Prometheus Mission I

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Abstract

Prometheus I is a hosted refueling payload for a small Venus X-Sat class bus that extends the life of microsatellites in low Earth orbit by transferring hydrazine through a compact, leak-tight docking magnet interface called Prometheus Quick Connect. The concept employs a magnetic connector on a telescoping arm to achieve self-centering capture at client fill-and-drain geometries, followed by controlled propellant transfer and safe departure. The paper presents the mission architecture, key trade studies, and preliminary design of the payload, including propulsion selection, structural integration, power and thermal management, guidance and control, and ground segment operations. A four phase concept of operations: rendezvous, docking, transfer, and depart, structures safety and fault management. Analyses indicate closure against mass, volume, power, and environmental constraints of the host bus and the feasibility of per-client top-off within operational constraints. The contribution is a standards-oriented refueling approach sized to a small spacecraft platform that can serve both new and legacy clients, offering a practical path to extending on-orbit assets and reducing replacement launches.

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I. Introduction and Problem Statement

This paper presents Prometheus I, a refueling payload hosted on a Blue Canyon X-Sat Venus class bus that demonstrates safe rendezvous, docking, propellant transfer, and departure for client microsatellites in low Earth orbit [1]. Many spacecraft retire early when hydrazine is depleted while hardware remains functional [2]. Prometheus I addresses this gap with a compact, standards-oriented hydrazine top-off capability sized to small bus mass and power envelopes. The baseline configuration uses a Venus class host, a compact telescoping arm with a magnetic lock that mates to a collaborative interface [3], and a piston driven transfer method for microgravity handling. Guidance, navigation, and control targets ±2 millimeters relative position and ±1 degree attitude at contact with defined abort and retreat behaviors for safety. Operations schedule high draw activities in sunlight to remain within the single array power budget. Command and data handling supports semi autonomous execution with event driven storage and downlink via ground passes such as Brewster [4].

Mission statement. Prometheus I will design, integrate, and flight demonstrate a Venus hosted refueling payload that safely rendezvous, docks, and transfers hydrazine to client satellites within defined mass, power, and thermal limits.

II. Existing Systems

Heritage efforts and commercial offerings have advanced in space servicing but do not simultaneously satisfy the Prometheus I constraints of compact packaging, safety with toxic propellant, legacy client compatibility, and small bus resources. Orbit Fab's RAFTI provides a commercial refueling interface with growing adoption and flight qualification, indicating market traction for on orbit propellant transfer [5]. RAFTI requires the client to carry the port from launch, which excludes the installed base of legacy satellites, and it does not meet this project's objective to publish an open, royalty free interface specification [6], [7].

NASA's Robotic Refueling Mission established tools and procedures for accessing caps, cutting insulation, and manipulating valve hardware on orbit, and it validated a four phase template of rendezvous, docking, transfer, and departure [8]. These were ISS technology demonstrations rather than compact, self contained payloads sized for a small host bus. OSAM-1 advanced blanket removal and multi tool servicing but was canceled after cost and schedule growth, with system complexity cited as a driver for risk [9]. SPHERES provided guidance, navigation, and control behaviors for small free flyers but did not address toxic propellant handling, leak tight fluid coupling, or docked load reactions [9].

Prometheus I targets the intersection that these systems do not meet at once. The Prometheus Quick Connect proposes a magnetic, bayonet style connector with an adaptor sleeve for common fill and drain geometries, enabling leak tight, bidirectional transfer without intrusive access sequences while keeping the interface specification royalty free.

III. Requirements

This section defines the top level "shall" statements that govern Prometheus I and the Venus host spacecraft. Requirements flow down from the capability gap and mission objectives through the four mission phases and are cross checked against bus interfaces, environments, and regulatory constraints. Verification modes listed here are single and specific.

Table 1. Top level requirements

ID	Requirement	Rationale	Verification
R1	Spacecraft shall deliver 30 kg hydrazine monopropellant during one servicing operation.	Core mission success metric and service value.	Analysis: Structure subsystem, tank capacity and refuel mechanisms.
R2	Payload shall fit entirely within the Venus X-Sat payload cavity of 20.5 × 16.4 × 27 in.	Ensures mechanical compatibility with the single-array host configuration.	Inspection: CAD envelope
R3	Total spacecraft power budget shall be ≤ 222 W in all operation modes.	Guarantees operability on the single-array power budget.	Analysis: Power budget
R4	Spacecraft shall autonomously service rendezvous, soft docking, propellant transfer, and depart to a ≥50 m	Demonstrates service and post ops safety.	Analysis: GNC and C&DH subsystems
R5	Space craft shall meet environmental survivability, orbital debris mitigation, and applicable RFP, export, and licensing requirements.	Enables safe flight.	Analysis

IV. Initial Concepts & Trade Study Down-select

The following are concepts that were worked out for use to meet the mission statement and requirements.

Concept 1: Prometheus Compact Robotic Refueler

The first proposed design concept is a Compact Robotic Refueler (CRR) which is seen in figure 1. The payload is inspired by NASA's OSAM-1 mission and starts its active process by grabbing the target using a robotic arm that is inspired by the SPIDER robotic arm [11]. This arm holds the target in place as another set of robotic tool's cuts into the target satellite's exterior and folds the multi-layer insulation back. This gives the CRR access to the internal systems of the target. The CRR would then gain access to the internal fill drain value [12]. The CRR would deliver the fuel with a pump while monitoring the state of the mission. After the fuel has been delivered the CRR would then replace the layers that it had peeled back and reseal the satellite for continued use.

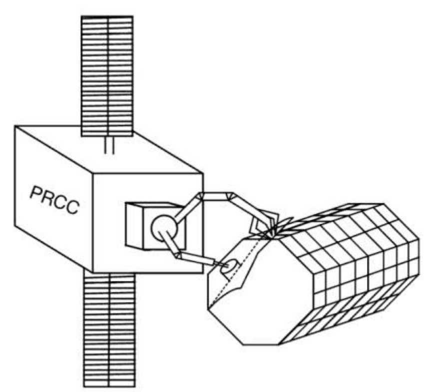

Figure 1. Prometheus Compact Robotic Refueler

Concept 2: Prometheus Quick Connect (PQC) Interface

The satellites that the Prometheus I mission aims to aid have an internal fill drain valve that can be difficult to safely gain access to. Companies like Orbit Fab have recognized this as a barrier to progress and have begun installing connections to make these ports external creating opportunity for refuel [13]. An example of this is the RAFTI interface.

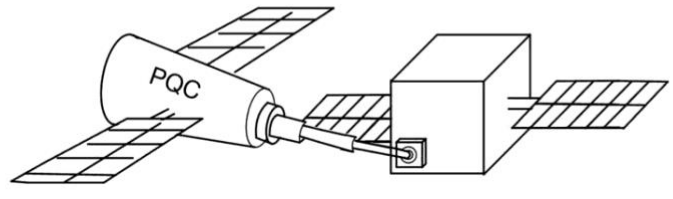

Figure 2. Prometheus Quick Connect

The Prometheus Quick Connect (PQC) works with these targets and incorporates lessons learned from NASA's SUMO/FREND tests showing the effectiveness of magnetic connections for precise guidance and connection. The payload pictured in figure 2 shows a interface connected to a telescoping pole that allows for extended reach to the target. The interface uses a collaborative design that works with systems that are already installed on satellites operating in LEO to access their FDV's and deliver the critical hydrazine fuel [14]. This design improves on physical mating methods to create a payload that connects more efficiency without creating wear on either target or payload.

Concept 3: Prometheus SPHERE

The Prometheus SPHERE pictured in figure 3 shows a multi part payload that incorporates a main body as well as multiple detachable spheres. The spheres are inspired by NASA's sphere initiative and in the SLOSH initiative were used to test fluids motion in space [10]. The Prometheus SPHERE design would use similar drones to transport a set amount of fluid over to the target and deliver it into a interface like Orbit Fabs' RAFTI port. When the payload is near the target a drone would be released for autonomous transportation and connection. After delivery it would return to the body and re-join the payload.

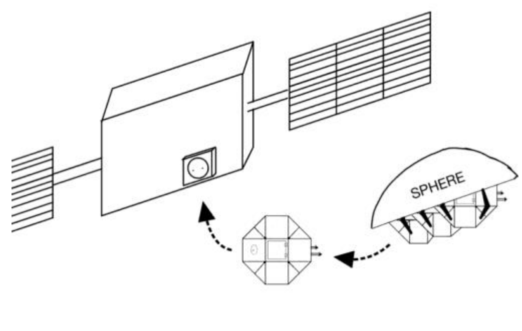

Figure 3. Prometheus SPHERE

Trade Study and Concept Selection Rationale

To choose the idea that was best to move forward with the team conducted a trade study based on autonomy, cost, maturity of technology, safety, adaptability and complexity of controls. These represented things that the team identified as most in alignment with the mission values and requirements. The rational between each weight can be seen in table 2.

Table 2. Trade study rational

	1	2	3			
Complexity: Autonomous 20%	More than 4 moving parts needed to communicate with each other for interaction.	2-3 moving parts needed to communicate for interaction	One or less moving part needs to comminicate	1	2	3
Cost 15%	Estimated over 2 billion.	Estimated between 1 billion & 2 billion.	Estimated under 1 billion.	Good	Better	Best
Maturity 10%	All completely new ideas. No basis in previous expirements.	Inconperates new uses and technology with proven concepts.	Majority of the system has been tested before but technology is used in new ways.			
Safety: Risk 25%	High opportunity for combustion or failure of the systems. Pressure could comprismise mission	Modererate opportunity for combustion. Systems are not put in high risk positions.	Low opportunity for combustion. Systems are safe from exposure.			
Adaptability 20%	Designed to only interact with one type of satalite.	Can interact with multiple kinds of satallites.	Can interact with any kind of satallite.			
Controls 10%	Needs greater than two complex controls system	Needs two complex controls system	Needs less one or less complex control system			

Using the weights seen in the far left column to prioritize the qualities of the team. The weights guaranteed that the values closest to the team were prioritized. Lower weights were determined as easier to overcome or less important. High weights like safety were critical to the success of the mission. The results can be seen in table 3.

Table 3. Initial concepts trade study

	Compact Robotic Refuler	Prometheus Quick Connect Interface (PQC)	Prometheus SPHERE
Complexity: Autonomous 20%	2	3	1
Cost 15%	1	3	3
Maturity 10%	2	2	2
Safety: Risk 25%	1	2	2
Adaptability 20%	3	1	1
Controls 10%	1	3	2
Prefrence equation calculation	1.7	2.25	1.75

As seen in the final row the selection identified the Prometheus Quick Connect as the idea that best accomplishes the mission objectives.

V.Mission Architecture and CONOPS

The Prometheus quick connect (PQC) will be carrying out a four year mission where it will carry out the Prometheus I mission which is depicted in figure 4.

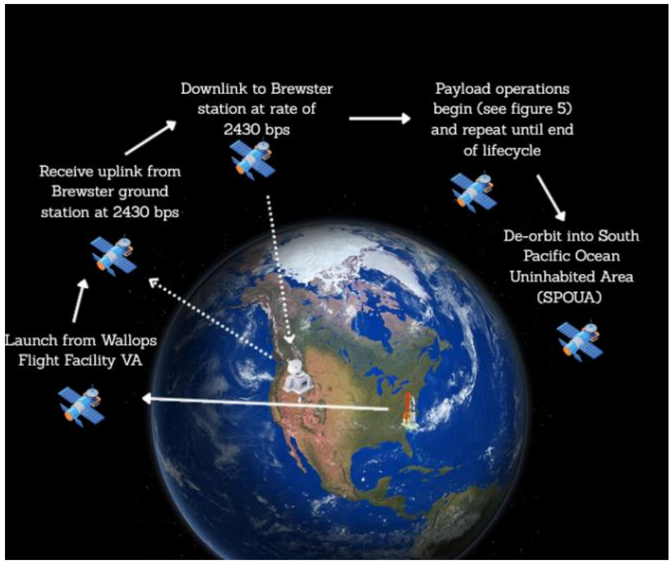

Figure 4. Mission architecture

The Prometheus I mission begins at the Wallops Island NASA facility in Virginia and launches the payload up in to low earth orbit (LEO). The payload will be launched using the Minotaur I rocket. During launch intense amounts of vertical, acoustic, and vibration forces will act upon the payload. To mitigate that the payload has been comprised of carbon fiber T700 material and tested to survive the expected 12g's of launch. Once launched the payload requires an adjustment to get on to the proper path of inclination of 55 degrees where the expected targets for the Prometheus I mission reside. Once on path the payload will begin to make contact with the Brewster ground station for uplink. This facility will be used for all communication operations. Due to the autonomous design uplink commands will be limited mostly to targets for the payload to refuel. When received the information will be transferred to the internal data handler, the Nanomind A3200. The command is then output to put the necessary systems in action. During operation update data from those subsystems are stored in the data handler and then transmitted down to the Brewster station upon next communication window. Downlinks will consist of reports on successful targeting, release and mission progress.

The next step is the beginning of the operations cycle. This is where once a target is identified the steps of figure 5 begin to occur. The payload orients itself to the target using its GNC systems. The payload will then extend its arm out to meet the collaborative interface that gives access to the fill drain valve. The PQC will use its magnetic connections to quickly guide itself to the port and make a lock. Once confirmed the fuel will begin to transfer to the target satellite in need of hydrazine fuel. The fuel will use a two part system to pump. A vain based PMD (propellent management device) will direct the fuel in zero gravity into the arm where an internal piston will move it to target [15]. Once the fuel has been transferred the PQC will disconnect from the target and the arm will once again retract into the payload. This cycle will continue until the payload has run out of supporting fuel. The payload will then use the last of its own reserves to push itself into a orbit that ends at the South Pacific Ocean Uninhabited Area (SPOUA) where the majority of the payload will burn up before impact and officially ending the Prometheus I mission [16].

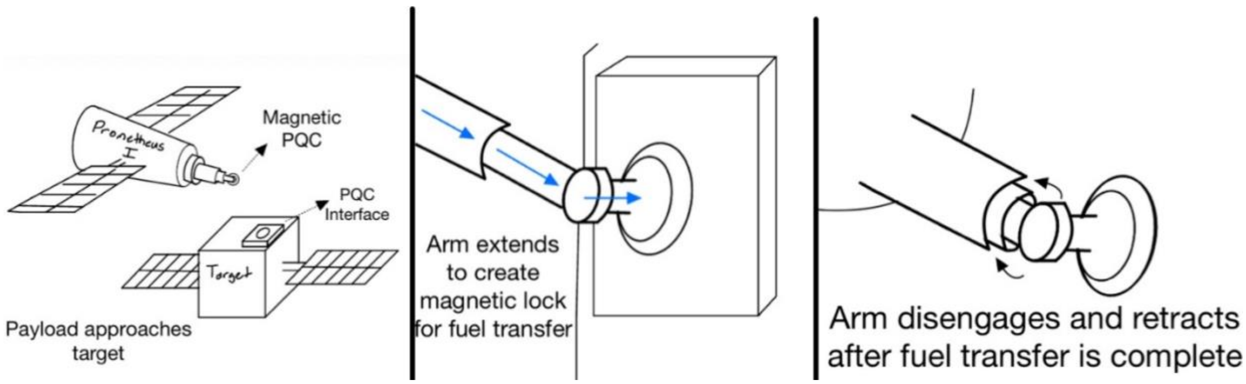

Figure 5. PQC operations phase

VI.Preliminary Design

Prometheus I is a Venus hosted refueling payload that performs approach, docking, controlled hydrazine transfer, and safe departure while remaining within bus limits for volume, mass, power, thermal, and interfaces. The payload resides fully within the Venus envelope and uses standard mechanical and electrical interfaces for power, command and data handling, communications, and attitude control. High draw activities are scheduled in sunlight to remain within the single array power budget.

Three custom elements were designed by the team to define their architecture:

1. Propellant tank. A Ti 6Al 4V rectangular tank with approximate external dimensions of 52 by 41 by 14 centimeters provides capacity for about 30 kilograms of hydrazine. An internal piston enables positive displacement for transfer and a vane propellent management system.
2. Refueling arm. A three segment foldable arm in Aluminum 7075 mounts above the tank. It deploys to place the docking PQC for soft capture.
3. Prometheus Quick Connect. A magnetic provides self centering capture, sealing, and a leak tight fluid path. Adaptor sleeves mate to common fill and drain geometries, which enables servicing of legacy clients that lack external ports.

Power management and survival heating are integrated with host functions. Store and forward commanding and telemetry keep average power within daily boundaries. Thermal design protects the tank, interface seals, arm actuators, and avionics through multilayer insulation, conductive paths to radiators, appropriate surface finishes, and localized survival heaters. Seal and propellant temperature ranges are controlled and verified through thermal vacuum testing targeted at confirming heater duty cycles and model correlation. Volume, mass, and power are tracked against Venus limits with margins held at component and payload levels.

Launch Vehicle

The launch vehicle is a part of the Prometheus I mission that is purchased for simplicity and accessibility. To select the most effective launch vehicle the team looked and three main competitors used for similar LEO missions. These were Minotaur I, Delta II, and Falcon 9 [17-19]. The priorities for selection were payload capability to make sure it could yes or no fit the size and volume of the payload, cost, success rate, and launch sites that gave access to valuable inclinations of orbit with the least fuel expenditure. Considering these things the launch vehicle that was chosen was the Minotaur I.

Ground Station and Communications Subsystem

A ground station must be established to communicate with the satellite during operation to give commands as well as receive relevant data. To be eligible for use the station must be open to outside contracting and be inside the communication range on the orbit path of the mission. Using STK (systems tool kit) to map the orbit path 3 ground stations in Washington state were evaluated for range and location. These three stations were the Brewster, Yacolt, and Echo Star ground stations. Using STK again all three stations had negligibly similar windows of communication. With similar results for location and range the antenna data was input into a LINK budge for both uplink and downlink to find the greatest data margin [20]. This led to the choice of Brewster ground station with a data rate of 2430 bps. The uplink and downlink margins are 63.77dB and 44.27dB respectively, which far exceed the required tolerance and can be seen in table 4 for uplink and table 5 for downlink.

Table 4. Uplink data

Freq.	f	Ghz	input	7.00
Xmtr Pwr	P	W.	input	1.0000
Xmtr Pwr	P	dbW	10 log(P)	0.00
Xmtr line loss	L_l	dB	input	-1.00
Xmtr Ant. Beamwidth	θ_t	deg	Eq. (13-19)	0.857
Peak Xmt. Ant. Gain	G_{pt}	dB	Eq. (13-20)	45.64
Xmt. Ant. Diam.	D_t	m	input	3.50
Xmt. Ant. Pointing Error	e_t	deg	input	0.10
Xmt. Ant. Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.16
Xmt Ant. Gain	G_t	dB	$G_{pt}+L_{pt}$	45.48
EIRP	EIRP	dB	$P+L_l+G_t$	44.48
Prop. Path Length	S	km	input	7.000E+02
Space Loss	L_s	dB	Eq. (13-23a)	-166.25
Prop. & Polariz. Loss	L_a	dB	Fig. 13-10	-0.10
Rcv. Ant. Diam.	D_r	m	input	0.10
Peak Rcv. Ant. Gain	G_{rp}	dB	Eq. (13-18a)	14.72
Rcv. Ant. Beamwidth	θ_r	deg	Eq. (13-19)	30.00
Rcv. Ant. Pointing Error	e_r	deg	input	1.00
Rcv. Ant. Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.01
Rcv. Ant. Gain	G_r	dB	$G_{rp}+L_{pr}$	14.70
System Noise Temp.	T_s	K	input (using Tal	135.00
Data Rate	R	bps	input	2430.00
Est. E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	66.27
Bit Error Rate	BER	--	input	1.0E-05
Rqd. E_b/N_o (2)		dB	Fig. 13-9 (BPSK	4.50
Implementation Loss (3)		dB	input (standard	2.00
Margin		*dB*	*(1)-(2)+(3)*	*63.77*

Table 5. Downlink data

Freq.	f	Ghz	input	7.00
Xmtr Pwr	P	W.	input	0.0100
Xmtr Pwr	P	dbW	10 log(P)	-20.00
Xmtr line loss	L_l	dB	input	-0.50
Xmtr Ant. Beamwidth	θ_t	deg	Eq. (13-19)	30.000
Peak Xmt. Ant. Gain	G_{pt}	dB	Eq. (13-20)	14.76
Xmt. Ant. Diam.	D_t	m	input	0.10
Xmt. Ant. Pointing Error	e_t	deg	input	1.00
Xmt. Ant. Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.01
Xmt Ant. Gain	G_t	dB	$G_{pt}+L_{pt}$	14.74
EIRP	EIRP	dB	$P+L_l+G_t$	-5.76
Prop. Path Length	S	km	input	7.000E+02
Space Loss	L_s	dB	Eq. (13-23a)	-166.25
Prop. & Polariz. Loss	L_a	dB	Fig. 13-10	-0.10
Rcv. Ant. Diam.	D_r	m	input	3.50
Peak Rcv. Ant. Gain	G_{rp}	dB	Eq. (13-18a)	45.60
Rcv. Ant. Beamwidth	θ_r	deg	Eq. (13-19)	0.86
Rcv. Ant. Pointing Error	e_r	deg	input	0.10
Rcv. Ant. Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.16
Rcv. Ant. Gain	G_r	dB	$G_{rp}+L_{pr}$	45.43
System Noise Temp.	T_s	K	input (using Tal	135.00
Data Rate	R	bps	input	2430.00
Est. E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	46.77
Bit Error Rate	BER	--	input	1.0E-05
Rqd. E_b/N_o (2)		dB	Fig. 13-9 (BPSK	4.50
Implementation Loss (3)		dB	input (standard	2.00
Margin		*dB*	*(1)-(2)+(3)*	*44.27*

Structure Subsystem

The most defining part of the structure was the material chosen to build the payload out of. The choice of this payload would become critical in determining the thermal system and structure necessary to survive the hostile process of space launch and travel. The team researched industry standard materials and down selected to 3 for a trade study. These were aluminum alloy 7075, carbon fiber T700, and titanium alloy Ti6Al7Nb [21]. They were evaluated on their tensile strength, elastic modulus, thermal conductivity, and density. These qualities were chosen because high tensile strength and modulus of elasticity would be the best to survive launch conditions such as fact accelerations, as well as intense acoustic and vibration environments. The thermal conductivity would resist changes in temperature to protect the payload from high heat during takeoff and aid the thermal systems during the freezing environments of space. Lastly the density indicates the weight of the structure and expands possibilities of design to keep under the mass ceiling. With all those things considered the chosen material was the carbon fiber T700.

Next in the process was the design of the PQC in CAD that would serve both functionality and strength. Figure 6 shows a CAD representation of the PQC. This highlights the tank holding the hydrazine fuel and the arm that will extend out of it to make contact with the target. Figure 7 shows the dimensioned layout of the payload when it is in its folded stage.

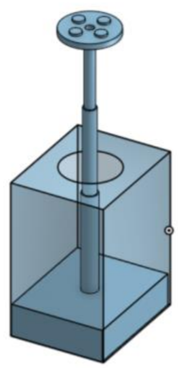

Figure 6. CAD design

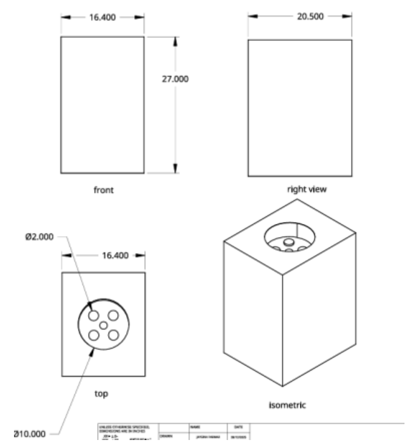

Figure 7. 2D layout

Using a representative mass was simulated was created and loaded with the qualities of the carbon fiber T700. Using a graph from the Minotaur I user guide the expected forces that the load would endure was 12g's [19]. Figure 8 and 9 show the stress tests ran to simulate these conditions.

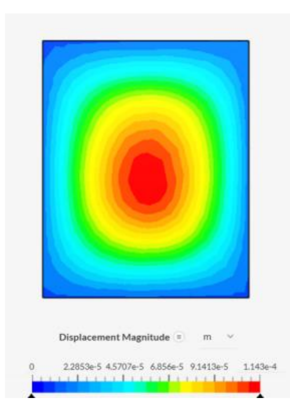

Figure 8. Displacement stress

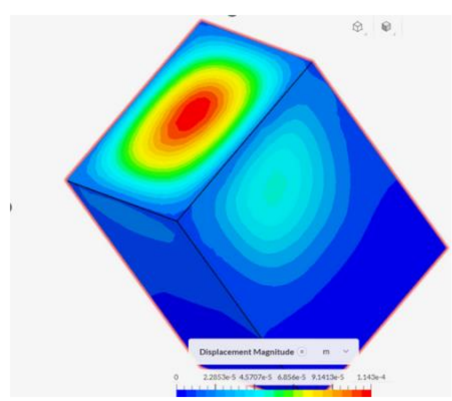

Figure 9. Von Misses stress

These stress tests show that the structure showed that it would far withstand the expected launch loads and confirm the payload safe for launch.

Command and data handling Subsystem

The Command and Data Handling (C&DH) subsystem on Prometheus I is responsible for managing telemetry, processing commands, supporting limited onboard autonomy, and interfacing with the spacecraft's attitude determination sensors. It handles four key types of data: housekeeping, attitude, payload/science data, and ground commands. These are generated or processed at rates ranging from 1 to 11 Hz, resulting in an estimated daily data volume of approximately 21 megabytes [22] and can be seen in table 6.

Table 6. Data volume

Data Type	Related Function (SMAD)	Execution Frequency (Hz)	Bits	Duration (s/day)	Total Volume (bits/day)	Volume (KB/day)
Housekeeping Data	Telemetry Processing	10.0	64	86,400	55,296,000	6750 KB
Altitude Data	Rate Gyro + Sun Sensor	10.0 + 1.0 = 11.0	64	86,400	60,544,000	7392 KB
Payload/Science Data	Simple Autonomy	1.0	64	86,400	5,529,600	675 KB
Commands	Command Processing	10.0	64	86,400	55,296,000	6750 KB

To meet the processing and memory needs of these functions, flight computer selection was based on capable of handling 12 KIPS (thousand instructions per second) and equipped with 64 KB of onboard memory, split between code and data. It also includes at least 40 to 50 MB of non-volatile storage, allowing the system to retain two days of operational data in case of delayed ground station contact [22]. Since Prometheus I follows an event-driven downlink model, data is stored during key operational phases and only transmitted during scheduled ground passes. The expected data requirements according to the SMAD [22] are seen in table 7

Table 7. Memory requirements

Function	Code (Kwords)	Data (Kwords)	Throughput (KIPS)	Frequency (Hz)
Telemetry Processing	1.0	2.5	3.0	10.0
Command Processing	1.0	4.0	7.0	10.0
Simple Autonomy	2.0	1.0	1.0	1.0
Sun Sensor (Attitude)	0.5	0.1	1.0	1.0

A trade study was conducted comparing three flight computers: the CubeSat Kit FM430, the Nanomind A3200, and the Raspberry Pi CM4. Evaluation criteria included processing capability, onboard memory, power consumption, spaceflight heritage, and radiation tolerance. The Raspberry Pi CM4 offered high performance but was not optimized for space environments. The FM430 was space-qualified but limited in processing power and memory. The Nanomind A3200 System offered the best balance of performance, low power usage, and proven flight history, making it the final selection.

GNC/Attitude Subsystem

For the Prometheus I mission the guidance navigation control (GNC) and attitude control systems are responsible for safely aligning the payload to the target and holding it into position during fuel transfer. It also must communicate with the central systems as it is responsible for halting the process if anything goes outside the expected tolerance. This is to prevent any damage to both the host and the payload which is of the highest priority. Standards for this process are set by IDSS [23] and ECSS [24] that define regulated amounts of certainty, speed, and jitter at three distinct stages of approach. The end of this requires that at contact the payload must be steady within 2 millimeters and 1 degree of motion at contact. The systems are comprised of GPS, and internal measurement unit (IMU) for navigation as well as star trackers, and gyroscopes for attitude. Lastly the payload will use reaction wheels and monoprop thrusters to act on the information from the attitude and navigation indicators. These are all sold in suites.

To decide the proper suite the team considered accuracy, mass and power consumption, cost, complexity, and reliability. This was to minimize the overall weight and power required by the payload while also prioritizing the precision needed to successfully execute the mission. The trade study can be seen in table 8.

Table 8. GNC trade study

Equipment and Sensors	Weight	Full Precision Suite	Simplified Suite	Balanced Hybrid Suite
Accuracy	0.35	3 (best)	1 (fails tolerance)	2 (meets tolerance)
Mass and Power	0.25	1 (heaviest)	3 (lightest)	2 (moderate weight)
Cost and Integration Complexity	0.15	1 (costly)	3 (cheapest)	2 (moderate)
Autonomy and Reliability	0.25	2 (complex with multiple failing points)	2 (needs more ground control)	3 (redundancy, robust autonomy)

After failing the simple suite for not meeting tolerance and requirements it was removed from consideration. The decision was made that the balanced suite was best for this mission. This system could deliver the movement necessary for the size of the payload within standard requirements at the best value.

Thermal Subsystem

The thermal system makes sure that all systems in the payload are at an operating temperature that allows them to complete their function. This includes things like the hydrazine tank, electrical systems, wiring, gauges, and sensors [25-26]. All of these are unable to perform in the freezing environment of space. Thermal systems are split into two categories. These are active and passive systems. Passive systems include coatings like MLI coatings and thermal straps. These require no energy and insulate or move heat. The active systems require power and physically distribute heat to the necessary regions in coordination with sensors implemented around the payload.

The selected systems were driving by the individual requirements for each system. Consideration was also given to the required mass and power added by excessive systems. To always guarantee functionality power requirements had to be met at eclipse where it would be both the coldest and have the least access to energy. The mission offered three possible methods of heating. These were passive, passive with the assistance of the Venus bus supported heating systems, and lastly an independent mini loop pump system. Under the mission requirements the decision was made to use a combination of passive systems as well as heating systems from the Venus bus. This met the requirements for each system and would provide the necessary support to sustain the payload through it service life without using extraneous amounts of power.

Propulsion Subsystem

The propulsion subsystem enables Prometheus I to reach the service orbit, conduct phasing and rendezvous, perform precise terminal approach and station keeping, and execute commanded retreat and disposal. It must provide promptly available thrust with predictable response during proximity and docking while imposing minimal electrical burden on the Venus host. Electric power on the single array Venus bus is limited to about 222 W with a 10.2 Ah battery, which constrains continuous heater and drive loads during eclipse [27][28][29]. Long, sunlight only burns that couple operations to power availability are undesirable near tight approach windows. A low electrical draw, immediately responsive chemical actuator is therefore preferred, since the electrical demand is largely limited to valves, controls, and small heaters [30].

A trade study was conducted across four candidate architectures and evaluated against responsiveness, safety, integration complexity, propellant mass, and operational coupling to the power system. Green monopropellant LMP 103S offers flight heritage, specific impulse near 230 to 235 s, and favorable density impulse that reduces tank volume and structural penalties while lowering handling hazards relative to hydrazine [31][32]. Bipropellant MMH and NTO in a cluster near 4 N provides specific impulse near 300 to 330 s and short burns, but imposes dual tank plumbing, higher toxicity, and programmatic complexity that are difficult to justify within the mission risk posture [33]. Electric propulsion provides very high specific impulse near 1500 s, yet at very low thrust it drives multi day burns and sunlight only operations with heater and power demands, which makes it unsuitable as the primary terminal actuator even though it can be useful for slow phasing [34][35]. Cold gas nitrogen near 0.3 N is simple and mature, but specific impulse near 65 to 85 s leads to prohibitive propellant mass and long maneuver durations at the required performance level [36][37]. The decision matrix summarizes the scoring and weights that led to the preferred option (see Table 9).

Green monopropellant LMP 103S with a 1 N thruster cluster is selected. This configuration balances responsiveness, safety, and integration burden, executes commanded maneuvers in practical hours rather than days, and avoids the toxicity and plumbing complexity of bipropellant while carrying lower operational coupling to the electrical system than an electric propulsion centric solution. The heritage base and density impulse advantage limit tank volume and thermal accommodation, which simplifies mechanical integration on the Venus bus and eases thermal control requirements.

Table 91. Propulsion decision matrix

Criterion	Weight	Goal	Green Monoprop LMP-103S	Biprop MMH/NTO	Green ACS	Cold Gas N_2
Prop Mass for Δv [kg]	0.3	Min	10.088	7.463	4.291	33.934
Normalized Value			*0.8044*	*0.893*	*1*	*0*
Time to Execute Δv [hr]	0.2	Min	6.25	1.562	140.958	20.833
Normalized Value			*0.9664*	*1*	*0*	*0.8618*
Peak Power [W]	0.1	Min	80	50	200	10
Normalized Value			*0.6316*	*0.7895*	*0*	*1*
TRL (0–1)	0.1	Max	0.9	0.95	0.85	1
Normalized Value			*0.3333*	*0.6667*	*0*	*1*
Safety/Toxicity	0.1	Min	0.2	0.8	0.2	0
Normalized Value			*0.75*	*0*	*0.75*	*1*
Complexity (1-3)	0.1	Min	2	3	3	1
Normalized Value			*0.5*	*0*	*0*	*1*
Integration Fit (0–1)	0.05	Max	0.9	0.8	0.9	1
Normalized Value			*0.5*	*0*	*0.5*	*1*
Tankage Fraction (dry/prop)	0.05	Min	0.15	0.25	0.25	0.5
Normalized Value			*1*	*0.7143*	*0.7143*	*0*
Totals:	**1**		**0.73109**	**0.649235**	**0.435715**	**0.62236**

Power Subsystem

The power subsystem provides generation, storage, and regulated distribution to the Venus bus and hosted payload. For the Blue Canyon Technologies X SAT Venus Class bus the primary source is a single deployable solar wing rated near 222 W and the secondary source is a rechargeable lithium ion battery listed at 10.2 Ah. The spacecraft uses a 28 VDC main bus consistent with small spacecraft practice [38]. Systems Tool Kit analysis for the service orbit yields a period of 1 hour 37 minutes 20 seconds with an eclipse of 34.411 minutes. The remainder of the orbit is illuminated. These geometry terms set the duty cycles that size array power during sun and battery energy during eclipse and they establish the cadence for state of charge recovery each revolution [39].

A realistic efficiency model is adopted to translate bus and payload loads into required generation and storage. The daylight path uses representative space grade converter efficiencies from vendor datasheets, a conservative harness efficiency, and literature values for lithium ion charge and discharge energy efficiencies [40]. The eclipse path applies the same distribution efficiency to battery discharge and adds the round trip penalty for battery charge in the next sunlit arc [50]. End of life array sizing references the specific power available after packaging factor, temperature, and incidence effects [41][42]. Battery sizing references a depth of discharge limit and assumes an end of life capacity fraction for margin. The lecture model provides a transmission efficiency target near 0.90 for the charge recovery path and that value is used here for conservatism [43].

Under these assumptions the average solar array power that must be supplied during illumination is approximately 132.5 W. The installed 222 W wing exceeds that requirement by about 90 W so the array can both run the spacecraft in sunlight and recharge the battery without stress [50]. On an energy basis the per orbit balance shows a surplus near 93.8 Wh after covering sunlit loads and fully replacing the eclipse removal. That headroom supports additional activities that are deliberately scheduled only in sunlight in order to keep heater duty and conversion losses modest. Battery capacity sized to the eclipse draw and the adopted efficiencies is approximately 8.35 Ah. The installed 10.2 Ah pack therefore provides margin and allows the depth of discharge to remain within life limits even after applying an end of life capacity fraction near eighty percent. The average charge power needed in sun to restore the eclipse energy is comfortably below the available array power, which confirms that the state of charge can be restored every orbit with margin.

Power Budget

Operations are analyzed in three modes that drive power allocation and scheduling see Table 10. In idle sunlight the payload draws about 35.65 W. In idle eclipse the demand rises to about 138.7 W as survival and line heaters dominate. During refueling the peak demand is concentrated in sunlight and totals about 217 W. The largest contributors in the refueling case are the foldable arm near 79.89 W [44][45]and the fluid pump near 38.3 W [46][47]with guidance and navigation near 22.55 W[48] and command and data handling near 0.9 W [49]. Communications is held at zero during transfer so that array power and thermal margin are reserved for the mechanism and the fluid system. Similar space operations were used to estimate operating power consumption [50]. Concentrating refueling in sunlight maintains a positive energy balance per orbit, keeps seal and tank temperatures within limits without excessive heater duty, and remains within the single wing capability identified for the Venus host.

Table 10. Power budget

Subsystem	Sun (W)	Eclipse (W)	Refueling (W)
Communications	12.2	12.2	12.2
Bus C&DH	0.9	0.9	0.9
GNC	22.55	22.5	22.55
Thermal	0	43.2	0
Power	0	0	0
Propulsion	0	0	80
Structure & Mechanical (Arm)	0	0	59.89
Fluid Transfer (Pump)	0	0	38.3
Payload Thermal	0	59.9	0
Payload C&DH	0.9	0	0.9
Total	36.55	138.7	214.74

Mass Budget

Creating the mass budget makes sure that the weight of all the systems does not exceed the values that the calculations for the other subsystems are based on. To guarantee this error margins and room for growth is factored in. This provides space for additional systems as the design gets further developed and any unknowns that may exist. The explicit margins can be seen in table 11. Team designed functions were calculated using mass times density equals volume. The results of this can be seen in table 11 and do confirm that the mission design is withing the constraints of the X-Sat Venus bus. Because all other maximum calculations were based on the maximum of the bus it verifies the other systems as well.

Table 11. Mass budget

Segment	Item	Mass (kg)	Margin	Allocated (kg)
Payload	Outer Structure Carbon Fiber 700	23,29	N/A	23.29
Payload	Hydrazine propellant	30	N/A	30
Payload	Tank	5.295	10%	5.8245
Payload	Arm structure	4.022	10%	4.4242
Payload	Prometheus Quick Connect	0.579	10%	0.6369
Payload	Propellant Pump	3	10%	3.3
Payload	Small subsystems margin (payload C&DH, sensors, valves, harness)	0.5	N/A	0.5
Total	Payload and its subsystems	66.686	N/A	67.6756

Risk Mitigation

Prometheus I risk management is concentrated on the five operations and threats that loom over mission reliability. Through the use of a possibility x impact risk mitigation matrix, autonomy faults, docking failure, contact damage, eclipse power deficit, and orbital debris were chosen as the primary risks. Table 12 lists the separate mitigations to address those risks: manual take over and hard emergency stop for autonomy faults, speed limited step and check approach for docking, compliant soft bumper and rounded edges to reduce contact damage during docking, sunlight-only refueling rule to avoid eclipse power deficit, and continuous screening with always on sensors to eradicate debris crashing accidents. These controls worked to transfer all of the risks from the red to green region in Table 13. It shows the expected movement in the matrix after these controls, with likelihood reduced for autonomy, docking, eclipse power, and debris, and consequence reduced for contact damage. This system explicitly connects risk reduction to design features and operations rules, and provides clear verification.

Table 12. Risk Mitigation Matrix

Risk	Mitigation
1) Autonomous failure	Add manual control and an emergency stop. If comms or software freeze for 5 seconds, auto-halt and back away to a safe pose.
2) Docking failure	Cap approach speed at 1 cm/s and move in short steps with a pause to check alignment. If latch fails twice, back out and retry later using a manual input
3) Damage during docking	Put a soft bumper ring around the end effector and round all edges.
4) Power not enough during eclipse	Refuel only in sunlight
5) Hit debris	Update GNC and C&DH subsystem with commands to make the GNC and sensors to be alert and always on

Table 13. Risk Matrix

		Impact				
		Negligible	Minor	Moderate	Significant	Severe
Possibility	Very Likely			1	4	
	Likely				2, 5	
	Possible		3			3
	Unlikely		2	1		
	Very Unlikely				4 5	

Future Work and Lessons Learned

Gantt chart is shown in figure 10 a that highlighting the planning and relationship between all of the significant work that was done since the SRR. The light blue bars represent the period of time that the team allocated for this work to be done. Thin lines between the bars visualize any events that had to be done before the next stage was able to be started. This chart displays the planning and communication required to bring the Prometheus I mission to this point.

D	Task Mode	Task Name	Duration	Start	Finish	Predecessors
1		**Milestones/Deliverab**	**29 days**	**Sun 7/6/25**	**Fri 8/15/25**	
2		SRR	0 days	Sun 7/6/25	Sun 7/6/25	
3		Preliminary design review	0 days	Fri 8/15/25	Fri 8/15/25	16
4		**Subsystem Analysis**	**34 days**	**Wed 6/18/25**	**Mon 8/4/25**	
5		Subsystems researc	19 days	Wed 6/18/25	Mon 7/14/25	
6		Requirments	6 days	Mon 7/7/25	Mon 7/14/25	
7		Trade studies	5 days	Mon 7/14/25	Fri 7/18/25	6
8		**Testing**	**9 days**	**Wed 7/23/25**	**Mon 8/4/25**	
9		STK	4 days	Wed 7/23/25	Mon 7/28/25	13
10		CAD Stucture/Str	4 days	Wed 7/30/25	Mon 8/4/25	7
11		**Budgeting**	**20 days**	**Mon 7/7/25**	**Fri 8/1/25**	
12		LINK/Mass/Power Budget Research	6 days	Mon 7/7/25	Mon 7/14/25	
13		LINK Budget	5 days	Mon 7/14/25	Fri 7/18/25	12
14		Mass Budget	5 days	Mon 7/21/25	Fri 7/25/25	7,12
15		Power Budget	5 days	Mon 7/28/25	Fri 8/1/25	9,7,12
16		**PDR development**	**18 days**	**Mon 7/21/25**	**Wed 8/13/25**	
17		CONOPS/Mission A	11 days	Mon 7/21/25	Mon 8/4/25	2,9
18		Addendum	9 days	Fri 8/1/25	Wed 8/13/25	17,13,14,15,7
19		Elevator Pitch	9 days	Fri 8/1/25	Wed 8/13/25	10
20		Presentation	9 days	Fri 8/1/25	Wed 8/13/25	
21		Requirments	11 days	Mon 7/21/25	Mon 8/4/25	2,6
22		Intro/Existing	11 days	Mon 7/21/25	Mon 8/4/25	2
23		Trade Study/Down Select	11 days	Mon 7/21/25	Mon 8/4/25	2

Figure 10. Penn Space Gantt chart

While bringing the mission to this point the three most critical technologies are the magnetic locking system, the collaborative interface, and the telescoping pole. The magnetic lock allows for effective and precise docking with less complications. The magnetic system aids in self-alignment rather than precise alignment and reduces the mechanical wear from an interlocking system. The interface is critical for any kind of effective fuel transfer as it gains simple access to the fuel tank rather than requiring a dangerous and invasive method. This increases the predicted success rate of the mission and removes unnecessary risk to the target satellite. Lastly, the pole gives extended reach to the payload while also remaining compact. This allowed the payload to maximize the fuel storage. Being compact aids the payload in stability as well as compatibility with both the Venus bus and the faring inside the Minotaur I.

The biggest challenges to this process were communication, realistic innovation, and research. Members of Penn Space were spread out over four time zones which made window for communication often small. Many parts of planning depended on each other and when broken up could struggle to proceed without information from another member. To mitigate this two time periods were set up for the hour periods that the most members could attend at one time. They would then alternate depending on need and availability to maximize participation. Another struggle was that of innovating in an advanced field while still trying to create a well researched and realistic plan. The solution to that was to make small but meaningful steps in the right direction. Instead of trying to reinvent the wheel Penn Space looked for inspiration in current design and looked for ways to improve, such as the more effective locking and fuel delivery system. Lastly research was the largest time consuming aspect of the mission. Finding reputable sources on specific qualities or data of already niche concepts proven difficult at times and sometime shaped the direction of the team. The method that proved the most effective was group research sessions. Having all team members research during meetings for similar results helped focus the team as well as increase the likelihood of finding credible results.

The next step of the design and verification would be the development of mock physical prototypes. By creating a small scale prototype of the payload, the team would be able to run more in depth practical tests of stress loads, thermal capabilities, and overall feasibility. It would give better insight into how the placement of systems would affect the success of the mission. Development of these could be done using 3D systems and coated or wrapped in the material of choice for that portion of the payload. Additional research into verification and subsystem requirements would also prepare the payload to go further along the process towards implementation.

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