Preliminary Design and Prototype for a 1P PocketQube Earth Observation Satellite

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***Abstract*—This paper describes a proposed design for a 1P PocketQube Earth observation satellite being developed by undergraduate students at Wentworth Institute of Technology. This satellite will push boundaries in satellite miniaturization, image sensors, and commercial off-the-shelf (COTS) components. By providing a new set of validated PocketQube hardware, future missions should carry less risk. Special care was taken to design measures to prevent, mitigate, or recover from the effects of ionizing radiation. An attitude determination and control system (ADCS) using embedded magnetorquer was designed to give authority over the pointing of the camera payload; allowing for more frequent imaging and live adjustments of image target.**

***Keywords*—PocketQube, picosatellite, nanosatellite, earth photography, STM32, satellite, aerospace, magnetorquer, CubeSat**

I. Introduction

Earth observation by space satellites has demonstrated numerous advancements to our society, improving safety, environmental awareness, and disaster management. Recently, there has been a significant increase in small satellite launches that inherit the responsibilities of older, larger satellites [1]. Improvements to electrical components now allow entire satellite subsystems to be miniaturized while maintaining the specification of previous designs. Consequently, small satellites such as CubeSats have exploded in popularity in both education and commercial sectors due to their inexpensive development and launch costs [2]. Since 2018, the PocketQube formfactor has become the new standard for satellites weighing less than 250 grams. As such, universities and industry have begun preliminary research into advanced mission profiles of PocketQubes.

The first camera-equipped PocketQube was designed by forecasting company MyRadar (Orlando, FL, USA) and launched by AlbaOrbital (Glasgow, United Kingdom) in 2022 [3]. This satellite has successfully downlinked an image from Low Earth Orbit (LEO), proving that cameras are no longer limited to CubeSats and larger satellites. Students at Wentworth Institute of Technology (Boston, MA, USA) are proposing a new PocketQube design with a similar mission profile to MyRadar-1 but as an open-source project. If successful, this new satellite design has the potential to be the smallest documented operational satellite with a camera aboard.

Satellite developers are also commonly looking to validate flight hardware for use on future missions. Thus, first-generation satellites with unproven hardware designs carry the most risk of in-orbit failure. A purpose of our project is to design and test prototype designs. Once completed and successfully tested in space, a new wave of PocketQubes could be developed using similar hardware with less risk. The satellite design proposed in this paper (Fig. 1.) is an in-progress, first-generation satellite herein referred to as Wentworth PocketQube.

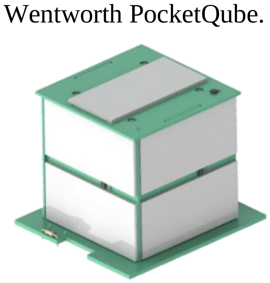

Figure 1: 3D Model of the Wentworth PocketQube, which adheres to the PocketQube Standard.

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II. Background

A. 'PocketQube' Standard

In small satellite design, a CubeSat unit is defined as 10x10x10cm, called "**1U**". Similarly, a PocketQube unit is defined as 5x5x5cm, called "**1P**". Multiple units can be stacked to form larger satellites if the mission profile of the satellite requires it.

The PocketQube Standard is a definition of physical constraints, particularly at the base plate of the satellite that mounts with the deployment spacecraft [4]. By adhering to this standard, the Wentworth PocketQube can take advantage of commercial launch opportunities and deployers [5], rather than developing a new deployment platform from scratch.

B. Nepal-PQ1

The Nepal-PQ1 was a training satellite designed by ORION Space Ltd. Nepal that served as an educational platform for picosatellite design [6]. The team at Wentworth received initial remote training from ORION Space that set a baseline for satellite functionality. The company provided Wentworth with their Nepal-PQ1 training prototype, a modular composition of printed circuit boards (PCBs) that provided essential foundational knowledge to this team.

C. Single Upset Event and Radiation Events

Spacecraft in low earth orbit (LEO) are exposed to significantly more radiation compared to on the earth's surface. Electronics bombarded with this radiation experience a number of single event effects (SEE) due to ionizing radiation particles traveling through their physical structure. These events are characterized as: single event upset (SEU), single event latchup (SEL), single event burnout (SEB), and single event gate rupture (SEGR) [7][8]. SEU occurs when a single ionizing particle strikes a sensitive element of a circuit and causes a state change. In logic devices, like microcontrollers, central processing units (CPUs), or memory, this can cause a "bit flip"; wherein the value of a register or memory cell randomly changes states. This event can be detrimental to the predictable operation of computer systems and oftentimes causes crashes or hangs. Fortunately, SEUs rarely cause permanent damage to the physical structure of the device. SEL, however, has the potential to develop potentially catastrophic damage. When SEL occurs, low resistance paths are developed between the power supply and ground, resulting in a short circuit. These short circuit conditions can cause silicon regions, copper traces, or bond wires in ICs to melt or vaporize.

Without careful design choices, these events can cripple or destroy satellite electronics. The proposed satellite design will focus on mitigating the effects of or recovering from SEU and SEL, as these are the most common in the radiation environment of LEO [8].

III. Mission Overview

The following list of mission objectives has framed the prototype design:

1. Establish uplink and downlink capabilities.
2. Monitor system for anomalies.
3. Take a photograph of Earth, store in memory.
4. Downlink preview image
5. If preview is acceptable, downlink high-resolution image.
6. Collect data of component performance.

Most small satellites are launched alongside other satellites on a commercial ride-share mission, or on a small rocket [9]. Companies such as SpaceX (Hawthorne, CA, US), RocketLab (Long Beach, CA, USA), and Arianespace (Évry-Courcouronnes, France) are competing to launch small satellites, which has resulted in decreasing launch costs in recent years. This satellite is expected to target a 500~600 km, sun-synchronous orbit as this is widely available from commercial PocketQube deployers [2].

Once in orbit, the primary mission of this 1P PocketQube is to photograph Earth from space and downlink the resulting image back to Earth for analysis. There appears to be little prior reports of research for performing this mission in a 1P formfactor. Therefore, the data acquired from this mission has the potential to springboard future research in the miniaturization of earth observation satellites.

The primary payload of this mission is expected to be a set of two 5MP (megapixel) image sensors, each supporting a maximum output resolution of 2592x1944 pixels, or a maximum file size of approximately 14.33MB (megabyte) uncompressed. However, image compression will be implemented to reduce both file size and image transmission length. Similar to a modern smartphone, a pairing of multiple image sensors has two distinct advantages: lens placement and redundancy. This satellite will leverage a telephoto lens, suited for ideal ground-scan distance, and a wide-angle lens, suited for ideal field-of-view. Together, this payload is expected to require robust supporting infrastructure to operate correctly. The preliminary system design is outlined in Section IV.

IV. System Design

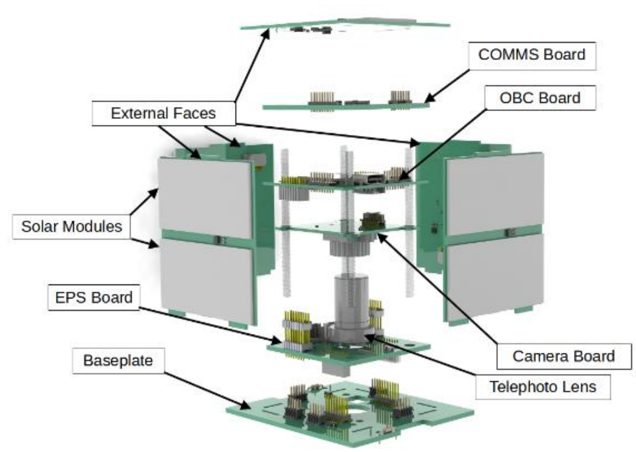

Figure 2: Exploded and Annotated View of Wentworth PocketQube

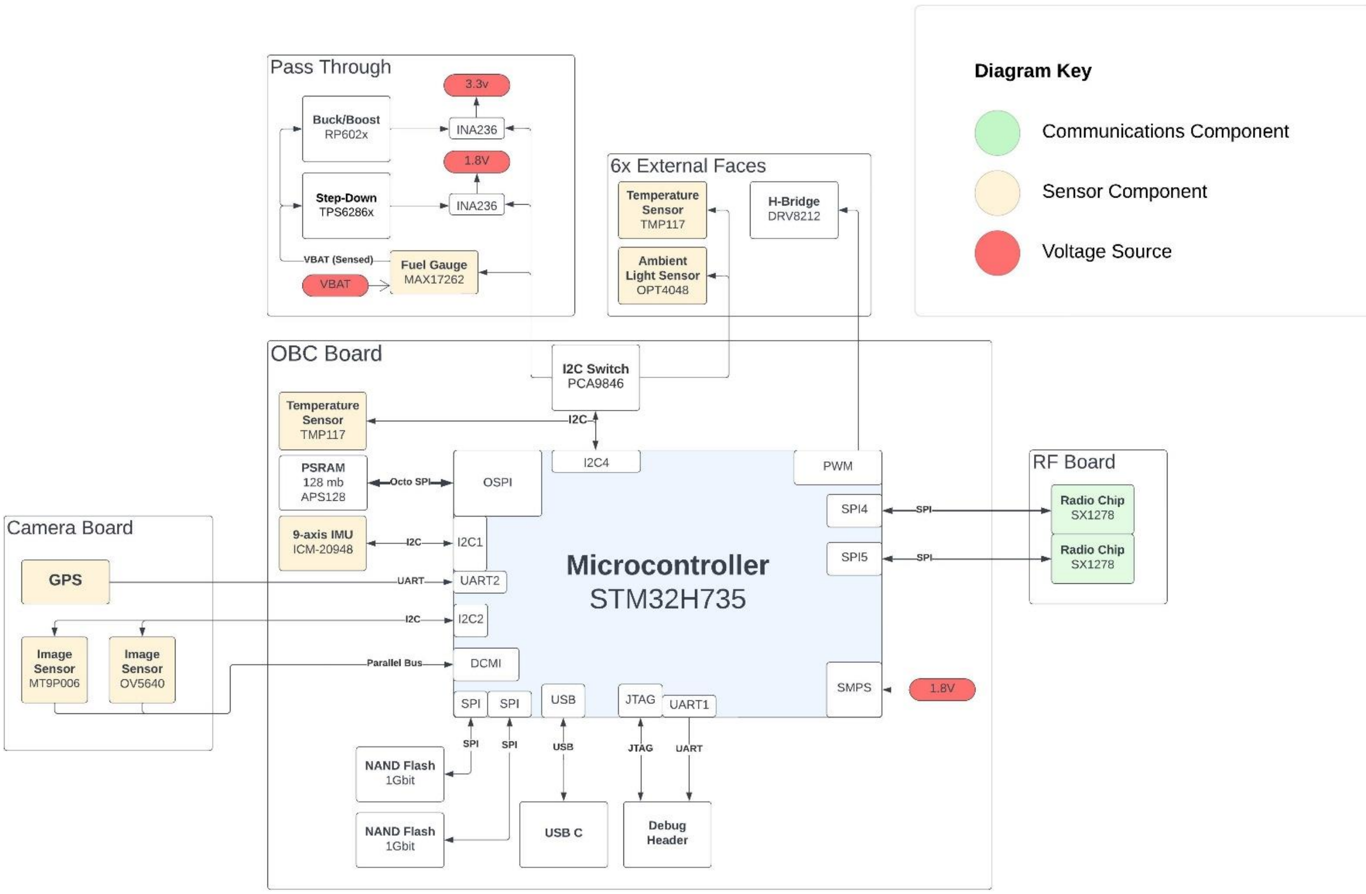

Figure 3: Block Diagram of Satellite

A. *On-Board Computer*

The on-board computer (OBC) (Fig. 3. and 4.) is centered around the STM32H723 microcontroller from ST Microelectronics (Geneva, Switzerland). This microcontroller contains an Arm® Cortex®-M7 core and a vast array of highly configurable peripherals. It is targeted for use by a variety of commercial and industrial customers. This chip is not manufactured using a radiation-hardened process node, which makes it susceptible to radiation upset events, but significantly cheaper; and thus within range of use by smaller groups. While the STM32H723 is not radiation hardened by process node, it does include a number of hardware features that significantly help mitigate, or at least detect and correct, the effect of radiation upset events. One of these features is error correcting code (ECC) memory protection on select random access memory (RAM) and flash memories using a single event correcting, double event detecting (SEC-DED) algorithm based on Hamming principles [17]. This algorithm is capable of correcting a single-bit error and of detecting two-bit errors. On the microcontroller this ECC implementation is available on the CPU cache, TCM, DTCM, most SRAM banks, and several flash banks. This should provide robust protection against SEU when implemented in software, ensuring predictable computer operation in the radiation environment of LEO [8]. The external SRAM that was selected does not implement this hardware level ECC algorithm and is therefore susceptible to SEUs. Therefore, only non-critical data will be stored on external SRAM and a hash code or cycle redundancy check (CRC) will be employed to check for data corruption.

The OBC will be responsible for making all command-and-control decisions aboard the satellite. To gather data for these decisions the OBC board needs several sensors. Temperature sensors will be placed on each internal and external board, allowing the OBC to switch on/off or disregard readings from components that are outside of their operational temperature range. Current sensors will be placed on each main voltage rail to monitor these components for any abnormal current consumption. If an external component goes into a deadlock, potentially caused by radiation, it may consume an abnormally large amount of current [8]. Once the OBC detects this it will hopefully be able to reset the components and fix the issue [16]. Current sensors will also allow the OBC to measure how much power is flowing into the battery, enabling better planning for future power intensive actions. Similar to this, the OBC design includes an advanced fuel gauge integrated circuit (IC) designed for lithium-ion (li-ion) batteries. This fuel gauge is capable of translating the non-linear voltage curve of a li-ion battery into a simple state of charge (SOC) percentage. This component will be continually active, making constant refinements to its measurement of SOC. The fuel gauge will also keep track of battery health, measuring how many times the battery has been cycled and estimating the degradation in capacity over time. This will help the OBC manage power consumption as the satellite ages.

The OBC will also have access to 6x4-channel Lux sensors. Data gathered from these sensors will feed into a sensor fusion algorithm for determining the satellites attitude with respect to the orientation of the sun. This data will also help OBC characterize the performance of the solar panels on each face and track how they degrade over time or under different temperature conditions. Another integral piece of attitude determination is the magnetometer. OBC is designed with a IC, ICM-20948, that integrates a magnetometer, gyroscope, and accelerometer all in one; saving space and power. At every point in LEO the geomagnetic field direction is roughly known. By comparing readings taken from the magnetometer to the known field directions in the satellite's orbit, attitude can be determined. One piece of data both of these attitude determination methods require precise data regarding the satellite's location along its orbital track at any given moment. This information could be obtained in a number of ways, for instance: dead reckoning from the moment of separation, calculated on the ground from data from the ascent vehicle and uploaded to the satellite later, or via utilizing GPS. For this design utilizing GPS was selected. This gives the satellite an advantage in the accuracy of determining it's orbit, as well as being able to track the inevitable decay of the orbit. The cost of using GPS is more power consumption, complexity, and space for components. However, the benefits of using GPS appeared worth the cost, as the goal of earth observation necessitates precise attitude control.

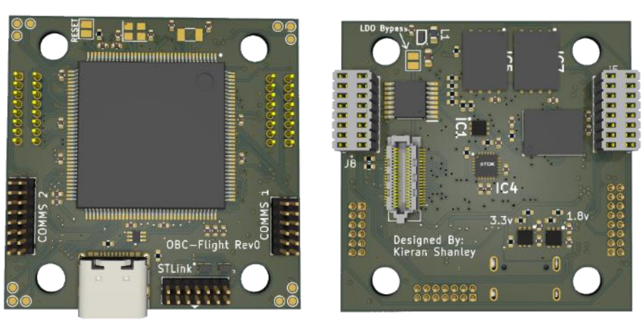

Figure 4: 3D model of OBC Top and Bottom

B. *Electrical Power System and Integration*

The Electrical power system (EPS) for this satellite is responsible for the solar energy harvesting, battery charging optimization, onboard voltage conversions, and on-board electrical protection.

The solar energy harvesting system is composed of two different types of solar cells, a main and a secondary, both monocrystalline in substrate makeup. The main solar cells are 45x22mm, and can fit two on each side face, and on the top face (none on baseplate). At its typical 25% efficiency, these generate 3.35V and 55.1mA when facing the sun on earths surface. This figure being generated on earth's surface is significant because the solar power in LEO is notably greater then that on the ground (1367 W/m^2 versus 1000 W/m^2, respectively). The secondary solar cells are much smaller, at 28% efficiency, these cells can squeeze into some left over space on the baseplate and top face that could not fit the primary panels, thus allowing us to generate more power.

The battery charging optimization is done through Maximum Power Point Tracking (MPPT) . This MPPT is done using the SPV1040 IC. This IC is used to regulate voltage output to a constant voltage, and increase the current level delivered to the battery to its maximum state. Keeping in mind that typically a lithium polymer (LiPo) battery is charged with a constant current, this is slightly counter intuitive to how to charge a LiPo battery, however, in this application, an "overcurrent" on the charging line will never be met, the battery is 1200mAh, and the maximum solar current output is far under that threshold.

The onboard voltage conversions occur through the DC-DC conversion of both 3.3V and 1.8V. The image sensors require unique LDO's, but because these are not supplying the main voltage lines, these are not individually protected.

The onboard electrical protection is essential to the integrity of the EPS system, and is the last line of defense against a harsh, space, environment. The electrical protection is focused into the voltage converters, and battery charging. The battery charging is monitored by a "fuel gauge". The fuel gauge measures the voltage and current in/out to determine battery charge state. This fuel gauge then communicates that data to the MCU via I2C, and if the MCU recognizes any issues with the data, measures can be taken to reduce or completely reset charging. This is done through a GPIO pin tied to a universal "MPPT SHUTDOWN" pin. The shutdown of the MPPT's will effectively halt charging, and the fuel gauge will still run to monitor the state of the battery. Then, when an OK state is met, the "MPPT SHUTDOWN" pin will be pulled back low and charging will resume.

Current measuring ICs that return measured data to the MCU via I2C are attached to all power convertors. These are connected to the MPPT voltage regulator and the DC-DC converters. These current sensors are the way the MCU can detect if a no-go state, then take action.

The design of this satellite utilized power supply from two voltage converters, a 3.3V, and a 1.8V. There is also a voltage "watchdog" IC that has the ability to reset the converters using a "reset" pin. This watchdog is inactive until the MCU pulls a MOSFET's gate low/open through a GPIO, sending battery voltage through the watchdog, allowing for a reset line to be pulled high, shutting down converters for 3 seconds. This can be done if the MCU recognizes that the converters are drawing too much current through a current sensor, which may occur due to a SEL.

Figure 5: Partially Assembled Prototype

The Integration subsystem for this satellite is responsible for the antenna deployment, thermal management, and subsystem electrical connections. Fig. 5 is a protoype model of the satellite interior.

Antenna design was inspired by the electrically conductive and springy metal that tape rulers are often made of. A design and prototype were developed and tested. There are two poles making a 1/2wave dipole at ~7 inches in pole length. The tape measure wraps around the satellite and back under, then, using holes drilled into the ends of the antenna, a thin wire is inserted. This wire is then tied and anchored down to two resistors. This arrangement was done for each pole of the antenna. The resistors were connected to a MOSFET that closes through a high GPIO output of the MCU. These resistors are low ohm value 1/4 watt resistors, at 12Ω each, they draw 0.35A. This equates to 1.26W at a battery voltage of 4.2V. This over wattage of the resistors makes for a hot, slow burn that effectively melts the wire wrapped around it. Once the wire melts, the tension force will snap the wire, and the tape measure will successfully deploy. Each resistor will burn for 12 seconds. It was concluded in a testing environment that this was the optimal burn time. The second resistor on each pole is a backup resistor, to be utilized if the first resistor fails.

The thermal management of the satellite is essential to the completion of the mission. Most components onboard have a temperature range from -45C to 85C, with some greater than +155 deg C. These temperature ranges will be adequate for the environment of LEO [12]. Additionally, this rugged component selection allows reduction in the amount of insulation and temperature monitoring. Each external face has a temperature sensor on it, which allows for the MCU to know which face is pointing at the sun, and if one side exceeds an expected temperature value. Using the Attitude Determination and Control System (ADCS), the satellite can change its orientation. The components that are most sensitive to thermal stress and variation are the battery and oscillators. The battery's fuel gauge has the capability to read an Negative Temperature Coefficient (NTC) thermistor, this thermistor will be adhered to the battery surface, and provide temperature readings necessary to know when to stop charging, enter a state of reduced current draw, or use the ACDS.

The electrical connections between boards are the backbone of the function of the satellite. The electrical connections throughout the internal PCB stack is done through header pins. The electrical connection to the external faces should be done through header pins on the bottom plate. This allows the mechanical integrity of the satellite to remain un-compromised. The connection to both thermistor and battery leads is done through off-shelf JST connectors.

C. Communications

The communication subsystem is responsible for receiving and transmitting data in a fast and efficient manner. There are several limiting factors to consider while designing the system. The form factor of the PocketQube puts constraints on the size of the antenna and the size of the EPS system for power which play a major role in developing the Communications Subsystem.

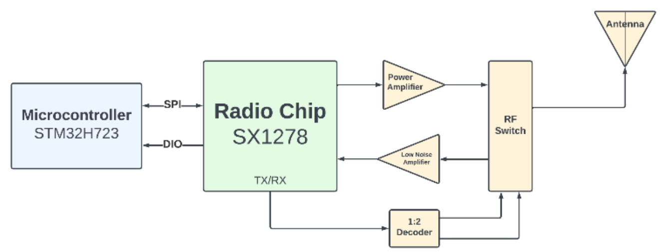

Figure 6: Communications Block Diagram

Off-the-shelf radio transceiver modules would not be optimal for use in the satellite because there will be unused hardware and it would be optimized for the specific use case. Because of this, a board was specifically designed for the payload (Fig. 6 and 7). The components picked for the communications board all are powered by +3.3V making the layout and routing simple. Common voltage also increases power efficiency because only a single converter is needed for the board. Signal integrity is increased by reducing the risk of components having voltage mismatches.

The PCB layer stack is utilized to mitigate noise in the system. Noise is a factor for radio frequency communications and eventually will affect the transmission rate. Thus, measures were taken to mitigate noise. The first step was making the Radio Frequency (RF) signal layer the very top layer with a ground layer right beneath it. This will allow for the ground layer to act as a shield for the sensitive RF line from other layers of the board, which carry power and other signals. Ground vias were also placed throughout the board, but especially near the RF path to provide a low impedance return to the ground. Moreover, there will be an aluminum electromagnetic interference (EMI) cage implemented to reduce interference from other satellites, solar storms, or any other forms of noise in space.

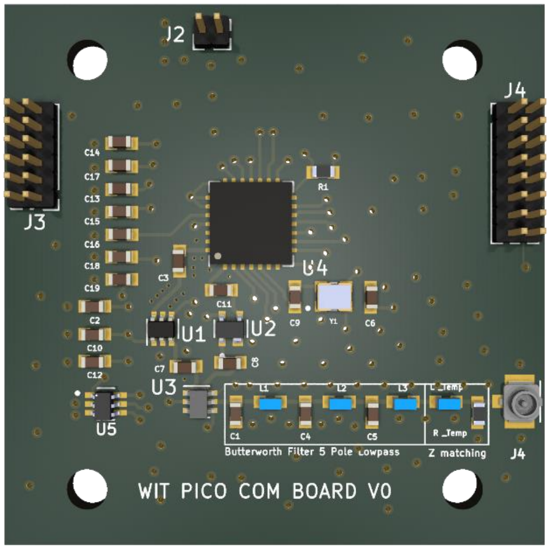

Figure 7: Version 0 Of the Communication Subsystem

The ground vias were also used as a means of impedance matching, and the board stack-up was specified to control the impedances. There will still be unwanted parasitic capacitances. To counteract the parasitic capacitances, the inductive nature of ground vias was explored to match the capacitive and inductive parts of the transmission line. The vias alongside each component having a 50 Ω input/output would ensure each part of the transmission line is well matched. This will be thoroughly tested once the PCBs have been delivered and assembled.

A link budget tool was developed to aid in visualizing the communication system and gain perspective on the design decisions. The link budget considers all expected losses and gains within the system to accurately calculate the signal strength that will be received by the ground station. The effective radiated power (ERP) is approximately 0.5 W. Our analysis predicted a signal strength of around -114 dB at the ground station. We accounted for a noise floor of -120 dB and considering a 10 kHz bandwidth and employing the Shannon-Hartley Channel Capacity equation, the maximum achievable data rate in the system was estimated at 28 kbit/s. The link budget provided insights into the system's performance and guided in making informed choices for optimal data transmission. Upon observing the channel capacity, a two-radio solution is being researched to attempt to increase the throughput of the system.

Since the main goal of the satellite is to send an image, a couple different modulation types were explored. Slow Scan Television (SSTV), used by many Amateur Radio operators to send images, was researched, but it was decided against because it could not support hi-resolution images. Long Range (LoRa) was also considered for its ability to send signals at a longer range with lower power. This is suitable for telemetry but limitations in bandwidth and means of error checking would hinder the ability to transmit an image. Frequency Shift Keying (FSK) was ultimately chosen due to its common use amongst the satellite community and ability to support both telemetry and image transmission. While FSK is a general term that describes the way digital information is modulated into an analog signal, there are different kinds of FSK that offer specific benefits for different use cases. The PocketQube will experience interference, as any spacecraft does, which will negatively impact the signal integrity. The type of FSK signal that seems most appropriate is MSK which will lower the throughput of the system but will allow for transmitted symbols to interfere less with each other making our signal more robust.

Most other picosatellites use the UHF band; specifically the frequency range 400-438MHz [18]. Using this band, Enthusiasts would also have the ability to decode our transmissions around the world, allowing us to gather more data. Our target range is 435-438MHz UHF band, with an exact transmit frequency still to be determined after frequency coordination.

The selection of a dipole antenna was chosen due to various key considerations. Not only is it the easiest antenna to design and build, but using a tape measure or steel cables creates a compact design for launch. Since the characteristics of a dipole antenna is omni-directional it is not necessary to point the satellite to achieve a connection. This also comes as a downside as the antenna will have lower gain.

After conducting thorough research, we opted for two chips manufactured by Semtech Corporation (Camarillo, CA, USA): the SX1278 and SX1268. These chips belong to the Semtech LoRa (long-range spread spectrum communication) transceiver family and offer a versatile range of programmable bit rates, with a maximum rate of 300 kbps. Notably, both chips exhibit excellent receiver sensitivity, enabling reliable signal reception over considerable distances. They also offer the modulation types OOK/(G)FSK/LoRa. Ultimately, the SX1278 was chosen due to its enhanced functionality, despite having examined and considered other options. It should be noted that while the satellite will not be using LoRa or On Off Keying (OOK), if desired in a future mission, a firmware update is all that is required to enable this modulation. This chip also communicates via SPI to the OBC which has up to 6 SPI busses, so there is potential to add more radios to the satellite if desired. This would be useful because having two radios on the satellite allows for full duplex communication.

The LoRa1278F30-433 module by G-NiceRF was selected for testing because it contains the SX1278 radio chip with a Low Noise Amplifier (LNA) and Power Amplifier (PA). Later the DL-SX1278PA module by DreamLink was picked to be used on a new breakout board which incorporates a better design of ESD protection, power filtering, reset button and vias to limit capacitance. Both of these modules were designed to have a 1-watt power output (Figure 8).

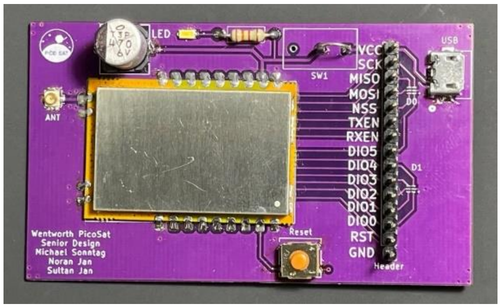

Figure 8: Breakout Board for DL-SX1278PA

D. Attitude Determination and Control System

To complete the mission objectives, it was determined that an active attitude determination and control system (ADCS) would be necessary. Various types of passive ADCS were evaluated [10] [11] [15] and were determined to give inadequate pointing accuracy, or a lack of "time on target" for the camera payload. Reaction wheels are also another popular option for primary actuator of an ADCS [11], however it was determined that the PocketQube form factor would not provide adequate space or mass to accommodate reaction wheels. The second most popular active control scheme is using active electromagnets to interact with Earth's geomagnetic field and generate rotational torque. These electromagnets, called magnetorquers, typically consist of a cylindrical ferrous core wrapped with conductive windings. These ferrous core magnetorquers are very power efficient but take up a lot of space and mass. A new emerging magnetorquer design is embedded magnetorquers, which are "embedded" inside of multi-layer PCB [13] [14]. These magnetorquers consume no extra space because they utilize copper layers in the external facing PCBs that host solar panels. Due to the decreasing cost of multilayer PCBs and the decreasing size of satellites these embedded magnetorquers are an attractive option. The satellite will utilize this technique on all external facing PCBs, except for the baseplate. The coils themselves will be square shaped, as that is the most efficient arrangement, and will occupy 6 of the 8 copper layers in the external facing PCBs (Fig. 9 and Fig. 10) [14]. The remaining 2 layers in these 8-layer PCBs will host solar panels and control circuitry.

Each of these magnetorquers will be driven by an H-bridge circuit, which will allow the field direction to be reversed at will. It is anticipated that these magnetorquers will provide enough rotational torque to detumble the satellite and point at designated points on the earth's surface. In this regard, the PocketQube form factor is beneficial as the 250g mass restriction limits the amount of torque needed.

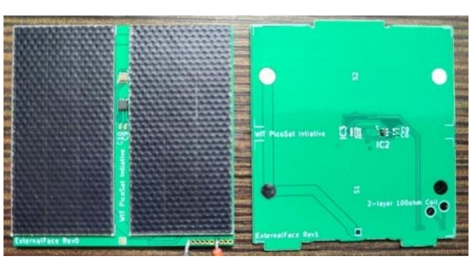

Figure 9: External Face PCBs with embedded magnetorquers

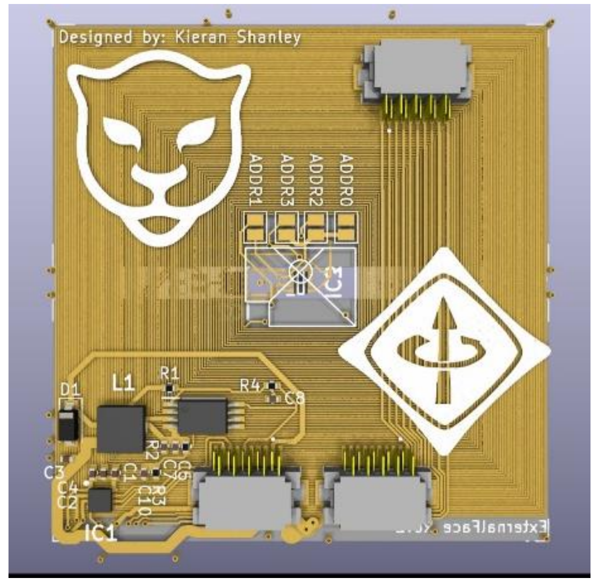

Figure 10: See-Through visualization of an External Face with visible magnetorquer coils

V. Conclusion

The prototype developed over a two-year period can likely lead to an eventual satellite launch, given additional testing and refinement. Each subsystem has hardware designs currently in-progress, with firmware continuing to be written that enables hardware functionality and failure prevention. After an engineering prototype is complete, extensive testing will be conducted to ensure the spacecraft is capable of surviving in the harsh space environment.

IV. ACKNOWLEDGMENT

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