HIGH RESOLUTION SATELLITE IMAGING OF NITROGEN DIOXIDE FROM LOW EARTH ORBIT

Presented to
The Faculty of the
School of Engineering and Applied Science
University of Virginia
In Partial Fulfillment of the Requirements for the Degree
Bachelor of Science in Aerospace Engineering

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Technical Project Team Members

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Approved:_______________________________________ Date:_______________
Chris Goyne, Department of Mechanical and Aerospace Engineering

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LIST OF TEAM MEMBERS AND COLLABORATORS

The team for this mission comprised 12 fourth year students enrolled in the Spacecraft Design I and II courses. All team members were either Aerospace Engineering or Mechanical Engineering students, and were divided into functional group sub-teams, as denoted in Table 1.

Table 1: Student Team Members

Sub-team	Personnel	Responsibilities
Program Management	Hannah Umansky	Manage project, provide technical guidance
Communications and Data Handling	Alex Brookes, Adelaide Pollard, William Schaefermeier	Design and implementation of subsystem and governing software, selection of space and ground segment communication hardware, FCC licensing
Power, Thermal, and Environment	Genesis Brockett, Noah DeMatteo, Matt Moore	Selection of EPS/Battery and solar panels, thermal analysis of spacecraft
Attitude Determination and Control, Orbits	Isabel Araujo, Sami Khatouri	Selection of ADACS and GPS systems; determination of orbit passes
Structures and Integration	Huy Tran, Max Diamond, William McNicholas	Assembly of components, partial design of spectrograph, implementation of solar panels, structural/thermal testing

ACKNOWLEDGEMENTS

The design of the spectrograph payload has been completed by John Wilson, Matt Nelson, and Mike Skrutskie of the U.Va. Department of Astronomy. Significant contributions to the preliminary electrical design, choice of on-board processors, and power budget has been completed by fourth year Electrical and Computer Engineering student Kathryn Wason.

The objective of this mission was formulated by Sally Pusede of the U.Va. Department of Environmental Sciences with the support of graduate student Angelique Demetillo. The satellite bus conceptual design was inherited from the Spacecraft Design Class of 2019, led by Ian Palmer. The project was originally developed within the 3Cavaliers Research Grant by Chris Goyne, Sally Pusede, and Mike Skrutskie. Development of the U.Va. ground station has been completed by Mike McPhereson.

INTRODUCTION

SCIENCE INVESTIGATION

Nitrogen dioxide (NO_2) is an anthropogenic pollutant generated by both mobile and stationary sources, and acts as a general pollutant indicator. Concentrations of NO_2 vary dramatically over short distances, therefore high spatial gradients require high resolution mapping to capture intra-urban variability. Environmental scientists use both ground-based air quality monitoring stations as well as remote sensing satellites to study the Earth's atmosphere and capture global NO_2 trends.

Current Earth-orbiting experiments such as the Ozone Monitoring Instrument (OMI), onboard the NASA AURA spacecraft, and the Tropospheric Monitoring Instrument (TROPOMI) onboard the ESA SENTINEL-5P satellite, have limited spatial resolutions that do not allow for a complete understanding of NO_2 spatial and temporal variability. Payloads OMI and TROPOMI are restricted to spatial resolutions of 13 km x 24 km and 7 km x 7 km, respectively, greatly smearing NO_2 concentrations across large ground swaths. Conversely, the proposed 3U CubeSat will have a spatial resolution of 200 m x 800 m, enabling a better image of NO_2 concentrations to be captured. An example of potential data collection by the novel CubeSat of NO_2 over Los Angeles can be seen in Figure 1, as captured by an aircraft-mounted instrument at 10 km. As the image shows, the NO_2 distribution has high spatial gradients, particularly near major roadways. The high spatial resolution of the payload will be able to more accurately map NO_2 distributions, and capture variability in NO_2 concentrations beyond the capabilities of current spacecraft. Obtaining higher spatial resolution information is crucial in understanding and locating emission sources.

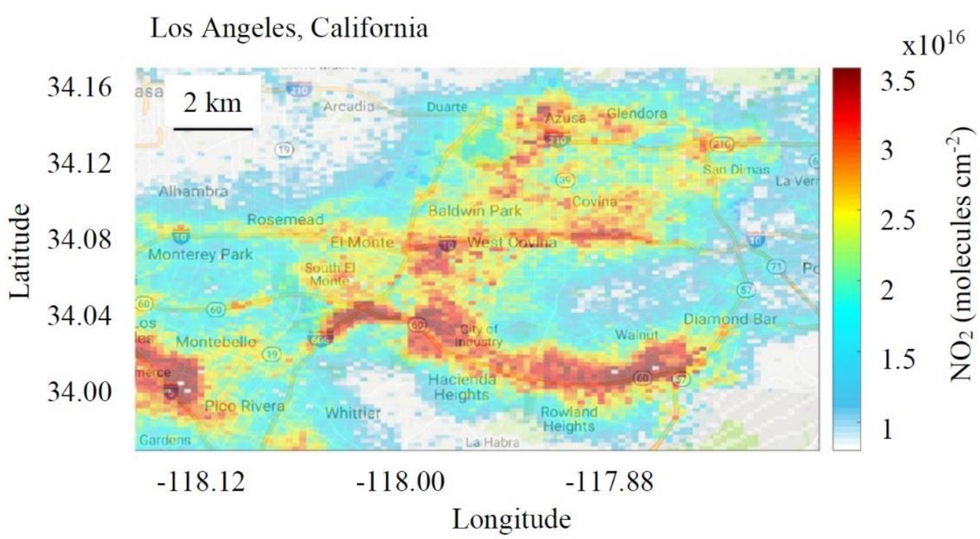


Figure 1: Expected NO_2 column density over Los Angeles, adapted from Pusede et al. (2018)

The creation of a novel CubeSat with a custom NO_2 -focused spectrograph payload will allow for remote sensing of the Earth's atmosphere at reduced cost and accelerated mission timeline. The resultant data will be analyzed by environmental scientists to better understand the emission and transportation of anthropogenic nitrogen dioxide in major cities around the world, as well as contribute to defining the connections between environmental phenomena and public health.

PREVIOUS WORK

This mission has been a part of the Spacecraft Design course for the past two years, during which it has reached two milestones: Conceptual Design Review (CoDR) and Preliminary Design Review (PDR). On April 15, 2019, the previous team presented the Conceptual Design Review to Professors Goyne and Pusede. The CoDR focused on establishing mission requirements and constraints, as well as presenting preliminary evaluations of commercial off-the-shelf (COTS) parts for the spacecraft bus. During the Fall semester of 2019, the team completed multiple reviews, expanding on the mission definition, objectives and constraints, and the evaluation of the mission architecture. This work contributed greatly to the PDR, presented on February 11, 2020. The PRD mapped the mission architecture to component choices and further developed the concept of operations for the satellite.

This report documents the work done over the 2019-2020 academic year, and the progress in the mission and spacecraft design since the CoDR. Unfortunately, further progress on the design work was hampered by the COVID-19 social distancing requirements during the second half of Spring 2020 semester, as the Spacecraft Design course continued remotely.

MISSION DETAILS

PRIMARY MISSION OBJECTIVES

This mission has two primary objectives: to develop a spectrograph suited to the constraints of a 3U CubeSat bus capable of measuring NO_2 columns at a spatial resolution better than 1 km x 1 km, aiming for a resolution of 200 m x 800 m, and to use this data to improve our understanding of NO_2 emissions and concentrations in urban landscapes.

SECONDARY MISSION OBJECTIVES

A consequential effect of this mission will be the expansion of the CubeSat and small spacecraft programs at the University of Virginia (U.Va.). Further, this mission will enhance the scope of the Aerospace Engineering degree program by including undergraduate engineers in satellite development projects. After the successful design, build, and launch of the Libertas satellite under the Virginia Space Grant Consortium-led Virginia CubeSat Constellation (VCC) mission, the continuation of the Spacecraft Design class and the subsequent products created by students will help attain these goals.

SYSTEM LEVEL FUNCTIONAL REQUIREMENTS

Expansion of the mission objectives has led to the creation of six functional requirements for the satellite. The specifications outlined in Table 2 act as overarching requirements for the mission and spacecraft design. These requirements are further broken down into subsystem level requirements in the following sections of this report.

Spectrograph analysis will be completed in the instrumentation laboratories within the Department of Astronomy. These tests will ensure the spectrograph is functioning according to its design specifications, will be able to withstand launch, and operate in focus while in the space environment. The orbital and lifespan analysis has been completed using numerical models and simulations, as well as referencing the lifespan of previous CubeSat missions. Much of the environmental testing will be completed by either the launch service provider or the integration service provider. However, theoretical CAD and FEA modeling, as well as strict adherence to NASA guidelines throughout the design and build process will ensure that the satellite will be able to pass physical inspection.

Table 2: System Level Functional Requirements

ID	Requirement	Specification	Verification Method
F1	Image Spatial Resolution	800 m x 200 m	Spectrograph Analysis
F2	Imaging Frequency	Nine cities, one city pass/day, daytime-only imaging	Orbital Analysis
F3	Operational Lifespan	Spacecraft must be operational for at least 12 months in orbit	Analysis

F4	Environmental	Must be able to operate in LEO, pass radiation, thermal vacuum, vibrations testing	Testing
F5	Safety	Must be compliant with all safety regulations in order to deploy from ISS, as well as during fabrication and testing on ground	Analysis
F6	Power	Must be able to provide power throughout entire orbit	Test and Analysis

SYSTEM LEVEL OPERATIONAL REQUIREMENTS

The system level operational requirements are derived from the mission objectives and address features of the mission design that are external to the physical components of the satellite. The specifications listed in Table 3 describe three major concepts driving the physical design.

Attitude verification will rely on the internal processing of the chosen attitude determination and control system (ADACS), and the ability to customize the control sequences for the mission.

Ground station testing will be performed throughout the design and built. Ideal testing would achieve successful communication between the satellite and the ground station before launch.

Table 3: System Level Operational Requirements

ID	Requirement	Specification	Verification Method
O1	Attitude	Nadir Pointing/Slewing capable of complex motion over target cities	Testing
O2	Orbit	Spectrograph will function in Low Earth Orbit (LEO)	Orbital Analysis, LV Choice
O3	Communication	Downlink and uplink data to U.Va. ground station across two frequency bands	Testing

MISSION CONSTRAINTS

The design of the satellite is constrained by multiple factors relating to mission elements external to the Spacecraft Design course, as listed in Table 4. The choice of following the design parameters of CubeSats allows for the use of COTS parts, however, in exchange, strict dimensions and mass requirements must be adhered to in order to comply with industry regulations.

This mission aims to use the NASA CubeSat Launch Initiative program to reach Low Earth Orbit, further constraining the design of the satellite with numerous NASA guidelines. Among these guidelines are mission design requirements that ensure compliance with launch and deployment services, general safety requirements, and federal regulations concerning small spacecraft.

While reliable customer service may seem to be a trivial requirement, the ability to discuss component choices, features, and opportunities for customization are crucial to ensuring a robust design. The selection of materials is a collaborative process between the team and the vendors, as their specific product knowledge is invaluable and necessary to ensure compatibility and success within the satellite architecture.

Table 4: Mission Constraints

ID	**Requirement**	**Specification**	**Verification Method**
C1	Form Factor	3U spacecraft form factor (10 cm x 10 cm x 30 cm), 1.5U allocated to payload	Measurement
C2	Budget	Material cost under $400,000	Financial Analysis
C3	Launch Opportunity	Compliant with NASA CubeSat Launch Initiative (CSLI)	Mission Analysis
C4	Vendors	Reliable vendors with customer service (response time)	Market Research

MISSION ARCHITECTURE

The mission architecture in Table 5 describes both the satellite itself as well as the larger mission elements. The required components are interlinked with the mission objectives, as the payload and ADACS both serve key functions for the spacecraft.

The lifetime of a CubeSat is variable, however, a mission timeline allocating 12 to 18 months in orbit is consistent with prior CubeSat missions.

The ground segment will consist of a single ground station (GS) at U.Va. capable of transmission over UHF frequencies, and reception of both UHF and S-band information. Flexibility in the current design leaves open the possibility for amateur radio involvement; choice of an experimental or amateur radio license has yet to be determined, and may allow for global interactions with the satellite over its lifetime.

The launch segment assumes a deployment from the International Space Station (ISS), as this is a common deployment scenario for NASA CSLI missions. These CSLI missions allow

CubeSats to travel as auxiliary payload on ISS commercial resupply missions, and employ an array of uncrewed launch vehicles, such as the Antares or Atlas V rockets. Once the CubeSat has reached the ISS, it is transferred to a satellite deployer, such as the Nanoracks CubeSat Deployer. This deployment mode provides the CubeSat with the initial orbital conditions of the ISS, which vary slightly as the satellite stabilizes and eventually begins to deorbit.

Table 5: Mission Architecture

Element	**Choice**	**Reasoning**
Subject	Nitrogen dioxide column densities	Mission Objective
Payload	Spectrograph	Required method of mission objective
Bus	1U electronics “stack” 0.5-0.6U ADACS 1.5U Payload	ADACS: Meet requirements for pointing accuracy, pointing determination, slew rate, and power usage
Ground Segment	U.Va. GS with S-band receiver and UHF transceiver	UHF transceiver to send commands, S-band receiver for data downlink
Mission Operations	Continuous operation over ~18-month lifespan	Lifespan of LEO
Command, Control, & Communications Architecture	Combination direct downlink to U.Va. GS + potential for Amateur operators	Have direct control over data as well as amateur back-up
Orbit	LEO, 92.5 min period, incl. of 51.6°, alt. of 400 km	Allows for multiple daily passes over targets
Launch Segment	Deploy from ISS	Most likely orbit from NASA CSLI

MISSION CONCEPT (CONOPS)

The satellite will have several modes of operation, as described in Table 6. The primary mode of the spacecraft will carry out its scientific operations, critical to the mission objectives. Complementary to the data collection mode is the ground station communication mode, wherein the satellite carries out two-way communication with the ground station over UHF frequencies and a one-way data downlink over S-band frequencies. However, the majority of the spacecraft lifetime will occur in Orbit mode, a passive state where telemetry is collected and stored.

Additional modes may be included in the satellite's operational protocols for infrequent scenarios, such as during stabilization immediately after deployment. A safe mode option would allow for an emphasis on power collection and the prevention of injury to the spacecraft in case of a malfunction.

The science collection mode is of central importance to fulfilling the objectives of this mission. When the satellite is approaching a target city, signaled by GPS location, the payload functionality becomes active. The detector completes a series of 1 second exposures, with an internal rapid coaddition of frames by the FPGA to prevent overexposing the spectrograph. As the satellite orbits at a velocity of approximately 7 km/s, the spacecraft must move within its own inertial reference frame to avoid blurring the image. A complex sequence of slew maneuvers will be carried out by the ADACS to continually adjust the pointing of the spectrograph, and by extension, the entire CubeSat at an average rate of 1.13°/s. Simultaneously to data collection, a small optical camera will capture the ground scene. This photograph of the target city will aid in verifying the coordinates of the associated NO_2 data and in mapping emissions to physical sources.

Table 6: Modes of Operation

Mode	Class	Description	Duration
Science Collection (Capture)	Active	Satellite performs custom slew maneuvers; payload is capturing images	14-23 passes per week ~60 s within 70° cone Local daytime only
Ground Station Communication (Send/Receive)	Active	GS sends commands to satellite, satellite transmits telemetry over UHF, transmits science data over S-band, no unique maneuvering necessary	5-7 passes per week Restricted to ~15° elevation Day or Night
Orbit (Passive)	Passive	Potential for minor adjustment to maximize power collection of solar panels	Majority of satellite lifetime

SCIENCE INSTRUMENT

The science instrument is tasked with imaging atmospheric NO_2 over the major cities passed by the satellite in orbit. It is designed to image at a spectral resolution of 0.05 nm/pixel, with a spectral coverage of 410-460 nm. The field of view will be able to cover major urban landscapes, with the slit width of 10 km on the ground over 500 pixels. The exposure time of imaging will have a maximum of 1s to allow for high frequency frame collection to achieve NO_2 detection of the expected column densities.

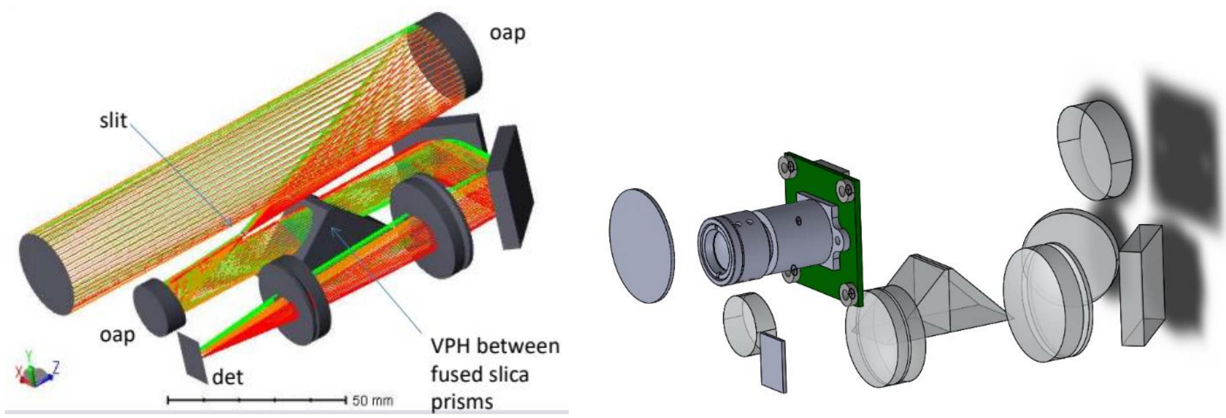


Figure 2: The Spectrograph with Ray Traces (Left) and Payload Configuration (Right)

As seen in Figure 2, the spectrograph will consist of many optical components. Specifically, there will be 2 plane mirrors, 2 off-axis parabolic mirrors, 4 lenses, and 1 dispersing element, alongside the spectrograph detector array and the optical camera. The spectrograph will occupy a 1.5U Endurosat frame, meaning it will fit within a 15 cm x 10 cm x 10 cm volume. Currently, the spectrograph only occupies about half of the allotted width, or about 5 cm. It is possible to explore options that would spread the two lens tracks out farther in the X-direction. This would free up space for the slit, which is currently hypothesized to be obstructed given the mount systems in place, as seen in the figure above. The mount for the slit will need to be designed in a way that doesn't obstruct the light path. In order to determine whether a mount system is viable, modeling in SolidWorks can be done to see if the mount system interferes with the light path. To confirm the viability of a mount system, benchtop testing can be done using 3D-printed mounts to experimentally determine if the light path will be obstructed.

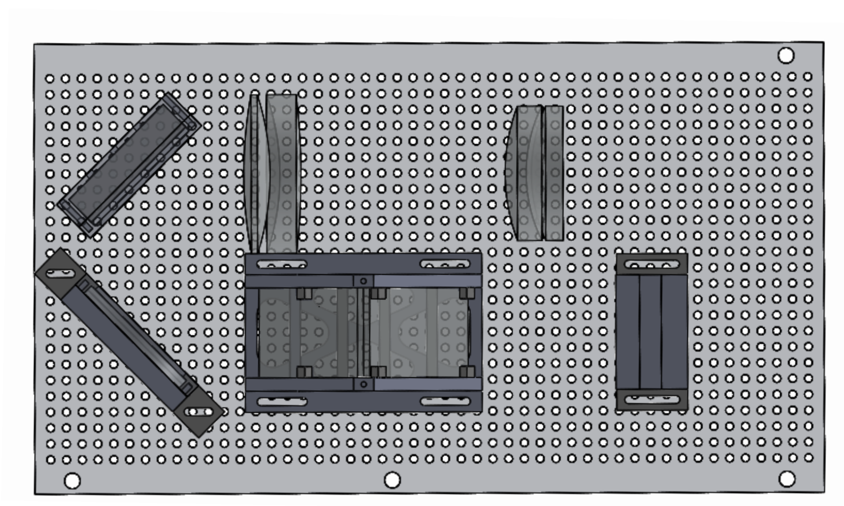


Figure 3: The Payload Mount System

The lenses will be mounted to breadboard-like plates that will run along the XZ and XY faces of the instrument, as seen in Figure 3. These plates will have uniformly dispersed holes so that the mounts may be adjusted to allow for changes in the spectrograph resolution. The plates will be fastened to the 1.5U chassis in a way that there are no fastener protrusions outside of the chassis frame that could interfere with the satellite's deployment.

As mentioned previously, the configuration of the lenses is not yet finalized. The slit location poses problems, as the light may be obstructed by the current mount configuration. By spreading the two lens tracks apart, this issue may be alleviated by providing more space for the light to pass through the slit without interference. The proof of concept is still under review, as testing of the spectrograph has been minimal.

SUBSYTEM DETAILS

STRUCTURES AND INTEGRATION

Requirements

The main requirement for the structures and integration of the spacecraft is to securely combine and attach all of the physical components of the spacecraft, as listed in Table 7. This includes safely securing all of the optical equipment for the spectrograph, integrating the payload structure into the CubeSat chassis, the arrangement of the solar panel array, and the assembly of the entire spacecraft.

Table 7: Structures and Integration System Requirements

ID	Requirement	Specification	Verification Method
F4.A.1	Fix Lenses	Lenses do not detach from mounts	Ground Testing
F4.B.1	Optimal Focusing & Resolution	Optics components are aligned and in focus	Thermal Modeling, Optics Modeling, Ground Tests
F5.A.1	Protection	All optics components are safe within the structure	Testing, Vibration Testing
F5.B.1	Ease of Assembly	Component locations can be adjusted upon assembly	Assembly Modeling
C1.A.1	Interface with Chassis	Payload structure can be inserted into CubeSat chassis	Assembly Modeling
C1.A.2	Meet Volume and Mass Constraints	Spacecraft has acceptable size compliant with CubeSat standards	Assembly Modeling
C3.A.1	Manage Acceleration Loads	Spacecraft can withstand the expected load factor and vibration due to launch	Simulation, Vibration Testing

Element Description

The three main components of the 3U spacecraft are the 1U electronics bay, the 0.5U ADAC system, and the 1.5U Instrument/Payload Bay. CubeSat vendor Endurosat will be used for both the 1U and 1.5U chassis and will be made of Aluminum 6061. The electronics bay consists mainly of the EPS and battery system, and will be integrated as a stack through threaded rods along the 1U chassis. The instrument bay will house the scientific equipment for the spacecraft, consisting of the mirrors and lenses that make up the spectrograph as well as the reference camera. A customized lens mount has been individually developed for each optical component. The optics will be adjustable in 3 dimensions. To mount the lens mounts to the spacecraft, an optics breadboard will be used. The breadboard allows for easy adjustment of the spatial position of the lenses along the plane of the breadboard, depicted in Figure 4.

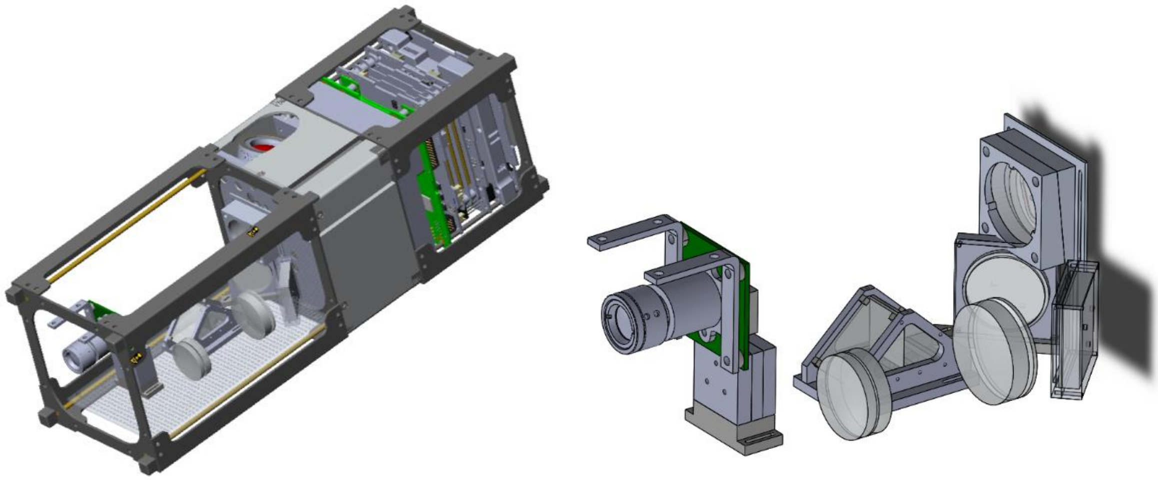


Figure 4: The Spectrograph Component Housings

The spacecraft will also employ a solar panel array to power the electronic equipment of the spacecraft. Solar panels will both be attached directly to the sides of the spacecraft and deployed along the long edge of the spacecraft. In order to avoid obstructing the camera for star tracker of the ADAC system, a 1.5U and 1U solar panel will be used along the sides with the star tracker apertures (+X, -X), leaving the 0.5U slot open for the star trackers. The other sides (+Y, -Y) will use a full length 3U solar panel, configured with the long edge deployable solar panels.

Table 8: Solar Panel Configuration

Item	**Quantity**
1U SOLAR PANEL X/Y	2
1.5U SOLAR PANEL X/Y	2
3U Single Deployable Solar Array	2

Mass Budget

The mass budget in Table 9 details the estimated weights of the various components of the spacecraft hardware.

Table 9: Mass Budget

Component (Qty)	Estimated Weight (g)
1U chassis (1)	98
1.5U chassis (1)	114
1U solar panel (2)	88
1.5U solar panel (2)	130
3U deployable panel (2)	600
Lenses	160
Spectrograph Housing & Lens Mounts	512
Communications (Antennas, UHF Transceiver, S-band Transmitter)	493
Navigation (GPS Antenna & Receiver)	80
Power (EPS I and Battery)	208
On-Board Processing (NanoMind Z7000 & Dock)	151
ADACS (Star tracker)	1000
TOTAL	**3634**

POWER, THERMAL, AND ENVIRONMENT

Requirements

The primary directive for the Power, Thermal, and Environment (PTE) subsystem is to ensure the functionality of all components throughout the spacecraft lifetime. This includes ensuring the solar panels provide enough power during sunlight to maintain all systems power requirements throughout the eclipse period, and that the electric power system (EPS) and battery is capable of supplying and regulating that power.

Table 10: PTE System Requirements

ID	Requirement	Specification	Verification Method
F6.A.1	EPS Power Storage	Provide required power throughout 1.54 hour orbit	Power budget estimations
F6.B.1	Solar Panel Power Generation	Generate required power in daylight to sustain all systems through eclipse	Predictive modeling of incident heat
F4.C.1	Radiation Protection in LEO	Ensure that electronics are protected from exposure to radiation in LEO by solar panels or radiation shielding	Thermal/Radiation analysis (ANSYS)
F4.C.2	Electronics stay within operating temperatures	Insulation and heaters ensure that all electronics stay within the range of operating temps; minimize thermal effects on structure	Thermal estimates and analysis

Element Description

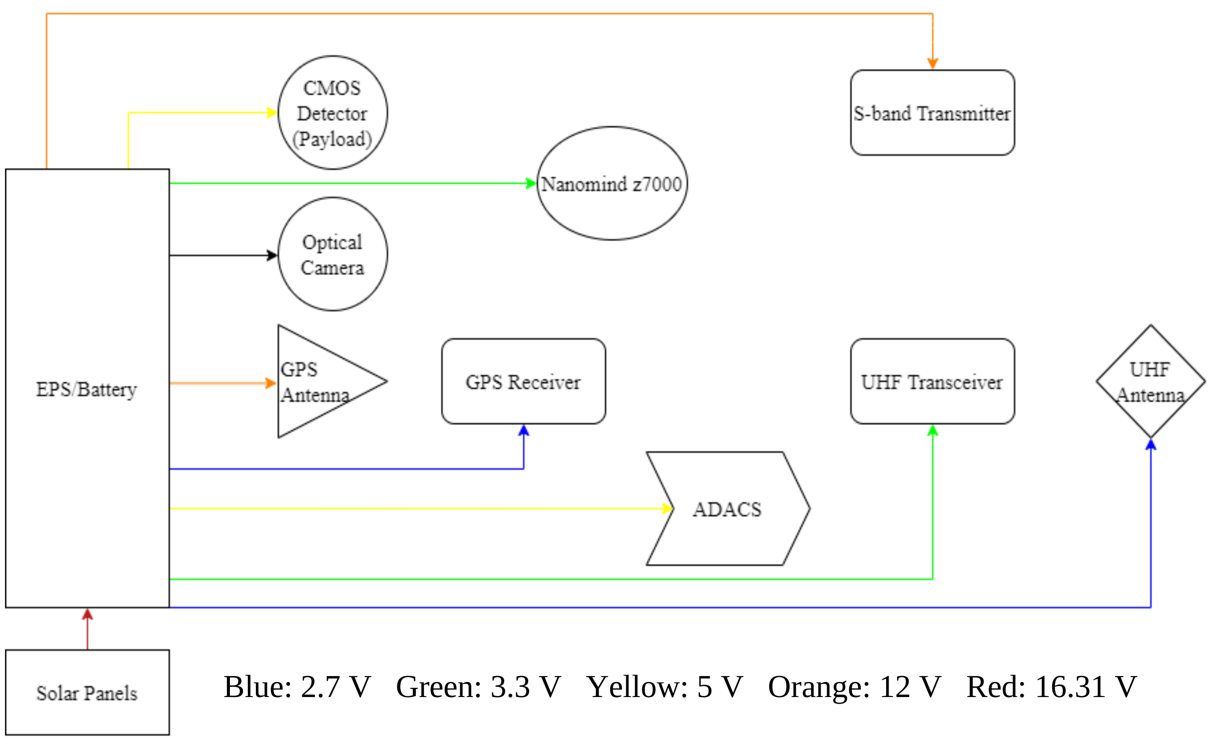
Based on our power budget estimations, we require less than 10 Watt-hours of power to run all necessary systems for one typical orbital period. This time includes both sunlight and dark transit times. Based on this requirement, we initially selected the Endurosat EPS I with a single built-in battery pack capable of storing 10.4 Wh of power. We considered using the EPS 1 Plus which would double the power storage to increase our margin to store power in case something goes wrong. This decision rested on the balance between the extra safety and the extra space it would take away from other components such as the ADACS. Both the EPS I and the EPS I Plus have length by width dimensions of 90.2 x 95.9 mm but their depths are 21.2 and 30.0 mm respectively. The EPS II only has a depth of 18.0 mm but requires an external battery pack containing at least 4 LiPo batteries. It is due to this space constraint that we elected to stick with the EPS I to make room for the ADACS system. However, we have recently realized that the input voltage of the 3U solar panels exceeds the capabilities of the EPS I and I Plus, which was only meant to accommodate 1 and 1.5 U solar panels. This issue will need to be addressed next to understand how it may impact the rest of the system.

We elected to use 2 each of Endurosat's 1, 1.5, and 3U solar panels to cover the faces of the CubeSat, with two long-edge deployable 3U solar panels to provide extra power. Based on orbital and orientation predictions, these solar panels will be able to produce 15.8 Wh of power during the daylight portion of one orbit, which is plenty power to sustain all systems. These calculations are based on the assumption that the ADACS system maintains nadir pointing while simultaneously spinning about the Z-axis to get even exposure on all sides. To get around the incompatibility of the 3U panels with the EPS I, we could try using two 1.5U panels in their place. With just six 1.5U and two 1U panels on the satellite while it rotates about the Z-axis, the solar panels could generate 9.8 Wh per orbit. This is a risky choice as it provides just enough power for all necessary systems, but it may be our only option. The solar panels also reduce the satellites vulnerability to radiation preventing both data anomalies and damage to electrical components. At this stage, we do not believe it is necessary to include any auxiliary heaters or radiation shielding as all systems should be protected using passive thermal regulation and the solar panels.

Future work would include selection of wires and connections for all electrical components. We would also like to conduct further thermal modeling to more accurately determine the expected temperatures of each component to ensure all components stay within rated temperature ranges. This will be easier after we have finalized all components along with their heat outputs.

Power Flowchart

The power flowchart in Figure 5 shows how power will be distributed from the EPS and battery to all the other components of the spacecraft. The different colored lines represent the different voltages that each component runs on and these values can be seen below in the Figure. The output voltages from the EPS I are 3.3 and 5V, and the EPS II has an additional 6-12 V bus. Input voltages from the solar panels are a maximum of 4.66V for the EPS I and 10-36 V for the EPS II.



Blue: 2.7 V   Green: 3.3 V   Yellow: 5 V   Orange: 12 V   Red: 16.31 V

Figure 5: Power Flowchart

Power Budget

Along with general current and voltage requirements, we calculated the time-dependent power requirements of the system. Table 11 below shows the accounting for the duration of a science collecting (capture) mode for one 5-minute data collecting pass. The total time and power used in each mode (capture, passive, and send/receive) is added to get the total power needed for each orbit, shown in Table 12.

Table 11: Time-sensitive Power Budget for a Science Data Collecting Pass

Component	Typ Voltage (V)	Max Current (mA)	Typ Power (W)	Typ Time (min)	Idle Power (W)	Idle Time (min)	Max Power (W)	Max Power Time (min)	Watt Hours per Orbit	% of Total Power
Payload									0.028	5.47
CMOS	3.5	75	1.2	1	0	4	1.2	0	0.020	3.86
Camera	3.3		0.5	1	0	4		0	0.008	1.61
Comm									0.032	6.19
S-band Tx.	12	800	9.6	0	0.30	5	19.6	0	0.025	4.83
UHF Ant.	5	1000	0.01	0	0.0025	5	5.6	0	0.000	0.04
UHF Tx.	3.3	800	1.3629	0	0.0825	5	2.64	0	0.007	1.33
ADACS									0.207	39.97
GPS Ant.	3.3	30	0.066	5		0		0	0.006	1.06
GPS Rec.	3.3	150	0.2904	5	0.1254	0	0.54	0	0.024	4.67
MAI-500	5	1603	2.13	1	1.82	3	3.05	1	0.177	34.23
Computers									0.199	38.37
NanoMind	3.3	700	2.31	5	0.99	0	3.96	0	0.193	37.16
EPS	3.7	230	0.075	5					0.006	1.21
Margin			0.0						0.052	10.00
Total			0.00	5					0.518	100.00

Table 12: Total Time and Power for Each Mode

Mode	Time (minutes per orbit)	Power (Wh)
Passive	77.4	7.161
Capture	5	0.518
Send/Receive	10	1.907
Total	92.4	9.586

ADACS/ORBITS

Requirements

The ADACS must meet the requirements for pointing accuracy, pointing determination, slew rate, power, and cross-sectional size. The GPS should minimize mass and size and must meet the location accuracy requirement. These subsystems must provide accurate enough nadir pointing and location reporting to allow for the high spatial resolution goal. The location accuracy, pointing accuracy, attitude determination, and slew rate requirements are derived from the pixel-size goal of 800x200m. The requirements are presented in Table 13 below.

Table 13: ADACS System Requirements

ID	Requirement	Specification	Verification Method
F2.A.1	Location Accuracy	Accurate within ± 100 m on ground	Analysis and Testing
O1.A.1	Pointing Accuracy	Accurate within 0.0072°	Analysis and Testing
O1.A.2	Attitude Determination	Reportable within ~ 0.0015°	Analysis and Testing
O1.A.3	Slew Rate	Ensure that pitch rate can correspond to a ground track over a given city of 0.8 km/s (Avg. Slew Rate: 1.13°)	Analysis and Testing
O2.A.1	Orbit Parameters	Collect images over 9 populated cities	STK Analysis
C1.B.1	Cross-Section	10 cm x 10 cm	Measurement
C1.C.1	Power	~ 3 W	Measurement

Orbit Determination

The CubeSat is planned to be released from the ISS into LEO, so the ISS orbit was used as a model of the CubeSat's orbit. This orbit has an average altitude of 400 km, an inclination of 51.6°, and an average period of about 93 minutes. Nine target cities for NO_2 data collection were chosen primarily by U.Va.'s Environmental Sciences Department, based on their pollution levels and cloud coverage. This orbit along with the cities chosen were modeled in AGI Systems Toolkit (STK) to analyze the passes of the satellite over each city. Figure 6 shows the chosen cities and the predicted orbit. For this analysis, a city pass was considered only when the CubeSat is at an elevation of at least 70°. This restriction is to prevent image distortion due to imaging at low angles and is represented by the red circles on Figure 6. Only daytime passes are considered for data collection, as the spectrograph requires daylight for its NO_2 measurements.

STK was used to calculate the CubeSat's pass data over the span of a year. Figure 7 shows the total number of passes that the CubeSat will see each month. It will consistently exceed 80 science passes per month, or an average of 2.7 passes per day at worst. This data confirms that the CubeSat will easily meet the goal of at least one pass per day. Figure 8 shows the number of passes that each city will see per month. This data confirms that all nine cities can be imaged frequently enough to meet the data collection goals. It was noted that Paris has a significantly higher number of passes per month than all of the other cities. This is due to its angle of latitude closely matching the CubeSat's angle of inclination. Figure 9 shows the average pass duration for each city, which varies very little between cities. This consistency allows a typical science pass duration to be safely approximated as 62 seconds, independent of city. This average pass duration is sufficient for the spectrograph to obtain adequate NO_2 measurements of a city during a typical pass, and allows it to track across a sufficiently large area. Knowing the pass time duration is also convenient for predictions of power usage for a typical science pass.

Figure 6: STK Model of Predicted CubeSat Orbit and Target Cities

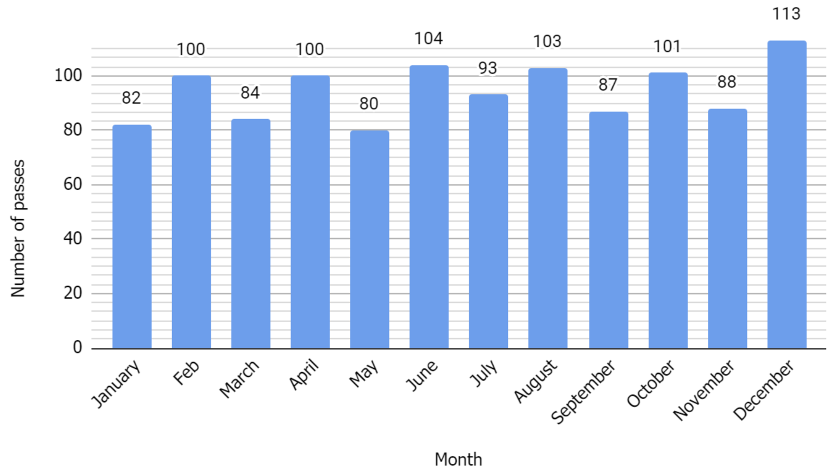


Figure 7: Total Number of Passes Per Month Over All Cities

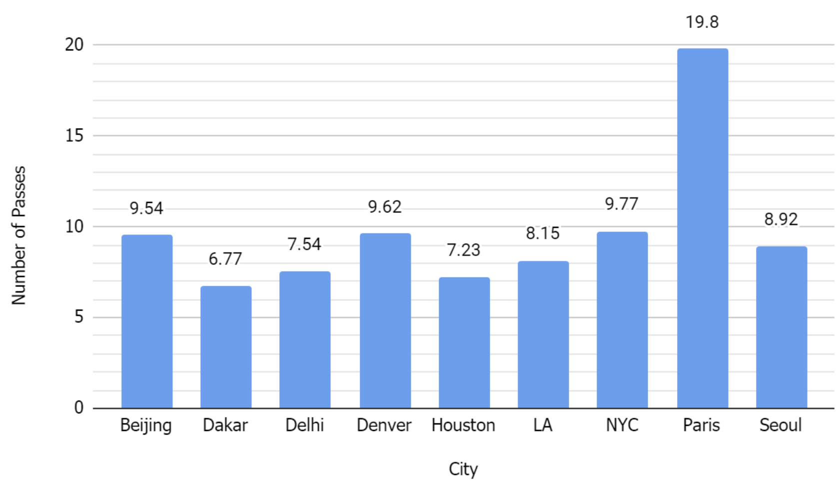


Figure 8: Average Number of Passes Per Month for Each City

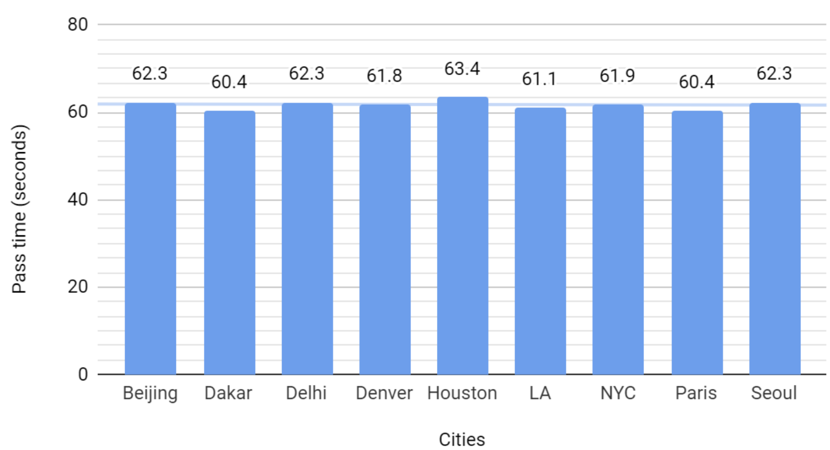


Figure 9: Average Science Pass Durations for Each City

Element Description: ADACS

The ADACS is used to achieve accurate pointing and swift slew rates to point the spectrograph at target cities for data collection. The units currently being considered are the Blue Canyon XACT-15 and the MAI-500.

The team has had communications with Maryland Aerospace to confirm the specifications of the MAI-500. The team has been unable to get in contact with Blue Canyon to confirm the specifications of the XACT-15, but most of the relevant specifications were able to be identified through studies and reports of the system. The attitude determination accuracy and cost still must be obtained for the XACT-15. From the data that is available, both the XACT-15 and the MAI-500 meet the minimum mission requirements, as presented in Table 14 below. Both have a pointing accuracy high enough to allow for precise imaging. The upper bound on the MAI-500's pointing accuracy range falls barely short of the requirement. However, for the relatively slow slew rates at which the ADACS will be operating, the pointing accuracy should remain near the lower bound, keeping it well within the requirement. Both products far exceed the slew rate requirement. They both comfortably meet the power usage requirement. They both fit within the chassis cross-section, using about 0.5U of space. The XACT-15 outclasses the MAI-500 in all categories: pointing accuracy, slew rate, power usage, and size. In particular, its smaller length would free up an extra 1.23 cm to be used for other subsystems. However, because the attitude determination accuracy and price are unknown, a definitive final selection cannot be made yet. Multiple studies on the XACT-15 have commended its pointing capabilities, so it is very likely that its attitude determination accuracy meets the requirement. Given the known specifications, the team is tentatively selecting the XACT-15 as the ADACS system, but will continue to try to contact Blue Canyon in order to obtain the cost and attitude determination accuracy and to confirm the specs that were calculated. If it is determined that its attitude determination meets the requirement, and the cost is similar or better than that of the MAI-500, the team will officially move forward with the XACT-15.

Table 14: ADACS Specifications

System Requirement		**Blue Canyon XACT-15**	**MAI-500**
Pointing Accuracy (°)	**0.0072**	0.003	0.004 - 0.008
Attitude Determination (°)	**0.0015 (within ½ pixel)**	TBD	0.0014
Slew Rate (°/s)	**1.113 (AVG)**	>10	3
Power (Average, W)	**~3**	1.9	2.13
Dimensions (cm)	**10 x 10 cross-section**	10 x 10 x 5	10 x 10 x 6.23
Cost ($)		TBD	100,000

Element Description: GPS

The GPS is necessary to acquire accurate location data throughout the mission. The GPS will be programmed to guide the ADACS in real time.

The SkyFox Labs piPATCH-L1 FM antenna module, with its corresponding piNAV-NG GPS receiver, was originally chosen due to its ability to meet the location accuracy requirement and previous positive relationships with the company. The previous U.Va. satellite Libertas had issues communicating and maintaining contact with their satellite with the use of the SkyFox Labs GPS system. Due to this, the 3U team has chosen to continue with a different vendor and system for the GPS system. The Surrey SGR-05 U and the NovAtel OEM615 were initially reviewed as new options for the GPS due to recommendations from the NASA CubeSat 101 paper. Since its publication, the NovAtel OEM615 has ceased manufacture, so a newer model, the NovAtel OEM7720 system was considered instead due to its precise positioning and small form factor. Both the Surrey SGR-05 and the NovAtel OEM7720 comfortably meet the location accuracy requirement and both systems are at a TRL 9 status (flight proven). Both systems require low power at about ~1 W of power. The performance of these systems has been confirmed through research and reviews of missions that have used these systems. The NovAtel OEM7720 has far better location accuracy and is smaller and lighter than the Surrey SGR-05U. The team is tentatively selecting the OEM7720 GPS system due to the aforementioned parameters. Before making a decision, the vendors of both systems still must be contacted to confirm the specifications and obtain the price for the NovAtel OEM7720, and the Orbits team must coordinate with the other subsystem teams to make sure that the GPS system can be properly integrated.

The specifications for the GPS units are presented in Table 15 below.

Table 15: GPS Specifications

Requirement		**Surrey SGR-05U**	**NovAtel OEM7720**
Location accuracy	**Within ± 100 m on ground**	± 10 m	± 1.5 m
Dimensions (mm)	**Minimized**	105 x 65 x 12	71 x 46 x 8
Mass (kg)	**Minimized**	0.04	0.029
Cost ($)		$17,675	TBD

COMMUNICATION AND DATA HANDLING

Requirements

The Communication and Data Handling subsystem includes the software, avionics, and radio elements of the spacecraft. The subsystem contains the CPU, which commands the peripheral scientific and spacecraft operations systems, and the radio architecture for communicating with the ground station. The specific operations requirements are outlined in Table 16 below.

Table 16: Communications and Data Handling System Requirements

ID	Requirement	Specification	Verification Method
F1.A.1	The FPGA will be responsible for managing the payload	Responding to input from the CPU, the FPGA will process data from the payload and pass it to the S-band transmitter to be stored.	Inspection
F3.A.1	Peripheral hardware will need to pass health telemetry to the CPU	The EPS, Solar Panels, ADACS, and GPS will need to pass pertinent health data directly to the CPU periodically.	Inspection
F3.A.2	The CPU will monitor the overall health of the spacecraft	The CPU will collect health telemetry from each subsystem and periodically produce a status report to be transmitted to the ground station.	Inspection
O3.A.1	The CPU will be responsible for managing the UHF radio	Periodically, the CPU will pass the stored images and health status reports to UHF for transmission to the ground station.	Inspection
O3.A.2	The FPGA will be responsible for managing the S-band radio	Responding to input from the CPU, the FPGA will command the S-band to perform data transmission to the ground station.	Inspection
O3.A.3	UHF will be responsible for the transmission for non-payload data	The UHF will periodically transmit digital image data and health status reports passed to it from the CPU.	Inspection
O3.A.4	The UHF will be responsible for handling transmissions from the ground station	Commands sent from the ground station will be received by the UHF and then be passed to the CPU to be processed.	Inspection

O3.A.5	S-band will be responsible for storing and transmitting payload data	Being passed payload data from the CPU, the S-band will store and then periodically transmit the data to the ground station.	Inspection
C4.A.1	The CPU will be responsible for managing the digital camera	Responding to telemetry from the ADACS and GPS, the CPU will turn the digital camera on/off and store the produced images in memory.	Inspection
C4.A.2	The CPU will manage the operation of the FPGA	Responding to telemetry from the ADACS and GPS, the CPU will send input to FPGA, communicating the operation cycle.	Inspection

Processor and FPGA

The Gomspace NanoMind Z7000 and NanoDock SDR were selected from available OBC options that matched the given requirements. The Z7000 consists of a combination ARM Core and FPGA, as well as all required clock, RAM, and storage components needed to function as the CubeSat's OBC; the NanoDock functions as a dock that the Z7000 must be slotted into in order to make external connections and write to removable memory, accessed through a USB connection or SD dock. It must be noted that the Z7000 is part of a modular system of chips and transceivers that can be slotted into the NanoDock, but for the purposes of the 3U CubeSat only the Z7000 is needed.

The Z7000 was selected for its combination FPGA and ARM core – the single-board shared hardware lowers transfer time between the two chips and streamlines communication between the central processor and peripheral hardware. While they share hardware, the arm Core and the FPGA are still functionally separate and data can be routed through each without disrupting the process of the other, allowing for completely discrete lines of data communication when necessary. For a full list of hardware components, architecture, and available communication protocols, consult the NanoMind Z7000 manual.

Radio and Ground Station

In order to support the mission's scientific objective, communication with the ground station was split into two different modes. For spacecraft health and handling data, UHF communication was selected, and for scientific data S-band communication was selected. The high volume of scientific data will be better supported by S-band transmission, and since the data only needs to go one direction (i.e. transmission only) then a simpler transmitter can be used in place of a transceiver. For the onboard communication hardware Endurosat was selected as the vendor due to alignment with system requirements and to standardize vendors across component areas. An Endurosat UHF Transceiver II with an Endurosat Antenna will be used for spacecraft health and handling data. An Endurosat S-band Patch Antenna and S-band Transmitter will be used to send the scientific data to the ground. The S-band transmitter will receive the scientific images and metadata from the CPU and can store up to 32 GB of data while waiting to downlink.

In addition to the onboard radio communication hardware, it is also necessary to have a functioning ground station. The University's current ground station is set up for UHF communication. Its major components include UHF antenna array, a USRP N210 software defined radio, as well as command and data processing servers. In order for the ground station to be used with this mission architecture, it must be updated to be compatible with S-band communications. This requires the purchase and installation of an S-band antenna, as well as the associated mounting and control hardware. The current USRP software defined radio can receive and transmit frequencies from 10 MHz to 6 GHz, so it can support S-band communications. Additionally, both the S-band and UHF radios would need to be tuned to the correct frequency, and adequate command software would need to be in place.

Data Communication Architecture

Pictured below in Figure 10 is a diagram of the communication between the CPU and the two radio modules, the S-Band transmitter (for payload data, sending only) and the UHF Transceiver (for sending system health data and receiving command updates).

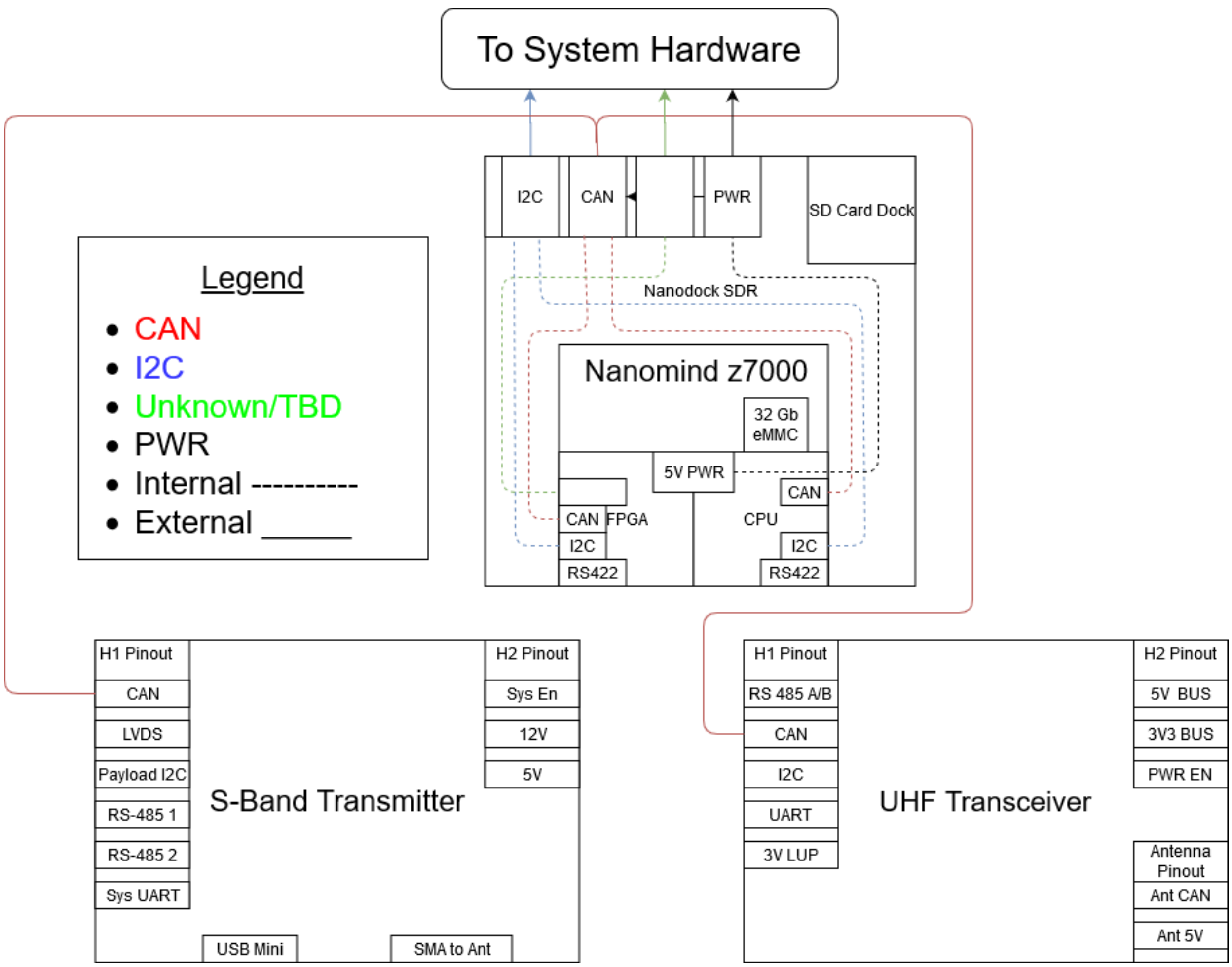


Figure 10: Data Communication Architecture

The CPU will communicate with the S-Band Transmitter and the UHF Transceiver, and vice-versa, by a CAN (Controller Area Network) communication protocol. The CAN protocol is a peer-to-peer, message-based protocol and was selected primarily for its simple hardware requirements (one clock wire and one data wire per network component) and redundant error-correcting data structure. Retaining the integrity of the data being sent and received by the satellite is a critical concern, especially over long distances and limited uplink/downlink windows.

The rest of the system hardware – Payload, EPS, Digital Camera, ADACS, and GPS – have been blackboxed in the diagram above due to uncertainties regarding some components. We have yet to confirm a specific model for GPS, Digital Camera, and ADACS; additionally, as the payload is custom-built, we have not yet confirmed which communication protocols it has access to. However, below is a list of the most common protocols we have been considering. Ideally a single protocol will be available between all system components and the CPU for the sake of hardware simplicity, but it is more likely that multiple protocols will be need to be selected from the following:

- **CAN - Controller Area Network BUS:** Previously Explained.
- **UART - Universal Asynchronous Receiver/Transmitter:** Short distance 1-wire physical interface, fast but not High Speed. No universal signal limit but most UART-equipped devices have their own baud rate cap. Simple digital data and physical structures. Asynchronous, not a good fit for devices with time-critical telemetry.
- **I2C (I-squared-C) Inter-Integrated Circuits:** Master/slave address-based protocol (essentially the inverse of CAN). Short distance physical interface with 2 wires (clock & data). Low hardware/pinout complexity, high digital/data complexity that is handled internally by each device. I2C has a higher theoretical baud rate than SPI or UART but this is balanced by more complex data package, more bits per transmission.
- **SPI - Serial Peripheral Interface:** 4-Wire, Master/Slave with a serial clock, two differential data wires, and a slave-select line (similar to a digital address). Bits are read by the voltage differential and data protocols must be implemented by the user. Not recommended due to hardware and software complexity.
- **RS422 and RS485**: Hardware-only signaling standards requiring 3 wires each to function, 2 entwined wires & 1 ground. Independent of a digital communication protocol, these only define the hardware setup – data is interpreted from the voltage differential between the 2 entwined wires. RS422 and RS485 are situationally compatible in 1 direction (RS422 can be configured to communicate with RS485 but not vice-versa). The hardware complexity makes this protocol a last-place candidate.

Data and Communication Flowchart: System Operating Images

During regular deployment operation, the CubeSat will be continuously powered and operate in one of three system images, detailed below.

1. **Passive:** The CPU monitors & regulates the onboard hardware and records telemetry from all components, which is written directly to the UHF transceiver hard drive.

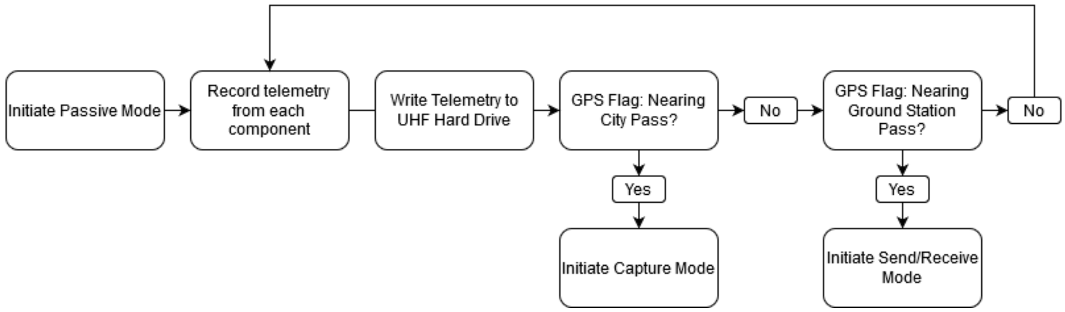


Figure 11: Passive Mode

2. **Capture:** The FPGA receives the images as well as time/location stamps from the payload hardware, pairs them with the photos from the digital camera, and stores them externally on the SDR

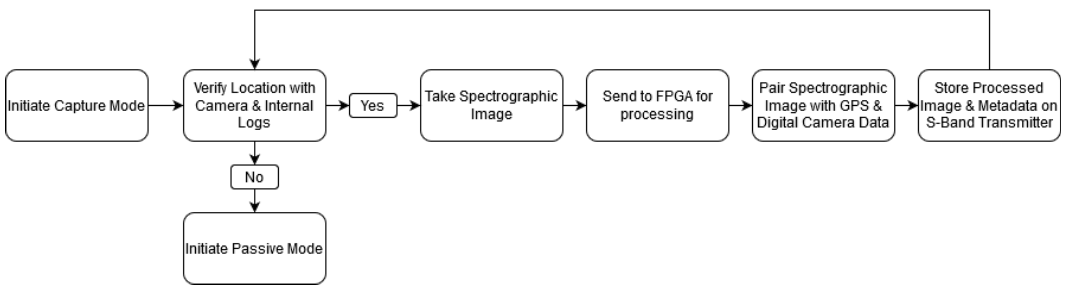


Figure 12: Capture Mode

3. **Send/Receive:** The CPU sends telemetry/system data to the UHF transceiver which transmits them to the ground station, and receives updates/ commands from the ground station via the UHF transceiver

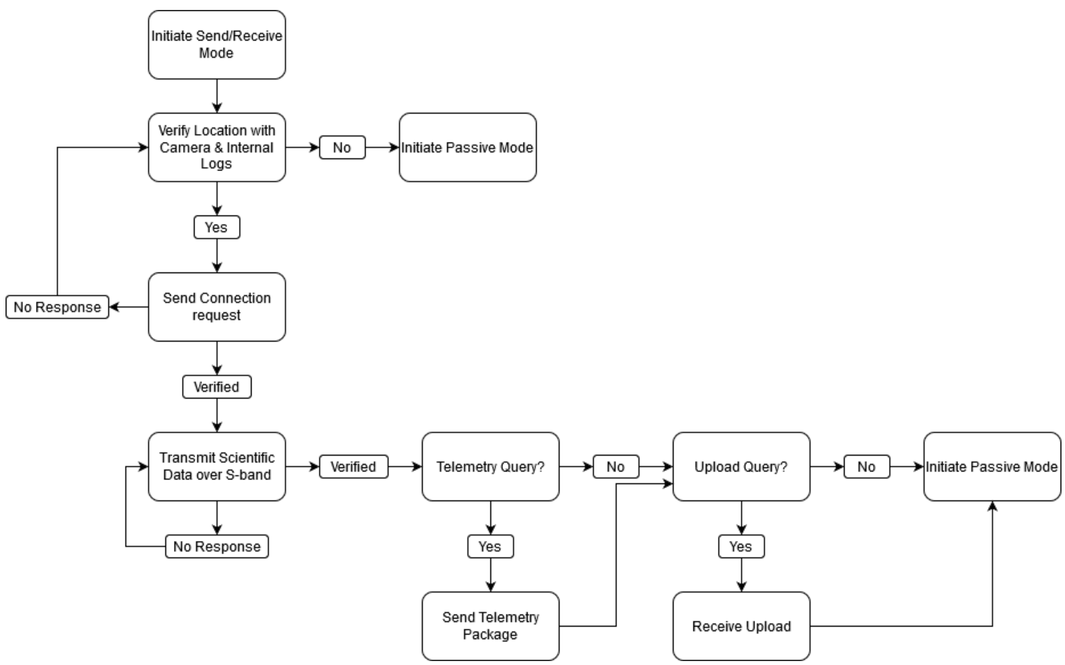


Figure 13: Send/Receive Mode

SYSTEM ASSEMBLY

The overall system assembly is depicted in Figure 14. The model illustrates the mission architecture, comprised of an electronics stack, the ADACS component, and the payload assembled within the 3U. The XY faces of the satellite are wrapped in solar panels, leaving only the star tracker apertures exposed. The nadir Z face will have the aperture for the spectrograph, as well as a smaller aperture for the optical camera. The zenith Z face will consist of the GPS and UFH antennas. The S-band antenna will be attached on a modular hinged surface mounted to the nadir edge, which will move into position after deployment.

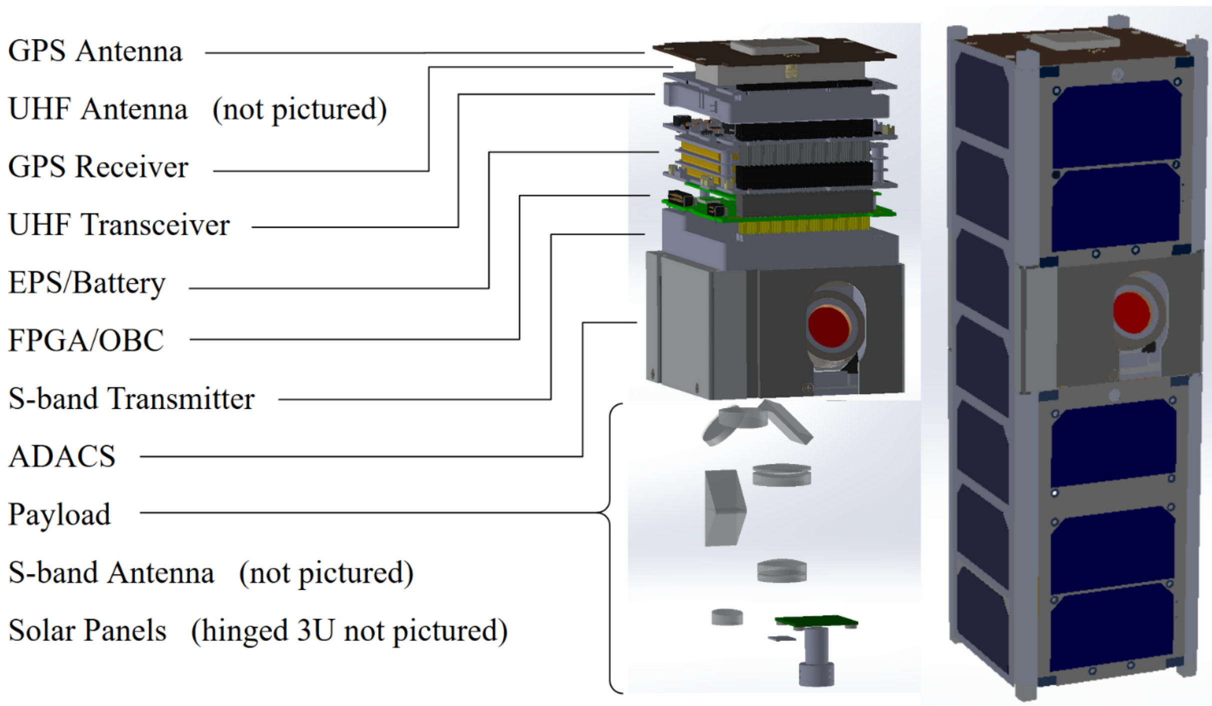


Figure 14: System Assembly

PROJECT MANAGEMENT

FINANCIAL BUDGET

The financial budget provided in Table 15 is a maintained list of all components within the spacecraft. Due to the ongoing evaluation of the GPS and ADACS components, various properties remain unknown. The payload integration is another major area that has yet to be finalized. The 3U long-edge deployable solar panels will be custom made by Endurosat; respective values are estimates based on the 3U short-edge deployable solar panels available from Endurosat. Fortunately, the budget shows that the current design is both within the mass limit for a 3U CubeSat (4 kg) and within the generous budget of $400,000 given within the mission constraints. However, this budget does not address the cost associated with the necessary upgrades to the ground station. Further, this budget is limited to material costs, and does not include the inevitable costs of labor, postage, licensing fees, or travel.

Table 17: Component Budget and Parts List

Component	Qty	Name	Manufacturer	Total Mass (g)	Price per Unit	Total Pri
1U Chassis	1	1U CubeSat Structure	Endurosat	98	1366.61	1366.61
1.5U Chassis	1	1.5U CubeSat Structure	Endurosat	114	1803.93	1803.93
GPS Antenna	1	TBD	TBD	**50	TBD	TBD
UHF Antenna	1	UHF Antenna II	Endurosat	85	3750	3750
GPS Receiver	1	TBD	TBD	**30	TBD	TBD
UHF Transceiver	1	UHF Transceiver II (Comm)	Endurosat	94	4375	4375
EPS/Battery	1	EPS I	Endurosat	208	2730.75	2730.75
FPGA	1	NanoMind Z7000	GomSpace	76.8	17000	17000
OBC Mount	1	NanoMind Dock	GomSpace	74.2	4000	4000
S-band Transmitter	1	S-band Transmitter	Endurosat	250	8500	8500
ADACS	1	TBD	TBD	**1000	**100000	100000
S-band Antenna	1	S-Band Antenna ISM Patch Antenna Type I	Endurosat	64	3000	3000

1U Solar Panel	2	1U Solar Panel X/Y	Endurosat	88	1875	3750
1.5U Solar Panel	2	1.5U Solar Panel X/Y	Endurosat	130	2750	5500
3U Solar Panel Deployables**	2	3U Single Deployable Long Edge	Endurosat	**600	**12500	25000
Optics	8	Lenses, Mirrors, Prisms	TBD	**160	**65	520
Detector Array	1	Custom	U.Va.	TBD	1000	1000
CMOS Board	1	TBD	TBD	**7	**100	100
Camera Lens	1	Standard Lens	TBD	TBD	TBD	TBD
Board Mount	1	Custom Made	U.Va.	TBD	TBD	TBD
Instrument Mount	1	Custom Made	U.Va.	**512	**5000	5000
TOTAL				**3634+**		**195029.68+**

SCHEDULE

The overall mission schedule is outlined below in Table 18. The schedule contains both previous milestones as well as future plans. The schedule for this mission has

Table 18: Mission Timeline

Date	Activity	Description
Fall 2018	Project Inception	Initial mission design completed
15 April 2019	Conceptual Design Review	First satellite design presented to collaborators
Summer 2019	Spectrograph Bench Testing	Creation and testing of a prototype payload
11 February 2020	Preliminary Design Review	Formalized design choices presented
15 April 2020	Spring Intermediate Design Review	End of academic year presentation

December 2020	Critical Design Review	Finalized design presented to collaborators
Spring 2021	Licensing and Manifest	Apply for FCC radio license and NOAA remote sensing license, seek additional funds, apply for NASA CSLI
Spring 2021	Build Phase	Part acquisition, create bus assembly and integrate payload, develop ground station
Summer 2021	Satellite Testing	Vibration, thermal, and vacuum testing mimic launch and the space environment
Fall 2021	Launch Preparation	Launch vehicle integration
Spring 2022	Launch	Launch of the spacecraft, followed by a deployment set by service providers

RISK ASSESSMENT AND RISK MITIGATION

As with any space mission, there are a multitude of risks associated with this CubeSat mission, and are listed divided by functional group in Table 19. This table contains both risk assessment and mitigation strategies. The potential issues posed within the Structures subsystem contribute a high risk to the overall mission success, as failures within the payload would severely hinder meeting mission objectives. Similarly, thermal fluctuations outside the operating temperature range may permanently damage critical components, leading to another high risk for the mission. Insufficient power poses a moderate to high risk, as a lack of power may prevent the spacecraft or payload from operating properly. However, the spacecraft may be able to recover from periods of low power by conserving energy within a passive state. Debris impact poses a moderate risk; though the consequences are severe, the probability of a catastrophic collision is low. Thorough component and environmental testing coupled with computer modeling will increase mission confidence and can decrease the likelihood of unforeseen or unmitigated risks.

Table 19: Potential Risks

Subsystem	**Assessment**	**Mitigation**
Structures and Integration	• Trusting of FEA model • Unknown and unclear material property/loads/optics behaviors • Potential change of payload configurations	• Well-meshed model, convergence study, used of 2D quad/3D parabolic elements • Used safety factor of 2, perform material testing/calibration, carefully study loads

	• Trusting of supplier specifications	• Removable, optic workbench influenced design
Power, Thermal, and Environment	• Insufficient power to components • Corrosion of connections • Possible overheating of equipment	• Detailed accounting and testing of power requirements • Thermal modeling of environment to determine heater/cooler necessity
ADACS/Orbits	• Stabilization after deployment • Impact with debris	• Constant monitoring through two-line element (TLE) data tracking • Preprogrammed maneuvers to stabilize upon deployment
Communications and Data Handling	• Potential difficulty communicating with ground station, loss of radio	• Minimize transmission volume • Test ground station and spacecraft radio before launch
Program Management	• Loss of information through project hand-off • External factors affecting timeline (grants, licenses)	• Proactive involvement of future team members to promote overlap • Early applications and constant checks on compliance to ensure approval

FUTURE WORK

Throughout this report, known action items have been alluded to which outline the next steps in the development of this mission. To complete the design of the payload, the arrangement of the optics needs to be finalized. This includes both the configuration of the slit and its housing, as well as an understanding of how the temperature fluctuations in the space environment affect the focus of the optical system. The choice of a GPS and ADACS component must be finalized, and they must be compatible with the avionics and onboard processing. As suggested above, further analysis needs to be completed to determine the capabilities of the EPS system with respect to the voltage demands of the larger 3U solar panels. Additionally, the wiring schematic must be completed. The optical camera and the S-band antenna have yet to be integrated into the final design, two critical components for validating and transmitting the scientific data collected throughout the mission duration. Along with finalized component choices, the budget can be updated to reflect actual material costs, and can be expanded to include non-material costs.

Many external logistics have yet to be developed. Once the Critical Design Review is completed, the mission will need to obtain licenses from the FCC and NOAA to communicate with the ground station and to capture photographic images of the target cites. Further, the mission must be submitted to the NASA CSLI program to be manifest on a launch vehicle. Within this timeline, additional funding may be necessary to meet the needs described in the financial budget, and grants may be applied to during the remainder of the mission. Completing these steps will increase mission readiness and prepare the spacecraft for the next phase of environmental testing and launch integration.

CONCLUSION

The design and development of a novel 3U CubeSat will allow for high spatial resolution spectroscopic imaging of the anthropogenic pollutant nitrogen dioxide from Low Earth Orbit. The custom payload will be able to capture the high spatial gradients of NO_2, allowing environmental scientists to better detect and identify mobile and stationary sources of air pollution in nine urban areas distributed worldwide. The proposed satellite greatly improves upon the capabilities of existing atmosphere-observing spacecraft, while reducing costs and size.

In addition to the payload, allotted to half the spacecraft volume, the satellite contains numerous electronics which both provide power and control the spacecraft's operations. The onboard processors manage the various functionalities of the satellite, such as data collection, maneuvering, and communication with the U.Va. ground station. Solar panels wrap the exterior of the spacecraft, protecting it from the space environment in addition to providing power.

With the work completed over the 2019-2020 academic year, the mission is nearing the close of its design phase. Future work will be dedicated to finalizing component choices, assembling and testing the spacecraft, and acquiring the necessary licenses prior to launch.

The unique capabilities of this 3U satellite does not only expand the potential of CubeSats – with its payload, this satellite will enhance our knowledge of local air pollution sources, as well as the expand the application of atmospheric-sensing spacecraft within the larger context of the global environment.

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