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Design and Analysis of a SmallSat as a Communication Relay for Venus Atmospheric Probes

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Abstract

This paper discusses a conceptual mission to Venus which utilizes the Lofted Environmental and Atmospheric Venus Sensors (LEAVES) to investigate the sulfur cycle and the unknown compound that absorbs near-ultraviolet light in the atmosphere. The mission uses two separate spacecraft coupled at launch, an autonomous bus, Demeter, carrying 144 LEAVES probes and a communications orbiter, Persephone, to relay data from the LEAVES to Earth. The LEAVES are estimated to be at a Technology Readiness Level (TRL) of 3, while the spacecraft consists of parts at TRL 9. The unique launch mechanism for the LEAVES is at a TRL of 1-2, and the mission meets the Concept Maturity Level (CML) requirements for a CML 4 classification.

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We would like to acknowledge our project advisor, Professor Ye Lu, for his immense support and advice over the past few months. We would also like to thank Jeffrey Balcerski and Maciej Zborowski for meeting with us and letting us use their LEAVES for this project. Finally, we would like to thank the team from Purdue University that developed the MATLAB code we used and edited for the LEAVES descent: Professor James Longuski, Sarag Saikia, and Eiji Shibata.

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1 Introduction

This project details the design of a small satellite, or SmallSat, that will deploy 144 Lofted Environmental and Atmospheric Venus Sensors (LEAVES) into Venus's atmosphere [1] along the day-night line. The LEAVES are grouped by eight and deployed at every 20-deg apart in latitude. The LEAVES are lightweight probes designed to investigate the atmospheric composition of Venus. The probes will be deployed by a spacecraft in orbit and enter the atmosphere at orbital velocity, then slow down rapidly and begin slowly falling through the atmosphere. The LEAVES are expendable probes not expected to survive long enough to reach the surface of the planet and offer an alternative method of studying Venus. This project's goal is to design a small spacecraft that can deploy the LEAVES on Venus, communicate with them during their science phase, and relay the collected data to Earth, all while ensuring the LEAVES are able to collect enough data towards the science goal of the mission.

2 Science Overview

This mission aims to study Venus, the closest planet to Earth. Venus is often referred to as Earth's twin, as the planets are very similar in size and shape, and both are within the habitable zone around the Sun. While Earth can support a wide variety of life and sustain flourishing ecosystems, however, Venus is a hot, inhospitable, barren landscape. Surface temperatures can reach 464 ºC and the planet has a thick, dense atmosphere that rains sulfuric acid [2]. Because of the planet's proximity to Earth, however, it is a primary target of scientific interest and research in the solar system.

2.1 Past Missions

The first mission to Venus and the first successful planetary science mission was Mariner 2, launched in August 1962 [3]. Mariner 2 performed a flyby of Venus, taking measurements of Venus's atmosphere. The probe found that Venus has dense clouds from 58-80 km altitude and little temperature difference at the cloud tops between the day and night sides of the planet. The probe also found no magnetic field around the planet because, as future missions would discover, Venus does not have a molten iron core and, as a result, does not have a magnetic field like Earth does [3]. Instead, the planet creates an induced magnetic field due to the movement of the atmosphere and the interactions between the upper layers and the solar wind. Mariner's older sibling, Mariner 5, also performed a flyby of Venus in 1967. This spacecraft got closer to the planet and took readings of the atmosphere, radio refractivity, and UV emissions, and measured the magnetic field [4].

The Pioneer Venus mission consisted of two spacecraft launched a few months apart in 1978. Pioneer Venus 1, launched first, was equipped with radar to image parts of the planet while also measuring the magnetic field of Venus [5]. It discovered that Venus was much more spherical than Earth and, while generally smoother, found peaks and canyons taller and deeper than any found on Earth. It was also discovered that Venus's magnetic field only extends a short distance into space. The second spacecraft, Pioneer Venus 2, consisted of four probes and a main bus [6]. There were three small probes and one large probe, each designed to enter the atmosphere and take readings of atmospheric concentration, pressure, and temperature. Against expectations, two of the smaller probes survived impact with the surface, and one continued to transmit data for 67 minutes.

The Magellan mission, launched in 1989, mapped the surface of Venus [7]. Using imaging radar, the spacecraft mapped 98% of Venus's surface in high detail. During the second half of the mission, Magellan mapped the planet's gravitational field in a lower orbit. From this, scientists learned that the surface of the planet had been resurfaced about 500 million years ago, recently in a geological timescale. The mapping of the gravitational field also provided evidence that the planet did not have an asthenosphere, a thin layer between Earth's crust and mantle, and because of this the planet has no tectonic activity [7].

A recent mission to Venus, and one that closely aligns with this mission's science goal, is Akatsuki, launched in 2010 by the Japanese space agency, JAXA [8]. Akatsuki is studying the global circulation and wind patterns of Venus with infrared and ultraviolet light. Scientists are already using the information from the mission to answer some of the unknowns about the planet.

2.2 Planned Mission to Venus

There are several planned missions to Venus within the next decade which seek to expand science's understanding of the planet. One of these missions, the Venus Emissivity, Radio Science, InSAR, Topography, and Spectroscopy (VERITAS) mission, is set to launch in 2031 and aims to study the geology and geography of Venus [9]. The orbiter is equipped with an array of sensors and cameras to take the most precise measurements of Venus's topography, surface composition, volcanism, and interior behavior to date. VERITAS will map the entirety of Venus's surface and, due to the sensitivity of the instruments, will be able to create the first active deformation map of another planet. The spacecraft will also be equipped with a variety of near-infrared sensors, enabling the spacecraft to determine the composition of the surface and search for outgassing from volcanoes. Using the spacecraft's connection to the Deep Space Network (DSN), scientists will be able to piece together a map of Venus's gravitational field three times more accurate than the one created by Magellan. Using this connection, scientists can create a model of the interior of the planet by measuring disturbances in the gravitational field.

NASA's Deep Atmosphere Venus Investigation of Noble Gases, Chemistry, and Imaging (DAVINCI) mission is set to launch in the late 2020's [10]. DAVINCI aims to gather data on the chemistry and composition of Venus's atmosphere during two gravity-assisted flybys, then plans to release an atmospheric probe into the atmosphere. The probe will investigate the same parameters as the DAVINCI orbiter in the Alpha Regio region of Venus. During the probe's hour-long descent, it will take thousands of measurements. While the probe is not expected to survive impact on the surface, it can theoretically continue broadcasting for 17 minutes, providing crucial data about the chemistry at the surface of Venus.

2.3 Unknowns of Venus

Despite Venus being a primary target for research over the past 60 years, scientists still lack crucial information about the planet's history, atmosphere, and composition. Much of this is due to the planet's nature. Missions to Venus have primarily been orbiters, using radar, infrared, and ultraviolet imaging to get information about the surface and atmosphere. Rovers and landers are nearly impossible to operate given the immense heat and pressure at the surface, and while probes have been launched into the atmosphere, they have short lifespans and are limited in the equipment they can carry because of entry conditions.

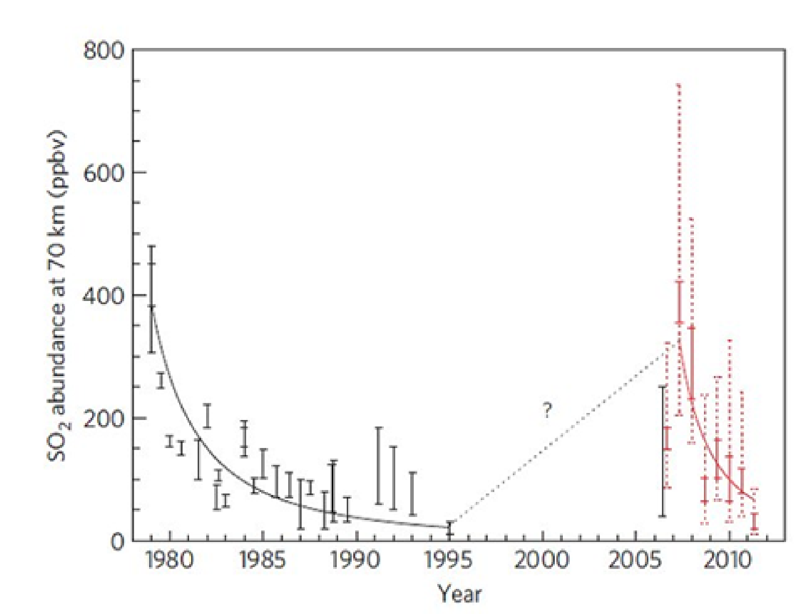

Fig. 1. Sulfur dioxide abundance at 70-km altitude from 1978 to 2011 [13].

Because of the challenges and limitations faced by research missions, Venus still holds many mysteries. For example, the Magellan mission, which mapped the surface of the planet in the 1990's, discovered that the surface of the planet is new, in geological terms. Evidence points to a resurfacing of the planet around 500 million years ago, likely caused by massive volcanic eruptions [7]. Because of this resurfacing, scientists do not know the early history of Venus, nor do they understand how the greenhouse effect started on the planet [11]. The leading hypothesis proposes that the harsh climate was created by volcanic eruptions, but when that happened is unknown. Additionally, scientists don't have a wind model for Venus. While there have been studies done on cloud top wind patterns, and the JAXA Akatsuki mission is currently studying global circulation patterns, data on how the wind and circulation patterns vary with altitude is still largely unknown [8,12]. While atmospheric composition is generally known, with the atmosphere consisting of 96.5% carbon dioxide and 3.5% nitrogen, along with other trace gases, there are compounds in the upper atmosphere that absorb the ultraviolet light that hits the planet [2,11]. Some compounds, such as sulfur dioxide (SO_2), have been identified as absorbers of ultraviolet light shorter than 320 nanometers, but the compounds that absorb the near-ultraviolet light are unknown. Additionally, there is a decade long SO_2 cycle on the planet where sulfur dioxide levels rise and fall, shown in Fig. 1. The cause of this cycle is unknown, and the presence of this pattern points to Venus' atmosphere being out of chemical equilibrium [13].

2.4 Science Goal of the Mission

This mission's goal is to make progress towards discovering what compound or compounds absorb the near-ultraviolet light in the upper atmosphere. Experts believe a key component of the formation of these compounds is the sulfur dioxide cycle. Current research indicates that the UV-absorbing compound is sulfur-based, either a species of S_2O_2 [14,15] or S_2O [16]. Either compound, S_2O_2 or S_2O, can be formed in chemical reactions with sulfur dioxide as a reagent. Because of this, understanding how sulfur dioxide interacts and changes within the atmosphere could give insight into the potential UV-absorbing compounds, as well as information on what the sources and sinks of the compound are on Venus.

The mission plans to deploy 144 Lofted Environmental and Atmospheric Venus Sensors (LEAVES), shown in Fig. 2, around the circumference of the planet along the day-night line. The LEAVES are lightweight probes designed to slowly descend from 100 to 30 km above the planet's surface, collecting sulfur dioxide and carbon monoxide concentration data and measuring temperature, pressure, and the 9-axis movement of the probes [1]. For a more in-depth overview of the LEAVES, see the Entry and Descent section. The hypothesis is that there is a catalyst of some sort created by the Sun's light that causes a reaction to occur that has sulfur dioxide as a reagent. By dropping the probes on the day-night line, the team believes the dawn side should see a minimum concentration of sulfur dioxide, while the dusk side should see a maximum concentration. LEAVES data drawn closer to the poles, where the circulation of the atmosphere from the day side to the night, should happen much quicker and could be used as a control for data gathered closer to the equator. If the predicted difference is shown in the data collected by the probes, then the day-night cycle could play a role in the sulfur dioxide cycle. If the hypothesis is not represented in the data, then the probes will still collect ample data on sulfur dioxide concentration. Additionally, the probes will be able to collect more accurate temperature and pressure data, and their movement data can be used to create models of wind and circulation patterns within the atmosphere.

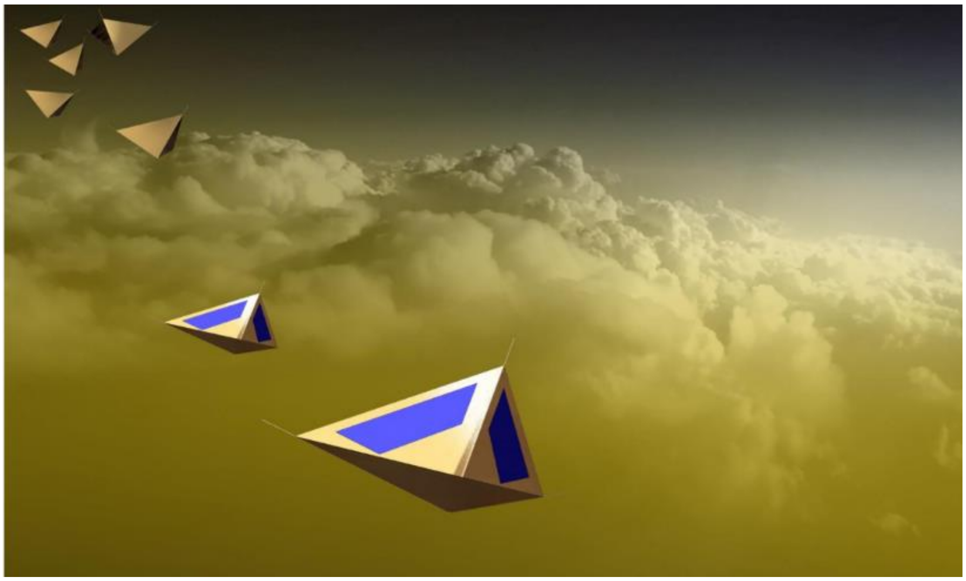

Fig. 2. Concept art of the LEAVES descending through the Venusian atmosphere [1].

This mission has the potential to provide incredible amounts of data that could further science's understanding of Venus. The concept is similar to the DAVINCI mission, taking measurements of compounds, wind, pressure, and temperature in the atmosphere. While DAVINCI's probe will likely provide more accurate data using its tunable laser spectrometer and mass spectrometer, the probe is limited to just the Alpha Regio region [10]. The LEAVES will be deployed around the entire circumference of Venus, covering vastly more space than the DAVINCI probe. Additionally, because of the number of LEAVES, the range of data collected by the mission will be much greater than the DAVINCI probe.

The team elected to not include any measurement or imaging devices on the orbiter to save mass, power, and volume. Any imaging or measurement device on the orbiter would not have high enough resolution or accuracy to provide data that doesn't already exist, and Akatsuki, VERITAS, and the DAVINCI orbiter, are able to image and measure Venus from orbit better than anything that could be fitted on the orbiter.

3 Mission Design

To deliver 144 probes to Venus, communicate and collect information from the probes as they descend, and send data back to Earth, a feasible mission trajectory design is required. The overall mission design is crucial for achieving the science objectives described above. To design the mission trajectory, ANSYS Systems Tool Kit (STK), specifically Astrogator, was used. The mission can be divided into four phases: Earth operations, interplanetary transit, Venus orbit operations, and finally, the LEAVES deployment or science phase. The third phase was subject to the most scrutiny. The location and motion of the probes and communications systems prior to deployment is important to ensure that data are collected and relayed to Earth, the scientific objectives are achieved, and probe failure is prevented or mitigated. Based on these requirements, the primary mission design objectives are: determine and minimize the ΔV requirement, achieve a low orbit around Venus to deploy the LEAVES, and maintain an orbit through the science phase that maximizes the data transfer between the LEAVES probes and Earth.

The series of maneuvers required to achieve these objectives are detailed in Table 1. For this mission design, the Earth escape duration and propellant used for heliocentric orbit insertion (HOI) are not considered because the maneuver is assumed to be impulsive and is not part of the primary design focus. The ΔV required for the Earth escape could be achieved with a particular launch vehicle, bypassing the parking orbit that would otherwise be required.

Table 1. Mission maneuver summary. All values are from STK.

	Date (mm/dd/yyyy)	Duration (s)	ΔV (m/s)	Initial Wet Mass (kg)	Fuel Used (kg)	Final Wet Mass (kg)
SmallSat						
Heliocentric Orbit Insertion (HOI)	05/30/2031	0 (N/A)	3684	N/A	N/A	254.2
Venus Orbit Insertion (VOI)	09/19/2031	533.4	1145	254.2	76.88	175.1
Initiate Aerobrake	09/20/2031	6.400	16.58	175.1	0.9220	174.2
TOTAL		539.8	4846		77.80	
Orbiter						
Circularize	10/20/2031	25.99	435.8	29.14	5.040	24.10

LEAVES Bus						
Exit Aerobrake	10/20/2031	9.469	30.05	147.21	1.365	145.845
Circularize 1	10/20/2031	20.00	64.41	145.845	2.883	142.962
Circularize 2	10/20/2031	20.00	65.75	142.962	2.883	140.079
Circularize 3	10/20/2031	20.00	67.13	140.079	2.883	137.196
Circularize 4	10/20/2031	20.00	68.58	137.196	2.883	134.313
Circularize 5	10/20/2031	20.00	70.10	134.313	2.883	131.43
Circularize 6	10/20/2031	20.00	71.68	131.43	2.883	128.547
Circularize 7	10/20/2031	4.074	14.80	128.547	0.587	127.96
TOTAL	N/A	133.5	452.5		19.25	

From Table 1 and the dry mass of the primary spacecraft (Table 8), the final mass of the LEAVES bus according to STK is 8.62 kg greater than that calculated:

$$m_{f,STK} - m_{f,analytical} = 127.96\ kg - 117.34\ kg - (2\ kg\ ACS\ fuel) = 8.62\ kg$$

This extra propellant does not include the 2 kg of fuel dedicated to the attitude control systems (ACS) thrusters in the mass budget (Table 8). Similarly, the orbiter has 0.12 kg of extra fuel, since

$$m_{p,STK} - m_{p,analytical} = 6.16\ kg - 5.04\ kg - (1\ kg\ ACS\ fuel) = 0.12\ kg$$

This mission design uses conservative propellant requirements; a contingency factor of 10% is included on all maneuvers and propellant is added exclusively for the ACS of the LEAVES bus and orbiter, 2 kg and 1 kg, respectively. From Section 5.6, this is more than enough fuel for all the planned attitude maneuvers. More details on the mass budget and associated calculations are included in Section 5.2 and Appendix A.1.

3.1 Phase I: Earth Operations

For this mission, a joint launch and transit with the NASA DAVINCI mission was considered [17]. Due to a lack of available data for this future mission, however, an independent launch and interplanetary transfer were chosen instead. For this mission design, the launch site was not considered. Design considerations instead begin from a parking orbit around Earth from which the interplanetary transfer can be commenced with a single maneuver in the direction of motion. This parking orbit is circular and at approximately 400 km altitude, shown in green in Fig. 3. The parameters of the parking orbit, including inclination and longitude of the ascending node, are determined based on the parameters of the Earth-to-Venus transfer and are shown in Table 2.

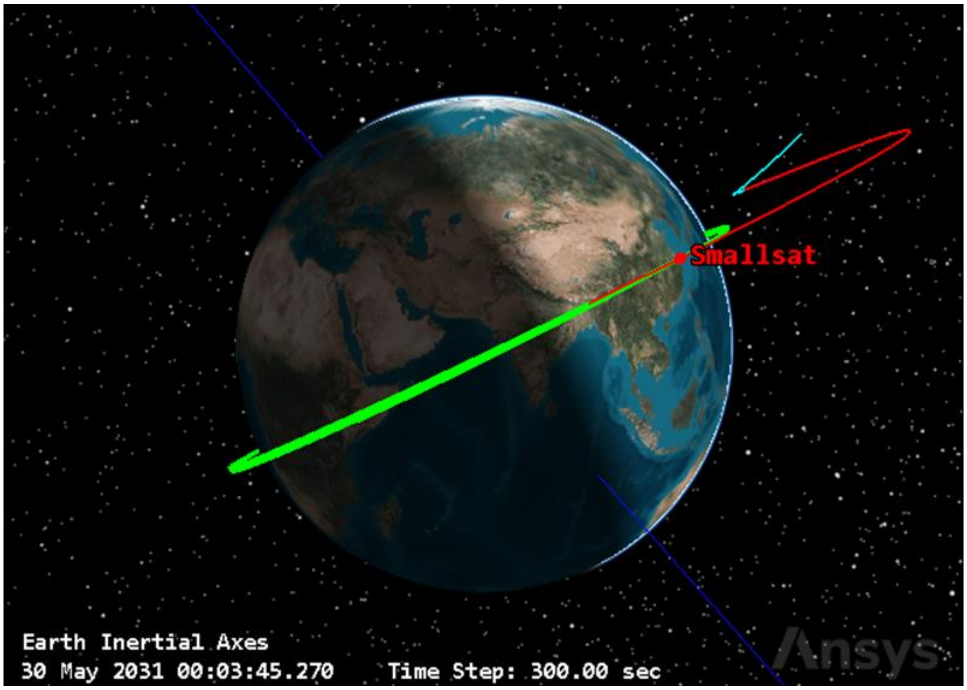

Fig. 3. Trajectory of SmallSat during Earth operations. In red is the escape trajectory of the satellite and in light blue the heliocentric orbit following Earth escape.

Table 2. Earth parking orbit parameters.

Parameter	**Value**
Semi-major Axis (km)	6776
Eccentricity	0.0009829
Inclination (°)	29.06
Right Ascension of the Ascending Node (°)	14.40
True Anomaly (°)	290.0

3.2 Phase II: Interplanetary Transit

To choose an optimal transfer for the mission, the NASA interplanetary trajectory database was used [18]. The chosen trajectory starts at Earth on 30 May 2031 and ends at Venus on 19 September 2031, taking 112 days to complete. The required target vector outgoing asymptote coordinates from Earth are shown in Table 3. This trajectory is validated using STK Astrogator to numerically solve for a trajectory that intersects Venus, using parameters from NASA's interplanetary database as initial guesses. The calculated coordinates are slightly different from those given in the NASA trajectory browser [18] for the same timeframe. There are a few possible reasons for this discrepancy. First, the NASA trajectory database assumes a parking orbit altitude of 200 km, half the altitude of the parking orbit simulated by STK. Next, B-plane targeting (

Table 4) is used in STK to achieve a trajectory from which a maneuver can accomplish an orbital insertion around Venus with a minimal ΔV cost, whereas the NASA database targets a Venus flyby orbit with no specified periapsis. Finally, the values from the trajectory browser are calculated using a different model and set of assumptions than STK. Because of these differences, the computed Earth-escape ΔV, 3.68 km/s, is slightly less than that from the trajectory browser, 3.73 km/s. The interplanetary trajectory is shown in Figure 4.

Table 3. Target vector outgoing asymptote coordinates (with respect to Earth) for insertion into interplanetary transfer to Venus.

Parameter	**Value**
Radius of Periapsis (km)	6778
Characteristic Energy C_3 (km^2/s^2)	12.09
Right Ascension of Outgoing Asymptote (°)	124.3
Declination of Outgoing Asymptote (°)	-12.35
Velocity Azimuth at Periapsis (°)	61.50
True Anomaly (°)	4.612e-14
Earth Escape/Transfer Orbit Insertion ΔV (km/s)	3684

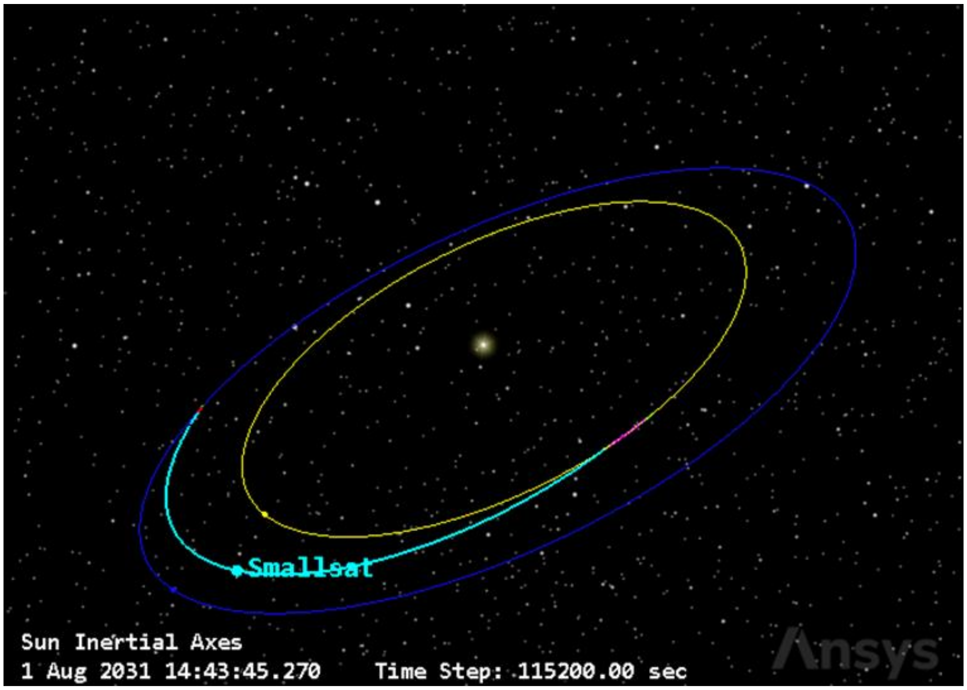

Fig. 4. Interplanetary trajectory (light blue) of SmallSat from Earth (blue) to Venus (Yellow).

Table 4. Venus b-plane target coordinates.

Parameter	Value
BDotR (km)	17000
BDotT (km)	0

3.3 Phase III: Venus Operations

After the interplanetary transfer of the SmallSat to Venus, several mission scenarios were considered and iterated in STK. For all these variations, aerobraking is used to reduce the orbit to an altitude from which the LEAVES probes can be safely deployed into the atmosphere. Aerobraking is a maneuver in which a satellite enters an orbit with a periapsis close to an object's surface, within the atmosphere. As the spacecraft travels through the periapsis, its velocity is reduced due to drag, lowering the apoapsis on each pass. The drag induced by the atmosphere is significant for planets with thick atmospheres such as Venus and can reduce the mission ΔV cost by reducing the propellant needed for orbit changes. The aerobraking duration and the number of orbital passes can be changed by varying the altitude of periapsis, but this is limited by the material and structure of the spacecraft.

A small variation in LEAVES deployment altitude drastically affects the time duration between deployment and atmospheric entry, which is why a low deployment altitude is important for minimizing the required satellite lifetime and mission duration. In addition, the velocity at which the LEAVES are travelling at deployment and the eccentricity of the deployment orbit affect the flight path angle of the deployed LEAVES probes which can cause them to "skip" out of the atmosphere or not enter the atmosphere quickly enough, resulting in an excessive delay between the LEAVES deployment and the beginning of the science phase.

For all Venus operations scenarios, the first maneuver is the Venus orbit insertion, a burn in the direction opposite to the velocity. This maneuver gives the spacecraft an elliptical orbit. It is executed as close to Venus as possible to minimize the ΔV requirement, per the Oberth effect. For optimization, the eccentricity is preferably large to reduce the ΔV cost of the insertion. This also decreases the cost of the aerobrake initialization maneuver, which is executed when the spacecraft reaches apoapsis. The maneuver lowers the orbit's periapsis such that the spacecraft is subject to drag effects from the atmosphere, which causes orbital descent and a decrease of velocity. The goal of aerobraking is to lower the satellite as much as possible without a burn to reduce the ΔV required for orbit insertion. Variations and iterations of this general mission plan are discussed below. The duration of the aerobrake changes drastically with a change in the periapsis, so precise altitude selection is crucial. Other limiting factors such as the maximum dynamic pressure and surface temperature must be considered as well.

Figure 5 shows the initial Venus mission trajectory generated in STK, in which the satellite enters a polar orbit upon arrival at Venus. This orbit is lowered with an aerobraking maneuver, shown in purple, to the green circular orbit. This trajectory is from an early design iteration and is not realistic because of the low number of aerobraking passes. Additionally, the final circular orbit is too high for LEAVES deployment within an allowable period of time.

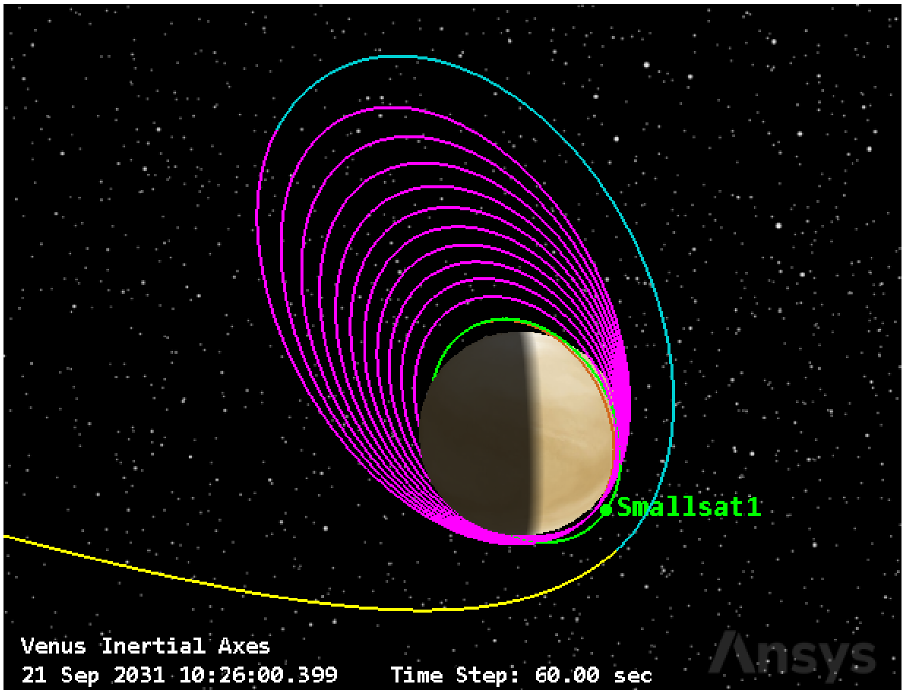

Fig. 5. Initial Venus mission trajectory.

3.3.1 Definitive Venus Trajectory Sequence

The chosen design for the Venus operations sequence takes advantage of the motion of Venus' day-night line as the planet revolves around the sun. Details such as the selected propulsion systems and total wet mass are also considered, since fuel consumption and maneuverability are dictated by the thrust and specific impulse of the selected thruster, and the total wet mass restricts the maneuvers that can be done by the satellite.

First, the satellite enters an elliptical orbit around Venus with an eccentricity of 0.9 and begins the aerobraking maneuver. After aerobraking, the primary satellite separates into two smaller satellites. The first of these is a communications relay satellite, Persephone, which serves as a relay between the LEAVES probes and Earth. The second spacecraft is the LEAVES bus, Demeter, which enters a circular orbit low enough to safely deploy the LEAVES into the atmosphere. The separation allows the two spacecraft to specialize in their roles without making compromises. Persephone can communicate to the LEAVES and be in a high enough orbit to easily communicate with Earth, while Demeter is at a low enough orbit to safely deploy the LEAVES. After deploying the probes, Demeter is obsolete. At the separation point, which occurs at apoapsis, Persephone's orbit is circularized while Demeter continues to decelerate to achieve a periapsis altitude of 235 km. After propagating to this altitude, Demeter performs several maneuvers at each subsequent periapsis to enter and refine a stable, low circular orbit from which the LEAVES probes can be effectively deployed.

Next, Demeter targets a low orbit aligned with the day-night transition of Venus, allowing the bus to deploy the LEAVES and fulfill the science goal of the mission. While in the polar orbit, Demeter is constantly exposed to the sun, which is beneficial for solar power generation, especially since solar intensity is greater at Venus than at Earth. Because the day-night transition is not stationary with respect to Demeter's orbit, the LEAVES have a short window to deploy. The motion of the day-night transition, however, can be exploited to reduce the mission ΔV cost as well as the duration of Demeter's orbit, which requires occasional maintenance maneuvers to remain stable. Approximately 33 days after the spacecraft arrives at Venus, on 22 October 2031, the polar orbit of the spacecraft coincides with the day-night line, or, quantitatively, the beta angle is around -90°. To optimize the mission cost, most of this "waiting period" is spent in the aerobraking maneuver described above, which ends about two days before LEAVES deployment. Longer-duration aerobraking is desirable because the maximum dynamic pressure and thermal loading experienced by Demeter decreases as the duration increases. Figure 6 shows the optimized trajectory with assumed spacecraft parameters, and Table 5 contains the characteristics of the aerobrake trajectory. The final design includes the implementation of finite maneuvers, which are more realistic than impulsive maneuvers for the chosen spacecraft thrusters. When using finite maneuvers in orbital analysis, achieving a circular orbit is more difficult than when using an impulsive assumption, since ΔV must be imparted before and after periapsis and apoapsis, resulting in orbit rotation. This is why a sequence of maneuvers, depicted in Table 1, is necessary for orbit circularization. By performing several small maneuvers, Demeter's orbit can be further circularized, reducing the difference between the apoapsis and periapsis radii. This also optimizes the orbit for LEAVES deployment, as the analysis of their deployment, discussed more in-depth in Section 4. As Demeter's orbit becomes more circular, the LEAVES flight path angle at deployment gets closer to zero, bringing the deployment of the LEAVES closer to ideal conditions and reducing the complexity of the deployment mechanism.

The primary tradeoff of this option is the longer mission duration. This is acceptable because overall mission time is not a constraint for this mission architecture. For other interplanetary missions, this can be an issue because of power requirements, but for this mission design the satellites are never eclipsed by Venus and, being closer to the sun, solar panels are much more effective, decreasing the need for batteries.

The greatest challenge to mission design is the constantly changing angle of the day-night transition with respect to Demeter's orbit. To collect atmospheric data along this line, the LEAVES probes must be deployed quickly, preferably within one to two days after the orbits are finalized. To make deploying 144 probes easier, they are deployed in 18 groups, or bundles, of 8 probes each. With a period of 5,499 seconds

(Table 6), a cluster of probes is deployed every 305.5 seconds, or 5 minutes, allowing Demeter to deploy all the probes in a single orbit. From Section 5.6, this was determined to be enough time for the required attitude adjustment maneuvers to ensure the probes are deployed properly. The orbits of Demeter and Persephone during deployment are shown in Figures 7 and 8.

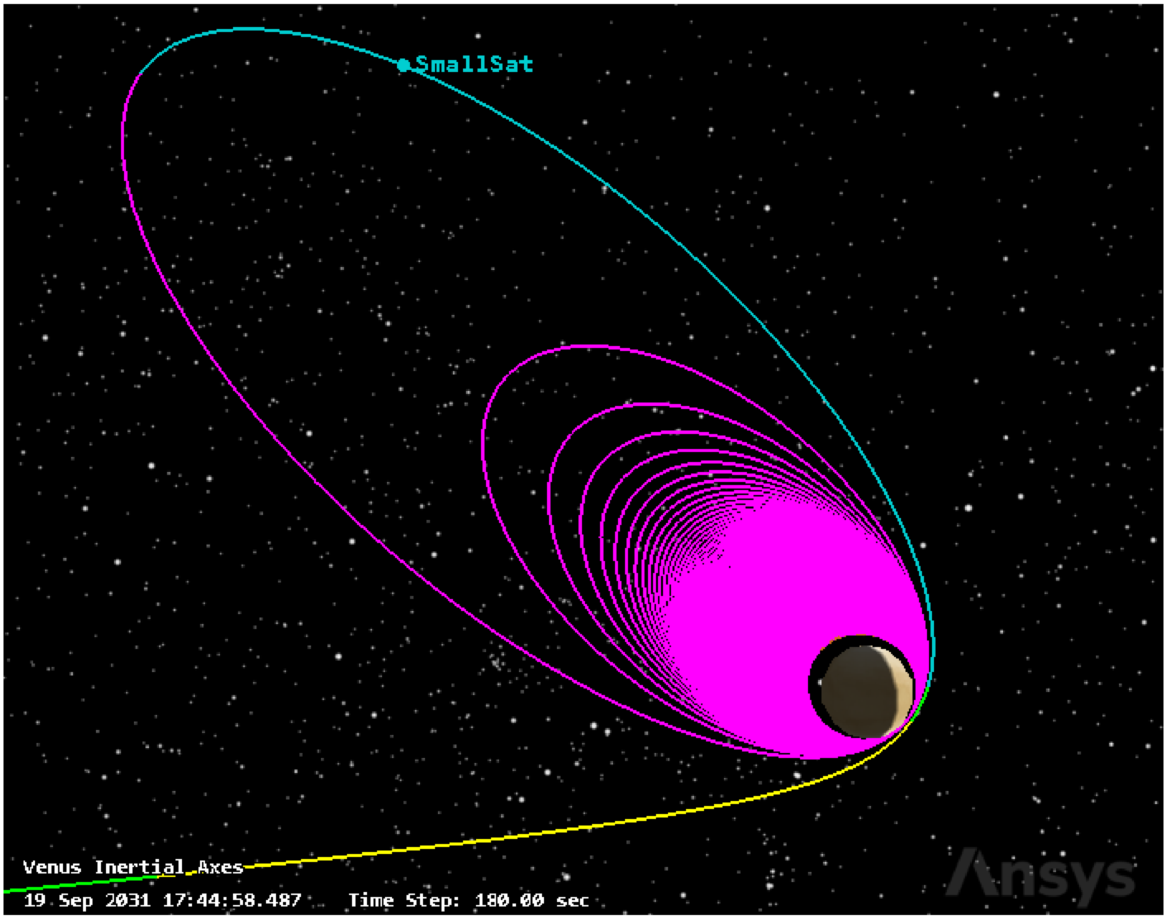

Fig. 6. SmallSat Trajectory during Venus operations phase. The aerobrake trajectory is pink.

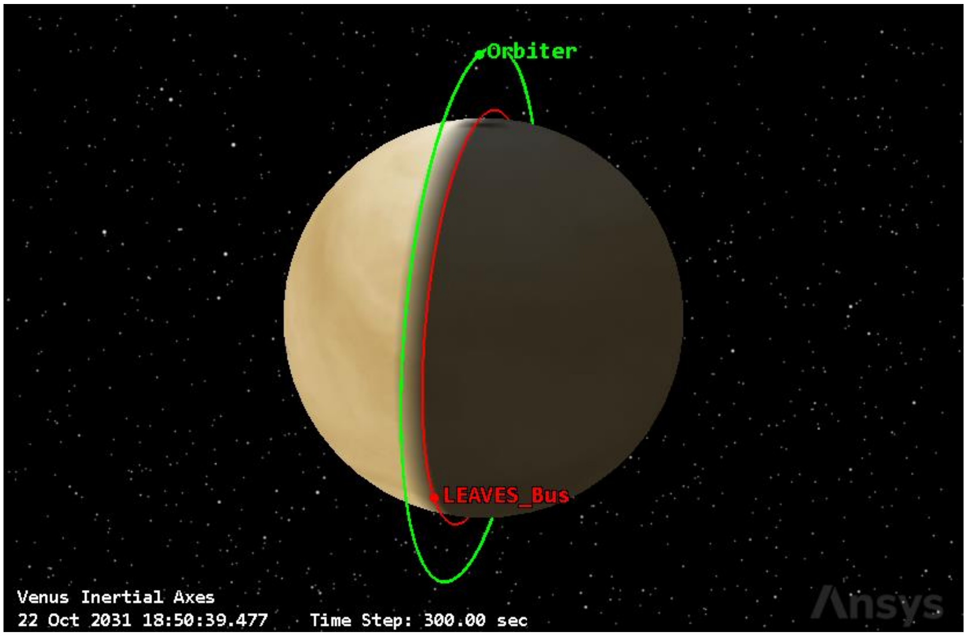

Fig. 7. Orbits of LEAVES bus and communications orbiter during probe deployment.

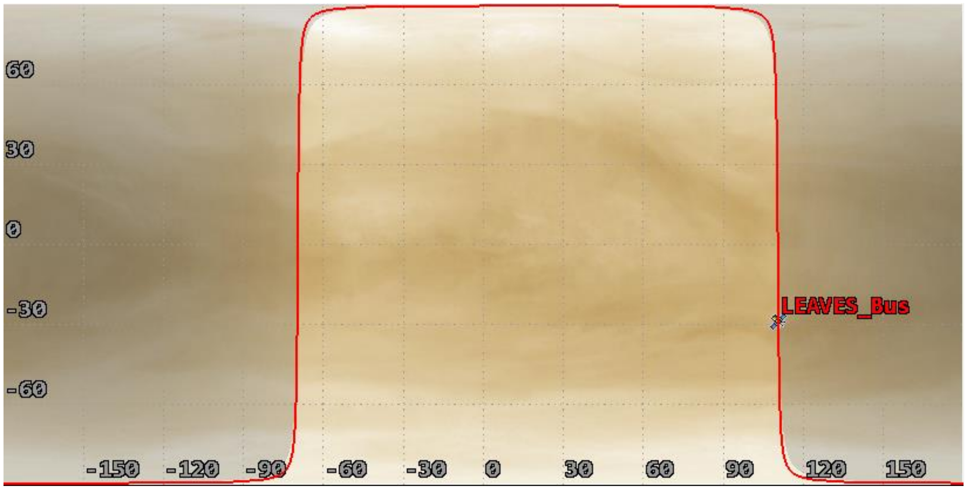

Fig. 8. 2-D view of LEAVES bus day-night transition orbit.

Table 5. Venus aerobrake trajectory characteristics. Spacecraft mass is assumed constant.

Parameter	Value
Altitude of Periapsis [km]	88.93
Spacecraft Mass (assumed constant through aerobrake) [kg]	176.34
Spacecraft Cross-Sectional Area (Drag) [m^2]	0.6257

Duration [days]	29.97
Initial Altitude of Apoapsis [km]	125,500
Final Altitude of Apoapsis [km]	1,986

Table 6. Day-night transition orbit characteristics.

Parameter	**Value**
Eccentricity	0.0006224
Period (s)	5498.8
Altitude of Apoapsis (km)	240.69
Latitude of Apoapsis (°)	-65
Altitude of Periapsis (km)	234.79
Latitude of Periapsis (°)	65
Velocity at Apoapsis (km/s)	7.183
Velocity at Periapsis (km/s)	7.190
Altitude Range (km)	5.90

3.3.2 Alternative I: Polar Orbit Rotation to Day-Night Transition

This scenario is like the final mission design but uses a different strategy to align Demeter's orbit with Venus' day-night transition. This scenario uses an orbit rotation maneuver at the apoapsis of the first elliptical orbit around Venus after orbit insertion, as seen in Fig. 9. This scenario was not chosen because of the large ΔV requirements of the maneuver. Orbit rotations have ΔV requirements similar to inclination changes, and this scenario was not considered feasible. The orbit rotation maneuver could be executed at a higher altitude than that in Fig. 9, perhaps during the initial orbit in light blue, to reduce the ΔV requirement dur to smaller initial and final velocities. However, the ΔV requirement would still be too high to justify.

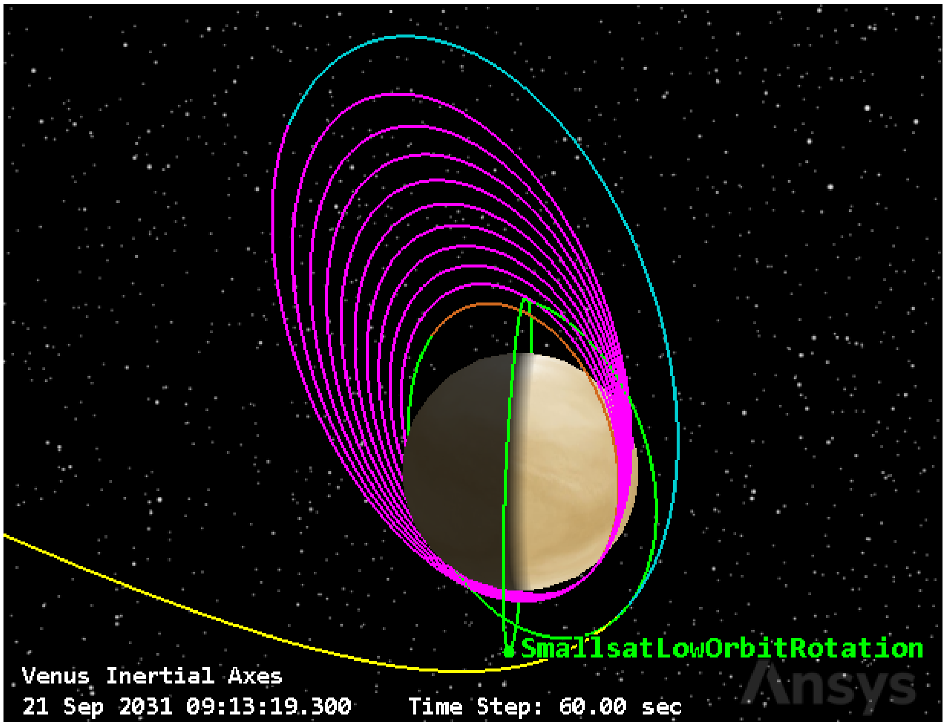

Fig. 9. Sample trajectory of Venus mission trajectory with orbit rotation maneuver after aerobrake to achieve day-night transition orbit.

3.3.3 Alternative II: No Separation of LEAVES Bus and Communications Orbiter

Another consideration for the mission design was whether a satellite separation should be incorporated into the Venus operations sequence. In earlier variations of the mission design, no separation occurred. This design's main benefit was the simplification of the spacecraft architecture. A single spacecraft uses a single propulsion system, saving mass and reducing complexity, and uses less fuel overall. A separable communications orbiter requires an independent propulsion system, by contrast, with propellant exclusively for maneuvering after separation from the LEAVES bus. Several other components, such as solar panels, propellant tanks, and structural supports, need to be present on both spacecraft so that they can operate independently after separation. These extra requirements increase the total wet mass of the primary spacecraft, which must carry the orbiter as additional dry mass until it separates. In addition to increasing the total wet mass, the separation affects the mass moments of inertia of the LEAVES bus, which must be accounted for in the ACS design.

Without a separation, mission requirements such as ΔV are decreased. Some mission objectives are less likely to be achieved, however, such as data collection from all probes. Having a communications orbiter at a high-altitude orbit increases coverage of Venus, which is important for communicating with all the LEAVES as they drift through the atmosphere. A low orbit is more difficult to maintain for a long period of time, so keeping an orbiter in a high orbit decreases the likelihood of total mission failure. This also makes communication with Earth easier and guarantees total sun exposure for solar panels throughout the mission. Having the LEAVES deploy from a high orbit, however, increases their entry speed into the atmosphere to the point where the stress and thermal load they experience exceeds their structural and material limits. Additionally, maneuvers for the separate spacecraft are more feasible because each spacecraft has less mass than the combined satellite.

4 Science Phase

4.1 Leaves Overview

The Lofted Environmental and Atmospheric Venus Sensors (LEAVES) are lightweight, trapezoidal probes designed to descend slowly through the atmosphere. Each probe has a mass of 130 grams and is 1.5 meters long by 0.8 meters tall [1]. The aerodynamic structure consists of three triangular panels made of Kapton®, a thin metalized polyimide film, with carbon fiber rods at the corners for support. At the apex of the tetrahedron is the instrument panel, which holds all the sensors, as well as the power supply and microcontroller. An example of the LEAVES size and instrument panel can be seen in Fig. 10. The LEAVES have batteries that supply 1.55 W of power during communication periods, and 0.17 W during science operations [1]. The batteries power the high and low-pressure sensors, temperature sensor, SO_2 and CO sensors, microprocessor, and 3-axis accelerometer, and the communications device. The SO_2 and CO sensors will be used to measure the atmospheric concentration of the compounds, allowing the mission to draw conclusions about reaction rates across the planet. The pressure sensors and temperature sensor allow for greater science gain from the probes. In addition to providing more accurate measurements of pressure and temperature, the pressure and temperature data will allow scientists to make more informed conclusions on observed reaction rates. The data the LEAVES collect is transmitted to the orbiter via an antenna that spans the top of the probes, with a 180-degree transmission angle, a gain of 3 dBi, and a maximum upload speed of 3.8 kbps. Each probe has 1 MB of program memory and 64 KB of flash memory [1]. The probes can experience a maximum deceleration during their descent of approximately 18.5 g's. This was calculated by linearly extrapolating from the stress analysis done in the LEAVES documentation [1]. The LEAVES are assumed to carry and operate an inertial measuring unit (IMU) that can give the probe location within a 10-km grid. In the LEAVES documentation, this device is stated to exceed the power limit and computational power of the probes, but this report is assuming that, with advances in technology, by the time of launch the technology will be ready to allow the device to be used. This assumption allows the orbiter to not require a tracking radar, as the LEAVES documentation describes, and narrows the accuracy of the LEAVES location from a 25-km grid to the previously mentioned value.

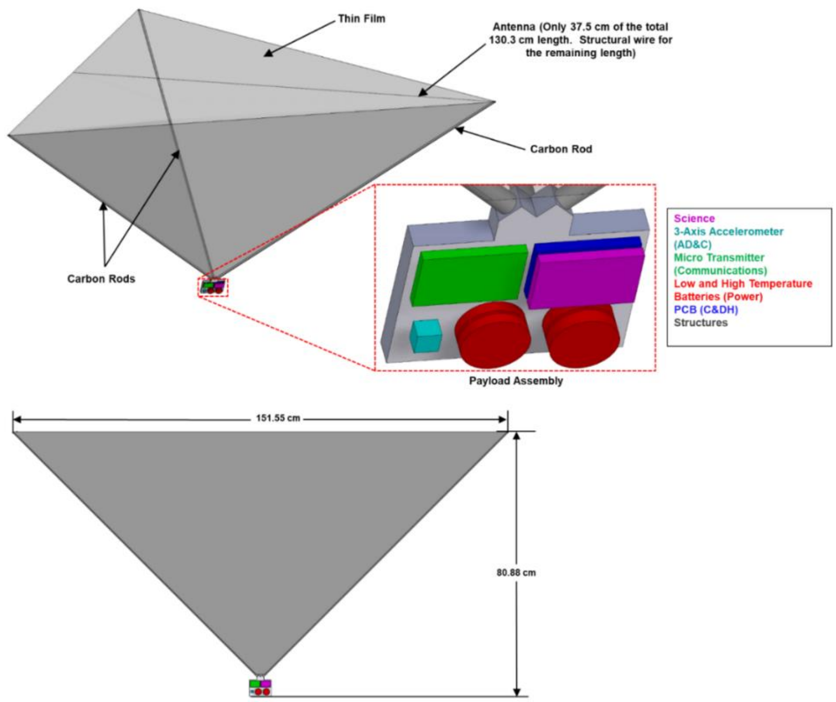

Fig. 10. Instrument panel on the LEAVES and the dimensions of the probes when deployed [1].

4.2 Deploying the LEAVES

The LEAVES will be deployed from orbit eight at a time every 20 degrees latitude around the day-night line for a total of 144 probes. The bus orbits the planet in a nearly circular orbit, with periapsis occurring at 233 km at 7.1907 km/s, and apoapsis at 240 km at 7.1834 km/s. The LEAVES will be deployed in clusters and will free-fall until around 150-km altitude, where a pressure sensor will unfold the LEAVES and they will begin to rapidly slow down.

To deploy the LEAVES from the bus, they will have a ΔV imparted on them in the direction opposite of the bus's velocity to change the orbit of the LEAVES. At the point of deployment, the LEAVES are at the apoapsis of their new orbit, and the ΔV imparted on them changes their periapsis from the periapsis of the bus to be inside the planet's atmosphere. Once the LEAVES make contact with thicker atmosphere in the 110-120 km range, they will start to experience drag effects and will slow down drastically. After 80 km the probes are effectively falling straight down, as seen in Fig. 11.

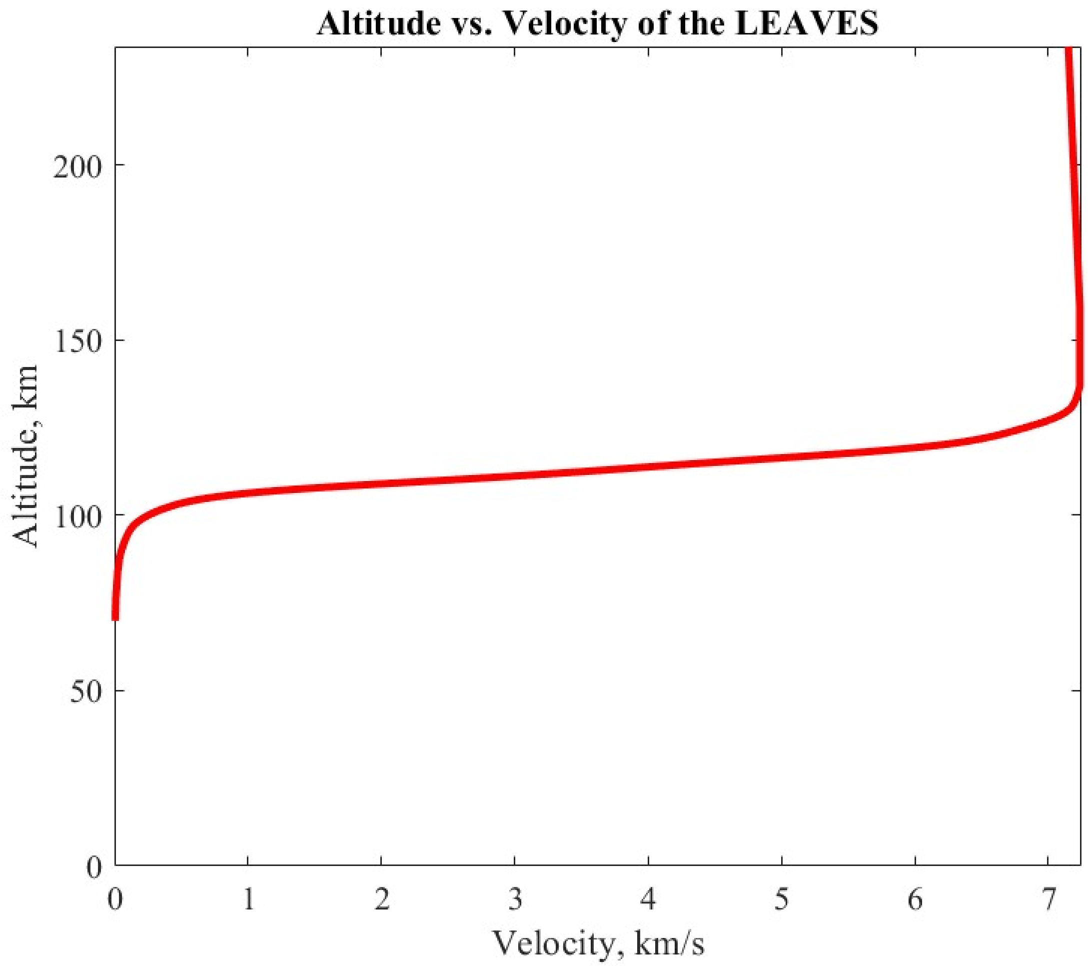

Fig. 11. Altitude vs. velocity of LEAVES entry for a ΔV of 40.7 m/s at periapsis.

There are several limiting factors on the deployment conditions of the LEAVES. If the LEAVES are deployed at too low a velocity, the periapsis would not be lowered enough, and the LEAVES will not enter the atmosphere in a reasonable time, or at all. Alternatively, the LEAVES could enter the atmosphere at too shallow an angle and "bounce" out of the atmosphere. If the LEAVES are deployed at too high a velocity, however, the periapsis may lower too much and the LEAVES will enter the atmosphere at too steep an angle, exceeding the g-limit of the aerodynamic structure, or the ΔV to reach the launch velocity would exceed the structural g-limit of the LEAVES in their storage configuration.

4.3 LEAVES Downrange Analysis

Using a MATLAB program, the team was able to model the LEAVES descent into the atmosphere [19]. The LEAVES are assumed to have a surface area (*S*) of 1 m^2, a mass (m) of 130 grams, and a drag coefficient (C_D) of 0.75 [1]. Using these values, the ballistic coefficient (β) can be calculated:

$$\beta = \frac{m}{S \cdot C_D} = 1.73\frac{kg}{m^2}$$

Using this value for the ballistic coefficient, along with atmospheric data from the Venus-GRAM model [20], the MATLAB program solves the equations of motion of the LEAVES and plots their descent into the Venus atmosphere. Using the apoapsis altitude of 240.375 km, the calculated ballistic coefficient, and assuming the LEAVES generate no lift, the ideal ΔV imparted on the probes ranges from 28.7 m/s to 40.7 m/s. At the lowest ΔV, the maximum distance traveled is 197 degrees around Venus before deployment, as seen in Fig. 12. At the highest ΔV, the LEAVES will travel a maximum distance of 129.9 degrees around the planet, as seen in Fig. 12. It is worth noting that the LEAVES will enter the Venus atmosphere with higher and lower ΔV's, but this range has been selected because it minimizes the acceleration the LEAVES experience during deployment while also keeping the deployment time down. Additionally, lower ΔV's cause the LEAVES to bounce in the atmosphere, as seen in Fig. 13. This bouncing could result in higher error of the LEAVES position, or, if the no-lift assumption is incorrect, the LEAVES exiting the atmosphere entirely.

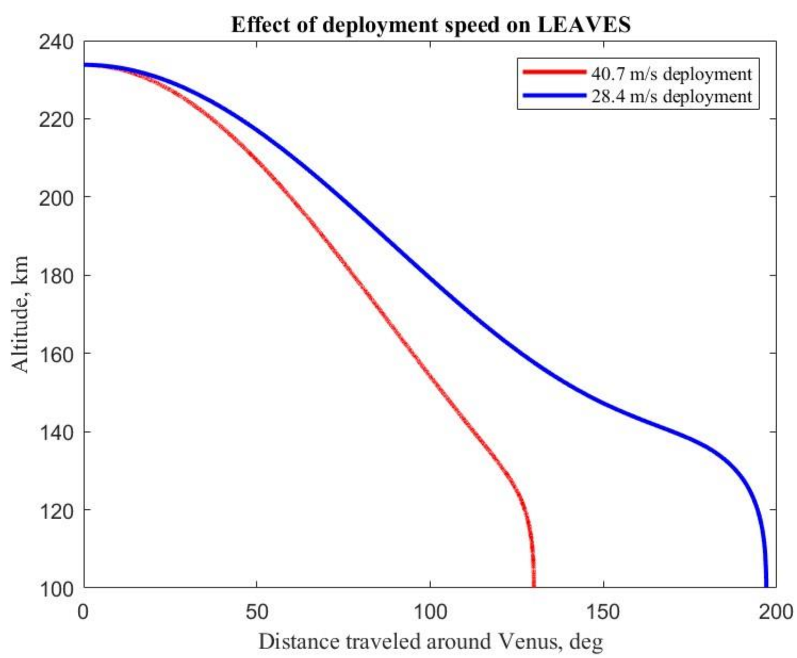

Fig. 12 Altitude vs. downrange of the LEAVES for two ΔVs.

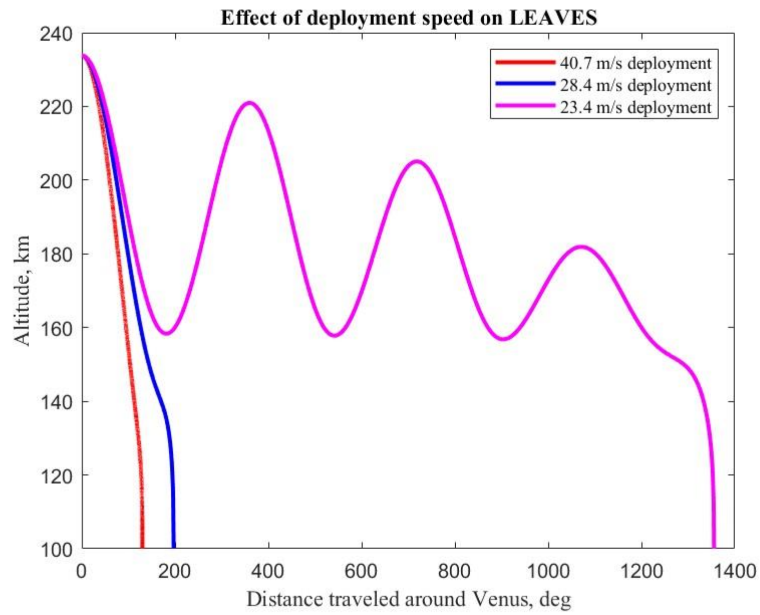

Fig. 13 Altitude vs. downrange of LEAVES, with ideal range (red and blue lines) compared to a too low ΔV (magenta line).

While the team has not integrated the wind effects on the probes, the LEAVES documentation assumes a constant westward wind velocity of 100 m/s that would cause the probes to drift by 4 degrees westward throughout their mission duration [1]. The team assumes the same in the calculations. It is worth noting, however, that the deployment conditions in this mission are different from the LEAVES documentation, and as a result the duration of the entry may change how far the LEAVES drift. It is also worth noting that the assumption that the wind is blowing at a constant 100 m/s across all latitudes and altitudes is not realistic as the zonal winds at the equator have been observed to be over 80 m/s faster than those at the poles [12]. Integrating wind data into the calculations, however, is beyond the scope of this project.

4.4 STK Validation of LEAVES Entry Trajectory

Prior to incorporating the LEAVES launch system in STK, the team wanted to verify that the MATLAB code produced replicable downrange distance results in STK. The team constructed an STK environment of similar conditions to the MATLAB simulation and ran it. STK should have similar results as the MATLAB code due to them both sharing the Venus-GRAM model, but this needs to be empirically demonstrated to ensure the validity of the simulation.

The STK platform, operating under the same assumptions with Demeter in a 240.375 km altitude orbit, 130 g mass, 0.75 drag coefficient, and 1 m^2 surface area, simulated multiple probes entering Venus's atmosphere after a fixed-ΔV launch using STK's Astrogator functionality. Two LEAVES probes are included in this simulation at the lowest and highest ΔV launch of the range – 28.4 and 40.7 m/s, respectively. Similar probes can be added to the simulation via copying a probe and modifying the launch ΔV, but the simulation is restricted to two probes for the sake of visual clarity. These simulations were done to verify the downrange distance the probes travel. While the data for altitude vs downrange distance can be exported to generate the plots like above, exact distances are not replicated at the present, rather a "close enough" verification for incorporation into mission planning and comms models.

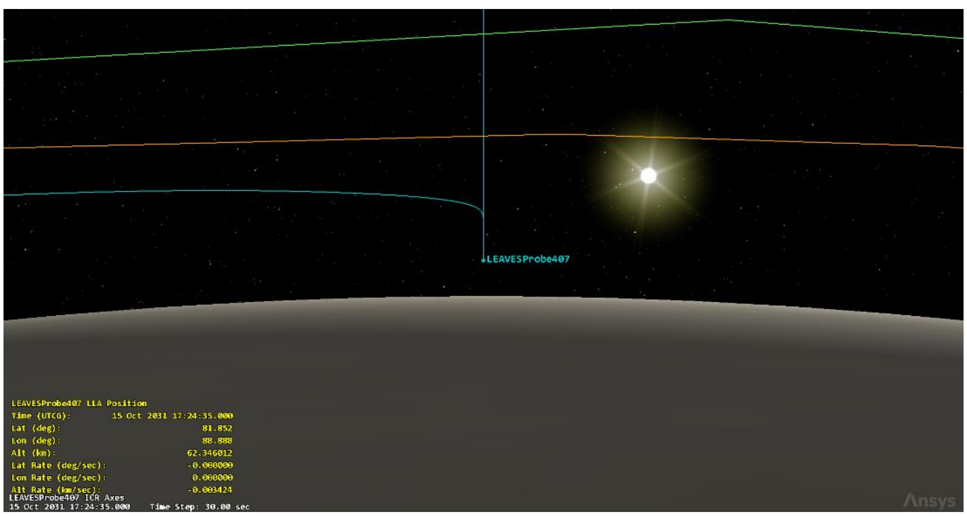

Fig. 14 LEAVES trajectory with 40.7 m/s Deployment Velocity.

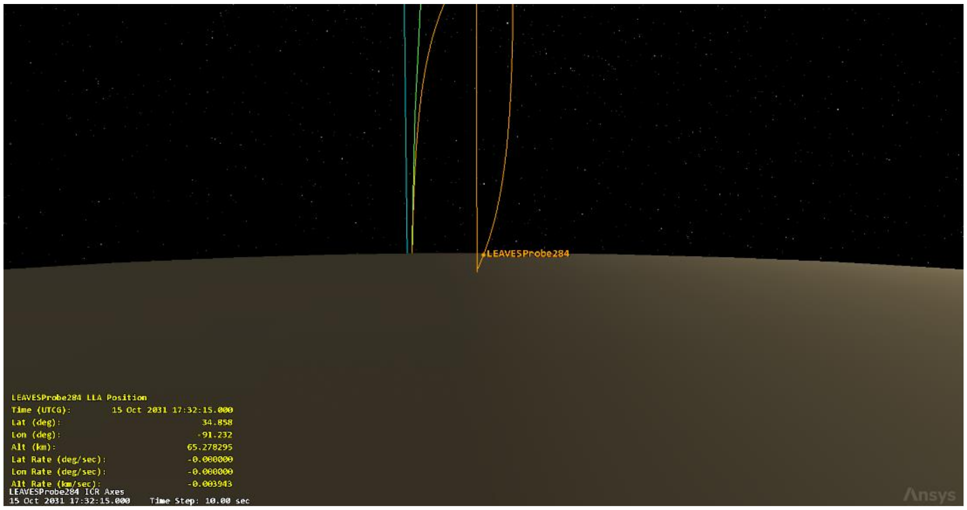

Fig. 15 LEAVES trajectory with 28.4 m/s.

Table 7: Downrange Distance Comparison

Method	28.4 m/s Deployment Drift (deg)	40.7 m/s Deployment Drift (deg)
MATLAB	192.85	129.95
STK	195.14	131.98
Difference	2.29	2.03

The STK simulation results, shown in Table 7 above, come close to the MATLAB results. STK simulations do not deviate more than 3 degrees of longitude compared to their MATLAB counterparts. The 40.7 m/s probe travels 131.98° in Fig. 14, with a roughly 2.03° difference to its MATLAB equivalent, while the 28.4 m/s probe travels a total of 195.14°, leaving a difference of 2.29° in Fig. 15. There are some limitations to the STK simulations, however. At roughly 60 km, the probes appear to approach a singularity. After passing 60 km, the probes are launched outward from Venus at 821 km/s, or 0.274% the speed of light. Because of the 60 km barrier, the probe trajectories cannot be modeled via the HPOP propagator alone, missing a sizeable fraction of the science operating period. Instead, the probe trajectories after the 60 km mark will have to be approximated using a different propagator. Despite this issue with the propagation, this indicates that the implementation of the MATLAB code is correct, and the probe reentry can be simulated adequately in STK.

4.5 Probe Release Mechanism

Initial designs for a probe release mechanism included several compressed gas cannons. The team did some basic feasibility studies assuming isentropic, adiabatic expansions. The isentropic assumption was justified because the muzzle velocity was significantly less than the speed of sound. The adiabatic assumption was justified because of the short time frames involved. The basic estimate for total propellant mass was 1.4 kg for a series of cylindrical cannons 10 cm in diameter with initial compressed lengths of 0.46 m and full barrel lengths of 1 m. Unfortunately, maximum acceleration was estimated at 157 g's.

This is an inescapable flaw of cannon systems, whether they be spring or air powered. For the best-case scenario, a constant acceleration of 41 g's over 0.1 s is required to accelerate a projectile to 40 m/s across a 2 m distance. The mechanisms need to become prohibitively large in order to safely accelerate the projectiles.

For this reason, Demeter uses small solid rocket boosters (SRB) to deploy the LEAVES proves. They impart a ΔV of 40.7 m/s at most. Different probe clusters requiring different launch velocities can be outfitted with less propellant. The design consists of 18 launch tubes shown in Fig. 16, three at each vertex of the hexagon, each containing a bundle of eight LEAVES probes. The design uses a solenoid release mechanism, with a pin through a hole on each bundle for retention. The custom motor has two nozzles oriented to rotate the LEAVES as they exit the launch tube for stability.

Fig. 16 CAD model of Demeter, showing the deployment tubes along the inside edge and a one meter bar to show scale.

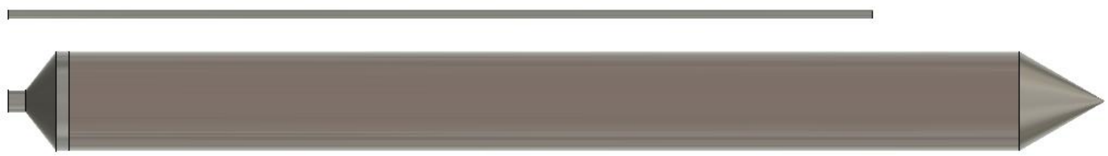

Fig. 17 LEAVES Probe Cluster with a one meter bar for scale.

The LEAVES probes are contained within a cylindrical bundle, seen in Fig. 17. Due to the folded and rolled arrangement of the LEAVES probes in their stowed configuration, the probes can survive much higher forces than when they are expanded during science operations. The carbon fiber struts of the LEAVES have a compressive strength of 1200 MPa, and the probes experience a maximum of 237 MPa during deployment, ensuring the LEAVES will survive deployment with a safety margin of 5.

As in Fig. 17 the LEAVES clusters feature conical fronts. This is not for aerodynamics in Venus's atmosphere, as the probes will have separated from the assembly by this point. The cone is to protect the probes from the exhaust of the other probe clusters. In their stowed configuration, the thin film that makes up the faces of the probes is not stretched and could be damaged by fast moving exhaust.

The solid rocket boosters (SRBs) are designed with an ammonium perchlorate composite propellant having a density of 2700 kg/m^3. The motor specific impulse is assumed to be 150 seconds. This is 95 seconds less than that of the NASA Space Transportation System boosters that used the same propellant. The SRBs have a propellant mass of 0.08 kg and a structure mass of 0.07 kg. They provide 49.8 m/s of ΔV and a final rotational velocity of 20.2 rotations per second. The SRB is 0.4 m long and 0.008 m wide. The nozzles are each centered 2.5 mm off the center of the booster, angled at 45° to the vertical. This imparts a final spin rate of 20.2 rotations per second on the payload to provide stabilization as the rocket is firing. The calculations behind these figures can be found in Appendix A.2. Both performance metrics are higher than optimal. This is to provide a margin of error to account for forces such as friction. The actual size of the SRBs can be decreased if they are too powerful. If they cause too much rotation, the nozzles can be rotated by a smaller amount.

The SRB produces a total impulse of 62.4 N-s. There exist COTS model rocket SRBs that produce 62.2 N-s of impulse with a mass of 101 g, validating the design [21].

The one major concern is that the rocket motor exhaust may damage the spacecraft. This concern is mitigated because the spacecraft is designed to withstand the heat from aerobraking and similar SRBs are used in hobby model rockets with minimal shielding.

5 Spacecraft Design

5.1 Structure

The design for the overall structure of the spacecraft is two hexagonal bodies stacked one on top of the other, as seen in Fig. 18. The two halves of the spacecraft separate during the mission to become the communications orbiter, Persephone, and the LEAVES bus, Demeter. Persephone, placed in front, carries equipment for communicating with the LEAVES probes, while Demeter carries the probes and release mechanism. The hexagonal shape of the vehicles allows for easy positioning and mounting of components such as the solar panels, which can be placed directly on the face of the spacecraft with no need for deployable arrays, saving mass.

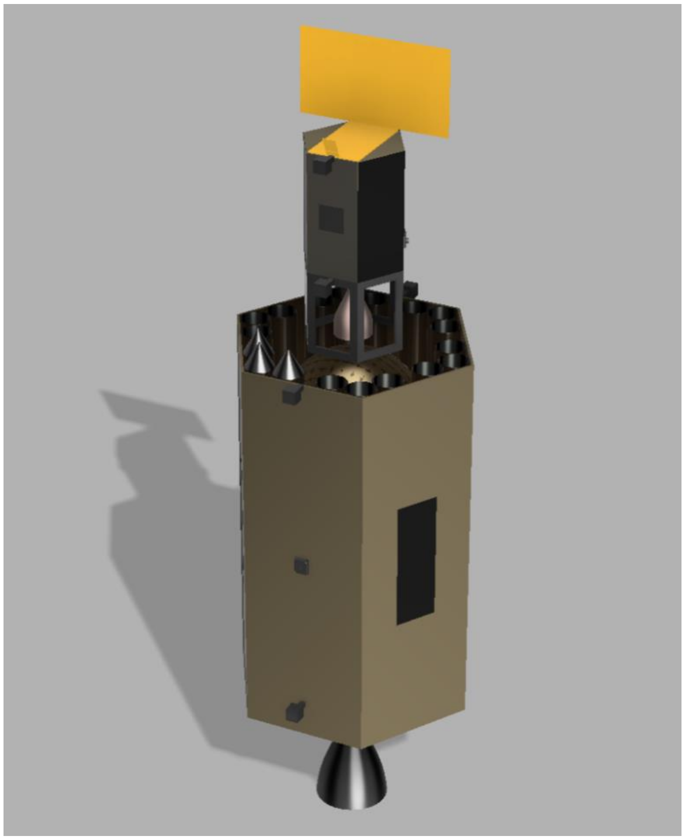

Fig. 18 Satellite assembly with LEAVES probe clusters in three of the eighteen launch tubes.

The initial design involved two spacecraft, each a triangular prism of similar size and mass, attached to each other side by side on one of their rectangular faces. The main engine on both spacecraft would fire to provide a thrust. This design was discarded, however, because the mass disparity of Demeter and Persephone created significant challenges in attitude control and balancing the coupled spacecraft during transit and orbital maneuvers.

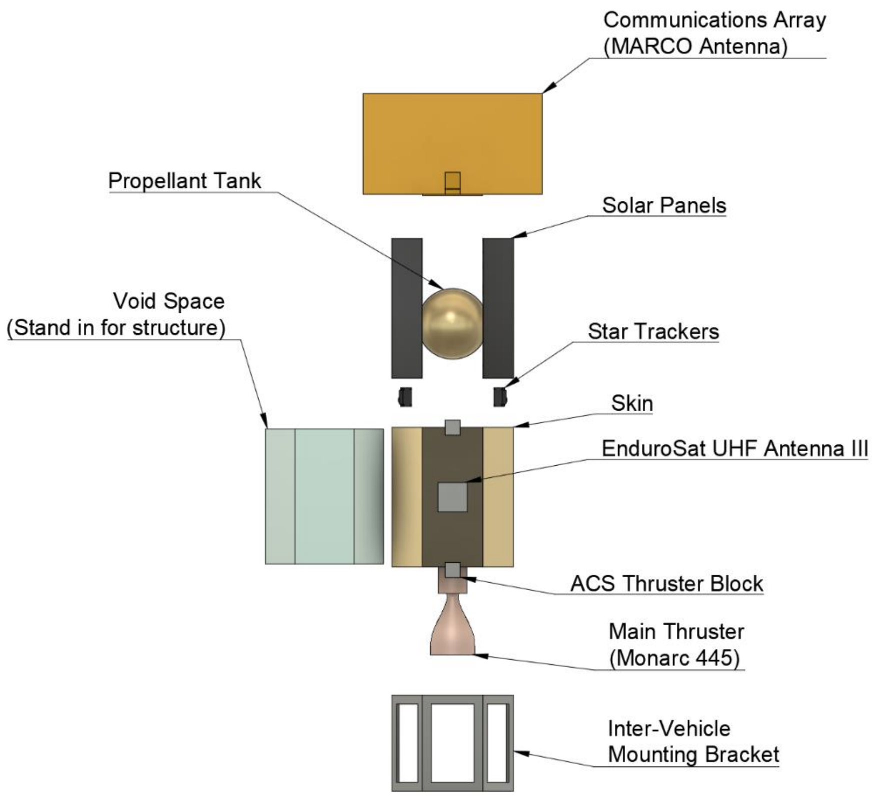

Fig. 19 Exploded view of Persephone with components labeled.

Figures 19 and 20 show an exploded view of both spacecraft layouts, with the component labeled "void space" representing the structural mass and mass of other components not considered, such as batteries, wiring, and tubing, during the calculations for moment of inertia. The LEAVES clusters are positioned forward in the launch tubes, as seen in Fig. 20. The bundles' positioning shortens the distance over which they are stabilized but prevents exhaust gas from one cluster entering the front of another cluster's tube, potentially damaging the LEAVES and spacecraft. The position of the fuel tanks, with the smaller pressurant tank at the rear, allows for more robust structure at the point where the engine is connected to the spacecraft, where the force is applied.

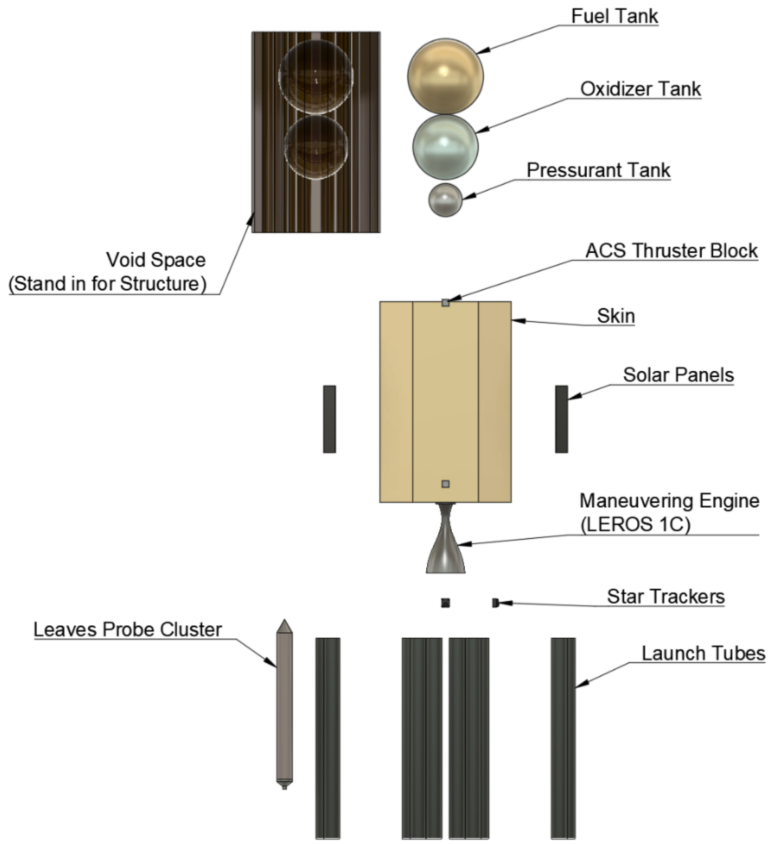

Fig. 20 Exploded view of Demeter with components labeled.

5.2 Mass Budget

The mass budget is detailed in Table 8. Further discussion of these results is included in Appendix A.1. The communications orbiter, Persephone, has a separate mass budget since it acts independently after separation (see Table 9), however, its total wet mass is part of the payload mass of the primary spacecraft, Demeter, before this event. The mass budgets were put together exclusively for the final definitive mission design, which is discussed in Section 3.3.1. Tables 8 and 9 also include the technology readiness levels (TRL) of each component, a rating from 1 to 9. For example, the LEAVES cluster deployment motors are TRL 2 in part because they have not yet been tested in space.

Table 8. Mass budget of primary interplanetary transit stage (Demeter and Persephone).

Demeter - Primary Spacecraft/LEAVES Bus			
Component	**TRL**	**Mass (kg)**	**Comments**
Payload			
• LEAVES Probes	3	18.72	144 LEAVES probes, 0.13 kg each
• Cluster Deployment Motors	2	2.70	18 clusters, 1 per cluster, 0.15 kg each
• Cluster Tubes	2	37.44	18 tubes, assumed 2× mass of probes
• Persephone satellite	6	29.14	Wet mass of Persephone satellite
Attitude Control Thruster	9	4.56	12 × MONARC-1
Star Tracker	9	0.60	2 × ST-16HV
Primary Thruster	9	4.30	LEROS 1c Bipropellant Thruster
Fuel Tank	9	5.70	NG 80364-1 (PMD)
Oxidizer Tank	9	3.90	NG 80353-1 (PMD)
Pressurant Tank	9	3.86	Custom
Pressurant (He)	--	0.34	--
Power Subsystem			
• Solar Panels	9	1.33	More details in Section 5.3
• Battery	9	0.75	--
• Regulators/Converters/PCU	9	5.75	Assumed mass
• Wiring	--	2.98	2.5% of total mass of components
General Structure	--	24.41	20% of total mass of components
Total Dry Mass			
• With Persephone	--	146.48	
• Without Persephone	--	117.34	
Propellant Required			
• Initial aerobrake	--	85.72	$\Delta V = 1219$ m/s,
• Post aerobrake	--	19.95	$\Delta V = 456.4$ m/s
• ACS thrusters	--	2.00	--
Total Wet Mass	--	254.15	With 10% contingency on propellant mass

Table 9. Mass budget for communications orbiter (Persephone).

Persephone - Communications Orbiter			
Component	**TRL**	**Mass (kg)**	**Comments**
MarCO Antenna	9	0.86	Flight heritage
UHF Antenna III	9	0.85	Endurosat
Attitude Control Thruster	9	4.56	12 × MOOG MONARC-1
Star Trackers	9	0.60	2 × ST-16HV
Primary Thruster	9	1.60	MOOG MONARC-445 Monopropellant
Propellant Tank	9	2.35	NG 80608-1 (Diaphragm/Blowdown)
Blowdown Pressurant (He)	--	0.03	--
Power Subsystem			
• Solar Panels	9	1.33	--
• Battery	9	0.75	--
• Regulators/Converters/PCU	9	5.75	Assumed mass
• Wiring	--	0.47	2.5% of total mass of components
General Structure	--	3.83	20% of total mass of components
Total Dry Mass	--	22.98	
Propellant Required (Hydrazine) ACS Thruster	--	5.16 1.00	ΔV_{req}= 426.3 m/s --
Total Wet Mass	--	29.14	With 10% contingency on propellant mass

There are inconsistencies between the values from the mass budget calculations and those computed with STK (Table 7). The discrepancy exists because STK is a higher-fidelity model which considers the dynamics of the spacecraft in more detail. Additionally, the calculations include a propellant mass contingency factor of 10%, and the ΔVs used in the mass calculations are from earlier STK simulations with different inputs. For this mission design, STK was initially used to design the trajectory and determine the associated ΔVs for a general spacecraft which has different parameters from the presented final design. The difference between the ΔV's is tolerable because the required ΔV does not vary much for a large change in mass. The STK simulation is used as the final check of the spacecraft design.

5.3 Power

Given the small size of the spacecraft and the high solar energy flux around Venus, solar panels were chosen for power generation. No alternatives were considered due to how well the solar panels fit the mission requirements.

Both spacecraft use body-mounted solar panels. Deployable solar panels were deemed unnecessary. Both spacecraft have enough available surface area that body-mounted solar panels can meet the power requirements with the help of battery power during periods of peak. The mission does not need the extra power available from deployable solar panels, so the extra complexity and mass they bring have no positive tradeoff.

The requirements for the power system were determined by the environment the spacecraft operates in. The spacecraft in orbits around Venus experience a solar flux of 2601.3 W/m^2 [2]. For the duration of the mission, including the transit, neither spacecraft is eclipsed by Venus, and the mission is scheduled to end a month before either orbit begins to be eclipsed by the planet.

The power system was initially designed to constantly generate enough power directly from the solar panels to supply all the instantaneous power required by the spacecraft. As the design of the spacecraft progressed, this became impossible. The system was initially designed to work on a 0.61 m × 0.71 m × 0.97 m spacecraft. As the project progressed and spacecraft, out of necessity, exceeded the ESPA-class size limitations. The initial design was also changed to consist of two coupled spacecraft, reducing the surface area available to mount solar panels.

Because of this, batteries are required to meet peak power demands. The batteries are designed to provide enough power to the spacecraft to run critical systems, such as basic attitude determination and control system (ADCS), computer control, and communications systems, for half the length of one orbit around Venus. This provides a backup in case of solar panel failures. The batteries are also designed to power the spacecraft at peak power for the duration of their longest primary thruster and ACS burns. These burns were 533 and 25 seconds, respectively.

Communications with Earth are considered critical because they are necessary to debug the spacecraft. Communication with the LEAVES is critical because of the limited communications windows; an unmitigated temporary power failure while the probes are in atmosphere would be disastrous for mission objectives. ADCS are considered critical for their role in orienting the spacecraft for communications and in properly orienting the solar panels.

Two different power systems were considered. The first was designed to be as light as possible. With the planned orbit, it is possible to maintain one face of the spacecraft pointing radially inwards towards Venus for communications while another face points radially inwards at the Sun for the solar panels. A third face can point close enough to Earth to keep it within range of the gimbal on the Earth communications antenna. Such a setup saves mass by requiring a minimum of energy collecting surface area. The downsides to this system, however, are that there are no backup panels and that it relies on constant attitude control.

The other option required solar panels on three of the six horizontal faces of both spacecraft. Because of the hexagonal structure of the spacecraft buses, as long as the horizontal axis of the spacecraft is normal to the ecliptic, the panel coverage is determined by:

$$\max(0, \cos(\theta - 120°)) + \max(0, \cos(\theta)) + \max(0, \cos(\theta + 120°)).$$

This leads to a maximum cosine loss factor of 0.866. Although this configuration had higher mass requirements, they were still very low. This configuration was the system chosen for the spacecraft. Peak power, nominal ACS power, and standby power requirements are needed to properly size the solar panels. The values are included in Tables 10 and 11. The nominal ACS power budget for Persephone includes more ACS thrusters than the corresponding budget for Demeter. This is to account for stricter pointing requirements brought about by the antennas on Persephone.

Table 10. Demeter power budget.

Component	Peak Power (W)	Nominal ACS Power (W)	Standby Power (W)
Payload			
• Leaves Probes	--	--	--
○ Deployment	1	0	0
• Persephone	--	--	--
Attitude Control Thruster	18 × 12	18 × 2	0
Star Tracker	1.0 × 2	1.0 × 2	1.0 × 2
Primary Thruster	58	0	0
Computer	10	10	10
Total	**287**	**48**	**12**

Table 11. Persephone power budget.

Component	Peak Power (W)	Nominal ACS Power (W)	Standby Power (W)
MarCO Antenna [22]	5	5	5

UHF Antenna III [23]	1.6	1.6	1.6
Attitude Control Thruster	18 × 12	18 × 4	0
Star Trackers	1.0 × 2	1.0 × 2	1.0 × 2
Primary Thruster	58	0	0
Computer	10	10	10
Total	**286.6**	**83.6**	**18.6**

The solar panels for Demeter and Persephone need to provide 12 and 18.6 W of power in total, respectively. The solar panels also needed to provide enough extra power over one orbit for a 22 s circularization burn. This amounts to an average extra 0.33 W over the orbit.

The longest burn by far lasts 533 s, occurring when the two spacecraft are connected. It requires 42.3 Wh of power, assuming all the ACS thrusters are active for the duration of the burn. This is not a likely scenario, as the thrusters should only be active for small portions of the burn, but this assumption gives a margin of error. The two batteries together need to be able to provide a minimum of 42.3 Wh of power to meet the power requirement for this burn. To function as a backup, each battery should be able to provide 84 W of power for 66 minutes, providing 93 Wh of energy total. This exceeds the power requirements for the burn.

With these requirements in mind, we designed the following power systems. The solar panel performance is based on a COTS solar panel designed for CubeSats [24] and the battery is a modular lithium-ion battery pack [25]. The same power system was used for both spacecraft, and the details of the power system are shown in Table 12.

Table 12. Power production system breakdown

Subsystem	Size	Mass	Power
Solar panels	0.088 m^2	0.442 × 3 kg	56.6 W (max)
Battery	0.492×10^{-3} m^3	0.75 kg	112.5 Wh
Total		2.08 kg	

5.4 Thermal

Because both spacecraft are constantly illuminated by the Sun because of the design of the mission's transit to and orbit around Venus, thermal analysis of the spacecraft was simplified. A heater is unnecessary. Because neither spacecraft experiences significant temperature swings during the mission; thermal cycling is not a worry. The largest thermal risk to the spacecraft during the mission is the heat generated by the aerobrake maneuvers. Due to the small size of the spacecraft, a passive radiator design was chosen, although the passive radiator's exact characteristics and the thermal analysis required to determine them are outside this project's scope. For designing the other subsystems, we aim for an operating temperature of 30 ºC.

In lieu of a full thermal analysis, the team identified several materials to be used on the spacecraft for thermal energy management. For each of these materials, the team determined how much net energy each material would dissipate in orbit around Venus. The materials are listed in Table 13. The absorbed solar flux and heat radiated were based on ideal blackbody radiation at 300 K and the solar flux at Venus of 2.6 kW/m^2 [2].

Table 13. Performance comparison of various coatings for use in passive temperature control systems [26].

Material	Emissivity	Radiated heat (kW/m^2)	Mass (kg/m^2)	Alpha	Absorbed solar flux (kW/m^2)
Aluminum coated FEP	0.47	0.216	0.169	0.14	0.364
Silver coated FEP	0.6	0.276	0.109	0.09	0.234
ITO coated aluminized polyimide	0.71	0.326	0.071	0.49	1.274
Glass cloth laminate	0.8	0.367	153	0.35	0.91

If cooling is needed, silver coated FEP tape, even when directly facing the Sun, radiates more heat at 300 K than it absorbs. Assuming a value of 0.5 m^2 of surface area facing away from the Sun or Venus, half of which is silver coated FEP tape, 69 W of heat would be radiated from the craft. ITO-coated aluminized polyimide is a useful coating because it has a high absorption and emissivity, so can be used to heat or cool the spacecraft during the mission by pointing it towards or away from the sun. The glass cloth laminate is an effective insulator, so it is used to protect the spacecraft from the heat of the aerobrake.

5.5 Propulsion

Propulsion systems will be primarily used for orbit maneuvers to put the combined Demeter spacecraft into the correct orbit. The Persephone orbiter will also be using it to maintain its orbit and perform all necessary attitude adjustments. For all stages of the mission, the team chose to use chemical propulsion. Due to the nature of this mission, many of these maneuvers require large changes in velocity. Additionally, some of these maneuvers are time sensitive, and need to be performed at a reasonable burn time. Most notably, the LEAVES probes must be deployed along the day-night line of Venus to meet the science goals, which gives the probes a small window of deployment. As such, the team has chosen to use chemical propulsion. When compared to electric propulsion, chemical systems boast higher thrust, and thus allow the spacecraft to perform faster maneuvers. The mission design also assumes the spacecraft reaches Venus independently from its parking orbit, as opposed to launching with another spacecraft, and as such requires more propulsive capability. While chemical propulsion systems are not typical among Smallsat missions, they have been used before and have been proved to be viable [22]. The team also adjusted various aspects of the mission design, most notably the aerobrake maneuvers and location in orbit, to minimize the need for propulsion systems. With these considerations, the team created the propulsion system to maximize the viability of the mission.

Since the spacecraft is split into two stages, each with separate orbit targets, both the Demeter and Persephone stages need a propulsion system. Each propulsion system will be required to perform orbit adjustments, so the propulsion system was designed to reflect these requirements. The first propulsion system considers the combined Demeter spacecraft while the Persephone orbiter is still attached. This segment of the mission has the highest ΔV requirement, at over 1000 m/s during the 10-minute Venus orbit insertion burn. For this, the team chose the Nammo Space LEROS 1c Apogee engine. This MON (Mixed Oxides of Nitrogen)/Hydrazine bipropellant chemical engine provides 458 N of thrust, with an Isp of 324 seconds [27]. A bipropellant engine was chosen due to their high thrust and increased efficiency over monopropellant engines. For this stage of the mission, high thrust is needed over a long burn. So, the efficiency of a bipropellant engine was found to have the greatest savings on mass by reducing the amount of fuel needed, and subsequently saving weight and volume on propellant tank. While the need for both oxidizer and fuel increases mass, the system was found to be lighter than monopropellant alternatives. Additionally, since the ACS system chosen also uses hydrazine as its fuel, no additional tanks are needed for this propulsion system. Careful wiring and valve control, as well as storing ample fuel, can allow the hydrazine tanks to fuel both systems. The LEROS engine will remain on Demeter once Persephone detaches and will continue to provide thrust for the bus as it performs circularization maneuvers in the Venus atmosphere.

Once Persephone uncouples from Demeter, it is placed in a high-altitude circular orbit for the remainder of the mission. To enter this orbit, it requires a moderate ΔV over a relatively short burn. For this, a Moog MONARC-445 engine was chosen. Unlike the LEROC 1c, the Moog engine is monopropellant, using hydrazine as its fuel. The engine can produce 445 N of thrust with an Isp of 234 seconds [28]. A monopropellant system was mainly chosen due to the mass saved by using an engine that does not need an oxidizer tank, and the very low weight of only 1.6 kg for the thruster. Calculations done by the team indicated that this system was overall lighter than bipropellant alternatives. The additional thrust that a bipropellant engine may provide proved to be unnecessary for the inexpensive maneuvers being performed. Like the LEROS 1c in the earlier stage, this propulsion system benefits from using hydrazine as its fuel, as the ACS system on Persephone also uses this fuel. Overall, the two systems chosen allow both Demeter and Persephone to perform all necessary orbit maneuvers, both independently and in tandem.

Once the propulsion systems had been chosen for each stage of the mission, fuel and tanking systems were considered. Before propellant tanks could be chosen, a mass estimate for the fuel required was established. An iterative process was used to determine the final mass of propellant needed, as outlined in Appendix A.1. This allowed the team to calculate the fuel propellant tanking volume required for both Demeter and Persephone, since both crafts require separate tanking systems. Additionally, gas volumes were considered for the fuel ejection systems (details in Appendix A.1). Fuel ejection systems included both blowdown and pressure fed systems. Tanks were then chosen to fit the calculated requirements. Table 8 shows a breakdown of the propellant masses and the chosen tank masses. Chosen tanks include off-the-shelf Northrup Grumman fuel tanks and a custom-sized tank.

For the combined Demeter system before separation, a Northrup Grumman NG 80364-1 was chosen for the hydrazine fuel, while a NG 80353-1 was chosen for the nitrogen oxide MON oxidizer [29]. These tanks were chosen primarily based off their volume. These tanks are constructed predominantly from titanium, which makes them marginally heavier than composite alternatives, but more capable of storing the fuel and oxidizer at the high volumes and required pressures. The fuel tank will also be capable of fueling the ACS system, since both will use Hydrazine. This propellant is fed to the LEROS 1c and ACS thrusters using a pressure-fed system, where Xenon gas is pushed from the pressurant tank into the fuel and oxidizer tanks, which pushes the propellant into the combustion chamber. This pressurant is stored in a custom-made titanium tank, like the Northrop Grumman tanks used in this system. A custom tank was chosen due to the high volume of pressurant required, as well as the high pressure needed to ensure the pressurant is able to maintain the operating pressure in the fuel and oxidizer tanks. Off-the-shelf tanks were either too small or unable to hold the required pressure.

Like the propulsion systems, two separate tanking systems are required, since Demeter will carry its propulsion and tanking system with it, leaving Persephone in need of a tanking system of its own. For Persephone, only one fuel tank is needed, since the Moog MONARC-445 thruster is monopropellant, using hydrazine. For this, a Northrop Grumman NG 80608-1 tank was chosen [30]. Like the other tanks, this tank is titanium due to its superior strength and storing capability. However, unlike the pressure-fed system used earlier, this system utilizes a gas blowdown system. The Xenon pressurant is stored in the fuel tank with the fuel, which allows the tank to maintain its pressure while fuel is being removed. In this way, Xenon helps “blow” the fuel out of the tank. So, no additional pressurant tank is needed. While a blowdown system uses more tank volume to hold the pressurant, the lack of a dedicated pressurant tank proved to be the most mass efficient option as proved by the calculations.

5.6 Attitude Control

The spacecraft must undergo several rotations to achieve its mission goals. When determining actuators for the attitude control system, or ACS, the team considered the types of maneuvers performed by the spacecraft and how to best execute them. Maneuvers such as telemetry repointing and reorientation for major burns are considered time insensitive and can be done in the order of several minutes, while repointing for LEAVES deployment occurs every few minutes and thus must be done more rapidly. These conditions reduce the viable ACS options to three choices: magnetorquers, reaction wheels, and monopropellant thrusters. Each of these options are assessed below for viability, with thrusters being the team's final choice for an attitude control system. The ACS is controlled by the spacecraft's RAD 750 flight computer, chosen for its flight heritage and ability to withstand radiation [31].

Magnetic torquers were considered but are impractical. Venus' magnetic field is around one tenth the strength of Earth's. This will diminish the maximum torque achievable. The short operational time frame precludes the use of such a slow attitude control mechanism for operations. Additionally, they would not work during the cruise phase, as the magnetic field during interplanetary transfer is too weak. However, given the reliability and small weight requirements, including a set might be useful as a backup. This way, if fuel reserves run dry, the spacecraft can still be pointed towards earth to transmit data back. In a sun-synchronous orbit, power should not be an issue. Given a stable orbit, neither will time.

Reaction wheels were also considered but have drawbacks. Like magnetic torquers, they do not use reaction mass, but they operate slowly and would not be able to perform the orientations required for LEAVES deployment in the necessary time. They could be used for repointing during the cruise phase for telemetry, but this also raises concerns about reaction wheel saturation, momentum dumping, mass requirements, and power allocation. Reaction wheel saturation would necessitate momentum dumping, which is typically done by magnetorquers or thrusters. Magnetorquers cannot be used to dump momentum due to the generally low magnetic field strength in all non-Earth phases of the mission. Thrusters, then, can be used to desaturate the reaction wheels, meaning that attitude control thrusters would still have to be incorporated in the spacecraft if reaction wheels are used. Reaction wheels can be lightweight, however for reaction wheels to be a viable choice of attitude control the sum mass of the reaction wheel assembly must be lighter than the total mass of any propellant that would be used for equivalent maneuvers in its stead.

Rocket Lab's suite of reaction wheels, notably the RW-0.003 and RW-0.01, are 50g and 120g respectively [34]. For the spacecraft to have three axis control, 6 reaction wheel units would be required: 3 for the Demeter spacecraft, and 3 for Persephone, totaling 300 g for a reaction wheel system using the RG-0.003, and 720 g for the RG-0.01. As shown in Table 14, the total mass of the propellant used throughout the mission using solely hydrazine thrusters is roughly 555 grams, meaning the RG-0.003 unit would be ideal to save some mass for the mission. Unfortunately, the datasheet does not include the power utilized by the reaction wheel. The RW-0.01 has that data, but the total mass of the reaction wheel system using these wheels would be greater than the 555 grams mentioned in Table 14. Other commercially available reaction wheel systems tend to be heavier than the RW-0.01 and are thus not viable for the mission's needs. Reaction wheels of similar mass with complete datasheets can be found, such as Astrofein's RW1 reaction wheels, but they provide several orders of magnitude less torque and are thus not suitable for this mission [35]. Reaction wheels also require more power than thrusters. Moog MONARC-1 thrusters, for example, require 18 watts but operate for short bursts, while the RW-0.01 would require 1.05 W for continuous operation [28]. Reaction wheel maneuvers would have a far higher actuation time due to their lower torques and would thus require more power overall. Considering the above, reaction wheels are not suitable as primary actuators for the spacecraft, leaving thrusters as the most viable choice.

Thrusters were deemed the most viable option for attitude control, as thrusters can perform all needed pointing maneuvers in the allotted times while meeting pointing accuracy requirements. Thrusters of this caliber are also often small and lightweight, and the slew maneuvers do not require a significant amount of propellant. As such, a thruster-based ACS system would be relatively light. For ACS thrusters, the team chose the Moog MONARC-1 thruster. While these thrusters only provide 1 N of thrust [28], that is enough for the required pointing maneuvers. Both Demeter and Persephone will utilize these thrusters, with each ACS system of each stage being attached to the respective tanking system. Before Persephone separates, the ACS systems for each individual spacecraft will be firing in conjunction to ensure proper attitude, pulse-width-modulated for equalized torque. This will ensure that the spacecraft is always able to rotate about the center of gravity without unplanned translations, no matter what stage in the mission the system is in. Once separated, Demeter is equipped with a separate hydrazine pressurant tank for ACS maneuvers, while Persephone uses its single hydrazine blowdown tank for both orbital and attitude maneuvers. With this ACS system, both spacecraft will be able to maintain their proper attitude.

The monopropellant thrusters are mounted on the spacecraft in T-shaped orthogonal blocks placed on its faces. There are 8 blocks total, 4 for each spacecraft. The blocks are mounted so that they are equidistant from the center of mass of their respective spacecrafts and are pulse-width-modulated during combined operation to maximize torque. As shown in Figures 19 and 20, the blocks are placed on the same and opposing faces which the antenna is mounted. The blocks would ideally be placed on the corners for maximum torque, but due to the presence of the LEAVES probes in the corners of the bus, there is no room for the necessary piping the ACS thrusters require.

Various maneuvers will be required during the operation. Multiple 180-degree slews will be required by the spacecraft to conduct operational maneuvers. In addition, during the deployment phase, the LEAVES bus will need to rotate for each individual deployment occurring every 20 degrees along the orbit, resulting in 18 distinct 20-degree rotations of the spacecraft. Table 14 details the maneuvers conducted by the spacecraft and the fuel mass required by each maneuver, and the number of times each maneuver is performed, as well as Table 15 detailing the overall missions performed during the mission.

Table 14. Mass required per slew and numbers of slews performed.

Maneuver	Mass Required per Maneuver (kg)	Times Performed	Total Mass Required (kg)
Whole Spacecraft X-Axis Slew	0.008583011	0	0
Whole Spacecraft Y-Axis Slew	0.022909	2	0.045818
Whole Spacecraft Z-Axis Slew	0.004392	114	0.500688
LEAVES Bus X-Axis Slew	0.000731456	0	0
LEAVES Bus Y-Axis Slew	0.000387	1	0.000387
LEAVES Bus Z-Axis Slew	0.000729	8	0.005832
LEAVES Bus Deployment Repointing	8.09392E-05	18	0.0014569
Relay Orbiter X-Axis Slew	4.98098E-06	0	0
Relay Orbiter Y-Axis Slew	1.22E-06	1	1.22E-06
Relay Orbiter Z-Axis Slew	4.98E-06	2	9.96E-06
Relay Orbiter Repointing	2.33803E-06	326	0.000762197
Total			0.55496

Table 15. Overall attitude maneuvers performed during the mission.

Maneuver	Instances Performed
Transfer Orbit Telemetry	114
Whole Spacecraft Y-Axis Maneuver Repointing	2
Venus Insertion Maneuvers	2
Probe Bus Orbital Maneuvers	8

Probe Bus LEAVES Deployment Repointing	18
Relay Orbiter Insertion	1
Relay Orbiter LEAVES Communication Repointing	326
Relay Orbiter Transmission	1

5.7 Communications

As a communication relay, Persephone will need to communicate with both the LEAVES probes in the Venus atmosphere and with the DSN to relay the collected data back to Earth. Because of the two different communications requirements, Persephone has two communications systems on board.

For communication with the DSN and Earth, an appropriately powerful antenna is needed. The MarCO mission proved that deep space communication using X-band frequencies is feasible with small spacecraft. X-band frequencies were chosen, allowing the communications system to transmit data in the 8-12 GHz range [22]. To transmit frequencies in this range, a reflectarray was chosen. While traditional SmallSat antennas, such as patch antennas, are smaller and save mass and power, a reflectarray is able to meet the frequency and gain requirements of the mission, while a traditional patch antenna is not.

The reflectarray chosen is based off the array used in the MarCO mission. This system was designed with several mission limitations in mind, including being size limited to fit on a 6U CubeSat, as well as power limitations [22]. The limitations faced by Persephone are not as severe. The orbiter is larger than a 6U CubeSat and will generate more power than the MarCO satellite, partially as a result of being much closer to the sun. As a result, the reflectarray falls within the size, mass, and power requirements of this mission. The reflectarray is 19.9 × 33.5 × 1.25 cm when stowed, making the antenna small enough to be stowed and deployed on top of Persephone. The antenna requires very little power to deploy and can transmit to Earth at 5 W. Because of the small amount of data generated over the mission and the months-long transmit time, data rate will not be an issue either. The antenna also has flight heritage from the MarCO satellites, giving it a TRL of 9. The antenna allows the satellite to transmit data at 8 kbit/s to Earth when properly oriented, operating at a frequency of 8.425 GHz and a gain of 29.2 dBi.

Communication with the in-atmosphere LEAVES probes is a multidimensional problem. Persephone needs a system that can receive data from the probes. For a given orbital altitude, the higher the antenna gain, the higher data rate Persephone can receive from the LEAVES. However, higher gain has the downside of decreasing the field of view of the antenna. This is a problem because the LEAVES are expected to disperse horizontally in the Venus atmosphere. The system needs to have a high enough gain to receive all the data generated by the LEAVES and a low enough gain to keep them all in view.

To determine the minimum gain required, we need to know the desired data rate. It is 2,120 bps. This is based on the LEAVES generating 40 bps over a 7950 s orbit during which there is a 300s comms window. Table 16 lists the link budget analysis used to determine the minimum allowable gain. Calculated values are underlined. Values not underlined were assumed or derived from mission requirements, such as orbital altitude or spacecraft temperature.

Table 16. Link budget analysis.

Transmitter		
Frequency	0.433	GHz
Power	0.15	W
Losses before antenna	1	dBi
Antenna Gain	3	dBi
Range		
Distance	4000	km
Pointing loss Rx	0.2	dBi
Pointing loss Tx	0.2	dBi
Absorption loss	5	dBi
Received power density	-154.6	dB(W/m^2)
Receiver		
Antenna gain	6.1	dBi
Receiver noise temperature	300	K
Data transmission		
Information data rate	1100	bps
Data symbols/block symbols	0.875	ratio
Transmitted data rate	1257	bps
Energy per bit/input noise	10.07	dBi
Required E_b/N_0	9	dBi
Margin	1.07	dBi
Spectral efficiency	1.12	bps/Hz
Filter factor	1.35	ratio
Bandwidth per LEAVES	1515	Hz
Total bandwidth	218.2	kHz

To get the required 1100 bps data rate with a margin of 1 dBi, Persephone needs an antenna of at least 6.1 dBi gain with a bandwidth of at least 218.2 kHz.

After the link budget analysis defined a minimum gain, we performed a FOV line of sight analysis in STK to determine a maximum gain. A patch antenna with a rectangular antenna pattern was chosen to maximize probe coverage. According to the LEAVES report estimates [1], the probes are expected to drift westward about 4 degrees after they reach 100 km. Given the uncertainty of atmospheric eddies and currents, the antenna is designed to cover the entire location the probes are expected to traverse at once, i.e., the antenna should cover a 3000 km swath in case the LEAVES do not all drift at the same rate. The rectangular geometry of the relay is expected to cover all the probes within the expected relative atmosphere. The curvature of Venus causes relay coverage to expand the farther along the probes are in their trajectory as the antenna's projection grows across the surface. It is important to collect the data from the farthest probes, as deviations in their expected trajectories indicate specific potential areas of improvement of atmospheric models.

Persephone's altitude of 2000 km requires a longitudinal beam width of 46.15 degrees, assuming the spacecraft rotates so the beam covers the entirety of the probe's trajectory. The lateral beam width, or the beam width along the orbit, depends on the required transmit time. The shorter the transmit time, the more the beam can be tightened. From the reference frame of the orbiter relay, a perfectly efficient transmission will relay all new cached data in one singular pass. The spacecraft will not be operating in perfect conditions, however, so some contingency coverage is included in the height of the beam. The current beam width is 50 degrees for a coverage time of 32.57 minutes. Assuming a frequency of 433 MHz, this results in a gain of 7.42 dBi, which is greater than the minimum required gain of 6.1 dBi. If necessary, the gain can be decreased to achieve a higher beam width at the cost of increasing transmission time, but the 7.42 dBi gain covers the entire potential range the probes could cover until impact.

The antenna used by Persephone is the EnduroSat UHF variable gain antenna, used with a gain of 7.5 dBi and frequency range of 435-438 MHz. It has the necessary frequency, bandwidth, and gain.

6 Summary and Conclusions

This section summarizes the objectives of our project and how our team worked to complete them. The goal of the project was to design a conceptual mission to deploy a swarm of lightweight probes, or LEAVES, to analyze the atmosphere of Venus. To achieve that goal, we had to design a spacecraft and its trajectory, a way to deploy the swarm payload of LEAVES, and a communications system to receive the data from the probe swarm. This resulted in the design of the Demeter and Persephone spacecraft, two conjoined spacecraft which separate after Venus insertion to perform their respective roles. Demeter deploys the probes, while Persephone relays the scientific data collected by the probes.

The Demeter and Persephone spacecraft are designed to examine the Venusian atmosphere using a swarm of 144 LEAVES probes. These probes carry instruments to analyze the temperature, pressure, and sulfur dioxide content of Venus's atmosphere along the day-night line. After interplanetary transfer, insertion, and aerobraking, Demeter and Persephone will orbit Venus in roughly circular polar orbits. Demeter will orbit Venus at 235 km altitude, and Persephone will orbit at 2000 km. After circularization, Demeter will deploy the probes in bundles of eight, launching the clustered probes every 20 degrees along the orbit with attached solid rocket motors. The probes will then enter Venus's atmosphere, and, once they reach 150 km altitude, unfold into a gliding configuration. At 100 km, the probes will begin to record data. Persephone will receive the data transmitted by the LEAVES probes and transmit the data to Earth at the end of the mission.

The LEAVES deployment mechanism, mounted on Demeter, consists of 18 clusters of LEAVES bundles nestled inside pods in the outer corners of Demeter. Each cluster consists of eight probes mounted on an ammonium perchlorate solid rocket motor. Each solid rocket motor has two angled nozzles, which propel the cluster while inducing spin to stabilize the clusters. After deployment from Demeter, the probes separate from the cluster and await the barometric switch activation at 150 km to extend into their glider form. The probes will then record and transmit data to the Persephone orbiter until they are destroyed.

Persephone's communication system is comprised of two antennas. The first is the MarCO reflectarray, a rectangular antenna initially designed for the MarCO mission, which is to be utilized for communication with Earth. The second antenna is the EnduroSat UHF variable gain antenna, which is used to receive data collected by the LEAVES probes.

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Appendix

A.1 Mass Budget Calculations

Table 17. Nomenclature for mass calculations. Brackets in subscripts indicate use of variable for different components/times.

Parameter	Symbol	Units
ΔV	ΔV	$\mathrm{m/s}$
Specific Impulse	I_{sp}	s
Contingency Factor	C	---
Earth Gravitational Acceleration	g	$\mathrm{m/s^2}$
Mass	$m_{[\,]}$	kg
Density	$\rho_{[\,]}$	kg/m^3
Pressure	$p_{[\,]}$	psia or Pa
Temperature	$T_{[\,]}$	K
Molecular Weight	MW	kg/mol
Gas Constant	R	$\mathrm{J/kg/K}$
Volume	$\forall_{[\,]}$	m^3

Calculating the required total wet mass for this mission is more involved than typical missions because of the separation of the communications orbiter from the LEAVES bus that occurs after the aerobrake. First, the total wet mass of the orbiter must be calculated and then used as a component of the payload (dry) mass of the primary spacecraft. The required propellant mass for the primary spacecraft must then be calculated in two parts: before and after the aerobrake, since the orbiter separation drastically decreases the mass to be maneuvered for probe deployment.

To calculate the entire mass budget for the orbiter, first add the known masses of the selected components, which are shown in Table 9. Of course, the values in this table correspond to the final iteration of this procedure, and initial values, such as the selected tank masses and general structure mass, are different for earlier iterations. This calculation uses the final iteration values to show that the design meets the mission ΔV requirements. Adding the masses of all the known components (exception of wiring, general structure, propellant mass), the following value is determined:

$$m_{components} = 18.68\ kg$$

Next, mass is added to this value to account for required wiring between components. The wiring mass is assumed to be 2.5% of $m_{components}$:

$$m_{wiring} = 0.025 m_{components}$$

$$m_{wiring} = 0.47\ kg$$

Similarly, mass to account for the general structure is added, that is, the mass of the structure that holds the components together and prevents spacecraft structural failure. This mass is assumed to be 20% of $m_{components} + m_{wiring}$. The sum of all these masses is the total dry mass:

$$m_{dry} = m_{components} + m_{wiring} + m_{structure}$$

$$m_{dry} = 22.98\ kg$$

With this dry mass, the required ΔV from STK for the orbiter circularization maneuver (only significant maneuver, value from simulation with previous values), and the specific impulse of the associated propulsion system selected, the required propellant mass can be calculated with the rocket equation. A contingency factor of 10% (1.10) is assumed to account for unplanned maneuvers and get a conservative minimum mass requirement. Also, an additional 1 kg of fuel ($m_{propACS}$) is added for use by the ACS thrusters.

The rocket equation is

$$\Delta V = I_{sp} g \ln \frac{m_0}{m_f}$$

This equation assumes a constant propellant mass flow rate, constant specific impulse I_{sp}, no drag during maneuvers, and no gravity. Solving for the final (dry) mass m_f,

$$m_f = m_0 e^{\frac{-\Delta V}{I_{sp} g}}$$

So the propellant mass is

$$m_p = m_0 - m_f = m_f \left(\frac{1}{e^{\frac{-\Delta V}{I_{sp} g}}} - 1 \right)$$

Adding the contingency factor and assumed mass for the ACS thrusters,

$$m_p = C m_f \frac{1}{e^{\frac{-\Delta V_{req}}{g I_{sp}}} - 1} + m_{propACS}$$

$$m_p = (1.10)(22.98\ kg) \frac{1}{e^{\frac{-(426.3\ m/s)}{(9.8\ m/s\char`^2)(234\ s)}} - 1} + (1)$$

$$m_p = 6.16\ kg$$

So the total wet mass of the orbiter is

$$m_{orbiter} = m_f + m_p$$

$$m_{orbiter} = 29.14\ kg$$

Next, the total propellant mass required for the LEAVES bus after the aerobrake and separation, m_{p2}, needs to be calculated. The procedure to do so is the same as that above for the orbiter but for the components on the LEAVES bus. The contingency factor as well as the assumed percentages for the wiring and structure masses are the same. This propellant mass is added to the calculated dry mass including the orbiter mass to get the required final mass before separation. With the required ΔV before the aerobrake, this value can be used in the rocket equation to calculate the total wet mass of the whole spacecraft, m_{wet1}. The total propellant mass required for the LEAVES bus propulsion system is the sum of the required propellant mass before and after the aerobrake. Another 2 kg of propellant is added to account for attitude control requirements for the LEAVES bus $\left(m_{p,bus} = m_{p1} + m_{p2} + 2\right)$. Since a bipropellant system is used for the primary spacecraft, the masses of the fuel and oxidizer need to be found for tank sizing. With an optimal oxidizer-to-fuel ratio of 0.85:1 [27], the propellant mass is:

$$m_{oxidizer} = \frac{0.85}{0.85 + 1} (107.7\ kg) = 49.47\ kg$$

$$m_{fuel} = \frac{1}{0.85 + 1} (107.7\ kg) = 60.20\ kg$$

To calculate the required tank volumes, the density of the fuel and oxidizer is used. For hydrazine, it is assumed that the density is $\rho_{hydrazine} = 1021\ kg/m^3$, and for the oxidizer, mixed oxides of nitrogen, $\rho_{MON} = 1400\ kg/m^3$ [32,33]. The required tank volumes are:

$$\forall_{hydrazine} = \frac{m_{fuel}}{\rho_{hydrazine}} = \frac{60.20}{1021} = 0.05896\ m^3 = 3598\ in^3$$

$$\forall_{MON} = \frac{m_{oxidizer}}{\rho_{MON}} = \frac{49.47}{1400} = 0.03534\ m^3 = 2156\ in^3$$

Based on these tank volumes, two tanks were selected with volumes greater but as close as possible to the required volumes. These tanks must also be able to operate at pressures within the specified thruster inlet pressure. For the final iteration discussed here, the calculated volumes are compared with the already chosen tanks to confirm their compatibility. For the case of an earlier iteration, different tanks may have to be selected, in which case the required propellant mass needs to be recalculated with the new tank masses.

The final step in the mass budget calculation process is the pressurant mass calculation. The communications orbiter uses a blowdown tank and the LEAVES bus has a pressure-fed bipropellant system requiring a separate pressurant tank. For both systems, the chosen pressurant is helium gas.

For the blowdown tank, the mass of the pressurant is calculated as a function of the propellant volume, properties of the pressurant such as the molecular weight and gas constant, the operating pressure of the tank, and the temperature in the tank. For this mission, all tanks are assumed to have an internal temperature of 20°C. For the selected tank, the Northrop Grumman 80608-1, the relevant parameters are

$$m_{tank,pressurant} = 2.35\ kg$$

$$\forall_{tank,pressurant} = 830\ in^3 = 0.01360\ m^3$$

$$T = 20°C = 293\ K$$

$$p_{BOM} = 400\ psia \text{ (operating, maximum pressure)}$$

$$MW_{He} = 4.003\ kg/kmol$$

The specific gas constant of helium is (where $\mathcal{R}$ is the universal gas constant)

$$R_{He} = \frac{\mathcal{R}}{MW_{He}} = \frac{8.314\ J/mol/K}{(4.003\ g/mol)\left(\frac{1\ kg}{1000\ g}\right)}$$

$$R_{He} = 2077\ J/kg/K$$

The first step in calculating the blowdown pressurant mass is to determine the volume of the pressurant gas at the beginning of the mission, or before any propellant is expended. The tank is assumed to be full at the beginning of the mission, so this volume is the difference between the tank volume and the required propellant volume:

$$\forall_{pressurant,BOM} = \forall_{tank,pressurant} - \forall_{p,orbiter}$$

Where, since the propellant is hydrazine,

$$\forall_{p,orbiter} = \frac{m_p}{\rho_{hydrazine}} = \frac{6.16\ kg}{1021\ kg/m^3} = 0.006034\ m^3 = 368.4\ in^3$$

So $$\forall_{pressurant,BOM} = 830\ in^3 - 368.4\ in^3 = 462\ in^3 = 0.008\ m^3$$

Next, assuming a natural gas, the natural gas law can be applied to determine the density of the pressurant gas at the beginning of the mission. For the chosen pressurant tank, the operating pressure is 400 psia or 2758 kPa.

$$\rho_{He,BOM} = \frac{p_{pressurant,BOM}}{R_{He}T} = \frac{2758000\ Pa}{(2077\ J/kg/K)(293\ K)} = 4.532\ \ kg/m^3$$

From the density and volume of the pressurant, the pressurant mass can be calculated:

$$m_{pressurant} = \rho_{He,BOM}\forall_{pressurant,BOM} = (0.008\ m^3)(4.532\ kg/m^3)$$

$$m_{pressurant} = 0.034\ kg$$

To confirm that the chosen blowdown tank is compatible with the rest of the propulsion system, the pressure of the pressurant at the end of the mission (after all propellant is expended) needs to be calculated. This is done to confirm that it is within the inlet pressure range of the orbiter thruster. Assuming all propellant is expended at the end of the mission, the volume of the pressurant is that of the tank, $\forall_{pressurant,EOM} = 830\ in^3 = 0.01360\ m^3$. From the ideal gas law,

$$p_{pressurant,EOM} = \frac{m_{pressurant}R_{He}T}{\forall_{pressurant,EOM}} = \ 1.534\ MPa = 222\ psia$$

This is within the specified inlet pressure range of the MONARC-445 thruster, so the blowdown tank is compatible with the spacecraft.

The procedure for determining the pressurant mass for the LEAVES bus is similar to that for the orbiter. Because the LEAVES bus uses a bipropellant system, the pressurant mass required for both the oxidizer and fuel need to be calculated individually and then added to determine the total pressurant mass.

This is the case because the oxidizer (mixed oxides of nitrogen) and fuel (hydrazine) have different densities and volumes. Unlike the blowdown tank, for a pressure-fed system the end-of-mission (EOM) and beginning-of-mission (BOM) pressures are known. The BOM pressure is set to the operating pressure of the chosen pressurant tank to minimize the mass requirement and the EOM pressure is the chosen operating pressure for the propellant tank. To simplify the propulsion system design, fuel and oxidizer tanks were selected such that their operating pressures are equal, $p_{pressurant,EOM} = 300\,psia = 2.068\,MPa$. The following formula is used to calculate the pressurant mass required for a particular volume of oxidizer or fuel:

$$m_{pressurant} = \frac{p_{pressurant,EOM} \forall_{propellant}}{R_{He}T - p_{pressurant,EOM}/p_{pressurant,BOM}}$$

For this propulsion system, a custom tank was designed because of the small size of the LEAVES bus. Available commercial-off-the-shelf pressurant tanks are typically designed for larger spacecraft. From the following calculations, the volume of a custom tank is much less than that of pressurant tanks produced by companies such as Northrop Grumman [29,30]. The required pressurant volume for the oxidizer and fuel can be calculated from their corresponding pressurant masses and the BOM pressurant gas density. Assuming an ideal gas,

$$\rho_{pressurant,BOM} = \frac{p_{pressurant,BOM}}{R_{He}T}$$

The BOM pressurant density is the same for the oxidizer and fuel because the pressurant is all stored in one tank, which has one uniform pressure at BOM.

$$\forall_{pressurant,BOM} = \frac{m_{pressurant,MON} + m_{pressurant,N2H2}}{\rho_{pressurant,BOM}}$$

$$\forall_{pressurant,BOM} = \frac{0.1286 + 0.2146}{50.96}$$

$$\forall_{pressurant,BOM} = 0.006735\,m^3 = 411.0\,in^3$$

Assuming a spherical pressurant tank, the internal radius and surface area are

$$r = \frac{3}{4}\left(\frac{\forall_{pressurant,BOM}}{\pi}\right)^{1/3} = 0.1171\,m = 4.612\,in$$

$$A = 4\pi r^2 = 0.1725\,m^2$$

And the wall thickness is, assuming a titanium tank (Ti-6AI-4V) with density $\rho_{wall} = 4430\ kg/m^3$, ultimate stress $\sigma_{ult} = 900$ MPa, and a safety factor S of 2,

$$t = \frac{Sp_{pressurant,BOM}r}{2\sigma_{ult}} = 0.004039\ m = 0.1590\ in$$

So the tank mass is, assuming a mass factor of 1.25 for other tank components such as mounting brackets:

$$m_{tank} = 1.25\rho_{wall}At = 3.86\ kg$$

Once the pressurant and pressurant tank masses are calculated, the final step in determining the mass budget is to substitute the values back into the equations above for the total wet mass. Several iterations need to be done in order to determine the total wet mass, since changing these masses changes the total propellant mass, which then changes the required pressurant mass. Iterations are continued until the values converge to a particular values, or the differences between the "guesses" and calculated values are zero. Microsoft Excel was used for all of these calculations, so the goal seek tool was used for this process. Once done, the total wet mass of the spacecraft can be found from the sum of the dry mass, propellant mass, and required pressurant mass. For the final iteration of this procedure,

$$m_{wet1} = 254.15\ kg.$$

A.2 Solid Rocket Booster Design

The equation for ΔV imparted on the LEAVES assembly was modified to account for the rotation of the nozzles by replacing the characteristic velocity term with

$$v_c = gI_{sp}\sin(\theta)$$

Where θ is the angle at which the nozzles are rotated.

The moment of inertia of the system was calculated to be 1.637 g/m^2. It was assumed constant throughout the burn due to the small propellant mass and the fact that it was concentrated in the center of the assembly. This made the equation for the final rotation rate out to be:

$$\dot{\theta}_f = \int_{m_i}^{m_f} v_c \frac{r}{I_{xx}} = gI_{sp}\cos(\theta)\frac{r}{I_{xx}}\left(m_f - m_i\right)$$