis simulator of spacecraft attitude maneuvering with nanosatellite dimensions». In: *Journal of Small Satellites* 4.1 (2015), pp. 315–328 (cit. on p. 59).