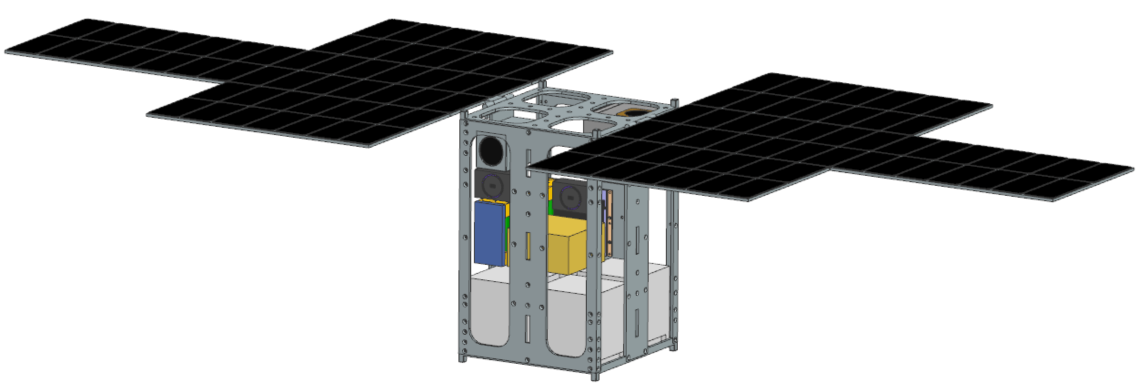
Venus Exploratory CubeSat Transfer Orbit Researcher (VECTOR)

Critical Design Review Report



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Abbreviations and Acronyms

ABET	Accredited Board for Engineering and Technology
ADCS	Attitude Determination and Control System
ASU	Arizona State University
BCT	Blue Canyon Technologies
CDR	Critical Design Review
COTS	Commercial Off The Shelf
CPU	Central Processing Unit
DSN	Deep Space Network
EPS	Electrical Power System
IMU	Inertial Measurement Unit
LEO	Low Earth Orbit
MEMS	Micro Electro Mechanical System
OBC	On Board Computer
PDR	Performance Development Review
PSG	Prime Solutions Group
SDR	System Design Review
SIR	Systems Integration Review
SOI	Sphere of Influence
SRR	System Requirements Review
SMD	Science Mission Directorate
TCS	Thermal Control System
TRL	Technology Readiness Level
VECTOR	Venus Exploratory Cubesat Transfer Orbit Researcher

1. Introduction

1.1 Overview

The Venus Exploratory Cubesat Transfer Orbit Researcher (VECTOR) project is a cost-effective CubeSat mission to Venus, intended to study lightning and weather patterns in the Venusian atmosphere while also increasing the technology readiness level (TRL) of commercial off the shelf (COTS) spaceflight components. The critical design review shows an overview of the mission along with all the necessary system requirements, subsystem requirements, system architecture, design considerations and choices, risk analysis, verification and validation, and future plans.

1.2 Mission Statement

Despite being one of the closest planets to Earth, relatively little is known about Venus's atmosphere, weather patterns, and toxicity. To continue to expand as an interplanetary civilization, we must increase our understanding of these aspects of Venus. Currently, however, the costs of spaceflight make Venusian research difficult. This mission will first be a proof of concept for how a small satellite can reach Venus and perform research at a much lower cost and weight than a conventional satellite. Secondarily, it will serve as a research mission to verify the existence of lightning and gather data about the atmosphere and weather patterns.

1.3 Constraints

The mission will be subject to two main constraints: a monetary budget and launch requirements. An initial budget of $25 million will be set for the project development and includes funding from government and private sector sources so that development can continue unhindered by finances. The launch segment will provide a few key constraints the mission must abide by, the first being that the system shall follow the 12U CubeSat size standard. This will allow easy stowing in the Photon spacecraft bus provided by Rocket Lab, which will also be the launch vehicle provider for the mission. Due to the interplanetary nature of the mission, the system will be subject to a launch window constraint, as Venus makes its closest approach to Earth once every 1.5 years (give or take .5 years). VECTOR is scheduled for launch on 12/10/2024. The final constraining factor is the total CubeSat mass, which will be fixed at a maximum of 24 kilograms.

1.4 Assumptions

The overall goals of this mission can be achieved given that two important assumptions are met. First and foremost, it is assumed that Rocket Lab will provide the launch vehicle, Electron, and the satellite bus, Photon. Electron's second stage, Photon, will perform the critical transfer orbit burn needed to set VECTOR on its initial trajectory towards Venus. Photon will house the CubeSat during this cruising stage, performing necessary course correction burns along the way while subsequently providing power and establishing preliminary communications to Earth. [11] This leads to the second important assumption: the system will utilize NASA's Deep Space Network for all of its communications capabilities. This is crucial for the mission's success as data can be inherently difficult to transmit from interplanetary distances. The mission can not afford to lose any valuable scientific data gathered while in orbit around Venus.

1.5 Societal Impacts

By successfully sending a CubeSat into Venus' orbit, this mission aims to inspire the next generation of scientists and engineers. We strive to complete this with a social campaign through NASA, ASU, and social media. The staff will provide updates, interesting graphics, and live streams which will reach hundreds of thousands of people. Additionally, taking an EDU or 3D printed model into local schools around Arizona will further push our goal.

Our team plans to implement programs within our local high schools to teach students about aspects of the mission and the importance of the data we will gather. These programs will vary based on the stage the mission is at. For example, before launch our outreach efforts will focus on university students and offering internships to acquire more hands on deck while simultaneously exposing our mission to top universities. During and after the mission, the team will go to local schools ranging from elementary to high school where we can take our findings and interact with students, teaching them how the mission was completed and what we have learned. This will allow our team to get the younger generations interested in space exploration in general, but also excited for our Venus mission specifically.

Furthermore, creating accounts or pages with social media outlets like Instagram, Twitter or Facebook will be vital for our team to teach people around the world about the mission. The ability to post status updates and share images we gather allows for the VECTOR name to be known around the world and allow for us to get more funding down the line. Social media can also allow for the engineers on our team to speak directly with the public and answer questions about our mission, CubeSat architecture, or the planet Venus for those who want to learn more about these topics.

The total outreach plan will use 0.1% of the mission budget with a 20% margin added in bringing the total budget allocation to $30,000. This money will not be pulled from the $25,000,000 mission budget, rather this will be an additional grant awarded by NASA under one or multiple of the outreach grant opportunities listed on the Grants.gov webpage The outreach plan will become operational following the approval of the CDR and will end two weeks after the launch.

In addition to inspiring the next generation, this mission aims to continue to add to scientific advancements around Venus lightning verification and Venus weather patterns. By using this proposed payload, it may become a standard payload which can detect and track lightning on any planet in our solar system! Therefore, this mission aims to increase TRL's of various components which will be used on board VECTOR. By being the first CubeSat to enter into a non-Earth planetary orbit, VECTOR will be a leader for future CubeSat missions to follow in our footsteps and continue to advance the smallsat industry.

1.6 Stakeholders

The mission will involve several key stakeholders who will benefit most from VECTOR's success. First and foremost, our primary customer will be NASA. They will be providing us with our largest portion of funding, the mission goal directly references the decadal survey of 2022 as well as follows NASA SMD. The secondary customer will be ASU as the CubeSat will be designed, built, and managed by ASU engineers and students. Also, the providers of the mission will also act as stakeholders in our mission and will include our launch provider Rocket Lab, communications user JPL, and smallsat companies providing hardware (Ibeos, BCT, MMA design, etc). Finally, end users such as planetary research scientists will take interest in the results, and will further propel university and industry missions.

2. Mission Level Evolution & Selection: Analysis of Alternatives

2.1 Analysis of Alternatives Description

An analysis of alternatives strives to explore the various ways in which the defined mission statements can be fulfilled while still abiding by our requirements and constraints. As part of the mission concept review MCR, a mission level trade was performed. This will be reviewed first before going into the selected Venusian mission.

For the mission selected, three tables were constructed, each concerned with an aspect of our mission that can be achieved in different but reasonable ways. These tables were divided based on the type of components the CubeSat will be composed of, the type of flight trajectory that the mission will take, the type of propulsion that will be utilized to reach the final destination, and lastly, the payload that will be onboard our CubeSat.

Furthermore, each table will be defined by selection elements determined based on the previously outlined requirements. As such, for the first three tables, the selection elements include cost, payload capacity, lifetime, and reliability. Each selection element was carefully weighed in relation to the other, with the lowest total score representing the best option. As a result, the cost was determined to weigh the highest and is therefore the most crucial deciding factor when comparing the alternatives.

It is important to note that in the last table, the one in which different payloads are analyzed, a new set of selection elements were created to compare and contrast each alternative properly. These selection elements include cost, size, future missions, and societal impacts. Cost is still weighed highly relative to the other elements. It is worth mentioning that future missions (defined by how likely the data collected by VECTOR will influence and possibly propel another Venusian mission) and societal impacts were weighed highly as these two elements fulfill the secondary mission statement.

The analysis of alternative tables was created to delve into the multitude of ways VECTOR can be achieved, pushing the mission toward reaching its full potential.

2.2 Initial Mission Selection

The initial trade matrix for selecting a mission is seen below. The left column is a list of potential mission ideas from each team member. The 5 selection elements and their respective weights are commercial value representing 10%, scientific value representing 10%, delivery difficulty representing 30%, payload difficulty representing 30%, and path forward representing 20%. The grids are color coded to represent the average and standard deviation between selections from each member. In the end, the Cislunar and Venus CubeSat received the highest points, but ultimately the Venus CubeSat was chosen based on general interest between the group.

Table 2-2-1 Analysis of Alternatives 1.0

Team Totals	For definitions, please see Report 1. Weights are 10%, 10%, 30%, 30%, 20% respectively.											
	Commercial value:		Scientific value:		Delivery difficulty:		Payload difficulty:		Path forward:		Total points:	
Project:	Average	Std	Average	Std	Average	Std	Average	Std	Average	Std	Sum	Std
Venus Cubesat	1.83	0.41	4.17	0.75	3.67	0.82	4.00	1.10	4.17	0.98	3.73	0.99
Cislunar Cubesat	2.00	0.63	4.33	1.03	4.67	0.52	3.00	0.63	4.33	0.82	3.80	1.13
Titan Lander	2.67	1.86	4.50	0.84	1.33	0.52	2.33	0.82	3.33	0.82	2.48	1.18
Enceladus Lander	2.17	1.17	4.17	1.17	1.33	0.52	1.67	0.52	3.00	0.63	2.13	1.14
Europa orbiter	2.00	0.89	3.83	1.33	1.67	0.82	2.67	0.82	2.83	0.41	2.45	0.84
Asteroid mission	3.33	1.37	3.83	0.75	2.00	0.00	2.33	0.52	3.17	0.75	2.65	0.75
Kuiper belt objects	2.00	0.89	4.33	0.52	1.67	0.52	2.67	0.82	3.33	0.82	2.60	1.07
Remote mining	4.83	0.41	3.33	1.37	1.83	0.41	2.33	0.52	4.17	1.17	2.90	1.24
Propellant depot	4.50	0.55	2.00	1.55	2.50	0.84	1.83	0.75	3.83	0.75	2.72	1.18
Observatory for moon	2.83	1.33	3.67	1.21	2.33	0.82	2.50	1.38	3.00	0.63	2.70	0.52
Modular propulsion asteroids	2.83	1.33	2.33	0.52	2.17	0.41	3.33	0.82	3.33	1.03	2.83	0.55

2.3 Mission Components

The trade show in Table 2-3-1 compares the methodology of subsystem/component composition for the VECTOR CubeSat. The options include constructing a CubeSat made of custom components or commercial off-the-shelf (COTS) components. When it comes to the cost factor, designing custom components for a satellite is notoriously more expensive than purchasing commercial off-the-shelf materials. Not only that but the licensing and testing of these components can significantly add to the overall cost, pushing well into the millions of dollars spent. For COTS components, climbing expenses often only stem from needing to redesign the system. When it comes to payload capacity and lifetime between custom and off-the-shelf, there is no significant difference. Finally, when analyzing the difference in reliability between custom and COTS components, the custom has a slightly higher reliability, but the increasing TRL of COTS is steadily improving this differentiation. Ultimately, the team chose COTS components as the primary methodology of obtaining hardware.

Table 2-3-1 Analysis of Alternatives 1.1

Selection Element:	Weights	Custom Components		COTS	
		Motivation	Score	Motivation	Score
Cost	**0.5**	New parts incur high R&D costs + procurement	4	Only procurement cost	2
Payload Capacity	**0.25**	No significant advantage	2.5	No significant advantage	2.5
Lifetime	**0.15**	No significant advantage	2.5	No significant advantage	2.5
Reliability	**0.1**	Slight significant advantage	2	Increasing the TRL	3
Decision Scores:	**1.0**		3.2		2.3

2.4 Mission Flight

The next trade shown in Table 2-4-1 concerns the type of orbital trajectory the mission will follow. The options include a fly-by mission versus an insertion into Venusian orbit. When considering the cost of each, it was noted how a fly-by would possess a reduced cost caused by the decrease in delta-V requirements, though it would result in a higher net cost per mission due to the reduced time spent observing the target. Furthermore, though an insertion into Venusian orbit would require higher upfront costs per launch, it is ultimately more cost-effective in the long term due to increased time orbiting and observing the target. It was also determined that insertion would reduce payload capacity due to the increased fuel requirement. However, despite these conclusions, it was found that a Venusian orbit insertion would have a much longer lifetime and would provide ample opportunity to fulfill the secondary mission statement. In the end, the ability to conduct scientific analyses outweighed the need to reduce cost and therefore served as the reasoning behind the final trajectory decision. Venusian orbital insertion was selected for VECTOR.

Table 2-4-1 Analysis of Alternatives 1.2

Selection Element:	Weights	Fly by		Insertion	
		Motivation	Score	Motivation	Score
Cost	**0.5**	Lower cost per mission, higher overall cost	3	Higher cost per mission, lower overall cost	2
Payload Capacity	**0.25**	Increased payload capacity by 10-15%	2	Decreased payload capacity by 10-15%	3
Lifetime	**0.15**	Limited to time of fly-by	4	Much longer time in service	2
Reliability	**.1**	Less reliable data Less time in proving tech	4	More data available Longer tech life-time	2
Decision Scores:	**1.0**		3		2.25

2.5 Mission Propulsion

Table 2-5-1 looks at the ways in which VECTOR can reach Venus. The options include a fully self-propelled mission and an assisted propulsion to Venus. When considering the cost, it was determined that a self-propelled mission would require more fuel and subsequently more power costs. At the same time, the payload capacity for the assisted propulsion would allow for a larger area due to the decreased power and fuel requirements. Lastly, an assisted mission would last less time due to the fact that there is no necessity to build and prepare for a longer mission. Ultimately, the benefits of an assisted mission greatly exceed those of a fully self-propelled mission and this was the VECTOR selection.

Table 2-5-1 Analysis of Alternatives 1.3

Selection Element:	Weights	Fully self-propelled		Assisted propulsion	
		Motivation	Score	Motivation	Score
Cost	**0.5**	More fuel & power costs	4	Less power and fuel required	2
Payload Capacity	**0.25**	Smaller area due to increased battery capacity and propulsion fuel reserves	4	Larger area for science mission due to less power or fuel requirements	2
Lifetime	**0.15**	Travel alone + science mission = Build for a Longer mission	2	Science mission will last less time due to no travel	4
Reliability	**.1**	More design/development risk	3	Less design/development risk	2
Decision Scores:	**1.0**		3.6		2.3

2.6 Mission Payload

As previously stated, the analyses of the payloads the VECTOR CubeSat could house differ from the preceding tables. Three different payloads were determined to be capable of collecting relevant data from Venus. The first payload was a radio, which would map dust and the different sulfur species within the Venusian atmosphere. Next was the magnetometer and electrostatic analyzer, which would catalog the weather surrounding the planet as well as the solar wind interaction with Venus. The last payload considered was the neuromorphic camera, which is capable of recording events such as lightning and meteors within the atmosphere. Each are shown in Table 2-6-1.

Table 2-6-1 Analysis of Alternatives 1.4

Selection Element	Weight	Radio		Magnetometer and Electrostatic Analyzer (ESA)		Neuromorphic Camera	
		Motivation	Score	Motivation	Score	Motivation	Score
Cost	**0.5**	Cost effective	3	Cost effective	3	New and not as cheap	4
Size	**0.1**	No significant advantage	2	No significant advantage	2	No significant advantage	2
Future Missions	**0.2**	Ability to understand the chemical composition of the planet	2	Predict the space weather around Venus, and how Solar Winds interact with the planet	2	Catalog the weather and environment of Venus	1
Societal Impacts	**0.2**	Interesting, but not wholly eye-catching	4	There may not be much interest about what occurs around Venus	3	Lightning and meteor strikes hold potential to excite	1
Scores	**1**		2.9		2.7		2.6

Table 2-6-1 displays the analysis of alternatives for the CubeSat payload. Radio has been built into numerous CubeSats, and the cost depends mostly on the product and supplier. Furthermore, the ability to radio-map different sulfur species and the dust on the Venusian surface ultimately means that scientists will be able to understand the planet's chemical composition and respond by designing missions that can go even further than VECTOR. This could lead to landers visiting Venus that can withstand the harmful environment. However, radio mapping may not be the most important or interesting scientific data collection for the average individual that our missions could provide. Though interesting, the results would not be incredibly eye-catching for the public.

The magnetometer and electrostatic analyzer are fairly cheap and often used in CubeSats within Earth orbit. The magnetometer and ESA would be able to inform scientists of what happens around Venus, compared to what happens in Venus–as shown with both the radio and neuromorphic camera options. Collecting data around the planet would help future missions be able to predict the space weather around Venus and how solar winds conduct a magnetic field around the planet. The public may also not be interested in what happens around another planet, and therefore the data would not have much of an impact on the average individual.

Neuromorphic cameras are relatively new technologies for their usage and implementation into CubeSats. As such, the cost of incorporating them into the mission would be more costly than the other two options. Since scientists believe that there is lightning on Venus, either coming from the thick cloudy layers above the surface or possibly from the volcanic clouds beneath, the neuromorphic camera would be able to catalog the data surrounding the weather of Venus and determine how hostile the Venusian environment is. This could better inform us how to approach Venus and prepare for future missions to the inhospitable, toxic planet. Lightning and meteor strikes hold the potential to excite the average individual. Such data, if determined to exist on Venus, would be interesting news to the public, especially if the images captured were published. The valuable scientific endeavors promised by the neuromorphic camera, therefore, provide the reasoning for the final decision.

2.7 Final Mission Selection

In the end, the VECTOR CubeSat will be built with COTS components, have assisted propulsion, be injected into a Venusian orbit, and have a neuromorphic camera as its payload. Table 2-7-1 demonstrates the conclusions reached through the analysis of alternatives process.

Table 2-7-1 Alternatives Selection

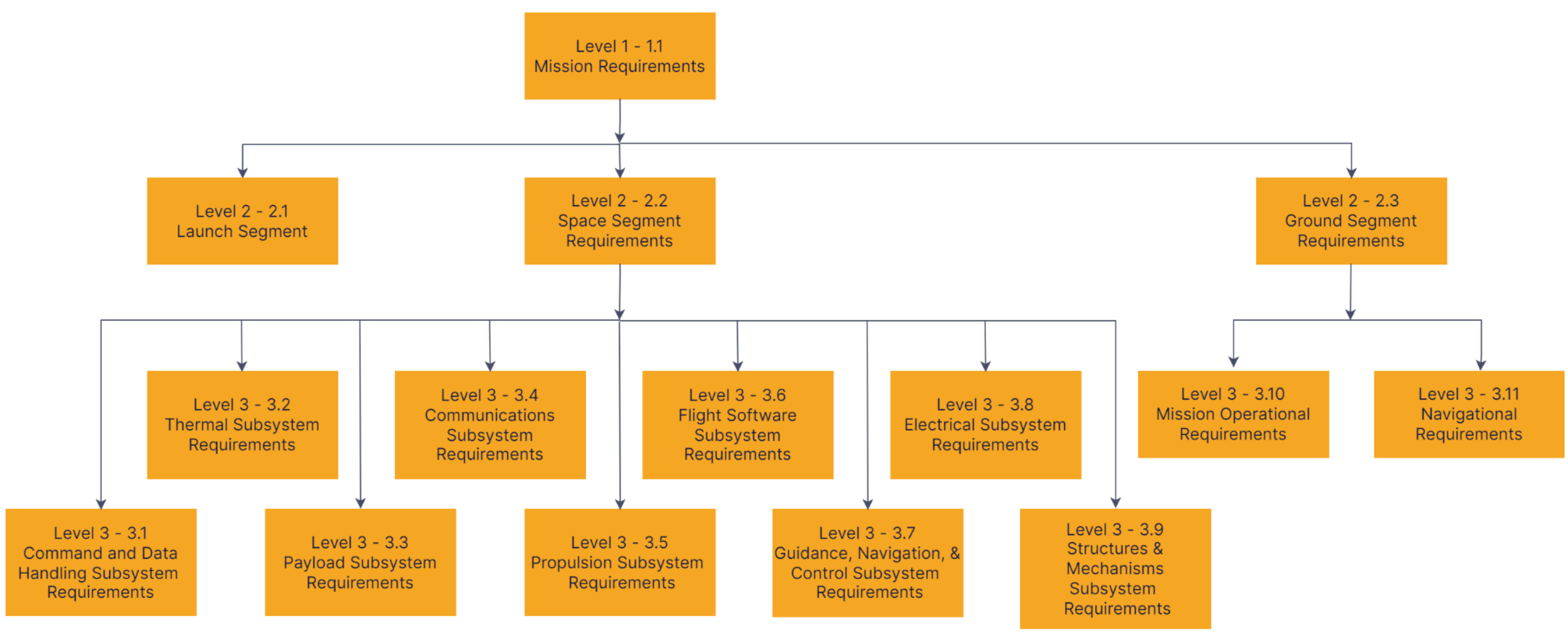
Trade Study:	Selection:	Motivation:
1.1 - Component design	COTS Components	Reduced R&D costs, higher compatibility
1.2 - Orbit	Insertion into Venus Orbit	Longer mission lifetime, more science capability
1.3 - Propulsion to Venus	Assisted propulsion	Higher payload capacity, reduced fuel requirement
1.4 - Payload	Neuromorphic Camera	Slightly more costly, but would engage public interest and fulfill secondary mission statement

3. System Level: Requirements

3.1 Requirements Description

The requirements for VECTOR are shown below in Figure 3-1, consisting of three levels of requirements flowing downward from Level 1 to Level 3. The Level 1, Mission Requirements, are shown in Section 3.3, these are the highest level of requirements for the mission to be completed, all the following requirements that are discussed in this report fall under this category. The Level 2, Launch, Space and Ground Requirements are shown respectively in Sections 3.4, 3.5 and 3.6, these demonstrate the specifics of VECTOR's high level system requirements. The Level 3, Sub-System requirements, are shown in Section 5.0, and go into greater detail of the crucial functions of our sub-systems, which in turn allow for VECTOR to meet the Level 1 requirements. VECTOR will also be held to the standards set forth by NASA-RP-1124-Rev-4, Cal Poly CubeSat Design Specification Rev. 14.1, NPR 8715.6, and AFSPCMAN 91-710, Volume 3. Refer to Appendix. Requirements tracking is shown by a status column on the subsequent tables. These requirements are continuously updated as the mission progresses.

Figure 3-1-1 Requirements Flow Down



3.2 Total Requirements

Table 3-2-1 breaks down how VECTOR's 78 different requirements are separated, ranging from system, subsystem, design to performance level requirements, all of which are crucial for the mission goals to be reached. Table 3-2-1 assists in visually understanding which aspects of the mission the team must focus on, as shown below there are 17 spacecraft requirements while only 4 launch and 4 ground requirements. This aligns with the fact that VECTOR's main tasks begin when the cubesat exits the Electron rocket. An in-depth analysis for the Level 3 requirements will be found in each individual subsystem section located in Section 5.0.

Table 3-2-1 Requirements by Type

Requirement Type	Amount
Mission	3
Launch	4
Spacecraft	17
Ground	4
Subsystem (i.e Level 3)	34
Critical Design	4
System Performance	6
Standards	7

3.3 Mission Requirements

The two requirements shown below flow down directly from the mission statement, that VECTOR will build a CubeSat using COTS components in order to get into the Venusian orbit and investigate the lightning patterns in the atmosphere, resulting in an increase of the TRL for the COTS components. The requirements simply call out the goals of our mission, all the requirements discussed in this report will connect back to the two general requirements. MR1.1 connects directly to the primary mission goal of reaching Venus with the COTS components. MR1.2 relates to VECTOR's secondary goal of performing useful science with an inexpensive instrument, which as previously stated we have chosen to investigate the lightning and weather patterns in Venus' atmosphere.

Table 3-3-1 Mission Requirements

ID	Requirement	Verification	Parent
MR1.1	System shall execute a Venus orbit insertion burn	Demonstration	Mission Statement
MR1.2	System shall perform a science objective based on the Decadal survey	Demonstration	Mission Statement
MR1.3	System shall increase TRL of cubesat COTS components	Demonstration	Mission Statement

3.4 Launch Requirements

The following four requirements lay out the basis of what needs to occur during the launch period of the mission. Due to the fact that VECTOR will be leaving Earth's atmosphere aboard Rocket Lab's Electron Rocket, this section is fairly short. LS-2.1 calls out VECTOR's presence in Electron, while LS-2.2 to LS-2.4 discusses VECTOR's ability to separate from Electron and insert into the Venusian orbit.

Table 3-4-1 Launch Requirements

ID	Requirement	Verification	Parent	Child	Status
LS-2.1	System shall be designed for launch onboard Rocket Lab Electron Rocket, as stated in "Payload User's Guide V6.6"	Inspection	MR1.2	TBD	Meeting
LS-2.2	Launch system shall have ability to insert spacecraft into Venus transfer orbit	Demonstration	MR1.1	TBD	Meeting
LS-2.3	System shall have primary vibrational and acoustical modes that do not correspond to launch vehicles primary modes	Testing	MR1.2	TBD	TBD
LS-2.4	Launch shall occur only when Venus is nearing closest approach to Earth	Inspection	MR1.1	TBD	Meeting

3.5 Spacecraft Requirements

The following requirements define certain tasks VECTOR needs to have the capability to accomplish, as well as basic definitions for what will occur while VECTOR orbits Venus. Many of these requirements will be discussed in the subsystems sections, where there will be a breakdown of how VECTOR plans to achieve the requirement. As well as, calculations for certain requirements like SC-2.2, and SC-2.17 will be shown in the respective subsystem sections, in this case 510 days was found after completing the needed propulsion calculations. The following discussion calls out a few of the crucial requirements:

Requirement SC-2.4 discusses the end-of-life plan for VECTOR, since the CubeSat will not be returning to Earth, there will be a certain amount of propellant left in order to complete a controlled deorbit plan, where VECTOR can fall into Venus' atmosphere, potentially allowing for more data to be gathered on the way down. Requirement SC-2.7 specifies that until VECTOR leaves the Electron rocket, the capabilities of the rocket will not be used, therefore saving power and systems for the crucial parts of the mission. Requirement SC-2.3 relates to Figures 6-2-1 and 6-2-2, which are breakdowns of our power budgets, those budgets are constrained by this requirement. These requirements will be verified through various means like inspection, analysis, demonstration or testing.

Table 3-5-1 Spacecraft Requirements

ID	Requirement	Verification	Parent	Child	Status
SC-2.1	System shall have a minimum 2U payload capacity	Inspection	MR1.2	TBD	Meeting
SC-2.2	System shall have a minimum operational life of 510 days	Analysis	MR1.3	TBD	Meeting
SC-2.3	System shall generate sufficient power for each subsystem	Testing	MR1.2	TBD	Meeting
SC-2.4	System shall have a controlled deorbit plan	Analysis	MR1.3	TBD	Meeting
SC-2.5	System shall have ability to power on and off	Demonstration	MR1.1	TBD	Meeting
SC-2.6	System shall be able to stay within allowable temperature range for each subsystem	Testing	MR1.2	TBD	Meeting
SC-2.7	System shall be powered off from time of delivery through orbit delivery	Testing	MR1.1	TBD	Meeting
SC-2.8	System shall wait a minimum of 30 minutes after deployment switches triggered before releasing deployables	Testing	MR1.2	TBD	Meeting

SC-2.9	System shall have attitude control accuracy to meet pointing constraints	Testing	MR1.2	TBD	Meeting
SC-2.10	System shall retain operational integrity throughout the mission duration	Testing	MR1.2	TBD	Meeting
SC-2.11	System shall be protected against radiation during entire duration of mission	Testing	MR1.3	TBD	TBR
SC-2.12	System shall have at minimum one deployment switch	Inspection	MR1.2	TBD	Meeting
SC-2.13	System shall be capable of communication back to Earth on average once every Earth day	Test	MR1.2	TBD	Meeting
SC-2.14	System shall have redundant downlink methods	Inspection	MR1.2	TBD	Meeting
SC-2.15	System shall incorporate mechanism(s) to prevent premature subsystem operations	Inspection	MR1.2	TBD	Meeting
SC-2.16	System shall provide flight system telemetry	Demonstration	MR1.2	TBD	Meeting
SC-2.17	System shall have enough fuel to remain in desired orbit for 510 days	Analysis	MR1.1	TBD	Meeting

3.6 Ground Requirements

The following four requirements define VECTOR's need to communicate back and forth with Earth and the ability to send data back to Earth. All these requirements will be verified through an inspection process. This will also be discussed in greater detail in the Communication and Data Handling section.

Table 3-6-1 Ground Requirements

ID	Requirement	Verification	Parent	Child	Status
GS-3.1	System shall be operational during the entirety of the mission	Inspection	MR1.3	TBD	Meeting
GS-3.2	System shall interface with the flight system to upload commands	Inspection	MR1.2	TBD	Meeting
GS-3.3	System shall interface with the flight system to receive and process telemetry and payload science data	Inspection	MR1.2	TBD	Meeting
GS-3.4	System shall be available for communication with the flight system at any point during a 24-hour period	Inspection	MR1.2	TBD	Meeting

3.7 System Performance Requirements

Table 3-7-1 shows the System Performance Requirements, which fall under the Level 2 requirements category, these define the necessities for the system to perform in order to meet our main mission goals. None of these requirements are new, they were gathered from the previous Level 2 requirements, specifically the Launch and Spacecraft sections. These requirements were chosen since they were found to be the main drivers relating to system performance.

Table 3-7-1 System Performance Requirements

ID	Requirement	Verification	Parent	Child	Status
LS-2.3	System shall have primary vibrational and acoustical modes that do not correspond to launch vehicles primary modes	Testing	MR1.2	STR-3.9.1 STR-3.9.2 STR-3.9.3 STR-3.9.4	TBD
SC-2.2	System shall have a minimum operational life of 510 days	Analysis	MR1.2	TCS-3.2.1 TTC-3.4.2 TTC-3.4.5 PRO-3.5.2 EPS-3.8.1 EPS-3.8.3 EPS-3.8.4	Meeting
SC-2.3	System shall generate sufficient power for each subsystem	Testing	MR1.2	EPS-3.8.1 EPS-3.8.3 EPS-3.8.4	Meeting
SC-2.5	System shall be able to stay within allowable temperature range for each subsystem	Testing	MR1.2	TCS-3.2.1 TCS-3.2.2 TCS-3.2.3 TCS-3.2.4	Meeting
SC-2.13	System shall be capable of communication back to Earth on average once every Earth day	Test	MR1.2	TTC-3.4.1 TTC-3.4.2 TTC-3.4.3 TTC-3.4.4 TTC-3.4.5	Meeting
SC-2.22	System shall contain enough propellant to satisfy on-orbit stationkeeping requirements	Testing	MR1.1	PRO-3.5.1 PRO-3.5.2 PRO-3.5.3 PRO-3.5.4 PRO-3.5.5	Meeting

3.8 Critical Design Requirements

Table 3-8-1 shows the Critical Design Requirements, these were the controlling factors that provides insight into the reasoning behind why VECTOR was designed in this chosen manner. Similar to the System Performance Requirements, none of these requirements are new, in this case there is a mix of Spacecraft Requirements and the Propulsion Subsystem Requirements, which will be discussed later in the report. These specific four were chosen since they were found to be the main requirements which constraint how the team could design the satellite.

Table 3-8-1 Critical Design Requirements

ID	Requirement	Verification	Parent	Child	Status
SC-2.1	System shall have a minimum 2U payload capacity	Inspection	MR1.2	TBD	meeting
SC-2.8	System shall have attitude control accuracy to meet pointing constraints	Testing	MR1.2	ADCS-3.7.1 ADCS-3.7.2 ADCS-3.7.3 ADCS-3.7.4	meeting
SC-2.11	System shall be protected against radiation during entire duration of mission	Testing	MR1.3	TBD	TBD
PRO-3.5.1	System shall produce a velocity increment of at least 1 km/s	Testing	MR1.1	PRO-3.5.1 PRO-3.5.2 PRO-3.5.3 PRO-3.5.4 PRO-3.5.5	meeting

4. System Design Elements

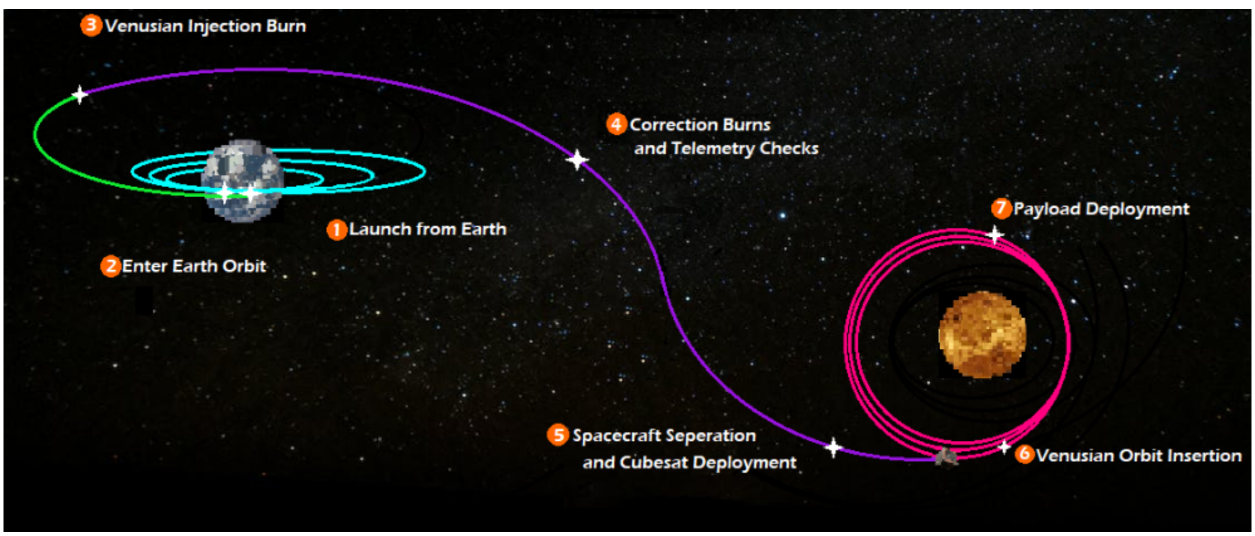
4.1 Description

VECTOR will be composed of a bus and a payload like many conventional and CubeSat-sized satellites. VECTOR's payload will consist of a neuromorphic camera. The rest of the system will follow a fairly standard construction layout. Below, system design elements will be examined. These will include the system's concept of operations, the functional block diagram of the system, the system's work breakdown structure, an N2 diagram of the system that depicts sub-system inputs and outputs, and finally, the physical architecture diagram where a model for the system will be shown.

4.2 Concept of Operations

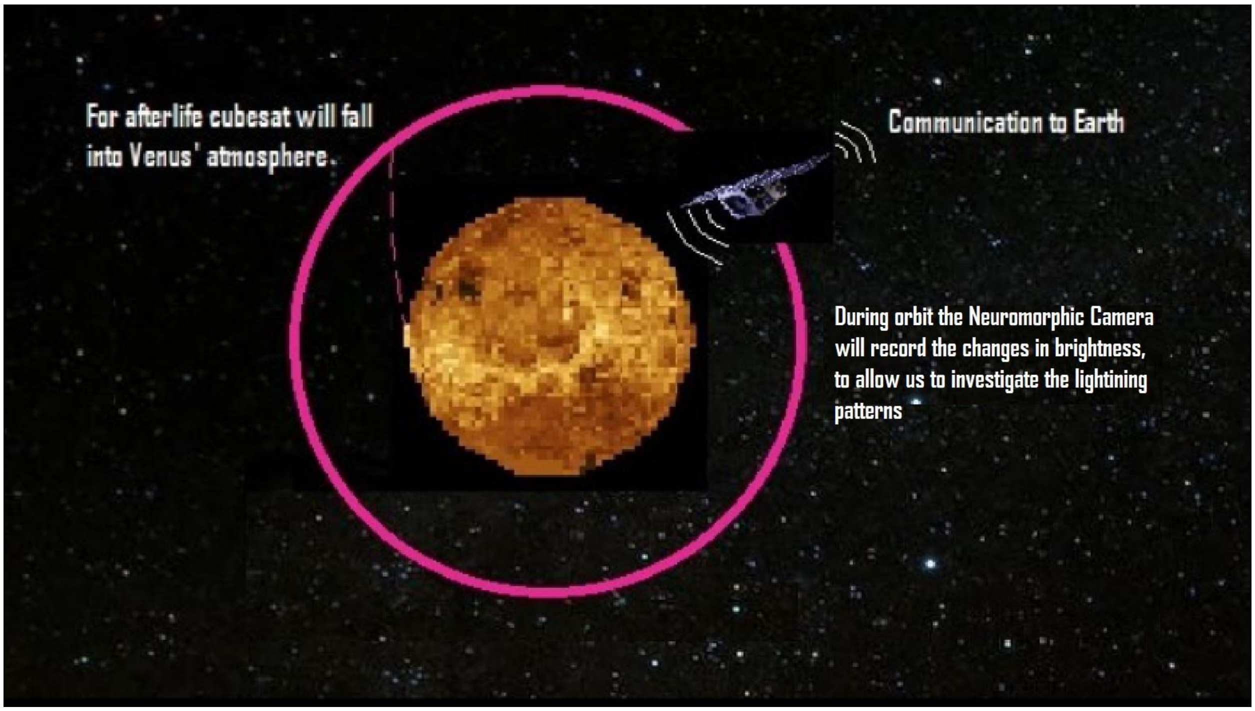
Depicted below is a two-part concept of operations for the system. The first section of the concept of operations will involve the system's arrival to Venus. Given that CubeSats have historically been limited to Earth orbit, and seeing that no CubeSat has ever successfully made it to Venus, a significant component of VECTOR's mission will simply be arriving at Venus.

Figure 4-1-1 Concept of Operations: Transfer



First, the system will launch from Earth. Following the launch, it will enter an Earth parking orbit. In this parking orbit, system integrity checks will be performed to ensure that the system survived the launch. Additionally, communication will be initiated with the ground system. Next, the system, along with the Electron rocket, will execute a Venusian injection burn to begin the transfer. Along the way, during the roughly three-month trip, VECTOR will continue to monitor trajectory and system integrity information. Electron will perform telemetry checks and correction burns as necessary to maintain the desired trajectory. Close to arrival at Venus, VECTOR will separate from Electron, and begin deployment. After deployment and separation, VECTOR will begin its insertion burn into Venusian orbit. Finally, once in a stable Venusian orbit, VECTOR will deploy its payload and begin collecting data. The second section of the concept of operations is depicted below.

Figure 4-2-1 Concept of Operations: Operational Orbit

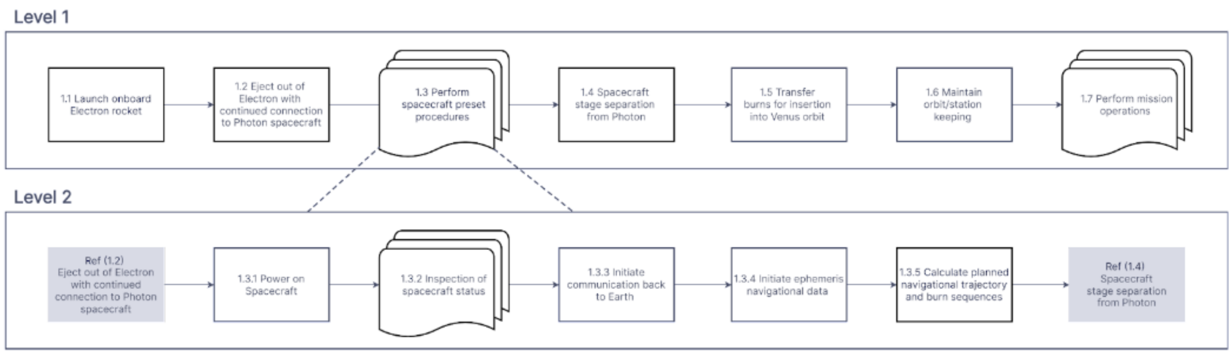


Once in orbit around Venus, VECTOR will collect data using the neuromorphic camera payload. Then, periodically, it will transmit the data it receives back to the ground station. In order to minimize the required data rate for communications, data pre-processing will happen on VECTOR before transmission. At the end of its lifetime, VECTOR will deorbit into Venus's atmosphere.

4.3 Functional Block Diagram

The system's functional block diagram shows the evolution of operations beginning at launch and ending at mission operations, as shown below in Figure 4-3-1.

Figure 4-3-1 Functional Flow Block Diagram.

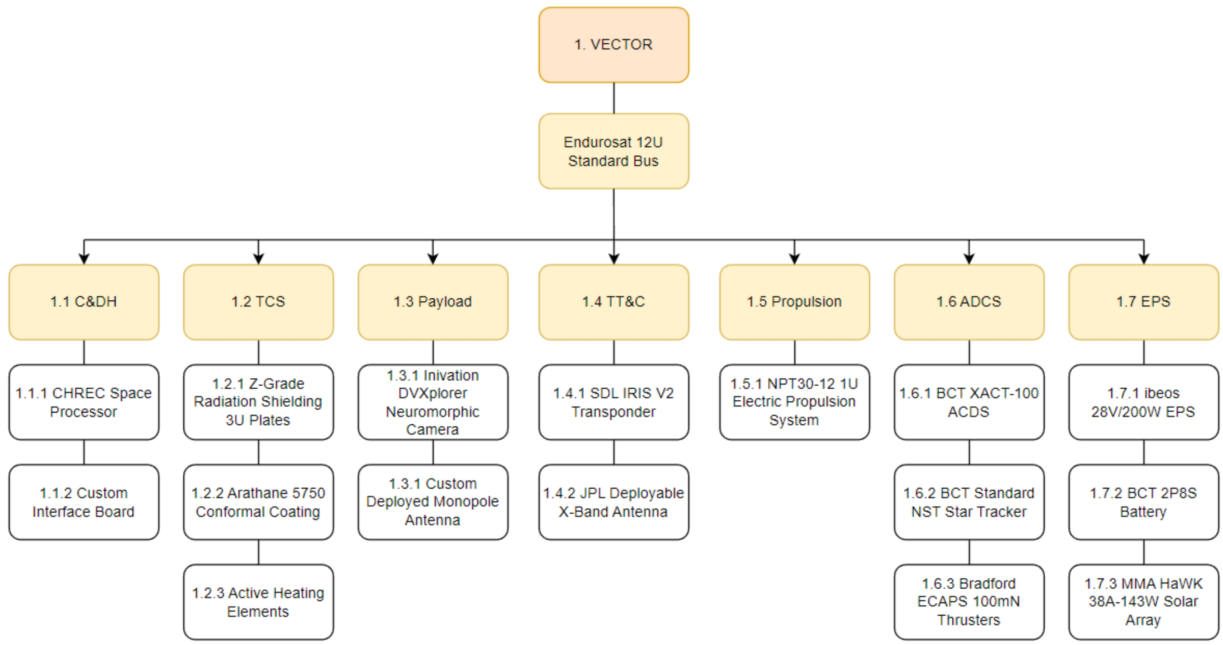


Level 1 shows the highest level of system operations, beginning with the launch. After launch, the system will disconnect from the first stage launch vehicle while remaining connected to Photon. Next, the system will perform preset procedures in order to become operational which are detailed in the level two blowout. First, the system will power on before it performs an internal status check. Then, it will initiate communication back to Earth. The system will then initiate navigational tracking and calculate potential transfer trajectories, in conjunction with the Photon delivery system. Moving back to level 1, after these procedures, the system will await stage separation from Photon. The system will then initiate an insertion burn into Venus orbit. Once in orbit, the system will perform orbit maneuvers and stationkeeping as necessary to maintain a stable orbit. Finally, the system will proceed with data collection operations.

4.4 Work Breakdown Structure

The work breakdown structure pictured below shows each of the main subsystems and their components that the overall system will be composed of.

Figure 4-4-1 Work Breakdown Structure

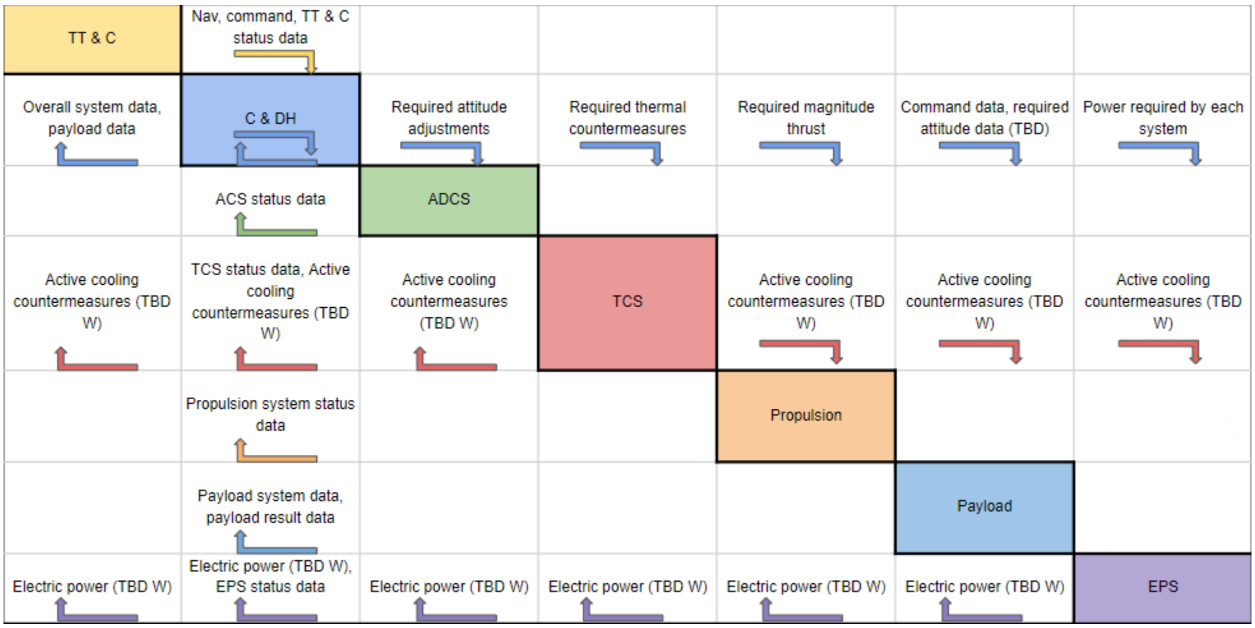


As seen above in Figure 4-4-1, the system will be composed of a number of subsystems. First, the command and data handling subsystem–which will be composed of a central processing unit–a data storage system, and a flight dynamics subsystem. Next, the thermal control subsystem will be composed of radiators, heating pipes, thermistors, and other active heating elements. Then, the telemetry and communication subsystem will be made up of command receivers and antennas. The propulsion subsystem will feature thrusters, propellant tanks, and propellant feed systems. The attitude determination and control subsystem will be composed of sun and star sensors in addition to gyros and reaction wheels. The electrical power subsystem will have solar arrays, batteries, and a power regulator. Finally, the system's payload will be a neuromorphic camera.

4.5 N^2 Diagram

With the subsystems defined, it is important to define what inputs and outputs to expect from each subsystem. To do so, the system's N2 diagram, an interfaced diagram, is shown below.

Figure 4-5-1 N2 Diagram.

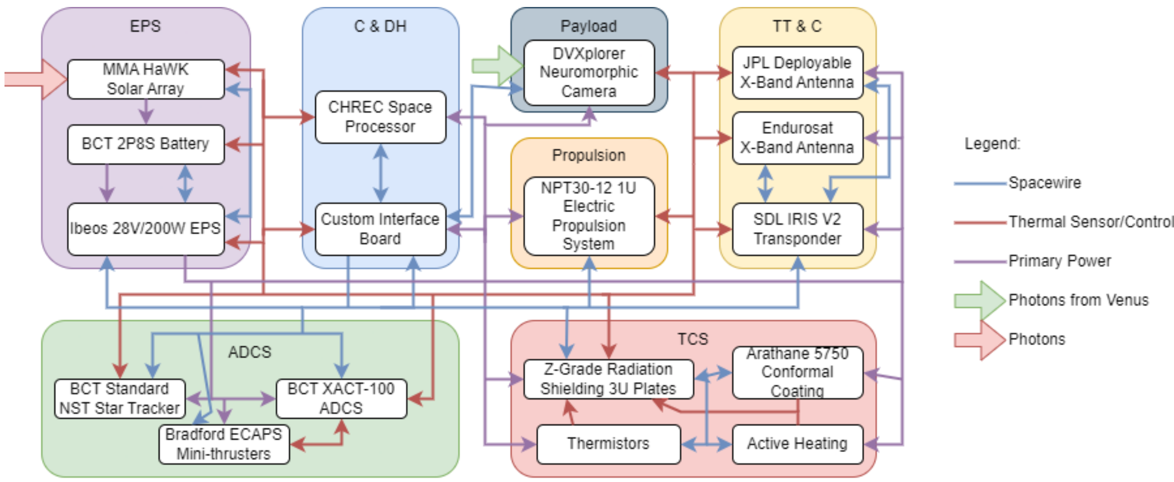


Overall, there are a few constants. First, each subsystem will take in active thermal control from the TCS and power from the EPS. Each subsystem will also export status data to the command and data handling subsystem. Beyond that, the telemetry system will export navigational and command data it receives and take in data to be transmitted back to the ground station. The command system will output command data to all other subsystems, and will also output processed payload data for transmission. Finally, the payload subsystem will output raw neuromorphic camera data to the command and data handling subsystem.

4.6 Physical Architecture Diagram

With these inputs and outputs in mind, the physical architecture diagram shown below will display the connections between each subsystem.

Figure 4-6-1 Physical Architecture Diagram

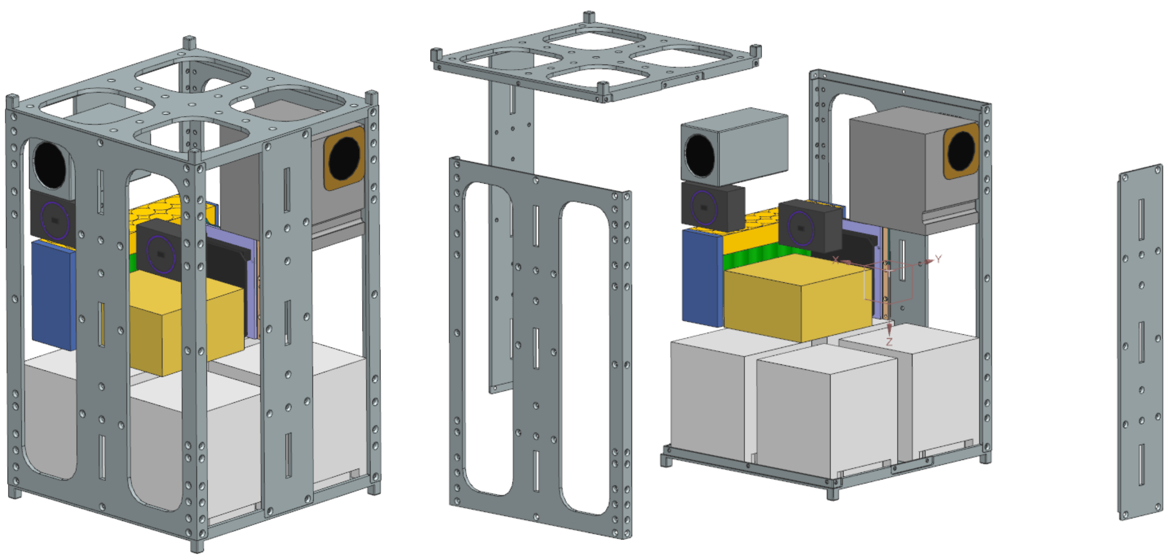


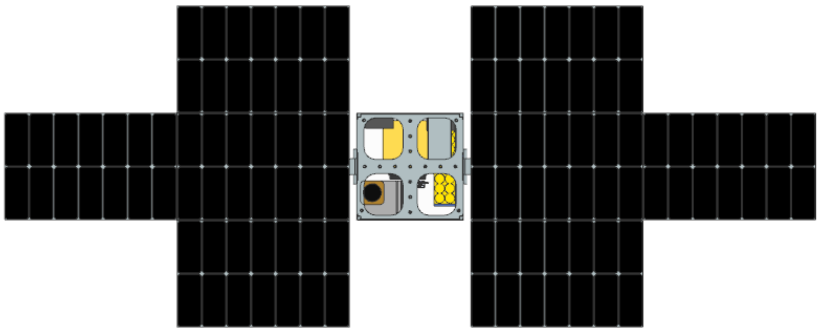
Each subsystem shown here has the same components as the work breakdown structure. There are three main types of connections made in the physical architecture diagram. First, there are the spacewire connections that will represent any connections in which data or command transfer will take place. These connections are prevalent and important as they mirror the relationships shown in the N2 diagram, Fig. (4-6). The next type of connection involves thermal control wiring. These will transfer heating and cooling to the different portions of the system, as required by the regulator, while the primary power wire brings power to the system. There are two external connections, specifically the photons from the sun hitting the solar arrays and photons from Venus hitting the neuromorphic camera. Additionally, there will be the reception of communication data in the form of radio waves from the ground station and transmission of communication and payload data in the same radio waves.

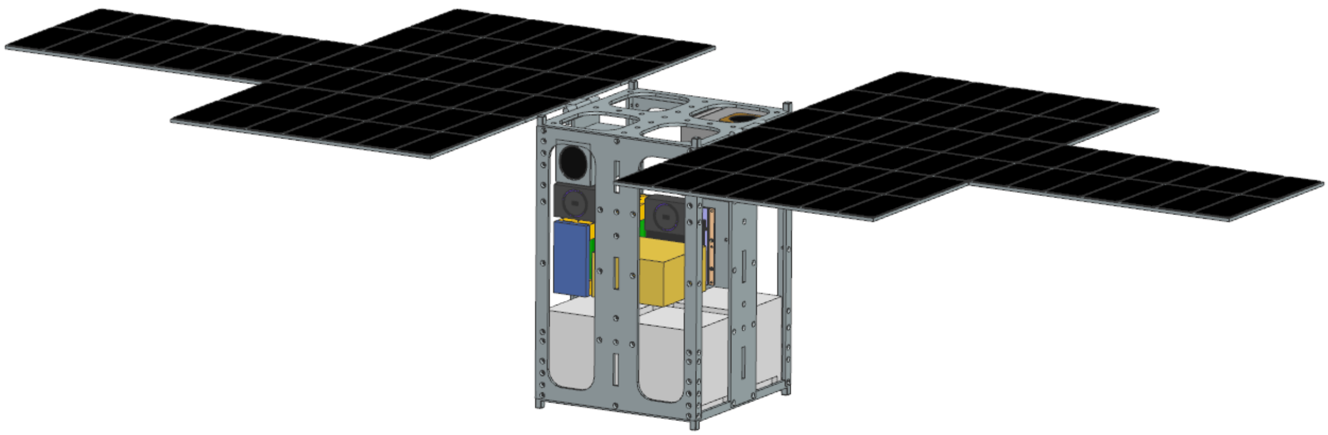
4.7 CAD Design

The CAD model was based on public facing information. A majority of the subsystems are simply mass and volume models. If we were taking this mission beyond the CDR, STEP files of these subsystems would be obtained.

Figure 4-7-1: CAD Design, Multiple Angles







5. Sub-System Design Elements

5.1 Description

Going deeper into VECTOR, the subsystem design is every system which makes up the satellite. This section will describe the subsystem requirements, analysis of alternatives, and technical details.

5.2 Computing and Data Handling (C&DH)

The C&DH subsystem is, essentially, the brain of the system. This subsystem will be in charge of electronic controls for the entire system, processing the data received from the payload and from all other subsystems, for storing this data, and for deciding what to transmit back via the TT&C subsystem. For these purposes, the following subsystem requirements were defined.

Table 5-2-1 C&DH Requirements

ID	Requirement	Verification	Parent	Status
CDH-3.1.1	System shall resist at least 40 krad.	Testing	SC-2.12	TBD
CDH-3.1.2	System shall collect telemetry data for communication at minimum every 12 hours.	Inspection	SC-2.12	TBD
CDH-3.1.3	System shall operate a neuromorphic camera during orbit.	Inspection	MR-1.2	TBD
CDH-3.1.4	System shall have at minimum 1GB of local storage.	Inspection	SC-2.12	TBD
CDH-3.1.5	System shall have at least 200 MHz of processing power.	Inspection	SC-2.12	TBD

First, the system must resist a considerable amount of radiation. The 40 krad radiation threshold was determined after investigating previous missions to Venus. Second, the system must be able to collect the data regarding system and subsystem performance and telemetry for the TT&C system. Third, the system shall be capable of operating the scientific payload, the neuromorphic camera, during orbit. Finally, the system shall have enough local storage and processing capability to manage the satellite system and record data from the scientific payload.

With these requirements in mind, an analysis of alternatives was conducted to determine which of three commercially available products would be the optimal solution.

Table 5-2-2 C&DH Analysis of Alternatives

Selection Element	Weight	Alen Space TRISKEL		EnduroSat OBC		CHREC Space Processor [9]	
		Motivation	Score	Motivation	Score	Motivation	Score
Cost	0.2	Unknown	3	$4,300-$10,400	2	Unknown	3
Weight	0.1	200 g	5	130 g	3	74 g	1
Radiation Resistance	0.5	No data, TRL 4	5	40 krad	3	100 krad	1
Storage	0.1	1 GB, room for more	4	8 GB	1	4 GB	2
Processing	0.1	280 MHz/32b	4	216 MHz	5	766 MHz	2
Scores			4.4		2.8		1.6

Out of the considered selection criteria, radiation resistance was by far the most important. Electronics failure due to radiation is one of the principal risks on our mission, and thus selecting a processor with high radiation resistance was paramount. In the end, the CHREC space processor was considerably ahead of the two other choices. In particular, its high radiation resistance was a key selling point, along with its considerable processing power and average storage capacity. Finally, the CHREC space processor already has space flight experience, and is homologated with NASA's GEVS vibration acceptance levels.

5.3 THERMAL CONTROL SYSTEM (TCS)

The Thermal Control Subsystem (TCS) is responsible for regulating the temperature of the CubeSat's internal components to ensure optimal performance throughout the mission. To achieve this, the TCS must meet several key requirements. First, the TCS must maintain the internal operating temperature of the CubeSat between 0℃ and 50℃. This requirement will be verified through testing. Additionally, the system must have a nominal operating temperature of 20℃, which will be verified through analysis. The TCS must also have the ability to modulate the internal temperature of the spacecraft from TBD ℃ to TBD ℃. This requirement will be verified through testing. Finally, the TCS must have a peak power usage of TBD W, which will also be verified through testing.

Table 5–3-1 TCS Requirements

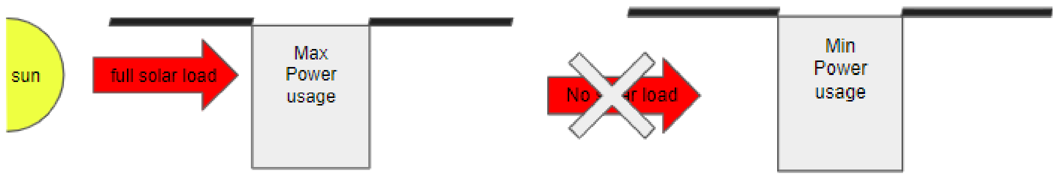
ID	Requirement	Verification	Parent	Status
TCS-3.2.1	System shall maintain internal operating temperature between 0℃ and 50℃.	Testing	SC-2.6	Meeting
TCS-3.2.2	System shall have a nominal operating temperature of 20℃.	Analysis	SC-2.6	Meeting
TCS-3.2.3	System shall be have the ability to modulate the internal temperature of the spacecraft by ± 25 ℃	Testing	SC-2.6	Meeting
TCS-3.2.4	System shall have a peak power usage of 154.7 W.	Testing	SC-2.6	Meeting

A first order thermal analysis was performed to show that a net zero thermal load can be maintained in steady state conditions. The analysis was bounded by two worst-case scenarios. The hot-case scenario occurs when maximum allowable power is being used while the spacecraft has maximum solar load. This happens while the ion engines are firing during the insertion burn phase. The cold-case occurs when there is minimum power usage and there is no solar occurs after the insertion burn and when venus is in between VECTOR and the sun.

The assumptions made in the analysis are:

1. If the system can achieve a net zero load in both the hot case and the cold case, then the system can survive any temperature within that range.

Figure 5-3-1: Hot Case and cold case



2. If the steady state NASTRAN simulation attains similar results as the hand calculations, then the model can be considered validated and the transient simulation can be trusted.
3. For internal loads; 1-Watt of power consumption is equal to 1-Watt of heat generation because of the inefficiencies of microprocessors.
4. Average distance between the spacecraft and the sun is 0.7 AU. (~ avg sun-venus radius)
5. Only radiating to space (venus/sun are sufficiently far enough away)

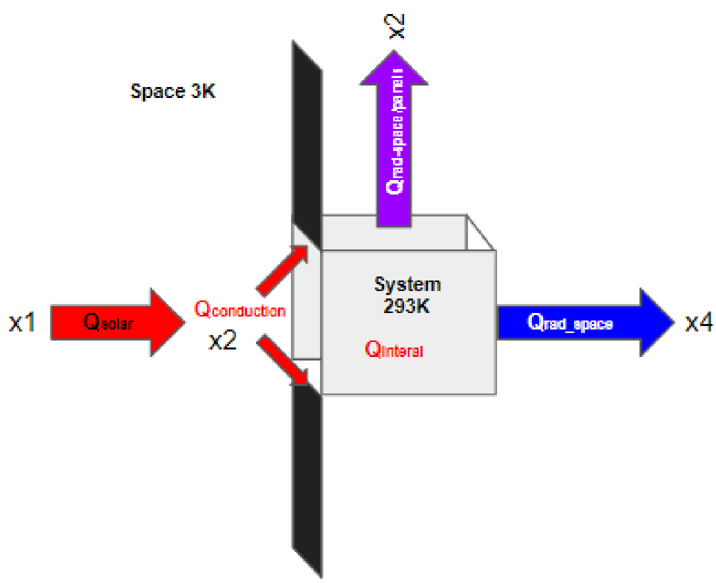
The two governing equations used in the analysis are the radiation and conduction equations.

Eq 5-3-1. $$Q = \epsilon\sigma FA(T_1^{\ 4} - T_2^{\ 4})$$

Eq 5-3-2 $$Q = KA(T_1 - T_2)/L$$

The thermal loads taken into consideration are solar loads, radiation from the spacecraft to space, radiation from the solar panels to the spacecraft, and conduction between the solar panels and the spacecraft. The free body diagram is shown below. The calculations were performed in a spreadsheet.

Figure 5-3-2: Thermal Loads FBD



The analysis procedure is as follows.

Step 1 - Perform the analysis in the hot case with no thermal controls implemented. This means the material is Anodized Aluminum-6082, there is no MLI, and surfaces are not painted.

Step 2 - determine the net wattage by summing all thermal loads. If this value is positive, then the system is heating up. If the the number is negative, then the system is cooling down. If the number is zero, then the temperature is not fluctuating.

Step 3 - Implement thermal controls such as coatings, material selection, mechanical controls, and material properties to either increase or decrease the net thermal load. If a net zero can not be obtained, then active heating/cooling elements are necessitated.

Step 4 - Perform thermal analysis in the cold case and repeat step 3.

Below is the list of known material properties and assumptions made and on the right is the calculations based on those assumptions in the hot case with no thermal controls.

Table 5-3-2: Hot Case No TCS Calculations

Knowns and Assumptions	
n radiating to space (per side type)	2
n radiating to space & solar panel	2
screw cross-section area, m^2	0.00023
screw length, m	0.006
thermal conductivity (Ti), W/mK	17
thermal conductivity (Al), W/mK	237
solar panel structure thickness	0.003
screw conductance (1/4-20), W/K	3.51
space temp, k	3
solar panel temp hot, k	393.15
solar panel temp cold, k	250
q_solar flux, W/m^2	2789.8
SA 4 sides, m^2	0.0828
SA end caps 2-sides, m^2	0.0512
emissivity base, e	0.4
abosportivity base, a	0.3
system temp, k	293.15
Stephan Boltzman Constant	0.0000000567

Hot Case, initial NO TCS implimented	
Q_solar, W	42.851328
Q_internal, W	154.7
Q_rad-space, W	-36.31
Q through Solar Panel	
Temp of SP faceing SC	
Q_cond_engine	57041.23
Q_rad-space/panels, W	72.59
Total Conductance	2.198494193
Q_cond-panels, W	314.7144437
Q_in, W	57589.78
Q_out, W	-36.31
Q_net, W	**57553.46**
if positive system is heating, if negative system is cooling. If 0 assume ss stable	

Notice the Q_net is 57.6 kW. This indicates that the system is heating up, really fast. The primary contributor of this is the conduction that arises from the heat generated from the ion engines while firing. This value alone is 57 kW. Furthermore, the next large contribution of heat is the heat transfer from the solar panels. This value is 314 W.

The first thermal controls to be implemented is to isolate the conduction loads from the engine bay and from the solar panels. This can be done through the deployment of mechanical controls such as insulations paneling. A 6mm thick sheet of aerogel will isolate the thermal loads from the engine bay and from the solar panel hinges. This brought the conductance down to a negligible number. Isolating the engine bay from the rest of the spacecraft reduces the internal power wattage to 38.9 W. Applying this thermal control and others like MLI and painting surfaces, the hot case with TCS is shown to be +2.56W. This is manageable. See the table below.

Table 5-3-3: Hot Case With TCS Calculations

Hot Case, with TCS		TCS Method
Q_solar, W	27.1391744	Paint sun-side white
Q_internal, W	38.9	
Q_rad-space, W	-0.45	Painted Black
Q_rad-space/panels, W	-63.02	wrap in MLI & paint black
Q_cond-panels, W	0.000	aerogell insulation
Q_cond_engine	0	aerogell insulation
Q_in, W	66.04	
Q_out, W	-63.47	
Q_net, W	**2.56**	
if positive system is heating, if negative system is cooling. If 0 assume ss stable		

This necessitates an active cooling component such as a radiator. Using the radiation equation, the total surface area for a radiator painted white would have to be 0.007m^2. This requires an 84mm by 84 mm radiator that can be mounted to one of the sides of the bus. The mounting surfaces for the patched antenna will be used for this purpose with heat straps.

For the cold case, the net thermal load is -25.5 W. This indicates that the system is cooling down. To manage this, electric heaters will be used. Polyimide Thermofoil™ Heaters made by Minco will be used. These have a TRL of 9 and are inexpensive.

Table 5-3-4: Cold Case With TCS Calculations

Cold Case	
Q_solar, W	0
Q_internal, W	38
Q_rad-space, W	-0.45
Q_rad-space/panels, W	-63.02
Q_cond-panels, W	0.000
Q_in, W	38.00
Q_out, W	-252.54
Q_net, W	**-25.47**
if positive system is heating, if negative system is cooling. If 0 assume ss stable	

Using Ohms Power Law

Eq 5-3-3. $$P = V^2/R$$

Where P is the 25.47 W, V is 28 V per the selected BCT batteries, the maximum required resistance is determined to be 31.36 Ohms. The HK6909 Polyimide Thermofoil™ Heaters were selected

Figure 5-3-3: Electric Heating Element

Table 5-3-5 Electric Heater Technical Details

Mounting	Acrylic Adhesive (PSA)
Thickness (in)	0.060
Min Temp (°F)	-26
Max Temp (°F)	212
Weight (oz)	0.080
Style	Etched Kapton/WA
X dim (in)	1.00
Y dim (in)	4.00
R (Ω)	23.59
AWG	26
Area (in^2)	3.3232
Volt	28.00
Watt	33.2
Watt Density (w/in^2)	10.00

5.4 Payload (PAY)

The secondary mission statement of VECTOR is to perform high-value, low-cost interplanetary science. The interest on Venus has been rather limited over the past decades, leading to a wide range of unanswered questions concerning aspects of its hazardous environment and toxic atmosphere. [13] Table 2-6-1 presents a select number of these areas of scientific intrigue in relation to the CubeSat's technological capabilities of gathering the necessary data. Ultimately, the CubeSat payload will intend to study the weather patterns of Venus and once and for all confirm the existence of lightning within the planet's acidic, cloudy skies.

Table 5-4-1 Payload Requirements

ID	Requirement	Verification	Parent	Status
PAY-3.3.1	System shall be an instrument sensitive to lightning	Test	MR1.2	Meeting
PAY-3.3.2	System shall consist of shielding to moderate radiation damage and background signals	Analysis	SC-2.11	TBD
PAY-3.3.2	System shall reside inside the spacecraft volume	Inspection	SC-2.1	Meeting
PAY-3.3.2	System shall have a method for confirmation of event detection	Demonstration	MR1.2	Meeting

In order for the CubeSat payload to confidently collect data regarding Venusian lightning, the subsystem requirements must be clarified. The requirements for VECTOR's payload instruments have been listed in Table 5-4-1. The payload must be capable of capturing the visual data of lightning strikes within the Venusian atmosphere. However, it was also determined that a secondary instrument must be included within the payload to confirm that the images gathered are indeed lightning. The goal of including both these instruments is to leave little room for doubt when analyzing VECTOR's collected data and subsequent findings. Furthermore, the instruments must also be capable of fitting within the dimensions of the CubeSat or else it cannot be included in the mission. Finally, given that the CubeSat will be exposed to intense levels of radiation on its orbit around Venus, it is vital that shielding is implemented to protect the instrument from prolonged exposure, otherwise operations would cease.

Table 5-4-2 Payload Science Traceability Matrix

Science Goals	Science Objectives	Science Measurement Requirements: Physical Parameters	Science Measurement Requirements: Observables	Instrument Performance Requirements	
There is strong evidence to suggest that lightning forms on the planet Venus	Determine if lightning forms from the sulfuric acid clouds within the Venusian atmosphere	Identify the images of lightning strikes in the Venusian atmosphere	Collect low-latency, high-speed, light-changing events with no motion blur over a large area	Latency Range:	<150ms
				Framerate:	>7,000 fps
				Dynamic Range	>70 dB
				Power Consumption:	30 mW
	Confirmation of the existence of lightning	Identify the radio signals emitted from lightning strikes on Venus	Collect radio waves in the very low frequency (VLF) range of 10-80 kHZ	Frequency Range:	1kHz-300MHz
				Dynamic Range	>/=50 dB
				Power Consumption:	3-26W

In prior missions to Venus, onboard systems detected radio spikes emanating from Venus and the most likely cause could be lightning strikes within the atmosphere. In order to assure that VECTOR's instruments are capable of detecting Venusian lightning strikes, it was decided that there must be a metric in which to break down the technological capabilities included as a part of VECTOR's payload. Thus, a science traceability matrix was employed, as shown in Table 5-4-2, to showcase the comparisons between proven instrument performance and known environmental observables. The STM describes the science objectives, observables, and parameters that define the necessary requirements of an instrument to specifically detect Venusian lightning. On Earth, lightning is a swiftly occurring weather phenomenon that illuminates dark storm clouds in a flash of brilliance. [2] It can be assumed that lightning on Venus is quite similar, and since VECTOR intends to visually detect these events, it is important that the instrument not only has a high framerate but also a wide dynamic range. Furthermore, a radio-detection instrument must hold the capabilities of capturing large but brief spikes in the radio waves emitted from the planet. Thus, the instrument must not only capture the correct frequency of lightning but also possess a large dynamic range to achieve the set scientific requirements. [2]

Table 5-4-3 Payload Technical Details

Instrument	Instrument Performance Requirements	Predicted Instrument Performance
Neuromorphic Camera: Inivation DVXplorer	Latency Range:	20ms
	Framerate:	10^6 fps
	Dynamic Range	120dB
	Power Consumption:	~ .7W
Radio: Red Pitaya STEMlab 125-14	Frequency Range:	DC - 50MHz
	Dynamic Range	~50dB
	Power Consumption:	~10W

As already determined within Table 2-6-1, the team determined that the neuromorphic camera would best fit the data-collection requirements needed to accomplish the scientific mission of VECTOR. Based upon the requirements detailed in Table 5-4-2, Table 5-4-3 intends to list the predicted instrument performance requirements. The payload instruments chosen to prove and verify the existence of lightning includes the Inivation DVXplorer neuromorphic camera as well as the Red Pitaya STEMlab 125-14. [16] [18] As demonstrated, both of these devices meet or exceed the performance requirements previously set within the STM. Figure 5-4-1 and Figure 5-4-2 display the selected neuromorphic camera and radio.

Figure 5-4-1 Inivation DVXplorer Neuromorphic Camera

Figure 5-4-2 Red Pitaya STEMlab 125-14

The neuromorphic camera is a technology whose design was inspired by the biological functions of the human retina. Though the neuromorphic camera does not incorporate color or texture into its images, it is capable of capturing events of a moving object while simultaneously reducing its motion blur. [16] The team acknowledged that such abilities would prove advantageous given that the CubeSat will be moving fairly fast along its orbit around Venus, which is further described in section 5.6. Additionally, the neuromorphic camera possesses a high dynamic range, and is able to distinguish between light and dark patches of a given scene. Such versatility will be incredibly important given that the team wants the payload to differentiate between the Venusian clouds and the bright bolts of lightning. Finally, the team decided that since previous missions to Venus held unintended success in their ability to collect radio wave transmissions, it was decided that an additional radio payload will provide an invaluable way to verify any images taken of Venusian lightning.

5.5 Telemetry Tracking and Control (TT&C)

The telemetry, tracking, and control subsystem seeks to ensure that a stable connection is forged between the VECTOR and groundstations on Earth. Essentially, this subsystem intends to update the status, location, and data communications of the CubeSat throughout the duration of the mission. As such, a list of requirements are needed to properly define all that is expected from the team.

Table 5-5-1 TT&C Requirements

ID	Requirement	Verification	Parent	Status
TTC-3.4.1	System shall communicate through DSN	Inspection	SC-2.16	Meeting
TTC-3.4.2	System shall withstand radiation up to particles with 20GeV	Testing	SC-2.10	TBD
TTC-3.4.3	System shall communicate back to Earth on average once per day	Analysis	SC-2.12	Meeting
TTC-3.4.4	System shall compile telemetry check data every 12 hours	Testing	SC-2.16	TBD
TTC-3.4.5	System shall include redundant communication systems	Inspection	SC-2.14	Meeting

As displayed in Table 5-5-2, the first requirement establishes that VECTOR must operate through NASA's Deep Space Network, which is a necessity for interplanetary missions and provides the only means in which the team can communicate with the CubeSat. Secondly, given that VECTOR will be orbiting around Venus, it is essential that there are measures to ensure that the system is protected against the harsh radiation of space. Next, the team wanted the CubeSat to not only maintain frequent communication and telemetry checks, but also include redundancies so as to mitigate the risks associated with communication. Further explanation of those risks is outlined in section 8.2.

Table 5-5-2 Antenna Analysis of Alternatives

Selection Element	Weight	MarCO-type Custom HGA/MGA/LGA (X-Band)		Syrlinks SPAN-X-T3		Endurosat Module (X-Band)	
		Motivation	Score	Motivation	Score	Motivation	Score
Weight	0.2	1.25 kg	4	65 g	2	53 g	1
Cost	0.1	~ $760k	4	Unknown	3	$8,400	1
Data rate	0.3	Max 1kbps up, 8kbps down	2	Unknown	3	Unknown	3
Radiation resistance	0.3	Unknown [DS Exp.]	2	Unknown [FH Unk.]	5	Unknown [FH Unk.]	5
Data loss	0.1	X-Band	1	X-Band	1	X-Band	1
Scores			**2.5**		**3.2**		**2.8**

As displayed in Table 5-5-2, three antenna types were analyzed against the selection elements of weight, cost, data rate, radiation resistance, and data loss. As shown, the MarCO-type custom antenna scored better than the Syrlinks SPAN-X-T3 and Endurosat Module. The Mars Cube One mission sent the first CubeSats on an interplanetary fly-by of Mars [5]. Although the MarCO-type is a custom made component instead of a COTS component, the team reasoned that it would be beneficial to the mission if an antenna that had prior usage in a deep-space mission was equipped to the CubeSat. Additionally, the team ultimately decided to equip VECTOR with both the MarCO-type as well as the EnduroSat 4x4 patch array. [20] The patch antennas will be added in an attempt to not only increase the TRL of the commercial off-the-shelf component, but also as a redundancy for the MarCO-type antenna [14].

Table 5-5-3 Radio Analysis of Alternatives

Selection Element	Weight	JPL IRIS v.2 [12]		GD Small Deep Space Transponder	
		Motivation	**Score**	**Motivation**	**Score**
Weight	0.2	1.2 kg	2	3.2 kg	4
Cost	0.2	$500k	2	~ $5 mil	4
Data rate [BPSK]	0.3	Max 8kbps up, 6.25 Mbps down	3	Max 4kbps up, 15 Mbps down	2
Radiation resistance	0.2	> 23.0 krad	4	50 krads	2
Data loss	0.1	X-Band	3	X-Band and Ka-Band	2
Scores			**2.8**		**3.0**

As demonstrated in Table 5-5-3, two different radios were compared and analyzed against the selection elements of weight, cost, data rate, radiation resistance, and data loss. The JPL IRIS radio was determined to be the better selection over the GD Small Deep Space Transponder. Another reason the team chose to incorporate the IRIS radio was due to the fact that it has already been established to operate proficiently well throughout the lifetime of interplanetary, deep space missions. [12] Though a crucial part of the VECTOR mission is to increase the TRL of COTS components, the team ultimately decided that the risks associated with CubeSat communications was too high to utilize a radio system that is not confirmed in its reliability. In the end, the team agreed that mitigating risk was more important than increasing the TRL of the radio aspect of the communications subsystem. Furthermore, given that the GD Small Deep Space Transponder costs approximately $5 million dollars and would exceed the mission budget, it was not added into the CubeSat as a redundancy like the EnduroSat patch arrays were. [19]

Figure 5-5-1 MarCO-Type Custom HGA/MGA/LGA (X-Band) Antenna

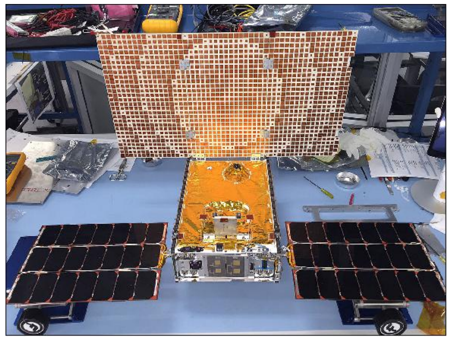


Figure 5-5-2 EnduroSat Module (X-Band) Antenna

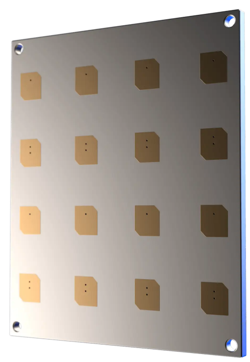


Figure 5-5-1 and Figure 5-5-2 displays the two selected antenna arrays for the VECTOR mission. The MarCO antenna has a mass of approximately 1kg while the EnduroSat patch antennas have masses of approximately 3g. Both antenna types consume about 4 watts of power. The MarCO-type antenna possesses a frequency range of 8-12 GHz, with gains that vary from a low gain of 6 dBi, medium gain of 9dBi, and a high gain of 29.2 dBi. These ranges will ensure that a connection is made between the VECTOR CubeSat as it orbits Venus and a ground station on Earth. Comparatively, the EnduroSat patch antenna has a frequency range of 8.25 - 8.4 GHz and a gain of approximately 6 dBi. [14] [20]

Figure 5-5-3 JPL IRIS v.2

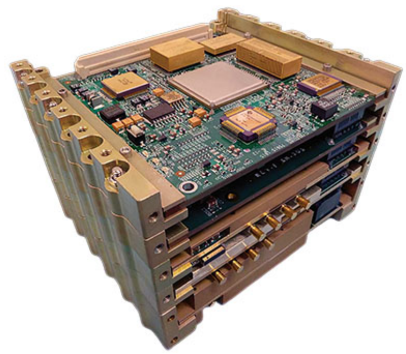
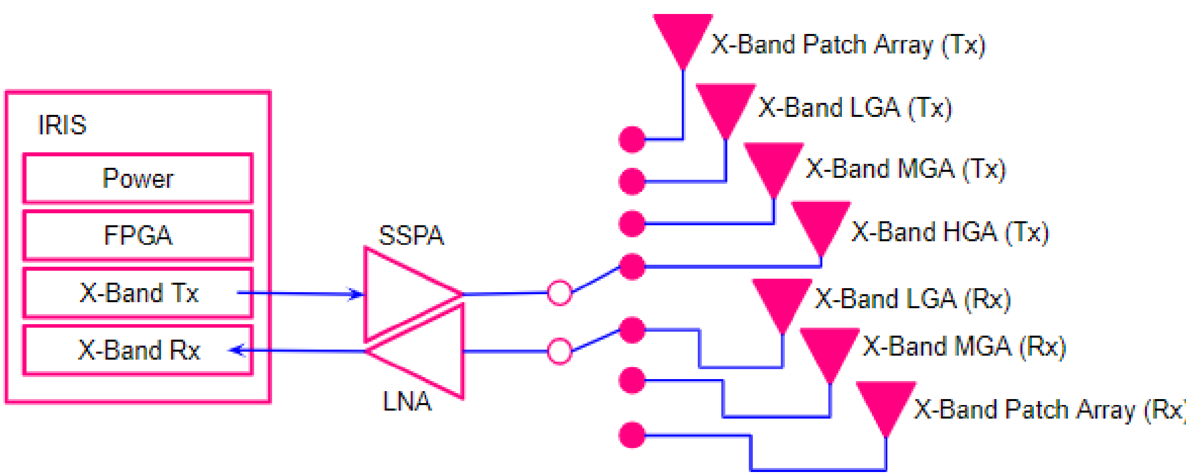


Figure 5-5-3 displays the selected radio for the VECTOR mission. The technical details for the JPL IRIS radio shall be explored. The X-band component is capable of 7.2 GHz uplink, 8.4 GHz downlink as well as a noise figure of 2.2 dB. Additionally, the IRIS radio has a carrier tracking signal range of -70 to -130 dBm. Furthermore, the IRIS radio possesses a dry mass of 1.1 kg, transponder volume of approximately 0.5U, an average power of 35.0 W DC power consumption at 3.8 W RF, operating temperatures between 253 K to 323 K, and a design lifetime of about 3 years. Effectively, this radio contains a unique architecture that allows for the VECTOR CubeSat to connect to NASA's Deep Space Network, which is vital in order for telecommunications to be relayed between Earth and Venus. [12]

Figure 5-5-4 TT&C Block Diagram



The connections between the IRIS radio and the MarCO antenna array as well as the EnduroSat patch arrays are displayed in the TT&C block diagram within Figure 5-5-4.

5.6 Propulsion (PRO)

VECTOR's propulsion system will be crucial to the overall success of the mission as it will be responsible for ensuring a successful capture into the target orbit around Venus; this is the primary need of the mission. A detailed set of requirements for this system have been laid out below, and the selected propulsion system must demonstrate the ability to satisfy each of them. Each of these system requirements are necessary and were carefully selected to ensure that VECTOR can satisfy its goals. The most important requirement is the required velocity increment shown in PRO-3.5.1, which is absolutely necessary if the system is to arrive in the proper Venus orbit. This value is slightly rounded up to account for any contingencies, and detailed orbital calculations are shown in the next sections.

Table 5-6-1 Propulsion Requirements

ID	Requirement	Verification	Parent	Status
PRO-3.5.1	VECTOR shall produce a velocity increment of at least 1 km/s	Test	MR-1.1	TBD
PRO-3.5.2	System shall retain at least TBD kg of propellant for deorbit burn	Demonstration	SC-2.4	TBD
PRO-3.5.3	System shall contain a propellant depletion gauge	Inspection	SC-2.17	Meeting
PRO-3.5.4	System shall contain at least one redundant thruster	Inspection	MR-1.1	Meeting
PRO-3.5.5	Propulsion system shall monitor and control the S/C thrust	Test	SC-2.16	Meeting

After departure from low Earth orbit and minor correction burns from Rocket Lab's Photon, VECTOR will be released at which point the propulsion system will turn on and begin to perform its primary functions. The spacecraft will have roughly 70 days after separation before arriving at Venus. Due to this long timeline, an electric propulsion system was selected that can perform continuous low thrust, high efficiency burns to slow the spacecraft down enough to capture into the target orbit. Alternative systems were analyzed as displayed in Table 5-6-2 below, however electric propulsion possesses many benefits when compared to typical liquid bi-propellant or monopropellant thrusters. First and foremost, the electric system is drastically more efficient in terms of specific impulse produced by the thruster, when compared to more typical liquid thrusters. The feed systems required for electric propulsion are also much simpler, requiring less moving parts and thus less overall mass. The weighted criterion for each system is shown below in Table 5-6-2, demonstrating the selection process.

Table 5-6-2 Propulsion Analysis of Alternatives

Selection Element	Weight	Electric Ion Propulsion		Bi-Propellant		Mono-Propellant	
		Motivation	Score	Motivation	Score	Motivation	Score
Mass	**0.4**	Does not require much propellant mass to achieve needed Isp	1	Will consume large amounts of propellant	3	Will consume large amounts of propellant	3
Complexity	**0.10**	Ionization process less complex as compared to liquid feed systems	2	Feed Systems more complex compared to mono-propellant	4	Feed System slightly less complex than bi-propellant	3
Cost	**0.25**	Cost effective compared to liquid propellant	2	Separate fuel and oxidizer increases costs	5	Less costly than Bi-Propellant	4
Efficiency	**0.25**	Produces small thrust but high Isp	1	Works best for medium to heavy satellites and spacecrafts	5	Good in-space propulsion for smaller maneuvers (end-of-life)	3
Scores	**1.0**	**Best Choice**	**1.35**		**4.1**		**3.25**

Table 5-6-2 shows a breakdown of the decision process the team went through while debating whether chemical propulsion or electric propulsion was the best choice for this mission. Through this process the team was able to find that due to the fact that the VECTOR will be ridesharing in order to leave Earth's atmosphere, this removes what would be a large amount of work for the propulsion system. For this mission VECTOR only needs to insert into the Venusian orbit and complete various maneuvers from there. After examining VECTOR's propulsion needs it was found that moving forward with electric propulsion was the ideal choice, it was then time to select specific hardware.

The actual thruster that was selected for this mission is the ThrustMe NPT30-I2 electric iodine thruster[3]. This unit is only a 1U volume weighing 1.2kg fully loaded with Solid Iodine Propellant, making it very mass efficient while still providing very adequate performance[3]. Technical specifications are listed in Table 5-6-3 below. In order to satisfy the propulsion system requirements, specifically PRO-3.5.1 and PRO-3.5.4, four of these thrusters will be attached to the aft section of the cubesat. This can be visualized by examining the CAD model present in Section 4.7.

Figure 5-6-1 NPT30-I2 1U Electric Thruster

Table 5-6-3 Electric Thruster Technical Details

Wet Mass	1.2 kg
Size	96mm x 96mm x 113mm
Propellant	Solid Iodine
Total Power	35-65 W
Specific Impulse	<2400s
Input Voltage	12-28 V
Operating Temperature	-40 - 50 C
Radiation Tolerance	>20 krad

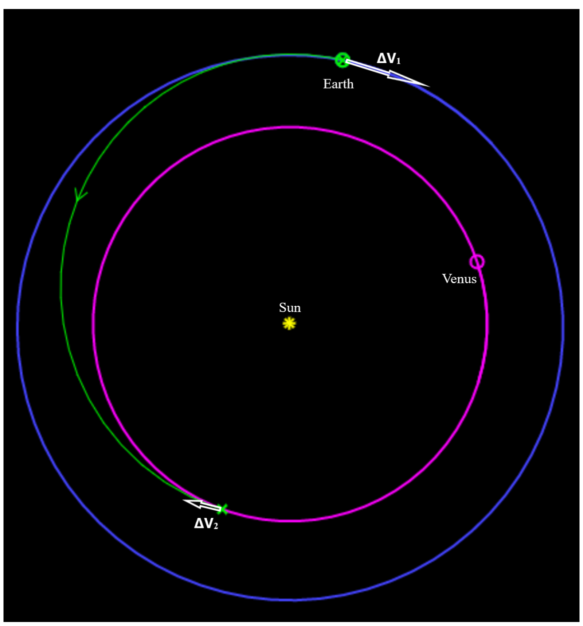
Now that the requirements have been laid out for the system, along with selection of an actual thruster, it is time to explore the technical details behind the mission trajectory, target orbit, and associated calculations.

Table 5-6-4 Delta V Requirements

ΔV Requirements	
ΔV_1 (Photon)	3.42 km/s
ΔV_2 (VECTOR)	0.4 km/s (minimum) - 0.97 km/s (nominal)

Table 5-6-4 above displays the velocity increments required for both departure from Earth orbit, and capture into Venus orbit. Rocket Lab's Photon spacecraft, as aforementioned, will carry VECTOR out of low Earth orbit and onto a nominal transfer trajectory to Venus. In order to do this, Photon must provide 3.42 km/s of velocity change, the first propulsive maneuver of the mission, to put VECTOR on a trajectory intersecting with Venus's orbit. Figure 5-6-2 below visually represents these propulsive maneuvers. The figure shows Earth and Venus in their nominal positions at the time of departure from Earth. An interesting effect of orbital mechanics is that since Earth is further away from the Sun, it is orbiting *slower* than Venus is. Thus, we must depart Earth's orbit when we are ahead of Venus so that we will rendezvous with it on the other side of the orbit. It is also important to note that the velocity vector for this first ΔV maneuver is pointing backwards with respect to the direction of our orbital motion. This is because the spacecraft must slow down, with respect to the Sun, in order to lower its orbit to that of Venus's orbit. Photon's 3.42 km/s of ΔV will slow the spacecraft down just enough to achieve this.

Figure 5-6-2 Transfer Orbit and ΔV Maneuvers



Once Photon has performed its burn and sent VECTOR on course to Venus, the cubesat will then have to perform its own propulsive burn utilizing the aforementioned electric ion thrusters. At the bare minimum, VECTOR will have to provide a velocity increment of 0.4 km/s to slow itself down enough to capture into a very loosely bound, highly elliptical orbit. While this would still satisfy MR-1.1, it would degrade the quality of data gathered from the neuromorphic camera and thus lower the scientific value of the mission. In the nominal case, VECTOR shall provide 0.97 km/s of ΔV in order to capture into the mission's ideal target orbit. The details for such orbit are displayed below in table 5-6-5. While this orbit is still elliptical, it brings the spacecraft very close to the surface for high quality observations with the payload and not so far away that the camera would not be useful at all. With this orbit, the cubesat will be able to perform 24/7 observations of the planet while also providing near continuous telemetry and data downlink back to Earth with NASA's DSN.

Table 5-6-5 Venus Orbit Parameters

Target Orbit Parameters			
Apoapsis	Periapsis	Orbit Period	Eccentricity
40,000 km	1,000 km	13.25 hours	0.734

Constants:

$r_E = 1.469 \times 10^8$ km (mean Sun-Earth distance)

$r_V = 1.0874 \times 10^8$ km (mean Sun-Venus distance)

$R_V = 6{,}051$ km (mean Venus radius)

$R_E = 6{,}378$ km (mean Earth radius)

$\mu_{sun} = 1.327 \times 10^{11}$ km^3/s^2 (gravitational parameter of the Sun)

$\mu_{Venus} = 324{,}860$ km^3/s^2 (gravitational parameter of Venus)

$\mu_{Earth} = 398{,}600$ km^3/s^2 (gravitational parameter of Earth)

$V_E = 29.78$ km/s (mean orbital speed of Venus)

$V_V = 35.02$ km/s (mean orbital speed of Earth)

With the planetary constants listed above, and utilizing the patched conics method for interplanetary trajectories, the departure and arrival ΔV's were computed along with the parameters for the target orbit. The ΔV values, while approximate, were verified against simulations and commonly accepted values to ensure accuracy.

ΔV_1 to achieve transfer orbit (Photon):

Transfer orbit parameters

$$V_{a,t} = \sqrt{2\mu_{sun}\frac{r_V}{r_E(r_E + r_E)}} = 27.72\frac{km}{s}$$

$$V_{p,t} = \sqrt{2\mu_{sun}\frac{r_E}{r_V(r_E + r_V)}} = 37.45\frac{km}{s}$$

Desired excess velocity after exiting Earth's SOI

$$V_{\infty,e} = V_E - V_{a,t} = 2.058\frac{km}{s}\,relative\ to\ Earth$$

ΔV_1 to initiate transfer is given by

$$\Delta V_1 = V_{after\ burn} - V_{park}$$

Velocity in 250 km LEO parking orbit (r_{park} = 250km + R_E)

$$V_{park} = \sqrt{\frac{\mu_{Earth}}{r_{park}}} = 7.78\frac{km}{s}$$

Required velocity after burn to achieve desired excess velocity

$$V_{after\ burn} = \sqrt{V_{\infty,e}^2 + 2\frac{\mu_{Earth}}{r_{park}}} = 11.2\frac{km}{s}\,relative\ to\ Earth$$

Thus, required ΔV_1 for Photon to initiate transfer orbit is

$$\Delta V_1 = V_{after\ burn} - V_{park} = \mathbf{3.42\frac{km}{s}}$$

ΔV_2 to capture into elliptical 1000km x 40,000km Venus orbit (VECTOR)

Excess velocity upon arrival into Venus SOI, relative to Venus

$$V_{\infty,V} = V_{p,t} - V_V = 2.43\,\frac{km}{s}\,relative\ to\ Venus$$

Semi-major axis of hyperbolic arrival trajectory

$$a = -\frac{\mu_{Venus}}{V_{\infty,V}^2} = -55{,}015.3\ km$$

Velocity at closest approach (1000km above surface)

$$V_{p,approach} = \sqrt{\frac{2\mu_{Venus}}{r} + \frac{\mu_{Venus}}{a}} = 9.9\,\frac{km}{s}\,relative\ to\ Venus$$

Desired orbit parameters:

$r_a = 46{,}051\ km, r_p = 7{,}051\ km$ (both including Venus radius)

$$a = \frac{r_a + r_p}{2} = 26{,}551\ km$$

$$e = 1 - \frac{r_p}{a} = 0.734$$

Velocity at periapsis of desired orbit

$$V_{p,desired} = \sqrt{\frac{\mu_{Venus}}{a}\left(\frac{1+e}{1-e}\right)} = 8.93\,\frac{km}{s}$$

Thus, ΔV_2 required to capture into elliptical Venus orbit is the velocity at periapsis of desired orbit minus our velocity at closest point of our hyperbolic approach:

$$\Delta V_2 = V_{p,desired} - V_{p,approach} = \mathbf{-0.97\ km/s}$$

5.7 Attitude Determination and Control System (ADCS)

The attitude determination and control system is essential for guidance and navigation towards and when in orbit around Venus. It is true that the propulsion system will be doing a lot of the heavy lifting during the transfer orbit, but it is only capable of pointing in the correct direction by the use of the ADCS and it's reaction wheels and sensors. Therefore, taking the time to establish the requirements and understand exactly what the system will need to do will overall lead to a more successful mission. These requirements can be found in Table 5-7-1.

Table 5-7-1 ADCS Requirements

ID	Requirement	Verification	Parent	Status
ADCS-3.7.1	System shall control the spacecraft attitude to reference frames to support default, propulsion, and science pointing modes	Analysis	SC-2.9	Meeting
ADCS-3.7.2	System shall autonomously control pointing of the solar array to within 2 degrees of the angle that provides optimal Sun illumination of the solar array in any attitude	Analysis	SC-2.9	TBD
ADCS-3.7.3	System shall provide attitude control of < 1 degree in all axes	Analysis	SC-2.9	Meeting
ADCS-3.7.4	System shall maintain and propagate on-board estimates of the spacecraft attitude and state vector	Analysis	SC-2.9	TBD

Now that the initial requirements are selected, an analysis of alternatives or trade matrix was established to determine which exact component worked best based on VECTOR's needs. A singular analysis of alternatives were performed for the ADCS, as shown in Table 5-7-2.

Looking at Table 5-7-2, the ADCS system needs to be very accurate for communication back to Earth and payload pointing. Additionally, the flight heritage and cost were decided to be equally important due to the high cost of ADCS systems and the fragility of systems in the small sat market. The three ADCS systems selected for review against each other were the XAT-100 from Blue Canyon Technologies (BCT), IADCS400 from AAC Clyde Space, and CubeADCS Gen2 from CubeSpace. Reviewing Table 5-8-2, BCT's XAT-100 ADCS system won based on the flight heritage in CubeSat and MicroSat missions, extremely high accuracy, and compact size.

Table 5-7-2 ADCS Analysis Of Alternatives

Selection Element	Weight	BCT XACT-100		AAC Clyde Space IADCS400		CubeSpace CubeADCS Gen 2	
		Motivation	Score	Motivation	Score	Motivation	Score
Mass	**.1**	1.52kg	2	1.3kg	1	1.25kg	1
Flight Heritage	**.2**	Multiple successful LEO and beyond flights	1	Multiple successful LEO missions	2	Just launched in LEO missions	4
Cost	**.2**	~$500k	4	Unknown	3	~$65k	1
Accuracy	**.2**	+/- .003 degrees (1-sigma)	1	< 1 degree (3-sigma)	4	< .1 degrees	2
Size	**.1**	.5U	1	.7U	2	1U	4

Taking a closer look into the XAT-100, this system features a main housing module which includes a star camera, a CPU, and MEMS IMU. A key feature is that the four reaction wheels are not connected into the main module, but instead as separate units. This is genius due to the fact that with a 12U CubeSat, our center of mass may not be in the middle of the spacecraft. Therefore, we can move these wheels to best control our satellite through testing and modeling! Additionally, the system will come with two pyramid Sun sensors which are tasks to find and detect the Sun. For reliability and confirmation, our team has opted to include a second star camera for redundancy and increased accuracy of our position and for the injection burns [6].

Figure 5-7-1 XACT-100 Housing

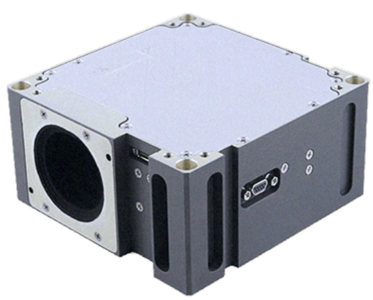


Figure 5-7-2 XACT-100 Reaction Wheels

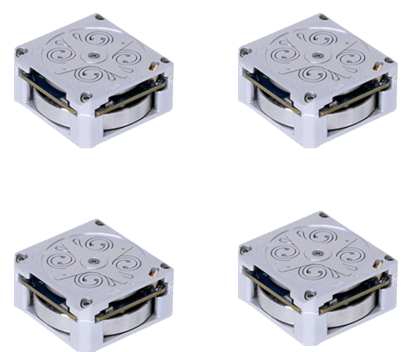


Figure 5-7-3 Pyramid Sun Sensor

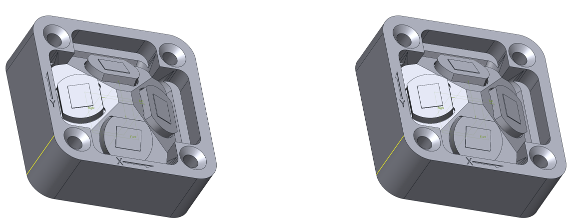


Figure 5-7-4 Star Camera

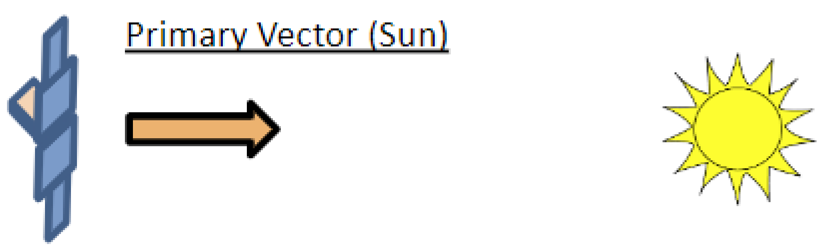
An issue discovered when performing research into the ADCS system was having to unload momentum build up in reaction wheels. In a typical LEO CubeSat, the wheels will be desaturated using a magnetorquer. However, the magnetic field is exponentially smaller at Venus than on Earth, and this solution will not be an option for VECTOR. A potential solution could be to use the propulsion system and three of the four wheels to desaturate a single wheel at a time. However, for redundancy and overall general accepted aerospace practice, we have opted to include Bradford ECAPS 100 mN HPGP thrusters. Having 8 thrusters in total, each being .04 kg and having .5 kg of LMP-103S propellant, they will be used to help unload the angular momentum. Essentially, when slowing the reaction wheel speed they get saturated and by performing an external torque this will reduce the saturation.

Figure 5-7-5 HPGP Thrusters

Now that the entire system has been selected, moving onto the different ADCS modes is extremely important. The main computer will communicate with the ADCS to determine which mode to go into. There are a multitude of different modes, but highlighted in this document are Safe Mode, Fine Reference Point: Thrusting, and Fine Reference Point: Venus Science.

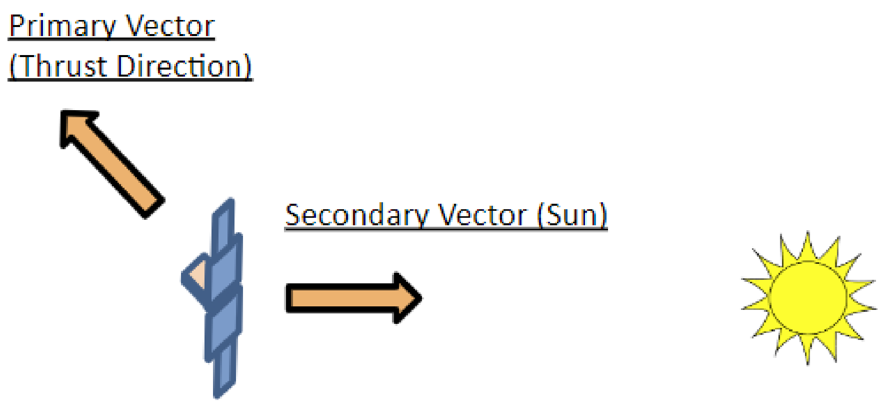
Beginning with Safe Mode, this will be the most basic and standard positioning for the ADCS. The entire system is defined on the vector pointed to the Sun. The objective is to use the Sun sensors to acquire the Sun. If the Sun is not initially in the line of sight, the spacecraft will rotate until the Sun is discovered. Once the Sun is acquired, VECTOR will rotate at a table defined rate about the Sun line. Therefore, this mode's main purpose is to charge the spacecraft and keep the spacecraft in view of the Sun for long duration charging.

Figure 5-7-6 ADCS Safe Mode



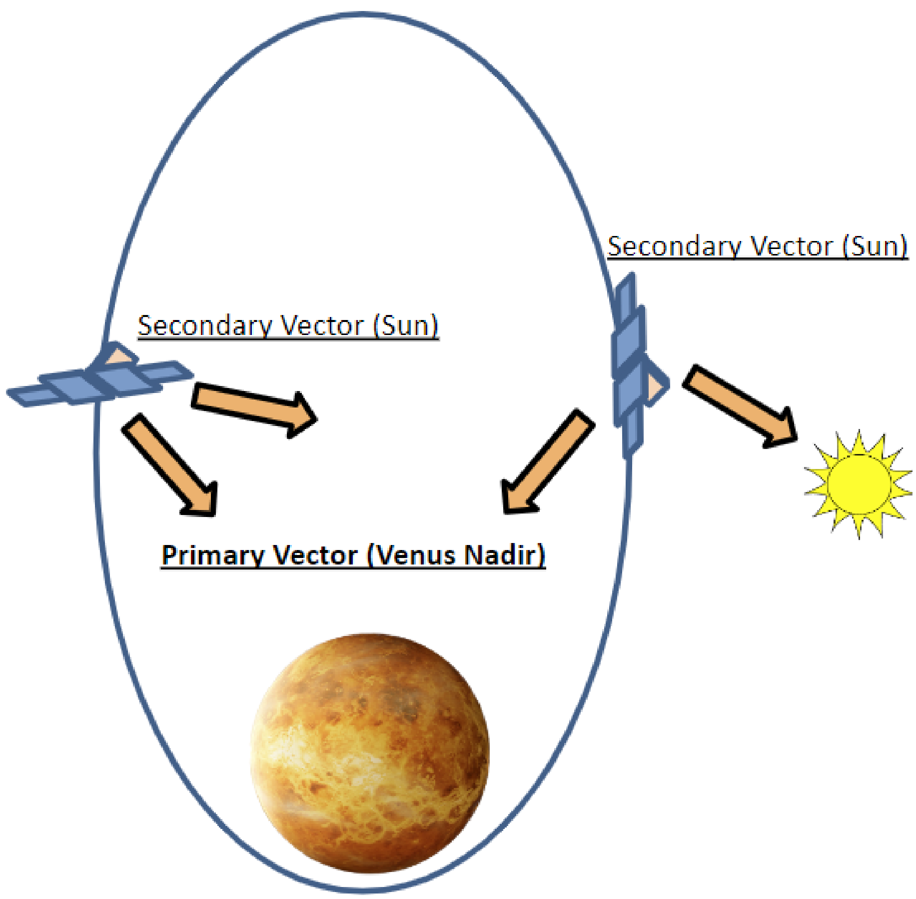
Continuing to Fine Reference Point: Thrusting, this mode will be defined by two vectors of the thrusting direction and the Sun vector. The primary vector is met by the control system and this then defines 2-axes of attitude. The secondary vector is met as closely as possible and defines the rotation about the primary axis. This method is used for when thrusting towards Venus or while in Orbit and performing orbit corrections. It should be noted that when thrust and the Sun vector become co-aligned, a yaw flip will automatically occur.

Figure 5-7-7 ADCS Fine Reference Point: Thrusting



Finally, the Fine Reference Point: Venus Science, will also be defined by two vectors. The primary being Venus nadir to point the radio and neuromorphic camera, and the secondary being the sun vector to point the solar array. The primary vector is met by the control system, and again, defines 2-axis of attitude. The secondary vector is met as closely as possible and defines the rotation around the primary axis. Just as with Fine Reference Point: Thrusting, when thrust and the Sun vector become co-aligned, a yaw flip will automatically occur.

Figure 5-7-8 ADCS Fine Reference Point: Venus Science



5.8 Electrical Power System (EPS)

The electrical power system is in charge of generating, regulating, storing, and distributing all the electrical energy in the spacecraft. When determining the makings of this important system, the first step is to determine the requirements based on other subsystem needs. These requirements can be found in Table 5-8-1.

Table 5-8-1 EPS Requirements

ID	Requirement	Verification	Parent	Status
EPS-3.8.1	System shall generate, regulate, store, and distribute power to each subsystem	Test	SC-2.3	Meeting
EPS-3.8.2	System shall be powered off prior to separation and, upon detecting separation, power on C&DH and TT&C	Test	SC-2.7	TBD
EPS-3.8.3	System shall use approved batteries based on NASA/TM-2009-215751	Inspection	SC-2.3	Meeting
EPS-3.8.4	System shall provide 8 solar array inputs	Test	SC-2.3	Meeting

Now that the initial requirements are selected, an analysis of alternatives or trade matrix was established to determine which exact component worked best based on VECTOR's needs. The two analyses of alternatives performed for the EPS were the solar panels (shown in Table 5-8-2) and the batteries (shown in Table 5-8-3).

Looking at Table 5-8-2, the EPS system will need to provide power to each of the subsystems and should have a sufficiently high peak power. For this reason, that selection element was rated the highest with half of the weight. Additionally, the cost and flight heritage were equally important due to our high risk of being a CubeSat and our low cost budget. The three solar panels selected for review against each other were the HaWK 38A-191 from MMA Design, PHOTON from AAC Clyde Space, and 12U-H Triple Wing from Blue Canyon Technologies (BCT). Reviewing Table 5-8-2, MMA Design's HaWK 38A-191 solar panels won based on the very high peak power and efficiency of solar cells, as well as the high success and flight heritage from high class missions such as MarCo and LunaH-Map.

Table 5-8-2 Solar Panel Analysis Of Alternatives

Selection Element	Weight	MMA HaWK 38A-191		AAC Clyde Space PHOTON		BCT 12U-H Triple Wing	
		Motivation	**Score**	**Motivation**	**Score**	**Motivation**	**Score**
Mass	**0.1**	No significant advantage	2	No significant advantage	2	No significant advantage	2
Flight Heritage	**0.2**	Multiple successful LEO and beyond flights	1	Flown mostly with 3U configuration	2	Flown mostly with 6U configuration	2
Cost	**0.2**	Aerospace Industry Standard	3	Aerospace Industry Standard	3	Aerospace Industry Standard	3
Peak Power	**0.5**	191W, 29.5% Efficiency	1	85W, Unknown Efficiency	4	118W, 30% Efficiency	2
Scores	**1.0**		**1.5**		**3.2**		**2.2**

Taking a closer look into the MMA HaWK 38-A-191, as shown in Figures 5-8-1 and 5-8-2, this system features 4 deployed panels per side. Each panel is a 2U x 3U, and the total system generates a maximum of 191 watts. This system also features a solar array drive assembly (SADA), which has the capability to rotate the solar panels a full 360 degrees to find the Sun in any orientation. Finally, these solar panels have such a high level of flight heritage and with a TRL of 9, these make for an excellent and reliable choice [15].

Figure 5-8-1 Generic 12U CubeSat Rendering

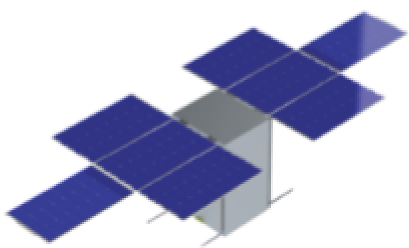
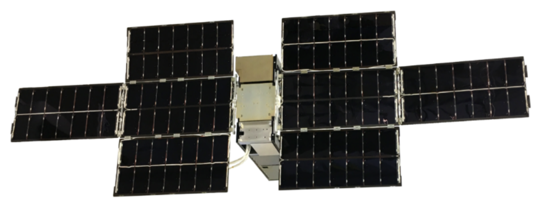


Figure 5-8-2 Flown 6U Cubesat



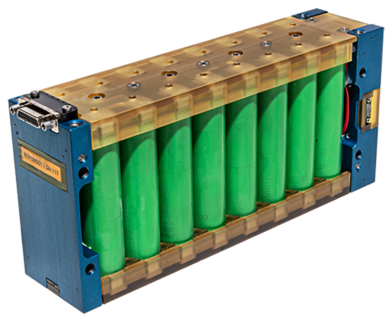
Looking at Table 5-8-3, batteries are essential for storing power during our housing inside the rocket before launch and throughout our orbit where we are hidden from the Sun. The three batteries selected for review against each other were the Nanopower BPX from GOM aerospace, B28-135 from Ibeos, and 2P8S from Blue Canyon Technologies (BCT). Reviewing Table 5-8-3, BCT's 2P8S was selected due to the flight heritage and the maximum energy storage with 198Wh. Additionally, they include over and under voltage protection and BMS protection.

Table 5-8-3 Battery Analysis Of Alternatives

Selection Element	Weight	GOM NanoPower BPX		Ibeos B28-135		BCT 2P8S	
		Motivation	**Score**	**Motivation**	**Score**	**Motivation**	**Score**
Mass	**0.2**	2 x .5kg = 1kg	2	.8kg	1	1.2kg	3
Flight Heritage	**0.3**	Mostly LEO missions	3	In development, never flown in space	5	Multiple successful LEO and beyond flights	1
Cost	**0.2**	Aerospace Industry Standard	3	Aerospace Industry Standard	3	Aerospace Industry Standard	3
Energy	**0.3**	2 x 86Wh = 172Wh	4	135Wh	3	198Wh	1
Scores	**1.0**		**3.1**		**3.2**		**1.8**

Taking a closer look into the BCT 2P8S batteries, as shown in Figure 5-8-3, this system features 16 18650 lithium ion batteries where 2 are in parallel and 8 are in series. The total system can store a maximum of 198 watt/hours. As mentioned above, these batteries have built in protection for ground testing and variation in voltages if in flight issues occur. Finally, these batteries have such a high level of flight heritage and with a TRL of 8, they meet the requirements for the EPS system [6].

Figure 5-8-3 BCT 2P8S Battery



Finally, the EPS system will need an actual power system board. The chosen system will be the Ibeos 28-200 EPS, as shown in Figures 5-8-4 and 5-8-5. This system, like the name suggests, will be a 28V board with a nominal power of 200 watts. However, the maximum power is 250 watts, which is quite a large margin and leaves room for discrepancies between data sheets and flatsat testing. Additionally, there are 3 main voltage rails fo 3V, 5V, and 12V with the VBAT being 28V. Additionally, the board features communication lines though I2C and SPI with a watch dog for protection. This system is estimated to be TRL 8 and meets our requirements. The block diagram can be found in Figure 5-8-6, which features the voltage rails, communication lines, and other internal systems [1].

Figure 5-8-4 Assembled EPS Board

Figure 5-8-5 EPS CAD Model

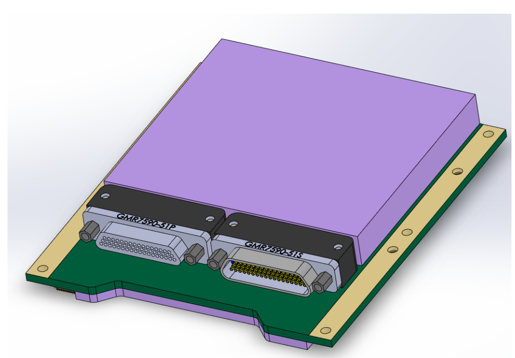
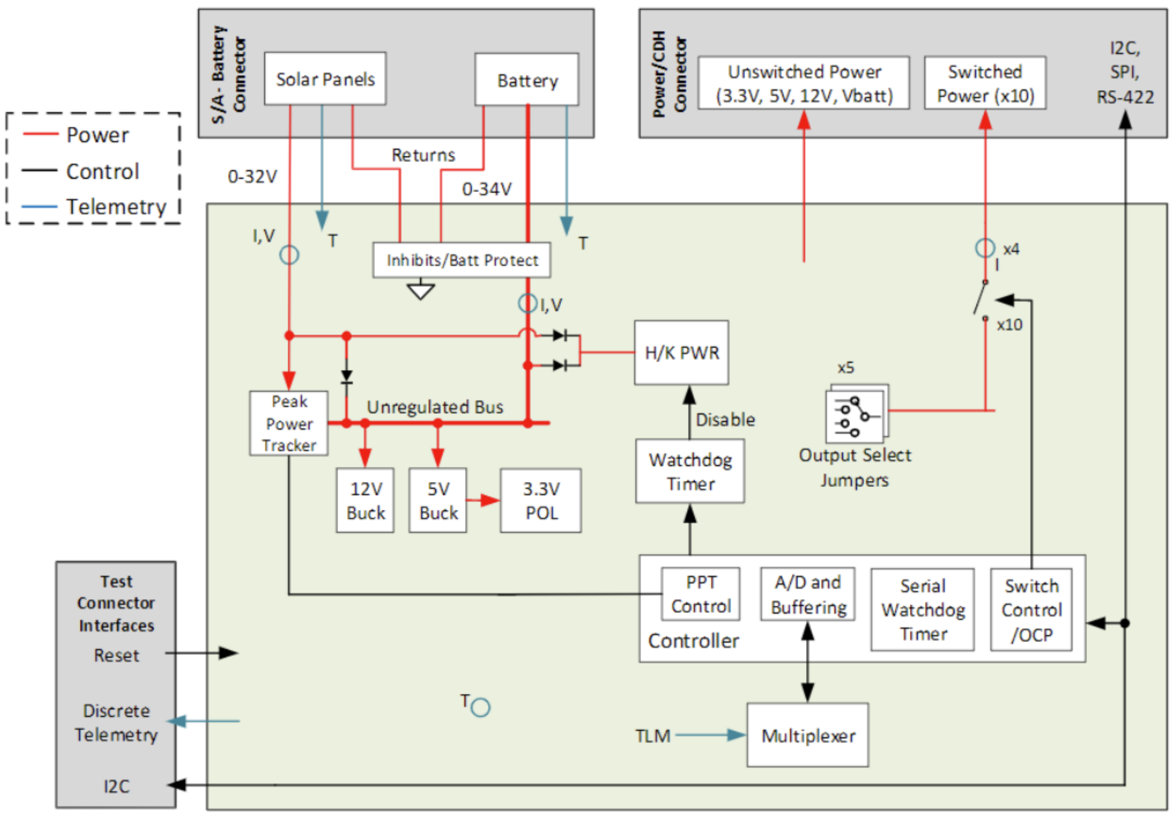


Figure 5-8-6 EPS Block Diagram



5.9 Structure (STR)

To meet the requirements of STR-3.9.1, the Bus subsystem must be designed with a sufficient internal volume to accommodate all required components. Additionally, the Bus subsystem must be compatible with a 12U deployer, as required by STR-3.9.2.

To satisfy the requirements of STR-3.9.3, the Bus subsystem must be designed to provide appropriate mounting interfaces and clearances for the solar panels and SADA. The specific design of the mounting interfaces and clearances will be informed by further analysis and testing.

Finally, the Bus subsystem must remain fully operational following a random vibrations test, as required by STR-3.9.4. This will require careful consideration of the structural and electrical design of the Bus subsystem to ensure that it can withstand the vibrational stresses that may be encountered during launch and operation.

Overall, the design of the Bus subsystem will be critical to the success of the CubeSat mission. The detailed design of the Bus subsystem will be informed by further analysis and testing, including inspections and verification tests, to ensure that all requirements are met.

Table 5-9-1 Structure Requirements

ID	Requirement	Verification	Parent	Status
STR-3.9.1	System shall accommodate a 12U internal volume	Inspection	LS-2.1	Meeting
STR-3.9.2	System shall interface with 12U deployer	Inspection	LS-2.1	Meeting
STR-3.9.3	System shall provide mounting interface and clearance accommodations for solar panels and SADA	Inspection	LS-2.1	Meeting
STR-3.9.4	System shall remain fully operational following a random vibrations test	Testing	LS-2.3	TBD

A trade study was performed to decide between buying a COTS bus or building it in house. Due to the mission statement, the COTS bus won out.

Table 5-9-2 Structure Analysis Of Alternatives

Selection Element	Weight	Enduro Sat		Custom Built	
		Motivation	Score	Motivation	Score
Cost	**0.4**	$12,500	**1**	~$6,000 machining + money spent designing/review	**3**
Schedule Risk	**0.3**	Immediately available	**1**	Needs to be designed	**4**
V&V	**0.3**	COTS tested	**2**	Needs to be internally vibed, temperature cycled, etc	**4**
Scores			**1.3**		**3.6**

6. Spacecraft Budgets

6.1 Description

After looking at each individual subsystem, the components can be incorporated to create a power, mass and cost budget for VECTOR. These will detail the generated and used power, consumed mass, and projected cost for each subsystem. This analysis will permit a more detailed analysis of VECTOR's feasibility with its current design, and whether or not mission requirements are being met.

6.2 Power Budgets

Starting with the power budget, this is a relatively simple but important calculation to determine how much power is being generated and consumed. For VECTOR, we created one budget for our insertion burn and one for standard orbit operations. Starting with the insertion burn, shown in Table 6-2-1, it can be seen that the power margin is very small. This is the reason for splitting the power budget into two different scenarios due to the high power consumption of the thrusters. Each thruster has a maximum power of 65W and with four firing, this becomes really costly. Simulations are currently being looked at to determine if we can go less than full thrust and adjust the power down from 65W to 50W. This will help increase our power margin significantly. Additionally, it can be seen that the instantaneous power of each system is multiplied by the duty cycle to get the average power. This duty cycle is the duration of the burn itself to get into our Venus orbit. Finally, during this burn a majority of the payload features will be modulated due to the small power margin and wanting to ensure the thrusters, CPU, and ADCS have the power available.

Moving on to the orbit operations power budget, shown in Table 6-2-2, this has a plethora of power margin. Due to the orbit being established, the thrusters are only needed for minor correction burns very rarely and therefore the power required by them is significantly less. The payload and main operation components take a relatively small amount of power and this leads to the large power margin available for VECTOR once in operation. It may be possible that the duty cycle will be less than .9 for the solar panels, but due to the highly elliptical orbit this was assumed that a majority of the time on orbit will be spent in the Sun. More simulations are required to adequately adjust and confirm values.

Table 6-2-1 Insertion Power Budget

Subsystem	Generic Component Name	Specific Component Name	Qty	Instantaneous Power	Total Instantaneous Power	Duty Cycle	Orbital Average Power (w/ Inefficiency)
GENERATED							
EPS	Solar Array	MMA HaWK 38A-191	1	191.00W	191.00W	0.9	154.71W
	Battery	BCT 2P8S	1	0.00W	0.00W	1	0.00W
						TOTAL GENERATED POWER	***154.71W***
CONSUMED							
C&DH	Mother board	CHREC Space Processor	1	2.85W	2.85W	1	2.85W
	Interface Board	Custom	1	1.00W	1.00W	1	1.00W
EPS	Power System	Ibeos 28-200	1	1.50W	1.50W	1	1.50W
TCS	Shielding	Z-Grade Radiation Shielding 3U Plates	8	0.00W	0.00W	0	0.00W
	Heaters	HK6909 Polyimide Thermofoil™ Heaters	4	25.47W	25.47W	0.25	6.37W
	Confirmal Coating	Arathane	2	0.00W	0.00W	0	0.00W
Payload	Camera	Inivation DVXplorer	2	0.70W	1.40W	0.25	0.35W
	Antenna	Custom Deployed Monopole	1	0.00W	0.00W	1	0.00W
	Radio	Red Pitaya STEMlab 125-14	1	10.00W	10.00W	0.25	2.50W
ADCS	ADCS	BCT XACT-100	1	5.00W	5.00W	1	5.00W
	Star Tracker	BCT Standard NST	1	1.66W	1.66W	1	1.66W
	Propellent	LMP-103S	1	0.00W	0.00W	1	0.00W
	Momentum Mangement Thursters	100 mN HPGP Thruster	8	7.00W	56.00W	0.015	0.84W
TT&C	Transponder	SDL IRIS V2	1	35.00W	35.00W	0.077	2.69W
	Antenna	JPL Deployable X-Band	1	0.00W	0.00W	1.000	0.00W
	Antenna	Endurosat X-Band Patch	1	0.00W	0.00W	1.000	0.00W
Structure	Bus Structure	Endurosat 12U Standard Bus	1	0.00W	0.00W	0	0.00W
	Cabling & Harnessing	Custom	1	0.00W	0.00W	0	0.00W
Propulsion	Thruster	NPT30-12 1U Electric Propulsion System	4	65.00W	260.00W	0.48	124.80W
						TOTAL CONSUMED POWER	***149.56W***
						POWER MARGIN	***5.15W***

Table 6-2-2 Orbit Power Budget

Subsystem	Generic Component Name	Specific Component Name	Qty	Instantaneous Power	Total Instantaneous Power	Duty Cycle	Orbital Average Power (w/ Inefficiency)
GENERATED							
EPS	Solar Array	MMA HaWK 38A-191	1	191.00W	191.00W	0.9	154.71W
	Battery	BCT 2P8S	1	0.00W	0.00W	1	0.00W
						TOTAL GENERATED POWER	***154.71W***
CONSUMED							
C&DH	Mother board	CHREC Space Processor	1	2.85W	2.85W	1	2.85W
	Interface Board	Custom	1	1.00W	1.00W	1	1.00W
EPS	Power System	Ibeos 28-200	1	1.50W	1.50W	1	1.50W
TCS	Shielding	Z-Grade Radiation Shielding 3U Plates	8	0.00W	0.00W	0	0.00W
	Heaters	HK6909 Polyimide Thermofoil™ Heaters	4	25.47W	25.47W	0.25	6.37W
	Confirmal Coating	Arathane	2	0.00W	0.00W	0	0.00W
Payload	Camera	Inivation DVXplorer	4	0.70W	2.80W	1	2.80W
	Antenna	Custom Deployed Monopole	2	0.00W	0.00W	1	0.00W
	Radio	Red Pitaya STEMlab 125-14	1	10.00W	10.00W	1	10.00W
ADCS	ADCS	BCT XACT-100	1	5.00W	5.00W	1	5.00W
	Star Tracker	BCT Standard NST	1	1.66W	1.66W	1	1.66W
	Propellent	LMP-103S	1	0.00W	0.00W	1	0.00W
	Momentum Mangement Thursters	100 mN HPGP Thruster	8	7.00W	56.00W	0.025	1.40W
TT&C	Transponder	SDL IRIS V2	1	35.00W	35.00W	0.077	2.69W
	Antenna	JPL Deployable X-Band	1	0.00W	0.00W	1.000	0.00W
	Antenna	Endurosat X-Band Patch	2	0.00W	0.00W	1.000	0.00W
Structure	Bus Structure	Endurosat 12U Standard Bus	1	0.00W	0.00W	0	0.00W
	Cabling & Harnessing	Custom	1	0.00W	0.00W	0	0.00W
Propulsion	Thruster	NPT30-12 1U Electric Propulsion System	4	65.00W	260.00W	0.038	10.00W
						TOTAL CONSUMED POWER	***45.27W***
						POWER MARGIN	***109.44W***

6.3 Mass Budgets

Moving on to the mass budget, shown in Table 6-3-1 and Table 6-3-2, VECTOR comes in at about 18 kg and 21.6 kg with a 20% margin. This falls below the 24 kg mass limit imposed by the CubeSat standard.

Table 6-3-1 Mass Summary Analysis

Total Mass	17.98 kg
Total Mass With 20% Margin	21.58 kg
Maximum Mass (Per 12U Standard)	24 kg

Table 6-3-2 Detailed Mass Budget

Subsystem	Generic Component Name	Specific Component Name	Qty	Mass	Total Mass
C&DH	Mother board	CHREC Space Processor	1	0.07kg	0.07kg
	Interface Board	Custom	1	0.02kg	0.02kg
EPS	EPS	Ibeos 28-200	1	0.14kg	0.14kg
	Solar Array	MMA HaWK 38A-143	1	0.71kg	0.71kg
	Battery	BCT 2P8S	1	1.20kg	1.20kg
TCS	Shielding	Z-Grade Radiation Shielding 3U Plates	8	0.20kg	1.60kg
	Confirmal Coating	Arathane	2	0.10kg	0.20kg
Payload	Camera	Inivation DVXplorer	2	0.10kg	0.20kg
	Antenna	Custom Deployed Monopole	1	0.50kg	0.50kg
	Radio	Red Pitaya STEMlab 125-14	2	0.25kg	0.50kg
ADCS	ADCS	BCT XACT-100	1	1.52kg	1.52kg
	Star Tracker	BCT Standard NST	1	0.35kg	0.35kg
	Momentum Mangement Thursters	100 mN HPGP Thruster	8	0.04kg	0.32kg
TT&C	Transponder	SDL IRIS V2	1	1.10kg	1.10kg
	Antenna	JPL Deployable X-Band	1	1.25kg	1.25kg
	Antenna	Endurosat X-Band Patch	2	0.05kg	0.11kg
Structure	Bus Structure	Endurosat 12U Standard Bus	1	2.40kg	2.40kg
	Cabling & Harnessing	Custom	1	1.00kg	1.00kg
Propulsion	Thruster	NPT30-12 1U Electric Propulsion System	4	1.20kg	4.80kg
				TOTAL MASS	17.98kg
				TOTAL MASS WITH 20% MARGIN	21.58kg

6.4 Cost Budgets

Finally, looking at the cost budget, shown in Table 6-4-1, Table 6-4-2, and Table 6-4-3, it can be seen that the total mission cost is estimated to be $25.8 Million. However, just looking at flight and EDU components with a 20% margin, VECTOR will cost $12.18 Million. Overall, the missions will be 3-4 years and be in operation for a minimum of 1.5 years.

Table 6-4-1 Cost Summary Analysis

Total Mission Cost	$25.8 Million USD
Spacecraft Components Cost With 20% Margin (W/ EDU units)	$12.18 Million USD
Development Time	3-4 Years
Mission Operational Time	1.5 Years (minimum)
Funding Mechanism	NASA SIMPLEx

Looking at the individual subsystems, it can be seen that the most expensive components are the solar panels, radio, ADCS system, and the deployable antenna. These were expected due to the high reliability, quality engineering, and successful flight history. When looking at the entire mission it is very obvious that the engineering time over the 3-4 years is the highest portion at an estimated $5.76 Million for a small team of twelve.

Table 6-4-2 Detailed Cost Budget

Subsystem	Generic Component Name	Specific Component Name	Qty	Price	Total Cost
C&DH	Mother board	CHREC Space Processor	2	$500,000.00	$1,000,000
	Interface Board	Custom	2	$10,000.00	$20,000
EPS	EPS	Ibeos 28-200	2	$500,000.00	$1,000,000
	Solar Array	MMA HaWK 38A-191	1	$1,000,000.00	$1,000,000
	Battery	BCT 2P8S	3	$100,000.00	$300,000
TCS	Shielding	Z-Grade Radiation Shielding 3U Plates	8	$10,000.00	$80,000
	Confirmal Coating	Arathane 5750	2	$500.00	$1,000
Payload	Camera	Inivation DVXplorer	4	$3,900.00	$15,600
	Antenna	Custom Deployed Monopole	2	$1,000.00	$2,000
	Radio	Red Pitaya STEMlab 125-14	2	$377.00	$754
ADCS	ADCS	BCT XACT-100	2	$500,000.00	$1,000,000
	Star Tracker	BCT Standard NST	2	$75,000.00	$150,000
	Momentum Mangement Thrusters	Bradford ECAPS 100 mN HPGP Thruster	8	$200,000.00	$1,600,000
TT&C	Transponder	SDL IRIS V2	2	$500,000.00	$1,000,000
	Antenna	JPL Deployable X-Band	1	$760,000.00	$760,000
Structure	Bus Structure	Endurosat 12U Standard Bus	2	$12,500.00	$25,000
	Cabling & Harnessing	Custom	2	$100,000.00	$200,000
Propulsion	Thruster	NPT30-12 1U Electric Propulsion System	4	$500,000.00	$2,000,000
				TOTAL COST	$10,154,354
				TOTAL COST WITH 20% MARGIN	$12,185,225

Table 6-4-3 Cost Budget Breakdown

C&DH	$1,224,000.00
EPS	$2,760,000.00
TCS	$97,200.00
Payload	$22,024.80
ADCS	$3,300,000.00
TT&C	$2,112,000.00
Structure	$270,000.00
Propulsion	$2,400,000.00
Launch	$2,000,000.00
Operations	$2,000,000.00
Ground System	$650,000.00
Integration & Test	$800,000.00
Software	$1,250,000.00
Mangement	$1,250,000.00
Engineering Time	$5,760,000.00
Total	$25,895,224.80

7. Verification & Validation

7.1 V& V Description

Verification and validation is a crucial step in any space flight mission, for the sake of completing the mission safely and ensuring to investors that the team is the appropriate steps to avoid any potential errors. All requirements discussed in the report will be verified and ultimately validated through the means of inspection, demonstration, analysis or testing. All of the system-level requirements have an identified verification method, as shown earlier in the report, each specific method is embedded in the respective requirements charts.

The main verification technique will be testing, since it is the best way to ensure the part is working to the level promised by the supplier. Any part that the team receives that can not undergo a physical test will be tested through analysis, inspection or demonstration, depending on the functions of each system. A lot of the tests that will occur will not just be for the life of the mission, the team plans to test in a fashion that ensures the system will not only be functional through the original life of the mission, but also through extreme circumstances and if the mission lasts longer than hoped.

The team is currently at the CDR level, after which will be the SIR level, where the team will start receiving parts, after which the verification and validation level will begin, followed by the flight readiness review, showing that the V&V step is the last crucial step before the CubeSat will be put together. It is crucial that the team is testing the parts to fully ensure there are no faults in the products received.

7.2 Critical Design Requirements

Table 7-2-1 lays out the Verification and Validation plan for the Critical Design Requirements that were discussed earlier in the report in Table 3-8-1. They have been aggregated in the form below to ensure that key requirements are being managed correctly.

The Momentum Control System Testbed relates directly to SC-2.8, which will be used to test our Attitude Determination and Control System after receiving all the parts for the suppliers. Ensuring the attitude control systems are operating accurately is crucial for this mission, and performing a test is the best way to ensure the system is working at the level the team desires. VECTOR's structure and sub-systems will endure Cobalt-60 radiation testing, where in the span of a few days, the test can simulate years of exposure, allowing us to ensure that the system will not only be capable to last the desired mission length, rather years past the time we hope, as we have seen many mission go long past the expected end date. The thrusters we will be receiving will undergo testing due to the fact they are off the shelf products, and we need to ensure that each thrusters is acting the same as the others.

Table 7-2-1 Critical Design Requirements V&V

Req ID #	Verification Method	Test
SC-2.8 ADCS-3.7.1 ADCS-3.7.2 ADCS-3.7.3 ADCS-3.7.4	Testing	Momentum Control System Testbed (Facility TBD)
SC-2.12 SC-2.17	Testing	Simulate a 5-year mission worst case scenario by exposing the satellites' subsystems to a Cobalt-60 radiation source (Facility TBD)
PRO-3.5.1 PRO-3.5.1 PRO-3.5.2 PRO-3.5.3 PRO-3.5.4 PRO-3.5.5	Testing	Thrust Test Bench (Facility TBD)

7.3 System Performance Requirements

Table 7-3-1 shows the Verification and Validation plan for the System Performance Requirements that were discussed earlier in the report and shown in Table 3-7-1. Again, they have been aggregated to ensure that key requirements are being managed correctly. Unlike the Critical Design Requirements where testing is an option for all the systems, some of the System Performance Requirements will be verified through analysis along with testing when possible. As seen in the first two methods, the team plans to use programs like NASTRAN to perform analysis of the CAD model, which behind testing is an efficient an trustworthy commercial method of verifying our system.

Table 7-3-1 System Performance Requirements V&V

Req ID #	Verification Method	Test
LS-2.3 STR-3.9.4	Testing	Analysis of the CAD model using **NASTRAN** will be performed to verify the preliminary design. A **viderations test(s)** will be performed to validate the system after the CDR. (facility TBD)
SC-2.2 SC-2.5 TCS-3.2.1 TCS-3.2.2 TCS-3.2.3	Analysis	Analysis of the CAD model using **NASTRAN** will be performed to verify the preliminary design.**Thermal-Vac lifecycle test** will be performed to validate the completed system. Unit thermal-vac tests will be performed on critical subsystems throughout the build process. (facility TBD)
SC-2.3 EPS-3.8.1	Testing	**Natural sunlight or a solar simulator tests** will be performed. The set of measurements is performed from the solar panel open-circuit point, where the output current will be at its minimum and the output voltage reaches its maximum value, i.e. Voc. From this testing condition, the output current is increased until its maximum value is reached, i.e. Isc, when the solar panel is short-circuited and the output voltage is at its minimum value, i.e. zero.
SC-2.12	Analysis	Computational Electromagnetics (CEM) Laboratory and Computational Electromagnetics (CEM) Laboratory facilities at NASA Johnson Space Center. (ideal)

8. Risk

8.1 Description

Engineering a satellite is difficult due to the seemingly infinite amount of risks posed by internal and external sources. In order to characterize and narrow this list, management of higher severity risks must be taken into account, which limits the risk posed upon a system. Through careful management and planning, these high severity risks can be mitigated or reduced in gravity.

8.2 Risk Management

Being a CubeSat, VECTOR will impose a larger risk than a normal satellite. This is accepted due to the nature of the mission and the limited budget. However, there are risks that should be considered and approached with either research (R), mitigation (M), watch (W), or acceptance (A). On the system level, seven risks were identified, with each being rated on a scale from 1 to 5 for both likelihood and consequence. Each of these risks can be seen in Table 8-2-1. Similarly, in Table 8-2-2 the risks are identified in a matrix showing the severity and position mapped according to Table 8-2-1.

To begin with, the environment is a major factor that needs to be considered and researched further. Going from Earth to Venus, there will be an increase in radiation that no other CubeSat has ever encountered. This will be rather difficult to handle, and research into previous Venus (or other planetary) missions should be looked into.

Moving then to Propulsion, CubeSats have a relatively short history with propulsion technology and specifically have never been tested performing an insertion burn around another planet. VECTOR will be a demonstration of this technology and for this reason, watching the risk and selecting a propulsion system which will successfully meet SC-2.21.

The most critical risk VECTOR will have is communication. Due to the small size (12U) of VECTOR, having a communication link from Venus will require a very high gain antenna, extremely accurate pointing, and an atomic clock. Each will be costly, and difficult to fit and implement. For this reason, the communication link will be rated at a very high likelihood and consequence at the current moment of SDR. Furthermore, any failure of the communication system, even partial, will completely interrupt the mission, potentially permanently. Delicate electrical systems such as communications are also highly susceptible to radiation.

Part of the purpose of a Cubesat mission is to decrease the cost of a typical satellite mission. Continuing with this standard, VECTOR will be less than 10% of the previous Venus satellite mission (Akatsuki) and result in around $25 Million in funding. Although this is a lot of money, space is expensive and due to the choice of COTS components, these may eat a majority of the budget (besides engineering time for small staff). Due to this, maintaining our budget will be difficult, but will be accepted due to the nature of our mission.

With the size of VECTOR, power will be rather limited solely based on the surface area. As the system components begin to be selected, a power budget will determine the power requirements and needs of VECTOR. Due to this, power is a relatively medium likelihood, but has high consequences. As determined by analysis in section 6.2, VECTOR as currently designed has satisfactory power margins.

Similar to budget, due to the nature of CubeSats, a schedule is rapid in comparison to most major missions. Some missions can take upwards of 10 years to develop and additional years for building and certifying flight readiness. With the low cost and higher risk, a much shorter timeline is required for CubeSats. VECTOR will need to meet its tight schedule and get commercial vendors to adhere to the timelines. This is a low likelihood, but again a high consequence as it might mean a missed launch window to Venus.

Finally, using COTS components will be high risk due to most of the components not being radiation hardened to the specs required and additionally flying in a condition not previously rated. Additionally, with these components not being developed to meet the specific needs of the mission, failure is at a higher risk.

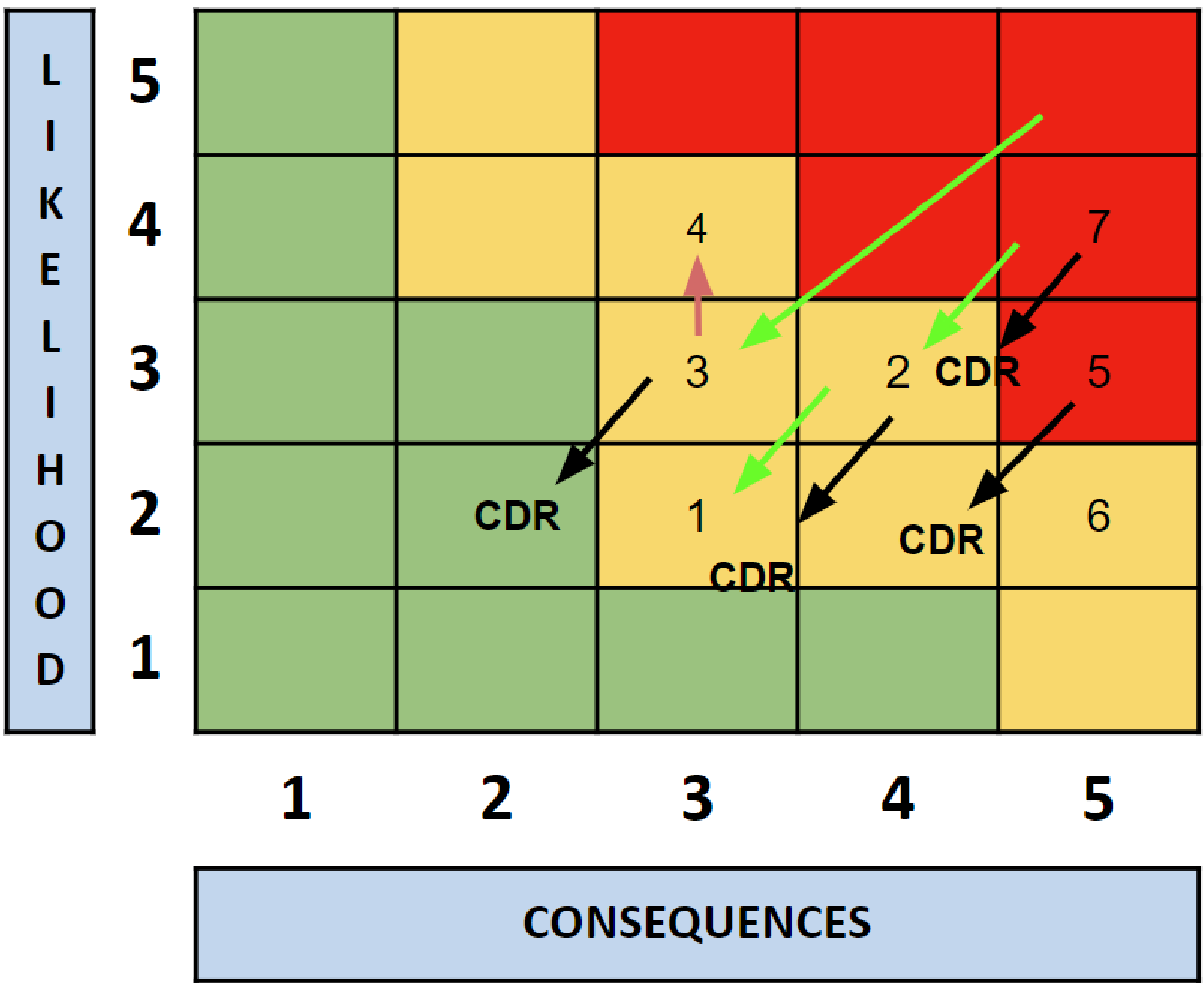
Table 8-2-1 Risk Management

ID	Summary	L	C	Trend	Approach	Risk Statement	Status
1	Environment	2	3	→	R	Charged particles and magnetic interference in Venusian orbit causing failure of subsystems	Active
2	Propulsion	2	4	↓	W	CubeSat fails to enter desired orbit around Venus	Active
3	Communication	2	2	↓	M	Unable to establish/maintain two-way communication due to receiver/transceiver failure or inconsistent pointing	Active
4	Budget	4	3	→	A	Missions exceeds $25 million cost cap	Active
5	Power	2	4	↓	M	System fails to generate TBD Watts or battery performance experiences premature degradation	Active
6	Schedule	1	4	→	M	On-orbit failure due to schedule slippage	Active
7	COTS Components	3	4	↓	A	COTS components failure	Active

1 = Low Likelihood/Consequence; 5 = High Likelihood/Consequence

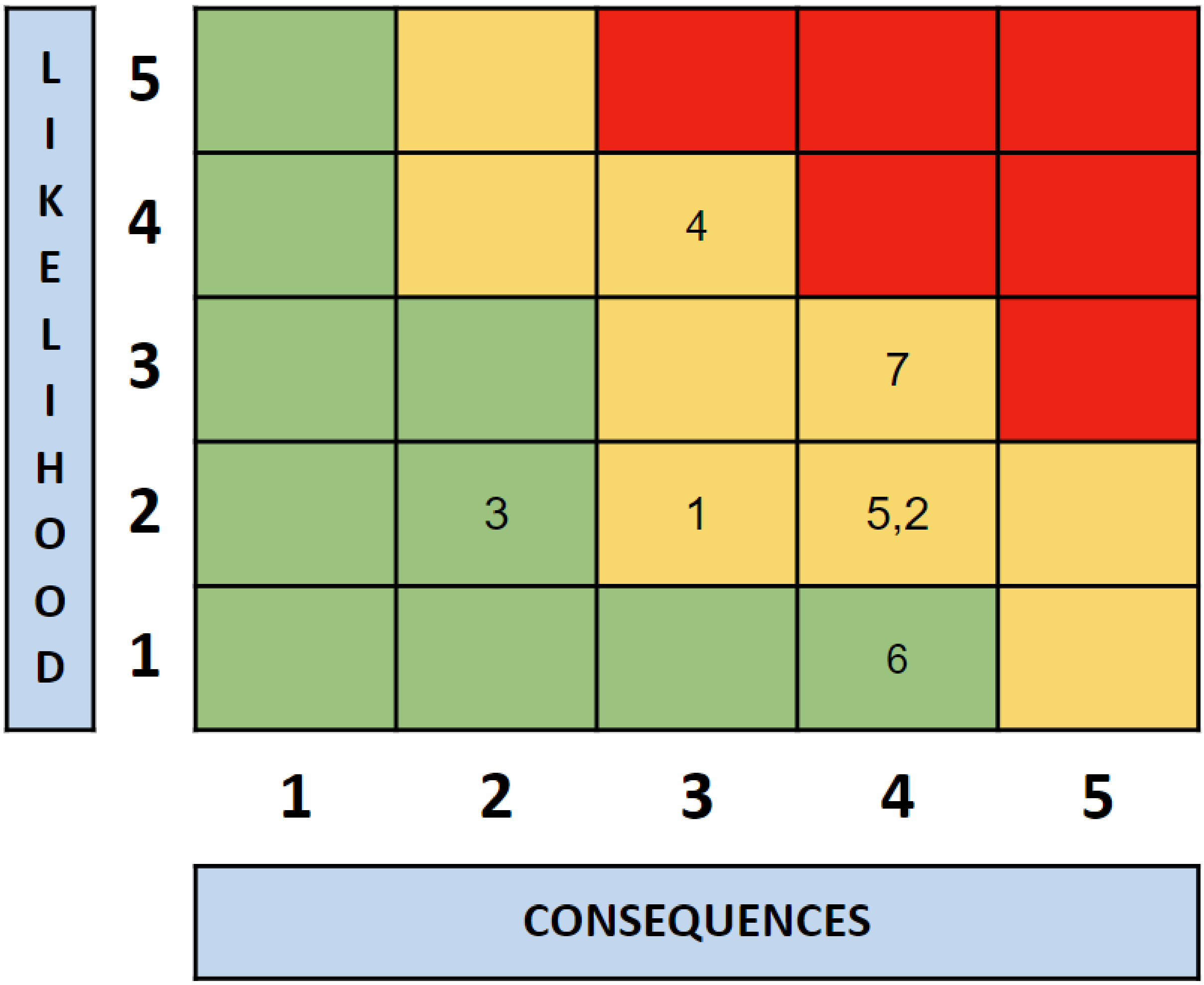
As the VECTOR team has progressed to the point of CDR, it is important to ensure that the risks are being actively managed and mitigated. In table 8-3-2, the PDR risk mitigation plan shows the progression before CDR. By CDR, it was planned that all risks would be out of the high severity red zone.

Figure 8-3-1 PDR Risk Mitigation Matrix



For CDR, an updated risk mitigation matrix was constructed. Shown below in table 8-3-3, it is immediately clear that all risks have been mitigated to the point of medium of low severity. While these risks will continue to be addressed as the VECTOR team moves forward towards SIR, this is a satisfactory standpoint.

Figure 8-3-2: CDR Risk Mitigation Matrix



8.3 Risk Mitigation

Upon review of VECTOR's Risk Management (Table 8-2-1, Figure 8-2-1, and Figure 8-2-2), a mitigation plan will be considered as a guide moving forward to minimize the risks previously addressed. This plan is addressed in Table 8-3-1, the previous risk mitigation from PDR is shown in Figure 8-3-1, and an updated path to minimizing the risk is shown in a risk mitigation matrix in Figure 8-3-2.

In order to combat the harsh environment VECTOR will be headed towards, a potential solution that requires further research includes the ability to implement a centralized critical area in the spacecraft that can protect from radiation with various materials. This then ensures less radiation cycling of critical flight components such as the OBC. A multitude of testing will need to be done to ensure this critical area will work and optimization will be required to meet mass requirements.

In order to limit the propulsion risk, testing will need to be completed to ensure the correct delta-V will be able to be reached with the mass of VECTOR.

Communication, as noted, is the most critical risk, and will need to be purchased using flight-demonstrated components, potentially using the most advanced research into Cubesat communications. Additionally, VECTOR should demonstrate communication links during ground testing, and have redundant downlink capabilities. The selection of widely used IRIS v2 radios will reinforce the communication system and mitigate risks.

Accepting the risk of VECTOR's budget, the team will need to minimize some costs associated with development by getting COTS components, simplifying the scientific payload and development costs, as well as follow the schedule to ensure no over-cost are added.

To mitigate power risks, testing will need to be completed to ensure charge and discharge capabilities are met. This is fairly standard, but specifically, with the power cycling required with radiation mitigations, this will be vital for mission success.

Continuing, in order to mitigate the risk of schedule overflow, it will be critical to meet key deadlines for SIR, FRR, and any other internal deadlines. Additionally, maintaining stock components will play a huge role as COTS are variable and repeatedly out of stock. Working with the suppliers to limit costs, and maximize supply will be key to success.

Finally, COTS components should be subjected to further testing to ensure they are able to meet the harsh environment requirements. This testing will inevitably raise the TRL of each of the components and a collaboration between the suppliers could lead to some discounts or special arrangements.

Table 8-3-1 Risk Mitigation Plan

ID: Summary	1: Environment	2: Propulsion	3: Communication	4: Budget	5: Power	6: Schedule	7: COTS Components
Mitigation Plan	Centralize mission-critical components with mass and shielding material coverage	Propulsion module will be purchased from a reputable supplier	Ensure quality of communications components	Acquire standardized materials (COTS)	Power system charge and discharge testing	Set the purpose and vision of the mission early-on to ensure success	Subject COTS components to rigorous testing
	Optimize central “critical” flight stack with research, analysis, and testing; power cycle “non-critical” components	Unit will be subjected to V&V tests	Perform a communication link testing with the ground station	Simplify CubeSat and payload		Stock spare components	
	Unit will be subjected to V&V tests		Redundant downlink capabilities	Follow NASA Project Life-Cycle to mitigate schedule slip			

9. Schedule and Future

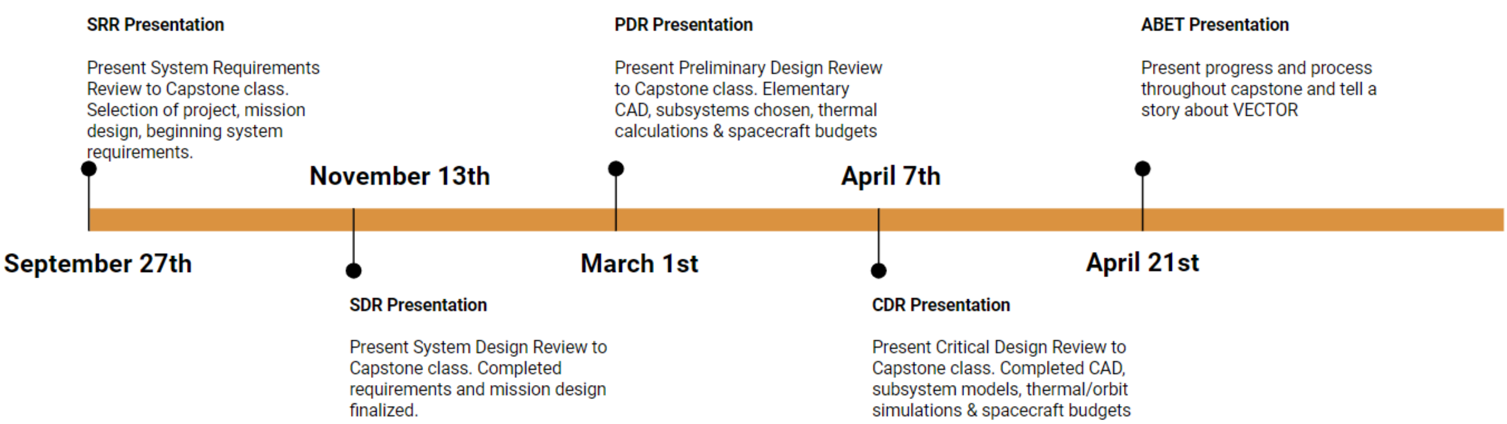
9.1 Description

VECTOR began as an idea, but through the team's hard work and dedication to engineering, became a comprehensive interplanetary CubeSat mission. As of April 2023, VECTOR has successfully completed the Critical Design Review. Therefore, it is important to look back upon all that has previously been accomplished, what is currently being presented, and finally what is expected in the future.

9.2 How We Got Here

VECTOR was first brainstormed in September of 2022, in the Fall semester of the Arizona State University 2022-2023 school year. Since then, the mission has been presented in the SRR, SDR, PDR, and CDR. At the end of April, the ABET presentation will be conducted and the senior aerospace Capstone of 2022/2023 will be completed. Figure 9-2-1 displays a detailed description of all that has or will occur at each stage of VECTOR's journey.

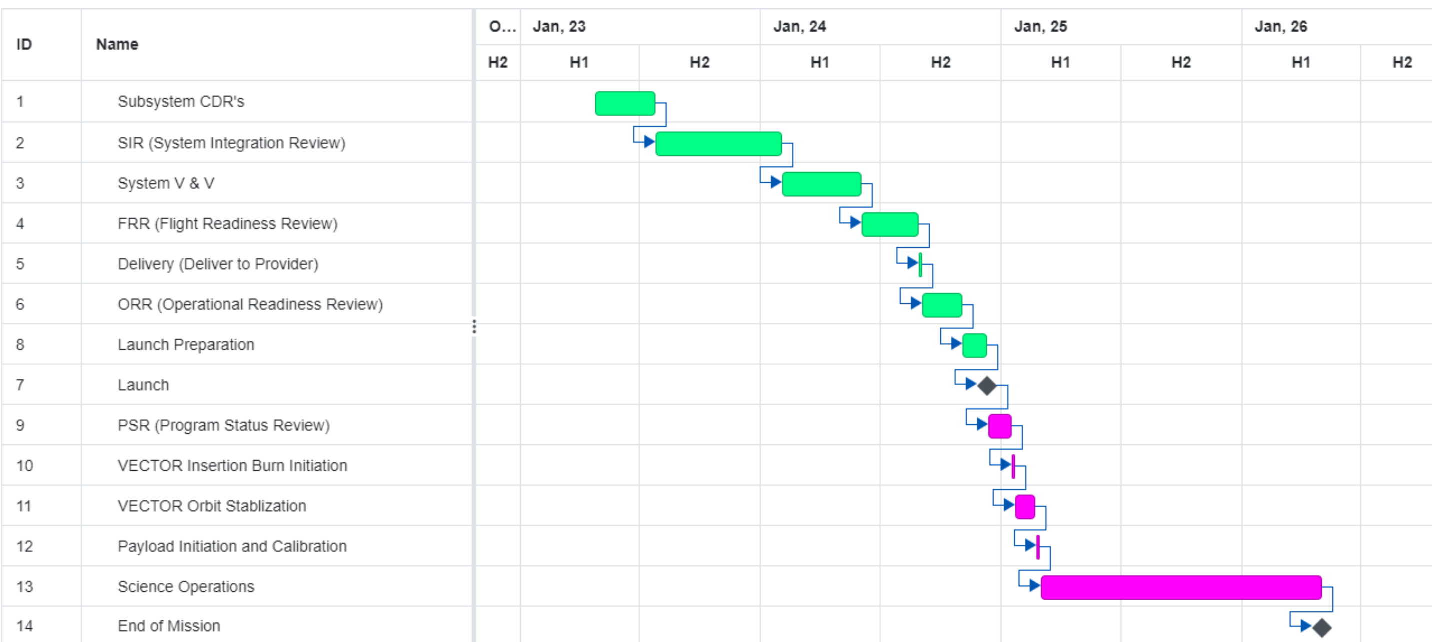
Figure 9-2-1 Capstone Schedule



9.3 Schedule Moving Forward

The team will push VECTOR forward towards future system reviews and eventually to launch until the mission's end of life. Figure 9-3-1 details the entire schedule outlook for the mission. The subsystem CDR's will begin after the completion of the ABET presentation and is scheduled to take place in July of 2023. Then, in 2024, the System Integration Review, System V&V, Flight Readiness Review, Delivery to provider will take place. Finally, in December of 2024, VECTOR will be ready to launch. From there, a Program Status Review will be conducted in the beginning of January of 2025, shortly followed by the VECTOR insertion burn initiation and orbit stabilization around Venus. Once that has been successfully completed, all that is left will be the payload initialization & calibration. From there, VECTOR will maintain science operations until the end of its life in 2026.

Figure 9-3-1 Schedule Outlook



9.4 Future Considerations

As VECTOR moves towards launch and operations, as shown in Figure 9-3-1, the team determined that there are a number of considerations that need to be taken into account before such steps are made. To begin, further simulation of our system is imperative in order to be properly examined in a realistic light. Essentially, examples of such simulations would include orbit simulations for both the transfer and operational orbits of the CubeSat. Furthermore, the team determined that additions need to be made to the thermal model as well as making a simulation on a thermal desktop for both transfer and operational orbits. Accomplishing these aforementioned steps will resolve any remaining questions concerning VECTOR's thermal and radiation components. The team also acknowledged that a Critical Design Review for both the CubeSat's mounting brackets as well as its subsystem must be conducted along with a modal analysis on the entire system in order to strengthen the mission design. Finally, all components need to be purchased so that flatsat testing can commence before investigations can begin into the mission's software and cabling architecture. Ensuring that all future considerations are started early will be key to a successful delivery of the team's CubeSat to our launch provider.

9.5 VECTOR's Team

Our team is committed to ethical exploration of space. We are composed by the following members in their roles:

Chandler Hutchens: Electrical Systems Lead

Jackson Manship: Computing Systems Lead

Tyson Hill: Thermal Systems Lead

Elysian Hudson: Scientific Systems Lead

Kelly Senanayake: Propulsion Systems Lead

Jacob Simonson: Control Systems Lead

Our team has met continuously throughout the design process to work on this project, keeping regular meeting notes and submitting meeting reports.

10. Conclusion

10.1 Summary

This report provides an in-depth analysis of Team 7's VECTOR mission and CDR presentation. In summary, VECTOR is a 12U CubeSat ridesharing mission that will be using low-cost COTS components to perform an insertion burn into a Venusian orbit. Furthermore, VECTOR will have a neuromorphic camera to detect lightning from Venus's atmosphere to collect valuable scientific data on the weather patterns and thus expand the scope of obtainable interplanetary research.

11. Appendix

11.1 Standards

Table A.1 Standards Chart

Document ID	Standard	Verification
Cal Poly CubeSat Design Specification Rev. 14.1	System shall be compliant with the 12U size standard	Inspection
Cal Poly CubeSat Design Specification Rev. 14.1	System shall be less than 24 kg as per 12U standard	Inspection
Cal Poly CubeSat Design Specification Rev. 14.1	System shall have a center of gravity which is compliant with 12U standard	Inspection
NASA-RP-1124-Rev-4	System shall satisfy NASA low-outgassing criteria	Testing
NPR 8715.6	System shall limit orbital debris	Demonstration
AFSPCMAN 91-710, Volume 3	System shall have hazardous materials conform	Inspection
AFSPCMAN 91-710 Volume 3	System shall have propulsion be designed, integrated, and tested	Inspection

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