CubeSat GNC Utilizing Photodiodes

a project presented to
The Faculty of the Department of Aerospace Engineering
San José State University

in partial fulfillment of the requirements for the degree
Master of Science in Aerospace Engineering

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June 2024

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Abstract

Attitude determination and control is a crucial subsystem for space vehicles. This subsystem is responsible for correcting the spacecraft or satellite attitude or to reorient if it were to stray from the mission goal. It also has the responsibility of reorienting for power if the system relies on solar panels as a main source of energy. Using photodiodes for attitude determination and reaction wheels for control, this project shows a conceptual design of the ADCS subsystem. Using the information from the photodiodes and given known dynamics, the controller can send commands to the actuators to control the CubeSat for reorientation or to hold the current attitude steady from outside disturbances. By utilizing off the shelf parts, the integration is simplified and development risks are reduced. This conceptual design was developed with scalability and modularity in mind. This allows for the design to be modified to fit mission specific requirements or design requirements on the whole CubeSat. It can also be scaled up for larger CubeSats given the modular nature of them.

Acknowledgements

I would like to thank my family, which consists of my mom, dad, and brother, for their unwavering support. Without them, I would not be where I am today.

- To Mom,

  Thank you for always supporting me no matter what.

- To Dad,
  Thank you for always pushing me to always achieve more.

- To Jeremy,
  Thank you for always being there as my brother.

Next, I would like to thank my many friends, who have joined me on my journey at various points, but still are crucial part of who I am today:

- To Jorge and Alexis,

  Thank you for making high school band and volleyball so enjoyable and continuing to be there for me whenever I need.

- To Joey, Hieu, Devin, Stanley, Jason, and Jesse,
  Thank you for being part of the end of my bachelors and throughout my masters. Without you, I would not have been able to get through all the hurdles.

I would also like to thank Dr. Periklis Papadopoulos for being my advisor for this project and for being my professor for several classes throughout my journey at SJSU. I also want to thank Dr. Long Lu for the amazing controls classes he taught while at SJSU and sparking my interest in controls.

On top of those mentioned above, I want to thank all the others who have been a meaningful part of my life. I wish all the best to everyone.

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1. Introduction

1.1 Motivation

We use satellites to gather information about the Earth. Satellites are also used for communication, sending signals across the globe, capturing pictures of the Earth, and taking deep space images. Satellites orbit the Earth outside of the atmosphere. These small bodies in space are subject to outside disturbances. Outside of the Earth's atmosphere, there is a lot of outside interference. These disturbances come from solar winds, gravity gradients, and aerodynamic torques. Disturbances have a noticeable effect on small bodies in space, like satellites, changing their attitude unfavorably. It is important for a satellite to have control of its attitude. The satellite needs to be able to resist the disturbance torque and reorient itself while in orbit if needed to function properly. CubeSats has seen an increase in popularity due to its small size. The smaller size leads to lower production costs and reduced launch costs. CubeSats can piggyback off of large spacecraft launches. CubeSats are measured in terms of a unit [U]. A 1U CubeSat is a 10 cm x 10 cm x 10 cm structure. CubeSats can be stacked together, forming structures of various sizes like 2U, 3U, 6U, and up to 24U. With the limited space a CubeSat has, components onboard need to be reduced in size. It sounds simple enough to increase the size to fit more components, but that would defeat the point of having a CubeSat.

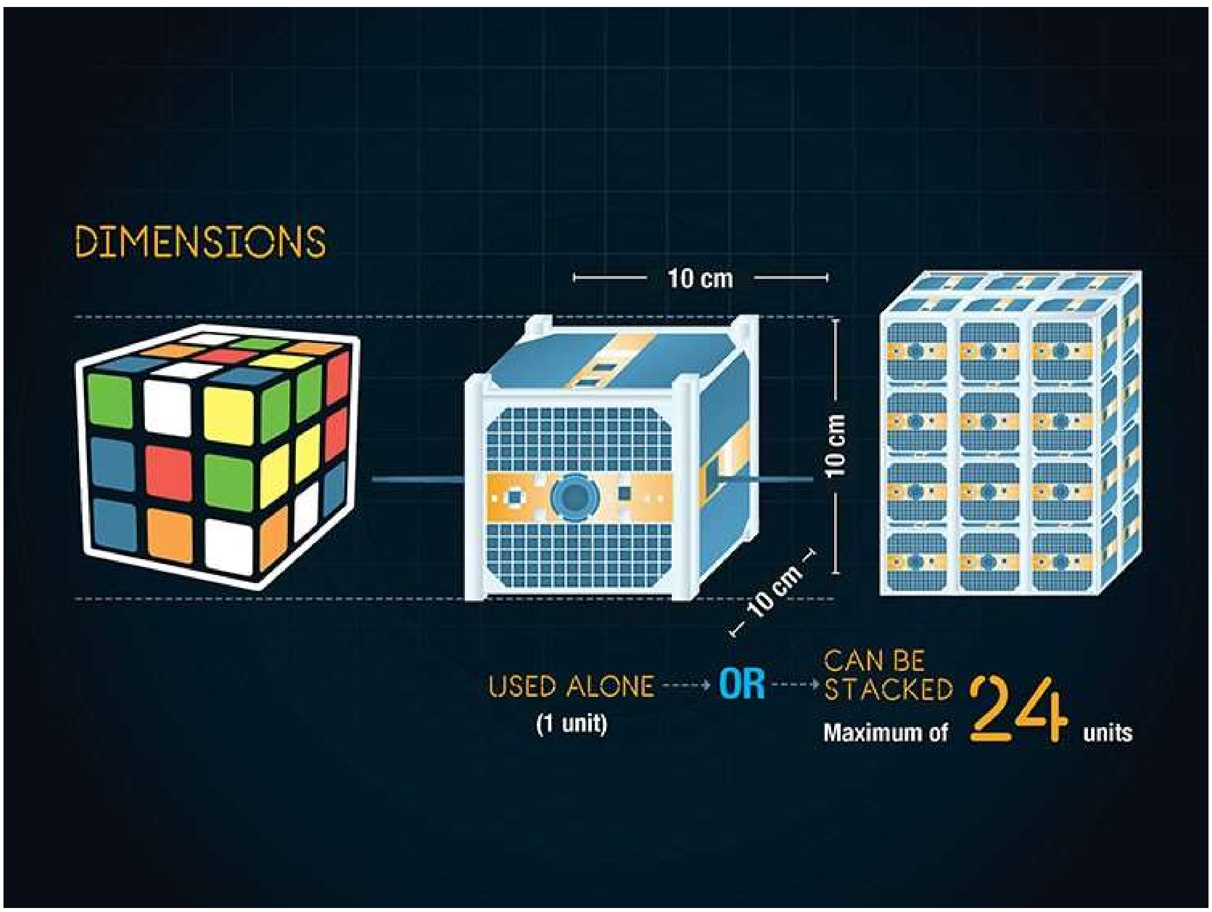

Figure 1.1: CubeSat size comparison and stacking [1]

1.2 Literature Review

Guidance, navigation, and control (GNC) is a subsystem which includes components used for position determination along with the components used by the Attitude Determination and Control System (ADCS). For Earth orbiting satellites, the position can be determined using a Global Positioning System (GPS) receiver. An alternative way to determine position would be to use a ground-based radar tracker.

ADCS requires sensors onboard to help with determining attitude and spin rate of the spacecraft. These sensors include star trackers, sun sensors, horizon sensors, magnetometers, and gyroscopes. ADCS is also used for trajectory correction maneuvers and terminating maneuvers using accelerometers when the desired velocity has been achieved. ADCS utilizes actuators for control and is often coupled with other subsystems

Hardware needs to be small to fit onboard a CubeSat. There is a trend of miniaturizing existing technology for CubeSats. The technology for stabilizing large spacecrafts in three-axis has been flown for decades, but the miniaturization has only been around for a few years. The table below summarizes current state-of-the-art performance for the NG&C subsystem.

Table 1.1: State-of-the-Art NG&C Subsystems [2]

Component	Performance	TRL (Technology Readiness level)
Reaction Wheels	0.00023 – 0.3 Nm peak torque, 0.0005 – 8 N m s storage	7-9
Magnetic Torquers	0.15 A m^2– 15 A m^2	7-9
Star Trackers	8 arcsec pointing knowledge	7-9
Sun Sensors	0.1° accuracy	7-9
Earth Sensors	0.25° accuracy	7-9
Inertial Sensors	Gyros: 0.15° h-1 bias stability, 0.02° h-1/2 ARW Accels: 3 µg bias stability, 0.02 (m s-1)/h-1/2 VRW	7-9
GPS Receivers	1.5 m position accuracy	7-9
Integrated Units	0.002-5° pointing capability	7-9
Atomic Clocks	10 – 150 Frequency Range (MHz)	5-6
Deep Space Navigation	Bands: X, Ka, S, and UHF	7-9
Altimeters	~15 meters altitude, ~3 cm accuracy	7

1.2.1 Sensor Selection

Before thinking about the control, the spacecraft will need a way to determine its attitude. The type of sensor used will have an impact on the whole system as each sensor offers its own benefits and costs. These sensors need to accurately depict which orientation the spacecraft is currently to notify the onboard computer if the orientation is incorrect and needs to be corrected. Below are some common sensors are but not limited to:

- Sun sensors
- Horizon sensors
- Star sensors
- Magnetometers
- GPS receivers
- Gyroscopes

1.2.1.1 Sun Sensors

Sun sensors have been utilized many times in the past for attitude determination, control, and the generation of switching and timing signals [3]. These sensors determine the attitude of the spacecraft by indicating the orientation of the Sun based on the intensity difference between radiation arriving from the solid angle determined by the Sun's boundaries and that arriving from adjacent regions within its field of view [3]. They are visible light detectors that measure one or two angles between their mounting base and the incident light. Sun sensors are a popular choice for attitude determination, are accurate, and reliable. However, they require a clear view to work.

Many sun sensors have been successful in past missions, however some did act irregularly and caused mission failures. The sources of these issues came from the thermal environment, radiation environment, handling and contamination, vibration and shock, interference, and interface problems. The temperature in space changes rapidly, cycling between hot and cold depending on if the object is facing the Sun or not. Radiation causes damage to sun sensors, changing the photo sensitivity. Sun sensors need a clean surface to function properly. Any sort of dust or debris can hinder the performance of sun sensors. Like many components, sun sensors are also subjected to vibrations and shock that can damage the component. Outside interference can also affect the performance. In one case, stray illumination from the Moon affected the sun sensor in the Mariner 5 mission [3]. Lastly, sun sensors have failed due to improper connection with the satellite.

The design of the sun sensor shall be consistent with the system specifications, mission objectives, and the test program [3]. Each individual characteristic of the sun sensor should be analyzed for compatibility with other components. The sun sensor's spatial filter shall demonstrate that it provides an angular subtense and resolution required by the mission objective. The spectral filter characteristics need to be compatible with performance objectives such that it will not degrade in the changes characteristics of components. The overall effectiveness of the sensor shall not be degraded by changes of the photosensitive conductor.

Each of these functions has its own associated parameters and hardware, shown in the table below.

Table 1.2: Sun Sensor Functional Elements and Related Components [4]

Functional Block	Associated Parameters	Hardware Components
Spectral filter	Spectrum of source Element spectral response Optical transmission Angle of incidence	Radiation detector Filters Lens (optical elements)
Spatial filter	Proportional range Field of view Linear range Resolution Cross-coupling Data format	Shields Masks Blades Aperture Detector shape Lens Reticles
Radiation sensitive element	Sensitivity Noise current (or voltage) Bandwidth Load impedance Hysteresis Spectral response	Radiation detector Electronic interface

1.2.1.2 Horizon Sensors

Another sensor spacecrafts use for attitude determination are Earth horizon sensors. These sensors can conveniently provide a way of determining the local vertical. The local vertical can then be used for attitude determination by referencing it to the orientation of the spacecraft's components, like onboard instruments and antennas. Some sensors are scanners, which scans a large area in space mechanically, electronically, or passively to gather information. Another type of sensor are edge-tracking sensors. Edge tracking drives a detector field of view to a location relative to the horizon. It is then dithered across the horizon using a servomechanism and it derives the error signal from the detector waveform [4]. Radiance-balance sensors require some setup. It assumes that when radiance balance has been achieved, the sensor's optical axis points at the center of the illuminated area. These are mostly designed to utilize thermal discontinuity towards the edges of an optically formed Earth image.

Horizon sensors have strict design requirements. They should be designed so that they can determine the angular position of the optical discontinuity. Some constraints the design needs to follow are the accuracy, lifetime, reliability, weight, and power based on the spacecraft the system will be onboard. The design also needs to minimize interference from the outside environment, like cold clouds, atmospheric scattering, and Sun and Moon interference (NASA, 1970). Some other design constraints to consider are scan mechanization protection, thermal design, detector life, alinement provisions, contamination and degradation of optical elements, and corona suppression.

Infrared Earth horizon sensors (EHS) can provide attitude knowledge for LEO satellites, also providing knowledge during eclipses [5]. Infrared EHS obtains the attitude information by detecting Earth's infrared electromagnetic field, determining which areas are concealed by the Earth from the sensor's field of view, and then computing a nadir vector estimation in the satellites body frame.

1.2.1.3 Photodiodes

Photodiodes are a semiconductor device. When it is exposed to light, it converts the photons to an electrical current. They output the "current as a function of light intensity and angle to the light source" [6]. Based on electrical current, we can determine how fast each photodiode is receiving the light, thus determining our attitude.

The photodiodes need to be placed in a specific architecture in order to properly determine the attitude. Simply put, the photodiodes can not be placed in a straight line as it would fail to determine the attitude of the spacecraft if they were facing away from the Sun.

1.2.2 Control Methods

Sensors are great for determining attitude, but the other aspect is the control. Spacecrafts and satellites have used reaction wheels, magnetorquers, and control moment gyroscopes for control.

1.2.2.1 Reaction Wheels

Reaction wheels are the most common system for active control. They are optimal due to the fact that they are highly reactive and can output a continuous feedback control. They create a torque on the spacecraft by creating equal but opposite on reaction wheels. Since they only offer internal torques, the spacecraft needs to be able dump any excess momentum from the reaction wheels.

Table 1.3: Reaction Wheel Comparison [7] [8]

Company	**Model**	**Momentum (Nms)**	**Mass (kg)**	**Volume (mm)**	**Max Torque (Nm)**
Blue Canyon Technologies	RWP015	0.015	0.130	42 x 42 x 19	0.004
Blue Canyon Technologies	RWP050	0.050	0.24	58 x 58 x 25	0.007
Blue Canyon Technologies	RWP100	0.10	0.33	70 x 70 x 25	0.007
Blue Canyon Technologies	RWP500	0.50	0.75	110 x 110 x 38	0.025
Blue Canyon Technologies	RW1	1.0	0.95	110 x 110 x 54	0.07
Blue Canyon Technologies	RW4	4.0	3.2	170 x 170 x 70	0.25
Blue Canyon Technologies	RW8	8.0	4.4	190 x 190 x 90	0.25
AAC Clyde Space	iADCS400	0.06	1.15 - 1.7	95.4 x 95.9 x 67.3	0.02
AAC Clyde Space	RW222	0.06	TBC	25 x 25 x 15	0.02
AAC Clyde Space	RW400	0.08	197 - 375	50 x 50 x 27	0.08

Reaction wheels offer many benefits. They have precise attitude control, offering fine tuned adjustments to small changes in orientation. This makes them ideal for missions calling for high pointing accuracy. They also provide continuous control, allowing for smoother and gradual attitude control. Reaction wheels are also fuel free. They are free from the limitation of needing the satellite to have fuel onboard, increasing the mission lifespan without the need for external support. Along with increasing mission lifespan, no fuel also means no waste is expelled, removing the concern of contaminating the environment or sensitive equipment. CubeSats already limited with space and reaction wheels not needing fuel is a plus. Reaction wheels operate quietly. Unlike thrusters, reaction wheels produce very little to no vibrations which can negatively affect onboard equipment that is sensitive to disturbances.

Reaction wheels have limitations as well. Over longer periods of time, reaction wheels store angular momentum from external forces, like Earth's gravity gradient. This leads to saturation, which requires an external source to desaturate them. They also require continuous electrical power to operate, putting a strain on the satellites electrical power system and reducing the mission lifetime. Reaction wheels are also more prone to mechanical failure due to moving parts. Depending on the design, some reaction wheels are more bulky, which is not ideal for CubeSats.

1.2.2.2 Magnetorquers

Another option for control are magnetorquers. Magnetorquers have many benefits. They are made of wire coils or magnetic rods. These are simple, durable, and have no moving parts, making them less prone to mechanical failure when compared to reaction wheels or gyroscopes. This makes them low maintenance, making them more ideal for longer missions. Another benefit is their lightweight and compact design. This is a plus for CubeSats as space is already limited as it is. Magnetorquers are also cost effective when compared to other control options. Magnetorquers, similar to reaction wheels, do not require any consumables, like propellant, to function, making them more environmentally friendly. Since they do need any propellant, no waste is ejected so there is no risk to any contamination of other sensitive hardware. Magnetorqers also require a relatively low electrical power to operate, which is another benefit for CubeSats.

However, there are some prerequisites and limitations for magnetorquers to function properly. Knowledge of the strength and direction of the ambient magnetic field must be known beforehand [9]. They need to maintain cleanliness in the satellite. Magenetorqers also rely heavily on Earth's magnetic field. This weakens at higher altitudes, so there is a limitation on what elevation magnetorquers will be effective. On top of these limitations, magnetorquers also require a significant amount of time to control the satellite, making them less ideal in situations where a satellite needs to reorient itself quickly for capturing an image or when tracking something moving at higher speeds.

1.2.2.3 Control Moment Gyroscopes

Control moment gyroscopes (CMG) are capable of fast maneuvering and have high precision. Having a high torque capability, CMG's can produce more torque compared to reaction wheels, allowing for more rapid changes to a satellite's attitude. Similar to reaction wheels, CMGs can provide continuous and precise control.

The downside to CMG's is that it has severe degradation, which can hinder the performance of the spacecraft. The rotor bearing is the core component, and the whole life of the gyroscope depends on that one component, which is not optimal as it increases the risk of failure significantly [10]. CMG' are also more mechanically complex when compared to reaction wheels, requiring more advanced control algorithms and increased external support.

1.3 Project Proposal

The goal of this project is to design a single board GNC system for a low earth orbit CubeSat. To determine the attitude of the satellite, four photo diodes will be in a pyramid formation. The attitude can be determined based on how fast each diode is receiving light from the Sun. For the control, the CubeSat will utilize reaction wheels. Reaction wheels have TRL of 7-9, react faster when compared to a magnetorquer, and there are many options that can fit onboard a CubeSat.

1.4 Methodology

The approach will be to simulate a CubeSat in a low earth orbit. Simulating will provide the advantage of doing it quicker and more cost effective. A light source, like a light bulb, will emulate the Sun. A structure will be designed and fabricated to act as the CubeSat. The layout of the photodiodes will get readings from the light bulb, thus giving the attitude of the CubeSat. A program will need to be developed in order to read the data gathered from the photodiodes and translate it into what the current attitude of the CubeSat is. Then the program will need to command the control of the subsystem to correct its orientation, if needed.

2. Photodiodes

2.1 Photodiode Basics

While there are many models of photodiodes, four common elements appear in them: a current source, a parallel capacitor, a parallel resistor, and a series resistor [11]. A normal pn junction is also commonly found in many photodiodes, depicted by a diode symbol.

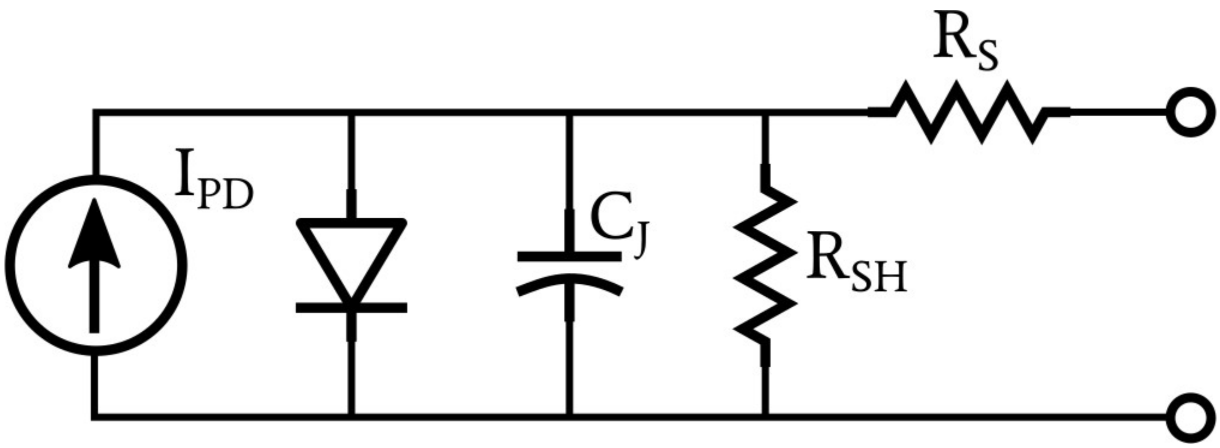

Figure 2.1: Model of a photodiode [11]

An ideal current source represents a photocurrent. The current is generated by the diode in response to a light source.The direction of the current corresponds to the flow from the cathode to the anode, indicating the use of zero bias or reverse bias [12].

2.2 Reading Photodiodes

A current to voltage converter is an optimal way to convert the photodiode current to a voltage [12]. Based on the diagram below, the current will flow through R1 when light hits the photodiode. It is important to note that the output voltage will be negative, and if the desired output is a positive value, then the diode polarity needs to be reversed.

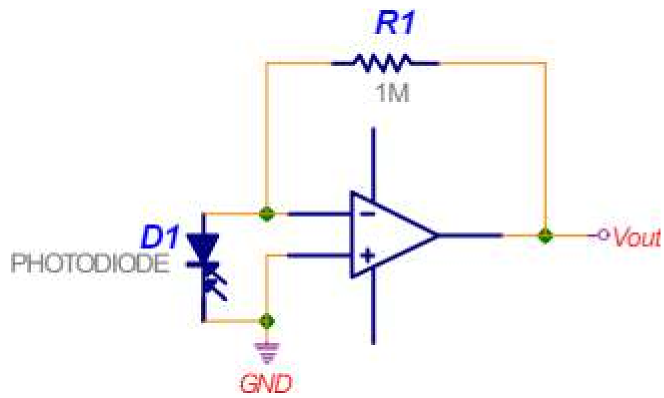

Figure 2.2: Photodiode circuit diagram [11]

Photodiodes are not perfect. Photodiodes produce a leakage current that is proportional to the intensity of the light. A small leakage current also appears when reverse biased/ The leakage current is also known as a dark current. It is temperature and voltage dependent, thus increasing the temperature and/or reverse bias would increase the dark current [11].

There are a few things to note when reading a photodiode datasheet: dark current (I_D), breakdown voltage (V_{BR}), noise equivalent power (NEP), response time (t_r), junction capacitance (C_j), short circuit current (I_{SC}), shunt resistance (R_{SH}) [11]. Dark current was explained above. Breakdown voltage is the maximum reverse voltage a diode can withstand. Noise equivalent power is the photon intensity required to equal the noise of a given reverse bias. Response time is how fast the diode responds to a step input on a given reverse bias. Short circuit current is the current that flows at given light intensity when the diode pins are shorted. Shunt resistance helps estimate thermal noise when a reverse bias is not applied. It is shown as a ratio of voltage to current near V=0 [11].

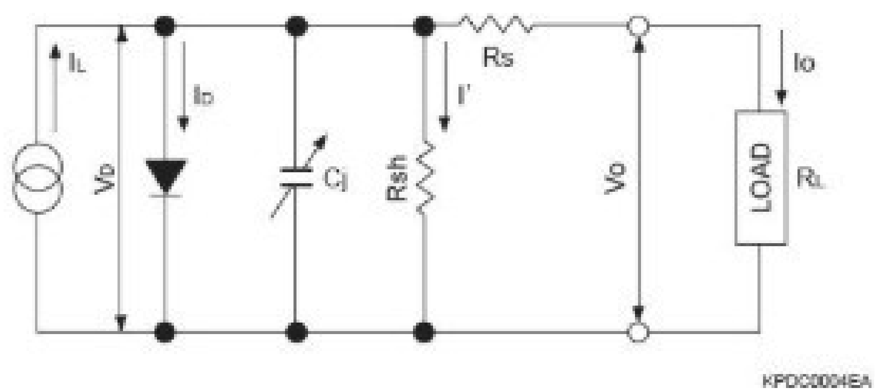

Figure 2.3: Photodiode equivalent circuit [11]

2.3 Junction Capacitance

Junction capacitance is the capacitance associated with the depletion region of a pn function. This is an important parameter since it has a big influence on the frequency response of the photodiode [12]. Having a lower junction capacitance allows for higher levels of frequency operation. The figure below depicts large reductions in junction capacitance by operating a photodiode in photoconductive mode

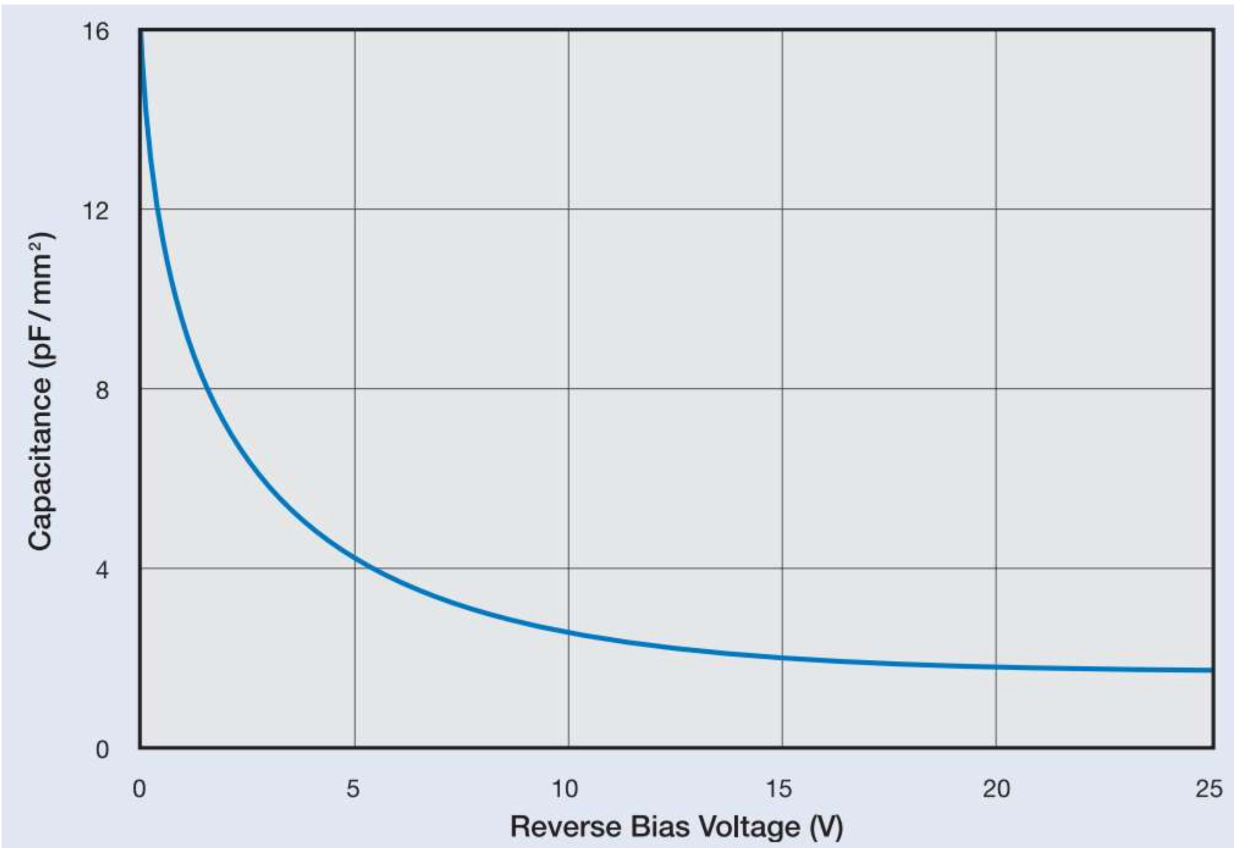

Figure 2.4: Junction Capacitance Graph [12]

2.4 Band-pass Filters

There are many situations in which a particular band, spread, or frequencies need to be filtered out from a wide range of frequencies. To accomplish this task, a low-pass and high-pass filter can be combined to form a filter circuit [13]. The properties of a low-pass and high-pass filter make the circuit a band-pass filter.

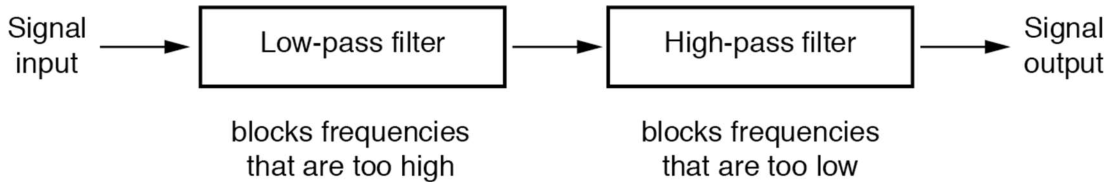

Figure 2.5: System Level Block Diagram of a Band-pass Filter [14]

Band-pass filters can also be created using capacitors. Using this type of filter circuit combined with the band-pass filter from above, the resulting circuit will only allow passage for frequencies that are neither too high nor too low [13].

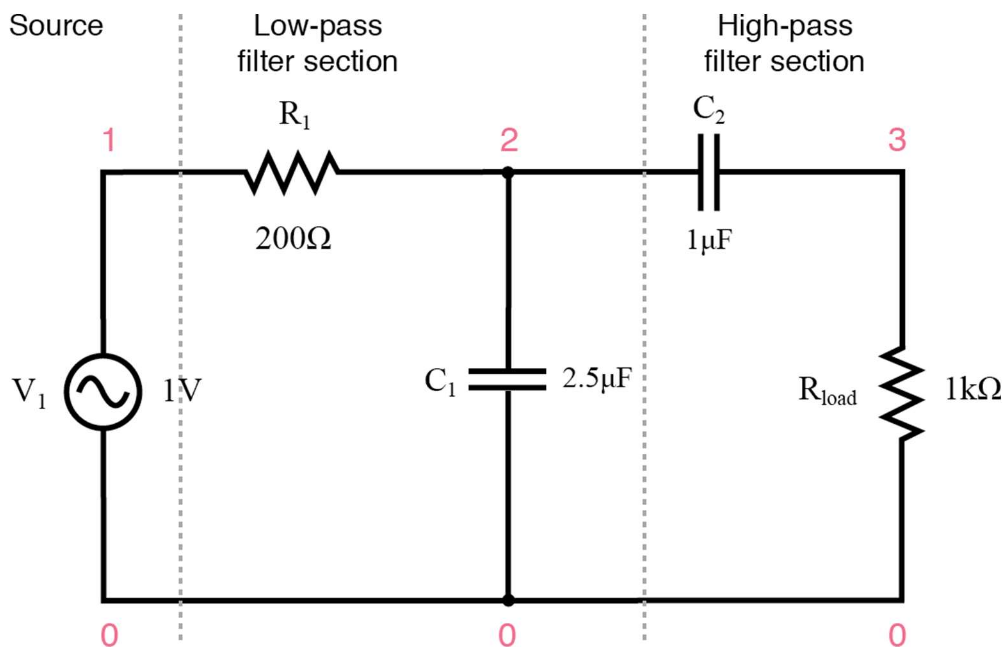

Figure 2.6: Capacitive Band-pass Filter [14]

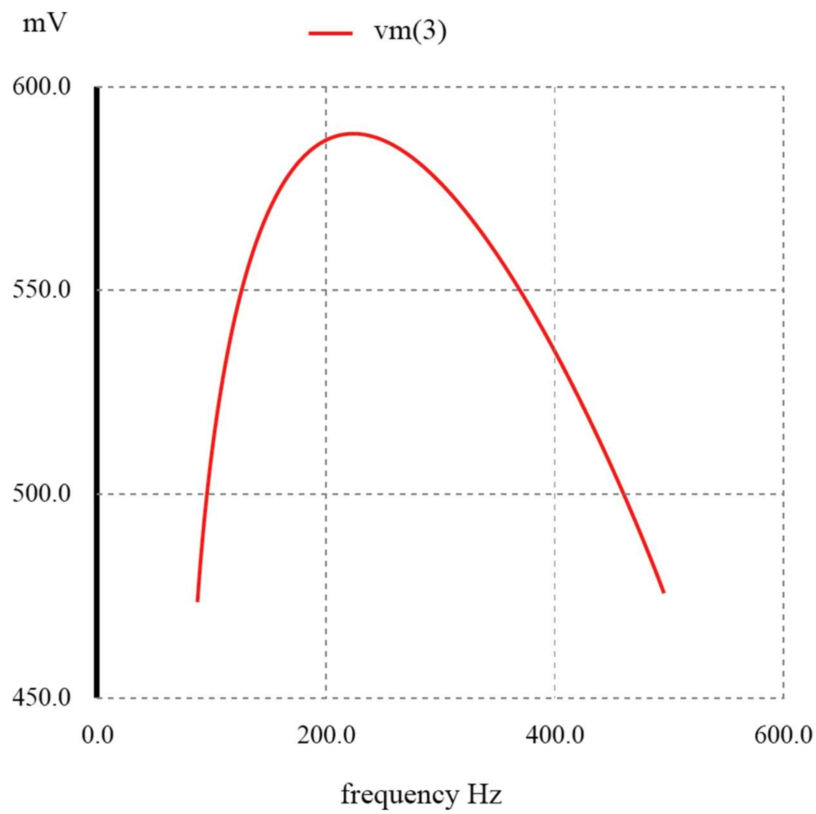

Figure 2.7: Response of a Capacitive Band-pass Filter [13]

2.5 Sensor Specifications

Sensor specifications will depend on the satellite's size and mission objectives. There is a diversity of how satellites are classified based on mass. Based on NASA, satellites fall into small or large spacecraft.

Small spacecrafts are any body with wet masses less than 500 kg. 500 to 1000 kg mass are generally considered medium satellites, but the medium classification is usually omitted to refer to them as either small or large. Small spacecraft are also split into smaller sub-categories: femto, pico, nano, micro, mini, and medium. CubeSats are satellites that fall into the micro and nano class.

Table 2.1: Satellite Class [14]

Spacecraft Class		Wet Mass (kg)
Small Spacecraft	Femto	0.01 - 0.1
	Pico	0.1 - 1
	Nano	1 - 10
	Micro	10 - 100
	Mini	100 - 500
	Medium	500 - 1000
Large Spacecraft		> 1000

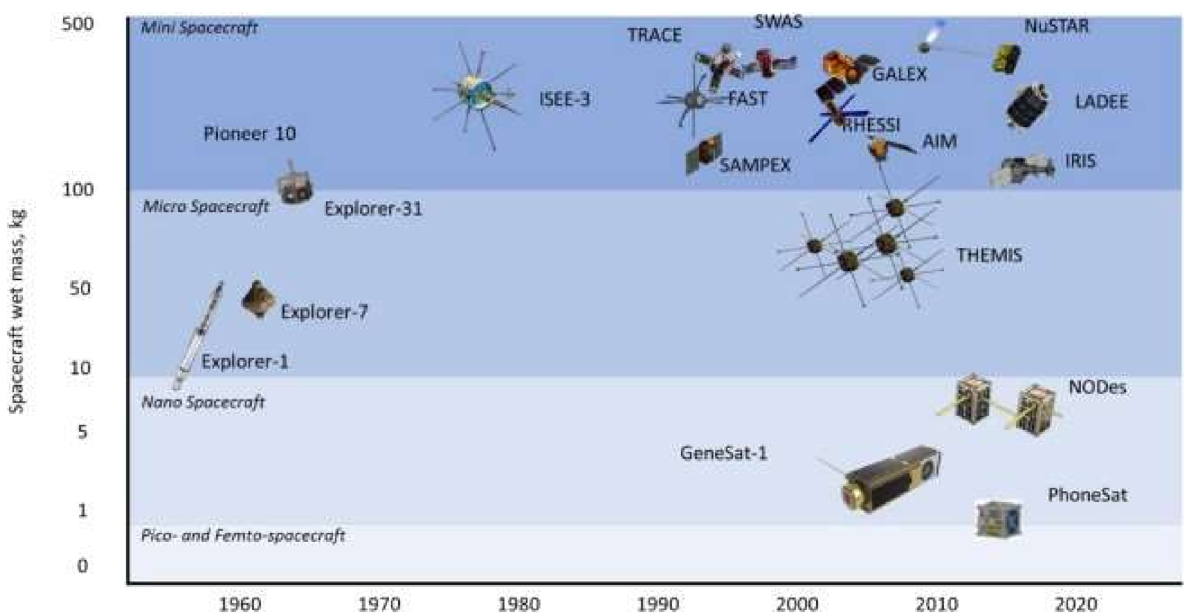

Figure 2.8: SmallSats Mass Over Past Six Decades [14]

When selecting a sensor, there are a few things to consider:

- Sensor type
- Resolution, usually based on ground sampling distance (GSD)
- Wavelength/Frequency
- Power Consumption
- Weight

Table 2.2: General Sensor Specifications Based on Satellite Size

Satellite Size	Sensor Type	Resolution	Wavelength/ Frequency	Power Consumption	Weight
SmallSat	Optical	1 - 10 m GSD	Visible (400 - 700 nm)	10 - 20 W	1 - 5 kg
MediumSat	Radar	3 - 30 m GSD	X-band (8-12 GHz) C-band (4-8 GHz)	150 W	50 kg
Full-scale	Hyperspectral	> 30 m GSD	400 - 2500 nm	500 W	150 kg

There are many types of orbits satellites are on when orbiting the Earth:

- Low Earth orbit (LEO)
- Medium Earth orbit (MEO)
- Geosynchronous orbit (GEO)
- Highly elliptical orbit (HEO)
- Sun-synchronous orbit (SSO)

Each has their own costs and benefits over the alternatives.

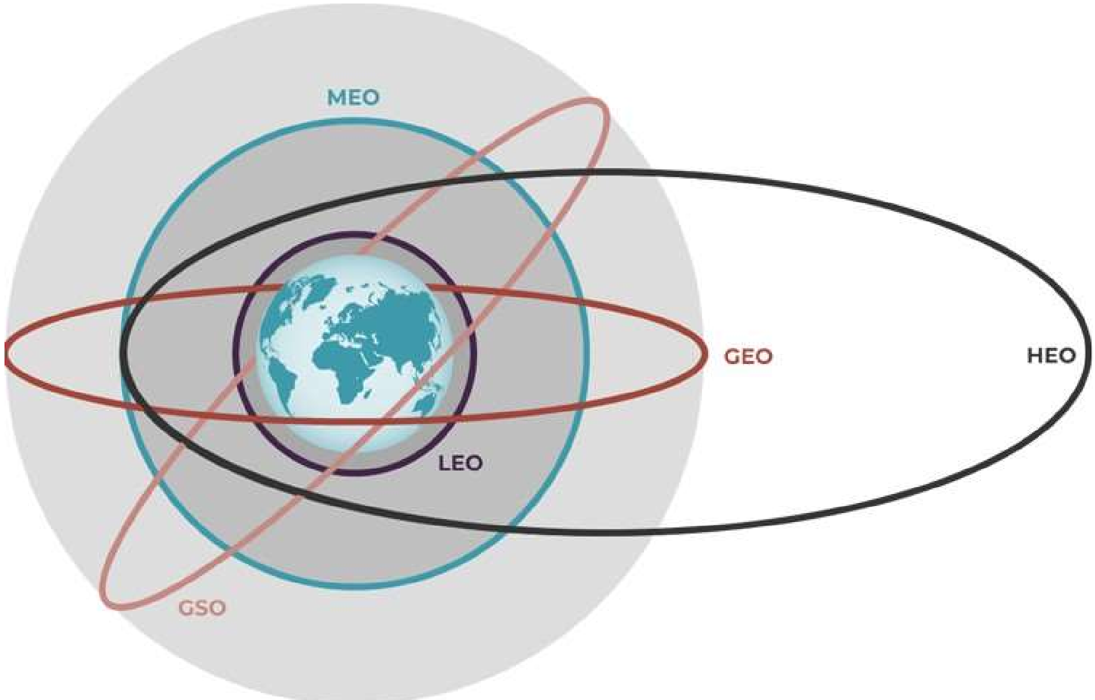

Figure 2.9: Type of Orbit Around Earth [15]

Table 2.3: Properties of Each Type of Earth Orbit [16] [17] [18]

Orbit Type	Altitude (km)	Satellite Path	Satellite Speed (km/s)
LEO	< 2000	No specific path	~ 7.8
MEO	2000 - 35,586	No specific path	~ 3.1
GEO	35,586 - 35,986	‘Fixed’ with the Earth	Equal to Earth’s rotation
HEO	Perigee: ~ 1000 Apogee: > 35,786	Elliptical	~ 3.1
SSO	200 - 1000	North to South, synchronous with the Sun	~ 7.8

Table 2.4: Orbit's Common Use [16]

Orbit Type	Common Use
LEO	Satellite imaging
MEO	Navigation and tracking
GEO	Fixed point above the Earth
HEO	Communications
SSO	Observing a point at a specific time

Along with having different properties, each orbit has a different set of mission objectives as shown in the table above. LEO orbits house the International Space Station (ISS), the Hubble Space Telescope, SpaceX's starlink satellites, and many others. With LEO being in close proximity to the Earth's surface, having the ISS orbit there is more optimal as astronauts do not need to travel as far [16]. Sending supplies would also cost less. The lower altitude of LEO also makes it favorable for imaging as satellites can take higher resolution images. LEO is also useful for inter-satellite communication and connecting to ground stations [19]. Satellites in LEO travel at about 7.8 km/s, circling the Earth in about 90 minutes [16]. Navigation and tracking satellites are commonly found in MEO. LEO satellites travel too fast for those types of satellites. Tracking ground stations at those speeds will require a lot more resources. SSO orbits are commonly used for missions that need to observe a point on Earth at a specific time of day. It is useful for when missions want the same images of an area across several days.

Within each type of orbit, there are different requirements sensors need to meet. Sensors in MEO need to be protected with shields as they need to pass the Van Allen Radiation Belts [20]. Generally, the farther a satellite is from the Earth's surface, the satellite needs a stronger transmit beam power [20]. This leads to larger hardware, increasing the size of the satellite.

3. Reaction Wheels

3.1 What are Reaction Wheels?

Reaction wheels are a flywheel type of device. They can store rotational energy which can then later be used to control the attitude of a spacecraft or satellite.

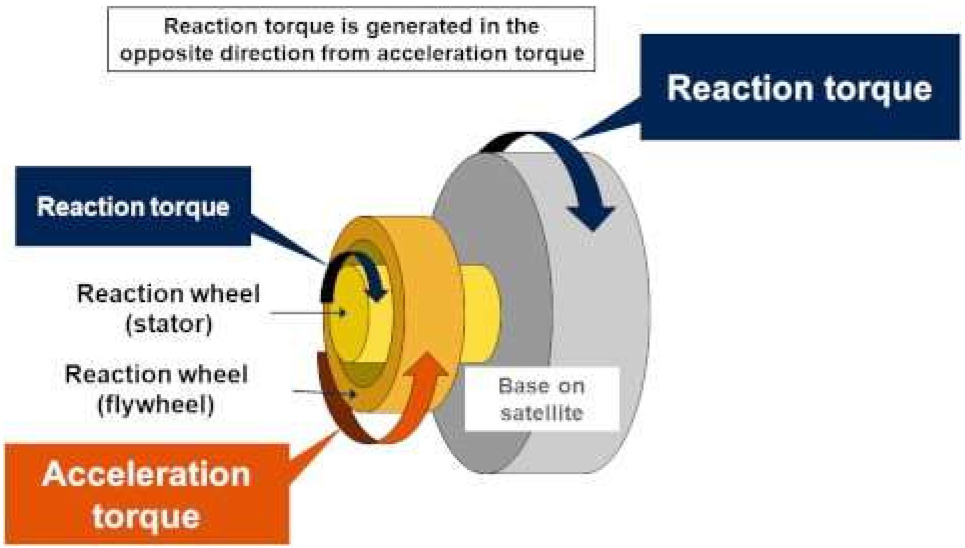

Figure 3.1: Reaction Wheel Behavior [21]

3.2 Why Reaction Wheels?

One major benefit of reaction wheels is that they offer three axis attitude control, without the need of external sources or torque. This reduces the amount of hardware the spacecraft or satellite requires, freeing up space for other components or simply reducing the size of the structure. This is critical when it comes to CubeSats as they are already very limited in space. Reaction wheels are electric actuators, so they can run off of the satellite's electric power that it gets from sources like solar cells. Thrusters require fuel to operate, adding more weight and taking up more space while reaction wheels are not dependent on fuel and do not require it. Even though they have some benefits over traditional thrusters, a downside to them is that they can only rotate the satellite. Reaction wheels can not apply any acceleration to the satellite and there is an upper limit to how much angular momentum it can store [21].

Another benefit of reaction wheels over traditional thrusters is the precision it has [22]. They can offer attitude control on the order of microradians. This is important for Earth observation which this project is focusing on. On top of precision, reaction wheels are also an internal system, power efficient, and have a long life time [23]. Being an internal system, reaction wheels are not exposed to the outside environment, avoiding any contamination that might interfere with sensitive hardware.

3.3 Momentum Requirements

Reaction wheels also need to scale with the satellite's size and weight. Larger satellites will require more momentum to control it. It is important to select the appropriate size reaction wheel, otherwise the satellite will not have any attitude controllability. The table below shows how much momentum is required to control each class of satellites.

Table 3.1: Satellite Class and Require Momentum [21]

Satellite Size	Weight (kg)	Angular momentum of reaction wheel
CubeSat	1-50	1-100 mNms
MicroSat	100	0.1-0.8 Nms
MiniSat	200	0.9-4 Nms

Based on that table, an optimal reaction wheel for this project would need to produce 1-100 mNms of angular momentum.

3.4 Failure Modes

Although reaction wheels sound perfect in terms of CubeSat application, there are few points of failure to consider. Some common issues reaction wheels face are bearing wear, motor failure, electronics failure, mechanical jam, and lubrication loss [23]. CubeSats generally stay in orbit for around two to five years. Reaction wheels are considered reliable, and failure any of those mentioned should not happen during the lifetime of the CubeSat. Many satellites are equipped for three axis control with three reaction wheels, but having a fourth reaction wheel for redundancy is a good practice [24].

4. Supporting Systems and Limitations

With the hardware in mind, supporting systems are required to ensure the GNC subsystem can operate as expected. A satellite generally has the following subsystems: attitude determination and control, telemetry, tracking, and command, command and data handling, power, thermal, structure and mechanisms, and guidance and navigation. This project is focusing on attitude determination. Other subsystems that need to be considered are the power and structure and mechanisms as the reaction wheels and photodiodes need electrical power to function and need a structure to live in.

4.1 Electrical Power Subsystem

The electrical power subsystem (EPS) is responsible for providing, storing, distributing, and controlling the spacecraft's electrical power [25]. The most important requirement to consider is the demand for average and peak electrical power. Another important factor to consider is the orbital profile the spacecraft or satellite will be in.

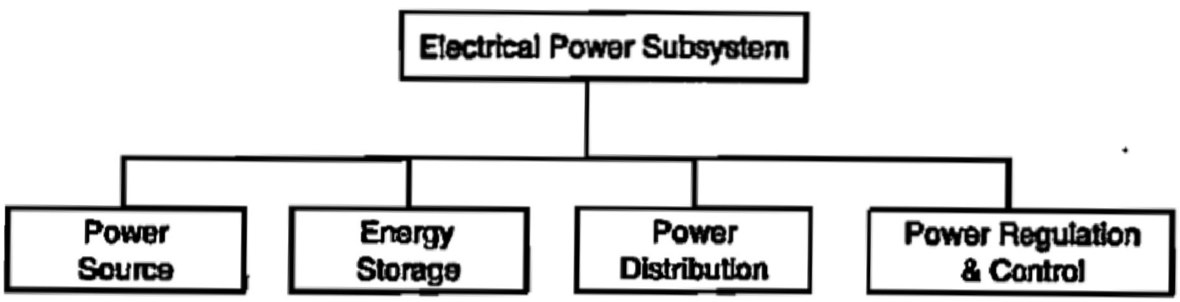

Figure 4.1: Breakdown of Spacecraft's Power Subsystem [25]

4.1.1 Power Sources

Launch vehicles primarily use batteries as their power source. Batteries are a simple solution when they are only needed for less than an hour. However, batteries are not optimal for missions that will run for weeks or months on end. There are generally four different power sources a satellite or spacecraft will use for these long missions:

- Photovoltaic solar cells
- Static power
- Dynamic power
- Fuel cells

4.1.1.1 Photovoltaic Solar Cells

Photovoltaic solar cells are the most common source of power for Earth orbiting spacecraft [23]. They convert incident light from solar radiation into electrical energy. The electrons move toward the front surface of the photovoltaic cell and it creates an imbalance of electrical charge between the cell's front and back surfaces [26]. The conductors on the cell absorb the electrons and when connected to an external load, electricity flows through, giving it power.

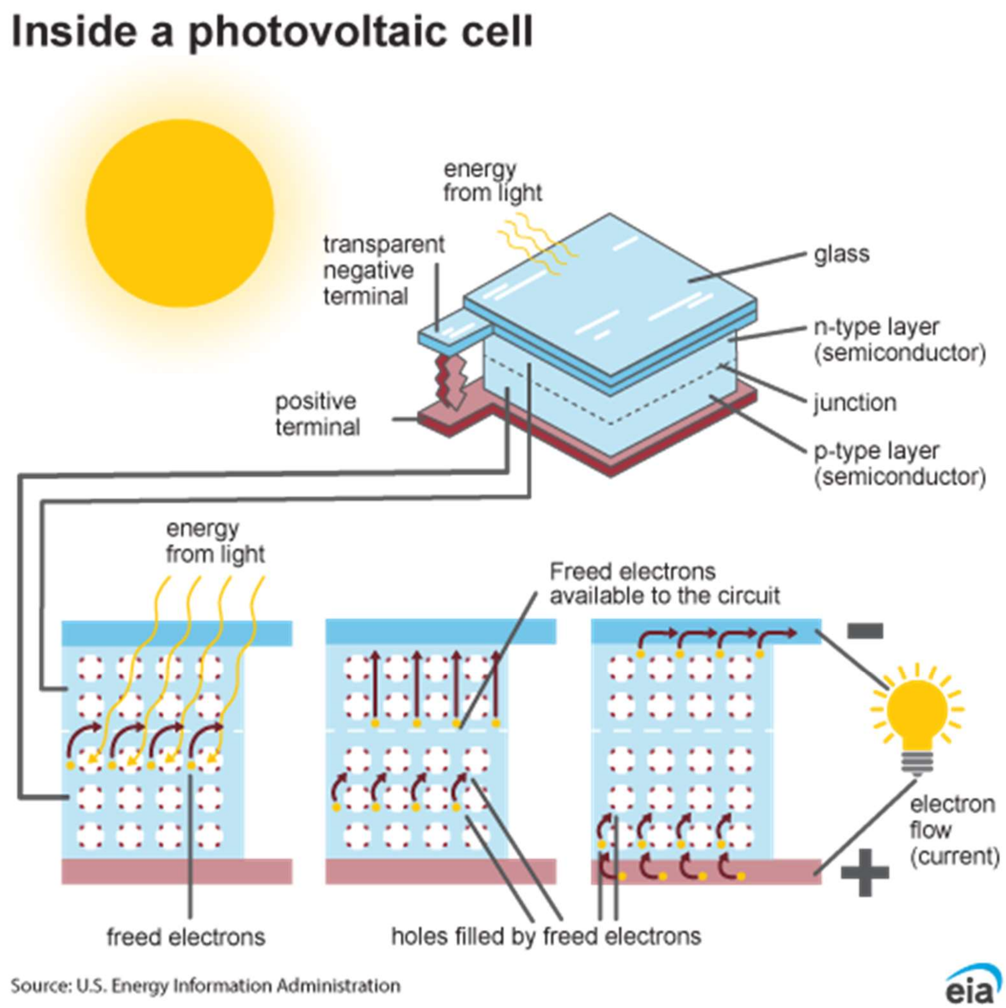

Figure 4.2: How a Photovoltaic Cell Functions [26]

4.1.1.2 Static Power

Static power uses a heat source for direct thermal to electric conversion [25]. Static power relies on one of two concepts: thermoelectric or thermionic. Thermoelectric is the more common of the two. The radioactive source decays at a slow rate, leaving behind a temperature gradient. Thermoelectric concept relies on a basic converter that uses the temperature gradient between the p-n junction of individual thermoelectric cells to provide a desired dc output to the converter [23]. Thermionic energy conversion generates electricity through a hot electrode that faces a cooler electrode. These components are housed inside a sealed enclosure, typically containing an ionized gas. Electrons flow from the hot emitter to the cooler collector across the interelectrode gap [23]. On the cooler collector, the electrons condense and return to the hot emitter through an electrical load.

4.1.1.3 Dynamic Power

Dynamic power also uses a heat source and a heat exchanger to drive an engine [25]. It typically relies on concentrated solar radiation in comparison to static power which generally uses a nuclear reactor. The heat source can also be radioisotopes or a controlled nuclear fission reaction. The balance of energy remains latent in the heat exchanger, providing a continuous energy to the thermodynamic cycle when the spacecraft is not in contact with solar radiation. Dynamic power utilizes one of three methods when generating electrical power: Stirling cycle, Rankine cycle, or Brayton cycle. Stirling cycle engines use a single phase working fluid as the working medium [25]. Rankine cycle uses a two phase system that employs a boiler, turbine, alternator, condenser, and a pump. Brayton cycle engines are dynamic devices, using a single, compressible working fluid as the working medium.

4.1.1.4 Fuel Cells

Fuel cells convert chemical energy of an oxidation reaction to electricity. They can operate continuously without the need of sunlight, allowing them to provide power at all times. However, fuel cells need to carry their own reactant supply. The size of the reactant tank scales with the length of the mission, making them less optimal for SmallSats that have longer mission requirements. A single fuel cell unit can generate many kilowatts of power. The more common type of fuel cell is a hydrogen-oxygen, also referred to as an alkaline fuel cell.

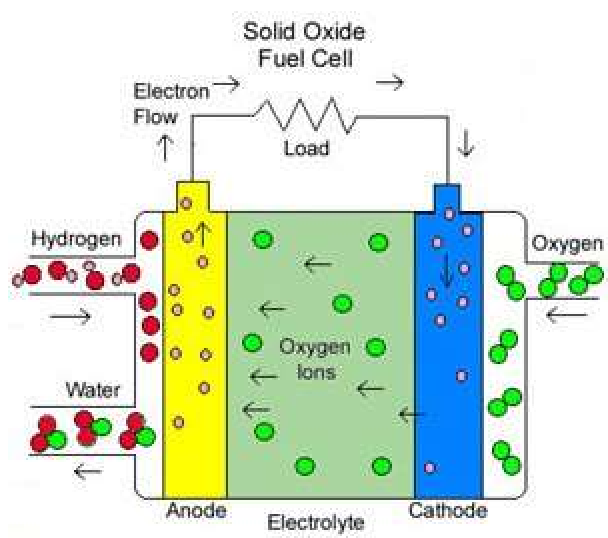

Figure 4.3: Alkaline Fuel Cell Model [27]

4.1.1.5 Batteries

Batteries are a common power method for CubeSats. In LEO, it provides a CubeSat with power even through eclipse periods where the CubeSat is not facing the Sun. They can also meet higher power demands, allowing for more power demanding components in other subsystems to function as intended. Batteries can be integrated with other power options, giving the CubeSat a redundant power source. They also provide the mission with flexibility, allowing operation if the CubeSat has a higher elliptical orbit or if it needs to travel further deeper into space. Being lightweight and compact is another feature CubeSats look for as space is limited. The table below shows a few common batteries that are used in CubeSats.

Table 4.1 Commonly Used Batteries in Satellites [28]

Battery	Pros	Cons
Ni-Cd	- adequate lifetime and fair capacity - lightweight and inexpensive - not as energy dense	- overcharging and extreme temperatures can cause damage - not ideal in prolong use cases
NiH2	- hybrid between fuel cells and battery - higher specific energy compared to Ni-Cd - can be overcharged without major issues	- high self discharge rate - low energy density - requires high pressure storage
Li-ion	- dense energy - longer lifetime - wide range of operating temperatures - can deliver short and high energy peaks	- develop internal resistance at low temperatures

4.1.1.6 Power Distribution

The power source of a spacecraft needs to be able to distribute power to the rest of the spacecraft. The distribution system consists of cabling, fault protection, and switching gear to turn on or off the power [25]. The power distribution system is a unique feature of the EPS, often reflecting the individual spacecraft loads it needs. The load profile of the spacecraft is an important factor when it comes to designing the specifications of the power distribution system.

4.2 Structure and Mechanisms Subsystem

Once all the hardware has been chosen, the parts need to be placed in a structure that can support them. The structures and mechanisms subsystem mechanically supports all other spacecraft subsystems, attaching them to the launch vehicle, and providing an ordnance-activated separation [25].

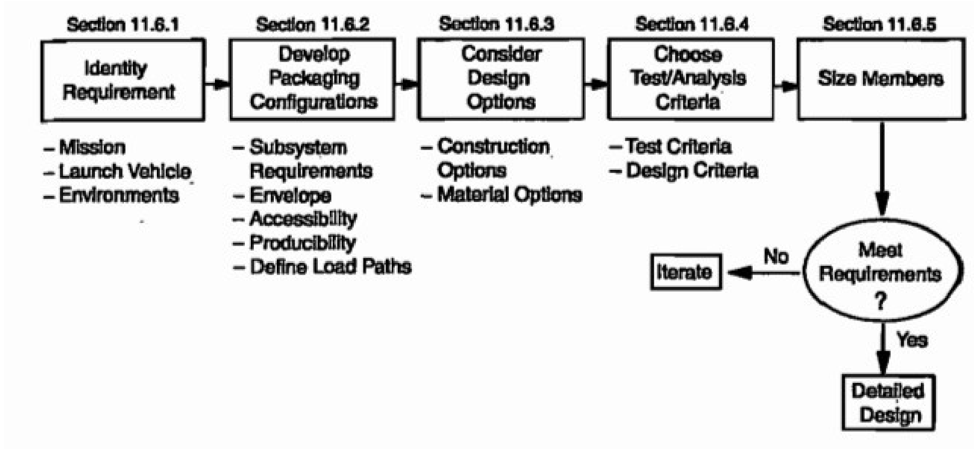

Figure: 4.4 Preliminary Design Process of Structures and Mechanisms [25]

The structure is the subsystem that the size of CubeSats affects the most. The structure needs to be able to support the CubeSat through the responsibilities above while meeting the size constraints. This structure and mechanism subsystem supports the other subsystems by providing a rigid frame. This protects the components of other subsystems from the vibrations of launch and the conditions while in orbit.

On top of simply providing a home for the other subsystems, it needs to integrate smoothly. It needs to secure all components in the structure, and keep sensitive equipment aligned and stable for proper functionality to meet mission goals and requirements. The structure must have the ability to integrate with the launch vehicle as well. CubeSats rely on larger launches as a way to save on launch costs.

5. Requirements

5.1 System Level Requirements

5.1.1 EPS Requirements

The minimum power required for a satellite is dependent on its size. Typically, satellites usually require 200 to 800 watts. The mission lifetime and requirements can also play into the power requirements. Fortunately, CubeSats are significantly smaller in size. They can operate in the range of 1 to 100 watts, depending on the size of the CubeSat.

Table 5.1 CubeSat Power Requirements Based on Size

Size	Power Requirement (W)
1U	1-5
3U	5-20
6U	10-50
12U	20-100

The numbers above are for the overall CubeSat. The ADCS typical power consumption is generally around 1 to 5 watts of power.

5.1.2 Structural Requirements

The structure needs to be able to endure the harsh environment of the outside world, starting from the manufacturing process all the way to the end of the mission. All disciplines: engineering, manufacturing, integration, test, and mission operations should contribute and collaborate together [23]. Since CubeSats are launched from a larger spacecraft, many concerns with surviving the launch can be neglected. The main concern during launch and leaving the Earth. The structure required to support the payload from any type of damage that can occur during this phase. The common types of materials used to build a satellite or spacecraft are aluminum, steel, magnesium, titanium, beryllium, and composites. The table below depicts the advantages and disadvantages of each.

Table 5.2 Materials and their Advantages and Disadvantages

Material	Advantages	Disadvantages
Aluminum	- High strength vs weight - Ductile; tolerant of -concentrated stresses - Easy to machine - Low density; efficient in compression	- Relatively low strength vs volume - Low harness - High coefficient of thermal expansion
Steel	- High strength - Wide range of strength, hardness, and ductile obtained by treatment	- Not efficient for stability (high density) - Most are hard to machine - Magnetic
Magnesium	- Low density–very efficient for stability	- Susceptible to corrosion - Low strength vs volume
Titanium	- High strength vs weight - Low coefficient of thermal expansion	- Hard to machine - Poor fracture toughness if solution treated and aged
Beryllium	- High stiffness vs density	- Low ductility and fracture stiffness - Low short traverse properties - Toxic
Composites	- Can be tailored for high stiffness, high strength, and extremely low coefficient of thermal expansion - Low density - Good in tension (eg. pressurized tanks)	- Costly for low production volume; requires development program - Strength depends on workmanship; usually requires individual proof testing - Laminated composite are not as strong in compression - Brittle; can be hard to attach

Given the limited resources available, the ADCS subsystem, which includes the sensors, actuators, and the controller, is generally 0.5U to 4U, depending on the overall size of the CubeSat. Depending on the size of the reaction wheels, CubeSats around the 3U range can comfortably fit four reaction wheels for 3-axis control and a redundant wheel as a backup.

5.1.3 ADCS Requirements

The ADCS is responsible for the control of the CubeSat. It needs to redirect the CubeSat for its mission, like reorienting the CubeSat to observe different parts of the Earth. It also has other responsibilities based on the other subsystems. If the power subsystem relies on solar panels for power, the ADCS needs to be able to reorient the CubeSat to face the Sun if needed. For imaging CubeSats, the main role the ADCS needs to fulfill is to counteract the disturbance torques the CubeSat encounters while in orbit. For CubeSats in LEO, the disturbance torques it can encounter are aerodynamic drag and solar pressure. The aerodynamic drag can be determined given by the following equation:

$$F_d = \frac{1}{2}\rho V^2 C_d A \tag{5.1}$$

Where:

- F_d = drag force (N)
- ρ = density (kg/m^3)
- V = velocity of the CubeSat
- C_d = coefficient of drag
- A = cross sectional area (m^2)

The control system needs to be able to overcome the disturbance torques the CubeSat will encounter and also need to be able to control the CubeSat when facing said disturbances. The reaction wheels need to overcome the force of the outside disturbance and be able to control the CubeSat, which has an angular momentum of 1-100 mNms requirement.

5.2 Lifespan Requirements

A CubeSat's lifespan is dependent on what kind of orbit it is in. Generally, satellites in a higher orbit have a longer lifespan, but are exposed to more radiation. CubeSats are usually in LEO. Within LEO, the altitude also has an effect on the lifespan. CubeSats that are closer to the surface of the Earth, approximately 500 km above the surface, have a significantly shorter lifetime. CubeSats orbiting at this altitude experience a lot more atmospheric drag, leading to orbital decay. This leads to a lifespan of a few years at best, however some can only last a few months. In LEO above 500 km, CubeSats orbiting in this range experience less atmospheric drag, increasing its lifespan to around 5-10 years.

The choices made for this project's CubeSat should last around 5 years. Lifespan of components are also tied to mission lifetime. If the mission lifetime is over 10 years, then the design should be able to last as long. However, this project is not about designing a GNC subsystem for a specific mission, but for CubeSats in general.

6. Hardware Selection

6.1 EPS Choices

Majority of CubeSats, like larger satellites, use solar panels as its primary source of energy. Solar panels can be stored easily, stacking up against each other before they are deployed. Solar panels also provide a sustainable source of power, relying on the Sun as a main source of power. Solar panels can also be integrated with batteries, charging them and utilizing them when the CubeSat is not in direct sunlight. They also are ideal for longer missions due to their resistance to the conditions of LEO. Solar panels can also be scaled up for larger CubeSats. Another advantage of solar panels is that they are proven as a reliable power source. They are the industry standard for satellites and many are available as off the shelf components.

Solar panels can also be integrated with batteries. Batteries can provide more power output, supporting higher demand systems. Batteries allow for the CubeSat to continue in full operation during eclipse periods where the CubeSat is not facing the Sun for the solar panels to function. Batteries can provide power reliability and redundancy, allowing the CubeSat to function for an extended period of time if the solar panels are to fail. They are also compact and lightweight, two ideal metrics for CubeSats. Including extra batteries for redundancy allows for extra operational time if the solar panels fail. Similar to solar panels, there are many off the shelf batteries. Lithium ion batteries would be an ideal choice as it can provide the CubeSat with the power it needs while having little drawbacks.

6.2 Material Choices

An aluminum alloy is an ideal material for the structure of the CubeSat. Aluminum is commonly used because of its lightweight but strong properties. The lightweight property is crucial for CubeSats to keep launch costs down. The strong properties provide excellent stability, which is key to ensuring hardware on board is not damaged from launch vibrations. It is also corrosion resistant due the nature of aluminum forming an oxide layer. Aluminum has the benefit of being easy to machine, being able to be constructed through CNC machining, 3D printing, or milling, providing many options and reducing production costs.

Aluminum is also more appealing for CubeSats due to their availability. Aluminum is a common material and it is readily available. Manufacturing methods are cost effective, giving it a nice performance to cost ratio. It also stands out more for this project as the focus is to use more off the shelf components to keep the production costs low.

6.3 Reaction Wheel Selection

The ideal choice of reaction wheel would be the AAC Clyde Space RW222 or the RW400. These two reaction wheels were designed with the intention of being used in a CubeSat. The RW222 is compact, offering a low mass and small footprint. On top of the small size, it offers a low power consumption. Even with its small size and power consumption, it can still output an adequate amount of torque to control a CubeSat. The RW400 is the bigger brother of the RW222. It is ideal for larger CubeSats, usually 6U to 12U The larger momentum capabilities of the RW400 are ideal for larger satellites.

Table 6.1 Comparison Between RW222 and RW400 [29] [30]

Model	RW222	RW400
Total Momentum Storage (mN.m.s)	+/- 3.0; +/- 6.0	+/- 15; +/- 30; +/- 50
Maximum Torque (nM.m)	+/- 2	+/- 8
Maximum Rotation Rate (rpm)	10000 / 15000	5000
Control Accuracy (rpm)	+/- 0.5	+/- 1
Outer Dimension (mm)	25 x 25 x15	50 x 50 x 27
Mass	TBC	197 / 210 / 375
Operating Temperature (°C)	-20 to +60	-25 to + 60
Radiation Tolerance (krad(Si))	> 36	> 36

Figure 6.1 AAC Clyde Space RW222 [29]

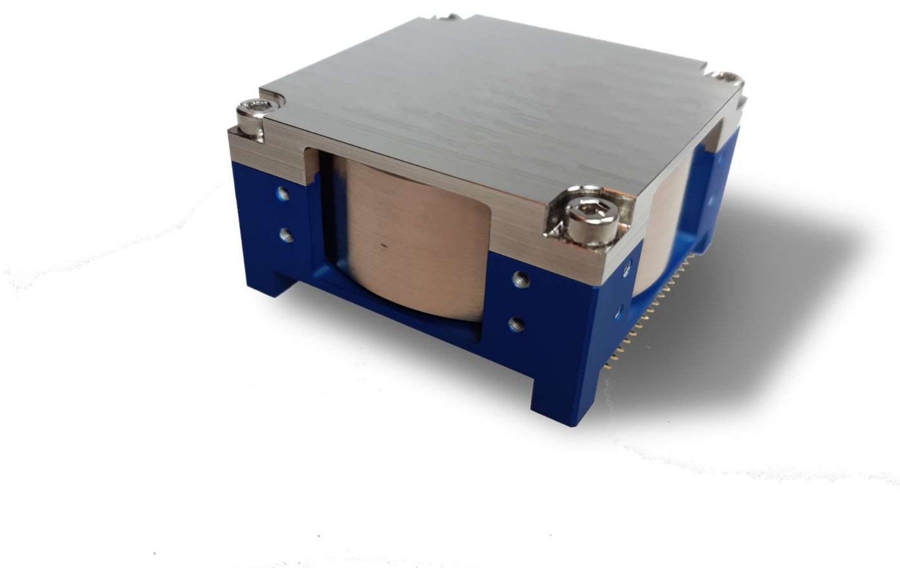

Figure 6.2 AAC Clyde Space RW400 [30]

Table 6.2 RW222 Electrical Specifications [29]

	Minimum	Typical	Maximum
Supply Voltage (V)	3.25	3.3	3.5
Bus Logic Level Voltage (V)	3.3 - 5.1	3.3 - 5.1	3.3 - 5.1

Table 6.3 RW400 Electrical Specifications and Power Consumption [30]

	Minimum	Typical	Maximum
Supply Voltage (V)	4.5	5.0	5.25
Logic Supply Voltage (V)	2.3	3.3	5.1
Bus Logic Level Voltage (V)	3.3 - 5.1	3.3 - 5.1	3.3 - 5.1
Idle Power Consumption (W)	-	-	0.075
Nominal Power Consumption (W)	-	1	-
Peak Power Consumption (W)	-	11.9	15

Since this project is not about a specific mission, the goal is to have the ADCS take up as little space as possible. With this in mind, the RW222 is the ideal choice with its smaller footprint and lower power consumption. Given that the RW222 has a dimension of 25 x 25 x 17 mm taking up 10,625 mm^3, or 10.625 cm^3, in volume, four of them will take up 42,500 mm^3, or 42.5 cm^3, in volume. A 1U CubeSat, with a volume of 1000 cm^3, can house four of these reaction wheels. The RW222 leaves plenty of space for the rest of the subsystems. Increasing the CubeSat to a 2U or a 3U will allow for more flexibility between the other subsystems. The RW222 also clears the momentum requirement defined above.

6.4 Controller

A control system requires the use of a controller. The controller takes the input of the photodiodes and other sensors to determine the attitude of the CubeSat. Using that information, it then sends commands to the reaction wheels to control the cubesat. Utilizing off the shelf controllers can keep the overall cost of the CubeSat down. When selecting a controller, there are a few things to consider:

- Size and weight
- Power consumption
- Radiation hardening

Table 6.4 Controllers with Specifications [31] [32] [33] [34] [35] [36] [37]

Controller	Processor	Radiation-Hardened	Power Consumption	Dimension (mm)	Weight (g)
STM32F4/F7	ARM Cortex-M4/M7	No	<1 W	28 x 28	10 - 20
TI MSP430FR series	16-bit RISC	No	<0.1 W	68 x 53	20 - 30
Microchip SAMRH71	ARM Cortex-M7	Yes	<0.5 W	95 x 75	30 - 50
GomSpace NanoMind A3200	ARM Cortex-M7	Yes	1 W	50 x 10	45
Raspberry Pi Compute Module 4	ARM Cortex-A72	No	3-5 W	55 x 40	8
NVIDIA Jetson TX2/TX2i	ARM Cortex-A57 + GPU	No	7-15 W	100 x 87	300
Cobham LEON3	SPARC V8	Yes	1 W	42 x 42	100
GomSpace NanoMind Z7000	Dual-core ARM Cortex-A9	Yes	1 W	58 x 58	50

The table aboveshows common off the shelf controllers that can be utilized as a controller for the subsystem. The dimension and weight shown are for the controller itself and the usual development board that accompanies it. An Arduino Uno is an ideal controller for this system. It has a small footprint, being 68.6 mm in length, 53.4 mm in width, and has a low weight of 25 grams. It also has a lower power consumption of 5 volts, making it efficient for space use as power is limited. The Arduino Uno was not built for space use, lacking important radiation resistance. Given this choice, an external source of radiation protection will be required. Aluminum shielding is an optimal choice, given that the structure is also going to be made from aluminum.

7. Controls

7.1 Attitude Determination

Attitude determination involves sensor input and previous knowledge of the CubeSat's dynamics. As mentioned before, photodiodes will be utilized for attitude determination. When exposed to a light source, like the Sun, the photodiodes will generate a current. That current is proportional to the intensity of the amount of light hitting the diode. It can also determine attitude based on the angle of the incoming light when referencing the position of the Sun. Angular momentum can also be determined by having multiple photodiodes placed in strategic locations around a satellite. As the satellite rotates while in orbit, the difference in the current produced from the photodiodes can determine the angle at which the light is hitting each diode.

7.2 Control System Design and Components

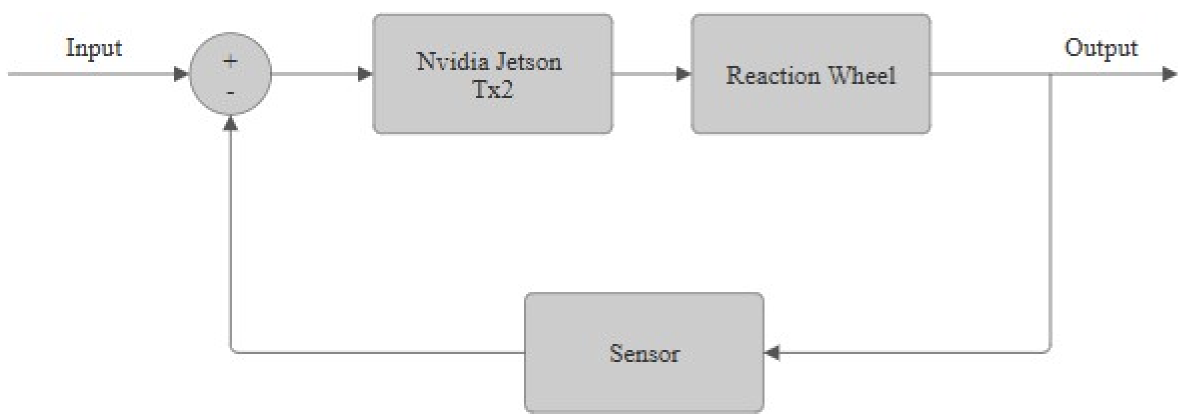

Figure 7.1 Control Block Diagram

The system will take the photodiode current as an input for the system. The Nvidia Jetson Tx2 will act as the controller of the system, taking the input to determine the attitude of the CubeSat. It then sends a command to the reaction wheels to control the CubeSat and reorient to its desired position. The output of the system will be the new orientation of the CubeSat. The output is also fed back into the system through a sensor to make adjustments as needed to correct any deviations from the output attitude compared to the desired attitude.

7.3 Active Method: Reaction Wheel Control

There are two types of control methods for CubeSats: active and passive. An active control method requires energy input to stabilize the system. Some examples of active control are thrusters, reaction wheels, and gyroscopes. As mentioned before, reaction wheels are a great choice for CubeSats due to their ability to control three axes without needing a lot of real estate. Adding a fourth reaction wheel as a redundant wheel is also an option due to their small footprint, easily fitting into smaller CubeSats.

The momentum of a system utilizing reaction wheels can be defined by the follow equation:

$$m = m_o + \sum_{i=1}^{n} m_i \tag{7.X}$$

Where:

- m: the mass of the entire CubeSat (mass of vehicle and the reaction wheels)
- m_o: mass of the body
- m_i: mass of the i-th wheel

Angular momentum of the body with respect to the center of mass can be represented by:

$$\vec{H}_G^{body} = I_G^{body}\,\vec{\omega} \tag{7.X}$$

Total angular momentum of the system about the center of mass can be represented by:

$$\vec{H}_G = [I_G^{body} + \sum_{i=1}^{n} I_G^{(i)}]\,\vec{\omega} + \sum_{i=1}^{n} I_{Gi}^{(i)}\,\vec{\omega}_{rel}^{(i)} \tag{7.X}$$

7.4 Passive Method: Gravity-Gradient Stabilization

Passive control does not require any energy to stabilize the system. A couple of examples of passive control methods are magnetorquers, gravity-gradient stabilization, drag stabilization, solar radiation stabilization, and mass shifting. One of the main things affecting the attitude of a satellite is gravity. Earth is constantly trying to pull these satellites back to the ground. The satellite's centrifugal force is keeping it orbiting around the Earth. The two balance each other out, keeping the satellite orbiting the Earth without falling to the ground or being sent off into space. This can negatively affect the attitude of the satellite due the forces wanting to the opposite, shifting the satellite ever so slightly from its intended orientation. Gravity-gradient stabilization is an optimal method to stabilize the satellite. It also has the added benefit of not needing to add any hardware to the already limited space available in a CubeSat.

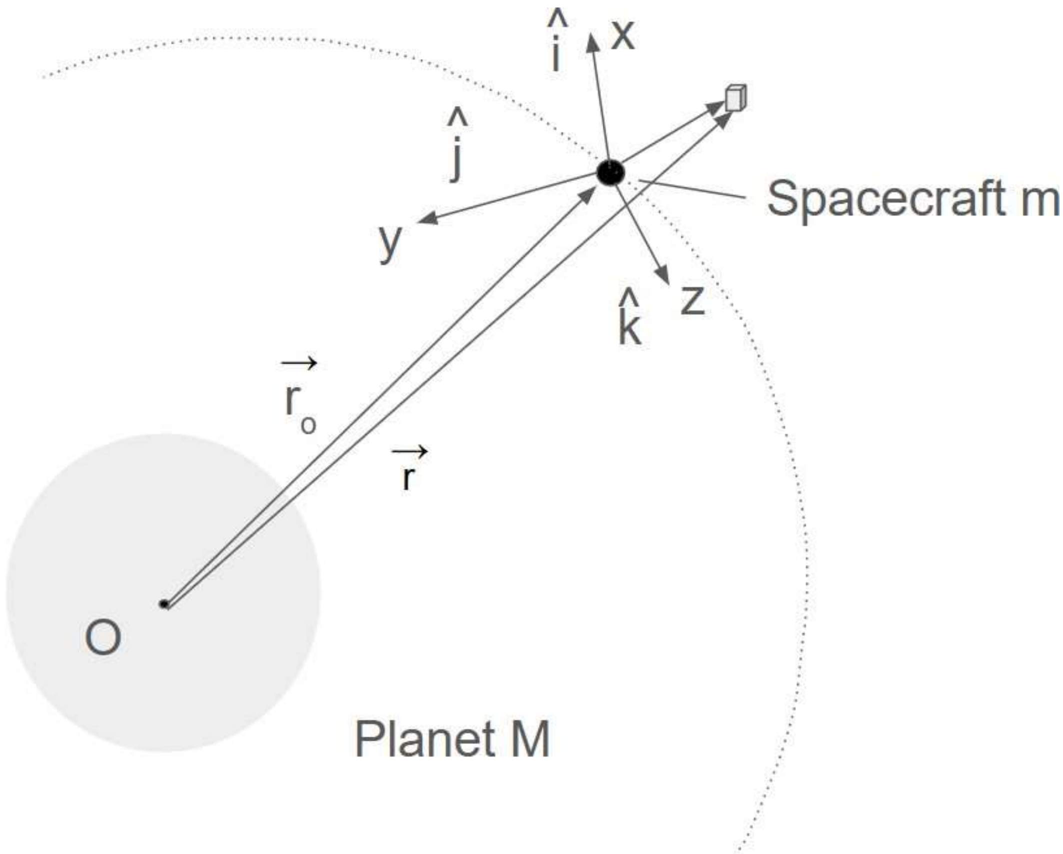

Figure 7.2 FBD of a Spacecraft Orbiting a Plane M

Assuming a circular orbit, the force of gravity acting on a satellite with a mass of dm given the free body diagram shown in figure 7.2 can be described by the equation below:

$$\vec{dF}_g = -G\left(\frac{Mdm}{r^2}\right)\left(\frac{\vec{r}}{r}\right) \tag{7.1}$$

Where G is the universal gravitational constant of $6.674*10^{-11}$ m^3/kgs^2, r is radius of the circular orbit, and $\vec{r}$ is the position vector of the satellite from the Earth's center and the satellite's mass center.

The net moment of gravitational force about the center of mass of a satellite can be described by the equation below:

$$\vec{M}_{GNet} = \int_m \vec{\rho} \times \vec{dF}_g \tag{7.2}$$

Breaking down the right hand side:

$$\vec{\rho} \times d\vec{F}_g = \vec{\rho} \times \left(-\mu \frac{\vec{r}}{r^3}\, dm\right) \tag{7.3}$$

Given $\vec{r} = \vec{r}_o + \vec{\rho}$:

$$\vec{\rho} \times d\vec{F}_g = -\mu \frac{dm}{r^3} [\vec{\rho} \times (\vec{r}_o + \vec{\rho})] \tag{7.4}$$

$\vec{r}_o$ and $\vec{\rho}$ can also be written as:

$$\vec{r}_o = r_{ox}\hat{i} + r_{oy}\hat{j} + r_{oz}\hat{k} \tag{7.5}$$

$$\vec{\rho} = x\hat{i} + y\hat{j} + z\hat{k} \tag{7.6}$$

Substituting (7.5) and (7.6) into (7.4), we get:

$$\vec{\rho} \times d\vec{F}g = \hat{i}\left[-\mu\frac{dm}{r^3}(r_{oz}y - r_{oy}z)\right] + \hat{j}\left[-\mu\frac{dm}{r^3}(r_{ox}z - r_{oz}x)\right] + \hat{k}\left[-\mu\frac{dm}{r^3}(r_{oy}x - r_{ox}y)\right] \tag{7.7}$$

Since $|\vec{\rho}| < |\vec{r}|$, we can use the following approximation:

$$\frac{1}{r^3} = \frac{1}{r_o^3} - \frac{3}{r_o^5}\vec{r}_o \bullet \vec{\rho} \tag{7.8}$$

We then get:

$$\int_m \frac{x}{r^3}\, dm = \frac{-3r_{ox}}{r_0^5}\int_m x^2\, dm \tag{7.9}$$

$$\int_m \frac{x}{r^3}\, dm = \frac{-3r_{oy}}{r_0^5}\int_m y^2\, dm \tag{7.10}$$

$$\int_m \frac{x}{r^3}\, dm = \frac{-3r_{oz}}{r_0^5}\int_m z^2\, dm \tag{7.11}$$

Converting to Euler's rotational equations of motion with gravitational moments, we get:

$$\hat{i}: A\dot{\omega}_x + (C - B)\omega_y\omega_z = \frac{3\mu r_{oy} r_{oz}}{r_o^5}(C - B) \tag{7.12}$$

$$\hat{j}: B\dot{\omega}_y + (A - C)\omega_x\omega_z = \frac{3\mu r_{ox} r_{oz}}{r_o^5}(A - C) \tag{7.13}$$

$$\hat{k}: C\dot{\omega}_z + (B - A)\omega_x\omega_y = \frac{3\mu r_{ox} r_{oy}}{r_o^5}(B - A) \tag{7.14}$$

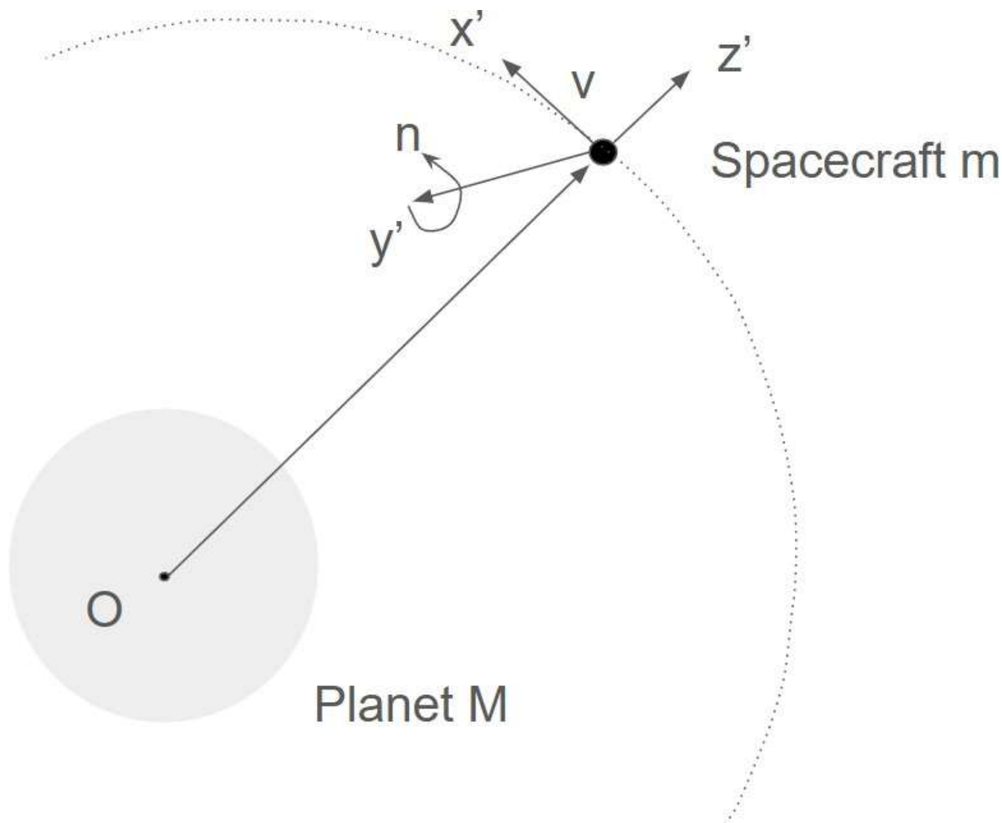

Figure 7.3 FBD with x', y', and z'

Given the free body diagram in figure 7.3, $\omega_{y'} = n$ and $\omega_{x'} = \omega_{'} = 0$. The position vector can also be defined by the equation below:

$$\vec{r}_o = \vec{r}_{ox}\hat{i} + \vec{r}_{oy}\hat{j} + \vec{r}_{oz}\hat{k} = 0\vec{i'} + 0\vec{j'} + r_{oz}\vec{k'} \tag{7.15}$$

Given equation 7.15, equations 7.12, 7.13, and 7.14 can be simplified to the following:

$$\dot{\omega}_x = 0 \tag{7.16}$$

$$\dot{\omega}_y = 0 \tag{7.17}$$

$$\dot{\omega}_z = 0 \tag{7.18}$$

This indicates the spacecraft is at an equilibrium, and if this equilibrium is stable, the spacecraft will be considered to be stabilized through gravity-gradient.

8. Conclusion

This project showed a conceptual design of an ADCS subsystem for a CubeSat using off the shelf parts. General design requirements are shown to allow flexibility in adjusting the design to fit missions specific goals or to adjust based on the hardware of other subsystems. The ADCS subsystem uses photodiodes for attitude determination, and displaying two methods for control: passive and active. It can passively control itself through gravity-gradient, but it has certain conditions it needs to meet before being stable. It can actively control itself using reaction wheels, but it will require additional power. Utilizing two control methods for attitude control gives the CubeSat some more reliability, allowing for one system to take over if the other fails. This project shows the design requirements needed for the CubeSat to be stable by gravity-gradient. The reaction wheel selected in this project is the Clyde Space RW222, allowing four of them for 3-axis control and a redundant wheel to fit comfortably in a 3U CubeSat.

The next step will be to simulate a CubeSat in a low earth orbit. Simulating will provide the advantage of doing it quicker and more cost effective. A light source, like a light bulb, will emulate the Sun. A structure can be designed and fabricated to act as the CubeSat. The layout of the photodiodes will get readings from the light bulb, thus giving the attitude of the CubeSat. A program will read the data gathered from the photodiodes and translate it into the attitude of the CubeSat. Then the program will send a command to the reaction wheels to correct its orientation, if needed. Once the system shows it can control the CubeSat, the GNC subsystem can be implemented with the rest of the subsystems required for a CubeSat. Given the modularity of CubeSats, the subsystem can be scaled up easily, utilizing the larger and more powerful Clyde Space RW400.

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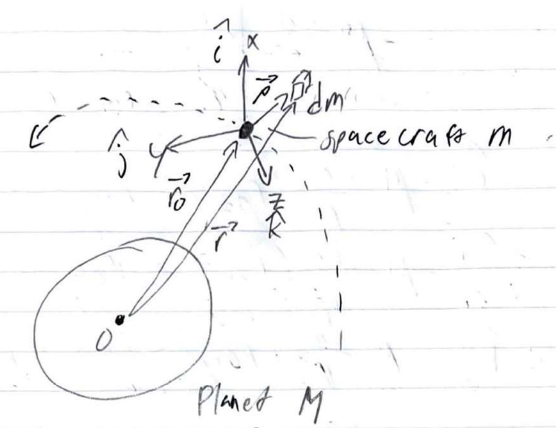
Appendix

Derivation for gravity-gradient stability



$$d\vec{F_g} = -G\frac{M\,dm}{r^2}\frac{\vec{r}}{r} \qquad \mu = GM$$

$$d\vec{F_g} = -\mu\frac{\vec{r}}{r^3}dm$$

$$\vec{M}_{G_{net}} = \int_m \vec{\rho}\times d\vec{F_g}$$

Let

$$\vec{r_0} = r_{0x}\hat{i} + r_{0y}\hat{j} + r_{0z}\hat{k}$$

$$\vec{\rho} = x\hat{i} + y\hat{j} + z\hat{k}$$

$$\vec{\rho}\times d\vec{F_g} = \vec{\rho}\times\left(-\mu\frac{\vec{r}}{r^3}dm\right)$$

$$= -\mu\frac{dm}{r^3}(\vec{\rho}\times\vec{r})$$

$$= -\mu\frac{dm}{r^3}\left[\vec{\rho}\times(\vec{r_0}+\vec{\rho})\right]$$

$$= -\mu\frac{dm}{r^3}(\vec{\rho}\times\vec{r_0})$$

$$= -\mu \frac{dm}{r^3} \begin{vmatrix} \hat{i} & \hat{j} & \hat{k} \\ x & y & z \\ r_{0x} & r_{0y} & r_{0z} \end{vmatrix}$$

$$\vec{\rho} \times d\vec{F_g} = \hat{i}\left[-\mu \frac{dm}{r^3}(r_{0z}y - r_{0y}z)\right]$$
$$+ \hat{j}\left[-\mu \frac{dm}{r^3}(r_{0x}z - r_{0z}x)\right]$$
$$+ \hat{k}\left[-\mu \frac{dm}{r^3}(r_{0y}x - r_{0x}y)\right]$$

$$\vec{M}_{G_{net}} = \int_m \vec{\rho} \times d\vec{F_g}$$
$$= \hat{i}\left(-\mu r_{0z}\int_m \frac{y}{r^3}\,dm + \mu r_{0y}\int_m \frac{z}{r^3}\,dm\right)$$
$$+ \hat{j}\left(-\mu r_{0x}\int_m \frac{z}{r^3}\,dm + \mu r_{0z}\int_m \frac{x}{r^3}\,dm\right)$$
$$+ \hat{k}\left(-\mu r_{0y}\int_m \frac{x}{r^3}\,dm + \mu r_{0x}\int_m \frac{y}{r^3}\,dm\right)$$

$|\vec{\rho}| << |\vec{r_0}|$

$$\hookrightarrow \frac{1}{r^3} \approx \frac{1}{r_0^3} - \frac{3}{r_0^5}\vec{r_0} \cdot \vec{\rho}$$

$$x\left(\frac{1}{r^3}\right) \approx x\left(\frac{1}{r_0^3} - \frac{3}{r_0^5}\vec{r_0} \cdot \vec{\rho}\right)$$

$$\int_m x\left(\frac{1}{r^3}\right)dm \approx \int_m \left(\frac{x}{r_0^3} - \frac{3r_{0x}}{r_0^5}x^2 - \frac{3r_{0y}}{r_0^5}xy - \frac{3r_{0z}}{r_0^5}xz\right)dm$$

$$\approx \frac{1}{r_0^3}\cancel{\int_m x\,dm}^{\,0} - \frac{3r_{0x}}{r_0^5}\int_m x^2\,dm$$
$$- \frac{3r_{0y}}{r_0^5}\underbrace{\cancel{\int_m xy\,dm}}_{0} - \frac{3r_{0z}}{r_0^5}\underbrace{\cancel{\int_m xz\,dm}}_{0}$$

$$\int_m \frac{x}{r^3}\,dm \approx \frac{-3\,r_{0x}}{r_0^5}\int_m x^2\,dm$$

$$\int_m \frac{y}{r^3}\,dm \approx \frac{-3\,r_{0y}}{r_0^5}\int_m y^2\,dm$$

$$\int_m \frac{z}{r^3}\,dm \approx \frac{-3\,r_{0z}}{r_0^5}\int_m z^2\,dm$$

$$\hat{i}: M_{Gnet_x} = \frac{3\mu\, r_{0y}\, r_{0z}}{r_0^5}\left[\int_m y^2\,dm - \int_m z^2\,dm\right]$$

$$\hat{j}: M_{Gnet_y} = \frac{3\mu\, r_{0x}\, r_{0z}}{r_0^5}\left[\int_m z^2\,dm - \int_m x^2\,dm\right]$$

$$\hat{k}: M_{Gnet_z} = \frac{3\mu\, r_{0x}\, r_{0y}}{r_0^5}\left[\int_m x^2\,dm - \int_m y^2\,dm\right]$$

Principle Moments of Inertia:

$$A = \int_m y^2\,dm + \int_m z^2\,dm$$

$$B = \int_m x^2\,dm + \int_m z^2\,dm$$

$$C = \int_m x^2\,dm + \int_m y^2\,dm$$

$\hat{i}: M_{G_{net\,x}} = \dfrac{3\mu\, r_{0y} r_{0z}}{r_0^5}(C-B)$

$\hat{j}: M_{G_{net\,y}} = \dfrac{3\mu\, r_{0x} r_{0z}}{r_0^5}(A-C)$

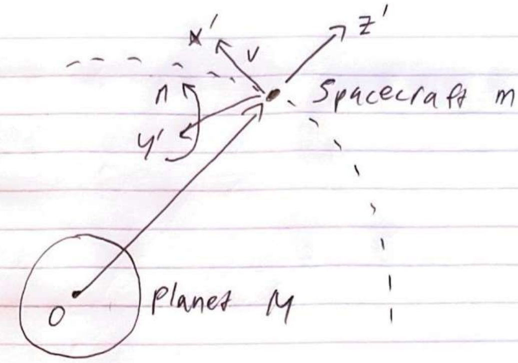
$\hat{k}: M_{G_{net\,z}} = \dfrac{3\mu\, r_{0x} r_{0y}}{r_0^5}(B-A)$

Combine w/ Euler's rotational EoM

$\hat{i}: A\dot{\omega}_x + (C-B)\,\omega_y \omega_z = \dfrac{3\mu\, r_{0y} r_{0z}}{r_0^5}(C-B)$

$\hat{j}: B\dot{\omega}_y + (A-C)\,\omega_x \omega_z = \dfrac{3\mu\, r_{0x} r_{0y}}{r_0^5}(A-C)$

$\hat{k}: C\dot{\omega}_z + (B-A)\,\omega_x \omega_y = \dfrac{3\mu\, r_{0x} r_{0y}}{r_0^5}(B-A)$



$$\begin{cases} \omega_{y'} = n \\ \omega_{x'} = \omega_{z'} = 0 \end{cases}$$

$$\vec{r}_0 = r_{0x}\hat{i} + r_{0y}\hat{j} + r_{0z}\hat{k}$$
$$= 0\,\hat{i}' + 0\,\hat{j}' + r_{0z}\hat{k}'$$

$$A\dot{\omega}_x + (C-B)\,\omega_y \cancel{\omega_z}^{0} = \frac{3\mu \cancel{r_{0y}}^{0} r_{0z}}{r_0^5}(C-B) = 0$$

$$B\dot{\omega}_y + (A-C)\,\cancel{\omega_x}^{0}\cancel{\omega_z}^{0} = \frac{3\mu \cancel{r_{0x}} r_{0z}}{r_0^5}(A-C) = 0$$

$$C\dot{\omega}_z + (B-A)\,\cancel{\omega_x}^{0}\omega_y = \frac{3\mu \cancel{r_{0x}}^{0}\cancel{r_{0y}}^{0}}{r_0^5}(B-A) = 0$$

$$\begin{matrix} A \neq 0 \\ B \neq 0 \\ C \neq 0 \end{matrix} \quad \Rightarrow \quad \begin{matrix} \dot{\omega}_x = 0 \\ \dot{\omega}_y = 0 \\ \dot{\omega}_z = 0 \end{matrix}$$