Mars Surveyor and Communications Orbiters for Positioning and Exploration (SCOPE)

Task Order #5
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TEAM 5: Mars Surveyor and Communications Orbiters for Positioning and Exploration

Team 5 proposes SCOPE, a mixed-constellation design to meet the LASSI task order using two distinct bus configurations: OPTIMUS and PRIME. OPTIMUS hosts imaging and spectroscopic payloads to determine Mars's mineral make-up and PRIME carries precision radionavigation and communications relay equipment to support operation of other assets at or around Mars. The SCOPE mission provides **scientific data at unprecedented resolution and scale** using mission-ready technologies and achieves **global relay and navigational coverage of Mars**. The constellation configurations are chosen to achieve task order priorities at a minimized cost for the 5-year lifetime.

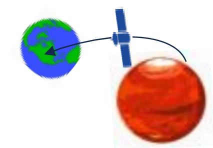
Figure 1a: Mission Summary

Science

Map and characterize minerals and ice on Mars

Comms Relay

Relay coverage at and near Mars to Earth for all missions.



Navigation

Global positioning of Mars for ascent/descent and ground missions.

Fig 1b: Driving Objectives

- Provide navigation for orbital and ground assets on ground or in orbit.
- Provide communications relay for orbital and ground assets on ground or in orbit.
- Determine the mineral make-up of Mars.
- The design lifetime shall be at least 5 years including test and operation.
- The mission shall include a bus capable of orbiting Mars.
- The mission shall include an imaging and spectroscopic payload.
- Propulsion requirements to achieve orbit at Mars shall be minimized.
- The installation of payloads late in the integration flow shall be facilitated.

Fig 1c: Task Order Requirements

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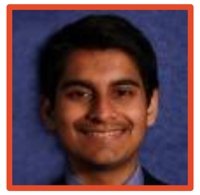


Ground Station
Pranav Banerjee

Figure 1d: Team Management

CONOPS

- Launch November 2026, followed by transfer
- Aerobraking & insertion into orbital plane(s)
- OPTIMUS maps ice and minerals in SSO, moving to polar orbit after 2 year
- PRIME provides precision positioning and communication relay
- Satellites decommissioned at end of mission

Trades

- Ground station coverage
- Flagship satellite trade
- Constellation configuration
- Interplanetary transit
- Material selection
- Propulsion architecture
- Launch vehicle selection

Develop

SCOPE will utilize COTS components to
- Increase TRL
- Lower cost and development timelines
- Flight proven = less risk

Non-COTS will be based on heritage items
- DSAC
- DSN
- Stereo cameras
- High-resolution camera

Figure 1e: Technical Approach

Cost

The following are expected to be major cost drivers for the SCOPE program:
- OPTIMUS payload sensors
- Deep Space Atomic Clock units Development and testing of bus and payload systems
- Launch vehicle, depending on launcher type and # of launches
- Power requirements for sensors and TT&C hardware necessitating larger systems overall

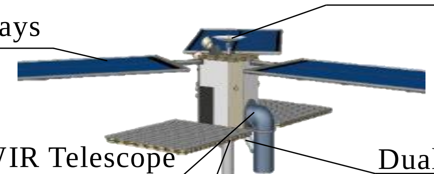
Figure 1f: Cost Drivers

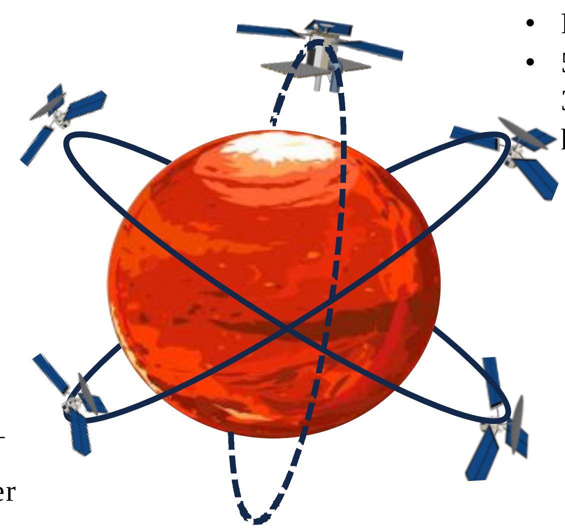
Launch Vehicle

Launch vehicles considered will be native to US launch sites and operational in the US.

OPTIMUS

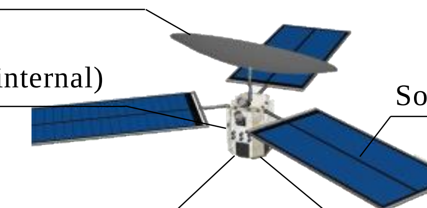
- Maps minerals, rocks, and ice on Mars
- 300 km by 92.7° sun-synchronous orbit in first two year obtains 75% coverage of Mars.
- Secondary, 300 km by 90° polar orbit reaches poles, achieving 99% coverage of Mars by year four.





PRIME

- Provides navigational positioning and communications relay.
- 5 shells, each at 55° inclinations equally spaced in RAAN, and 3 satellites in each shell. The PRIME constellation achieves high-precision 4-fold coverage of 100% of the surface.



Ground Control

The Deep Space Network receives and uplinks data as well as distributing received data to stakeholders.

Figure 1g: Mission Design

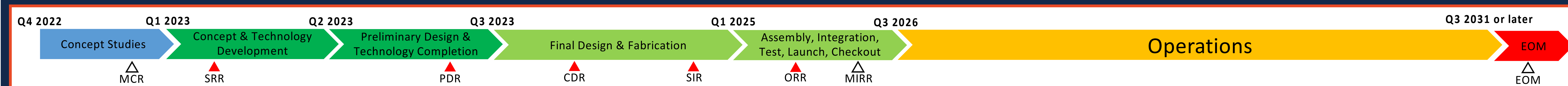


Figure 1h: Timeline

2. Executive Summary

SCOPE is a mixed-constellation design to satisfy the LASSI task order of determining the mineral composition of Mars and standing up communication and navigation systems 5 years or more period. This is done via two distinct bus configurations: **OPTIMUS** hosts imaging and spectroscopic payloads to determine Mars's mineral make-up and **PRIME** carries precision navigation and communications relay equipment to support operation of other assets at or around Mars. Together, these satellites satisfy all mission requirements set forth.

In Figure 1b, the major driving objectives are split into three key operational goals: science, communication relay, and navigation based on the LASSI task order requirements as reflected in Figure 1c. The SCOPE mission achieves 99% coverage of Mars to determine the mineral make-up of the planet. To support orbital and ground assets on or around Mars, SCOPE acts as a communications relay to Earth for all future Mars missions. SCOPE also provides global positioning of Mars to support the same assets.

Figure 1e describes our technical approach to the task order. To facilitate a low-cost option for the provider, SCOPE utilizes COTS components which lower the cost and development timeline, provide already tested components, and increase TRL. Any developed items to support the mission are based on heritage items and sized appropriately. Major mission critical systems undergo a rigorous trade study to find the optimal cost and design.

Cost drivers in Figure 1f include a high-power requirement, which may manifest as an increase in mass and hardware needed. The costs of launching space segment components, the development and testing of bus and payload systems, and use of custom payload and instrument hardware are also expected to contribute significantly to costs. Design choices take account for these drivers to minimize cost.

Figure 1g shows the design of the SCOPE mission comprises ground segment, launch vehicle, OPTIMUS, and PRIME as major systems of systems. Ground segment enables downlink of science data and Mars-to-Earth communication relay as well as uplink of commands, synchronization messages, and Earth-to-Mars communication relay. The launch vehicle delivers space segment components on a trajectory towards Mars. OPTIMUS satellites focus on mapping of minerals and ice in support of scientific goals, hosting an array of remote sensing instruments, and surveying from two distinct orbits to maximize coverage. The PRIME constellation carries dedicated radio communication equipment and an atomic clock to satisfy requirements for navigation at Mars and relay between Earth and assets on and near Mars.

Figure 1h illustrates the mission timeline and shows major milestones and phases. Key dates include launch in 2026 and planned end of mission is 2031 or later.

Figure 2: OPTIMUS payload summary.

Payload	Characteristics
P-Band SAR / Dual Frequency Sounder	1st SAR imaging of Mars at 30 m spatial resolution. In Sounder mode, dual frequencies probe up 20 m depths of buried ice at 1 m vertical resolution, or down to 40 m at reduced resolution. Order of magnitude increase in sounding resolution.
SWIR/LWIR Spectrometers	Hyperspectral SWIR sensor at 10 nm spectral resolution and 5 m spatial resolution. Multispectral LWIR sensor (1 µm bandpass) over 20 channels and 10 m spatial resolution. Shares a common telescope to reduce size, and maps spectral response of minerals and rocks.
High Resolution Color Imager	Visible and VNIR imaging, 20 channels (10 nm bandpass) for high-res (1 m) imaging. Spot checks and provides context to all other data.
Dual Stereo Cameras	High resolution (15 m) stereo imagery for digital elevation models, improved topography, and radar clutter mitigation.

3. Mission Overview

SCOPE fuses a science mission and utility mission into one using OPTIMUS and PRIME orbiters. SCOPE meets mission requirements and stakeholder values with an effective concept of operations and mission timeline while appropriately addressing relevant risks.

This section provides an overview of the SCOPE mission. In addition to the overview provided in the executive summary, this section goes into details regarding requirements, waivers, the concept of operations, the mission timeline, the interface control diagram, and relevant risks.

3.1. Mission Requirements

The mission requirements described in Figure 3 define the science mission that SCOPE is to achieve. These requirements are satisfied using a science orbiter, OPTIMUS.

The SCOPE mission architecture, through OPTIMUS, satisfies the mission requirements of determining the mineral make-up of Mars and inclusion of imaging and spectroscopic payloads through onboard science instruments such as the SAR sounder, LWIR/SWIR spectrometers, high-resolution color imager, and dual stereo cameras, all of which study the Martian surface. The OPTIMUS satellite is equipped with electric propulsion, which minimizes propulsion requirements to achieve orbit at Mars in conjunction with a low-thrust trajectory design.

Installation of payloads late in the integration flow is enabled by the design of the bus, where science payload instruments are mounted externally. Installation of the payload is therefore minimally constrained by the integration flow. As later discussed, the bus is designed with sufficient propulsion, power, storage, communications, and attitude control for operation in Mars orbit, and these characteristics are appropriately sized such that OPTIMUS can carry out its mission for a minimum of 5 years.

Figure 3: Science mission requirements

Identifier	Requirement
MR-05	The mission architecture shall determine the mineral make-up of Mars.
MR-07	The mission architecture shall enable a minimum design lifetime of 5 years, including ground test and operations.
MR-08	The mission architecture shall include a bus design capable of orbiting Mars.
MR-09	The mission architecture shall include an imaging payload.
MR-10	The mission architecture shall include a spectroscopic payload.
MR-11	The mission architecture shall minimize propulsion requirements for achieving orbit at Mars.
MR-13	The mission architecture structure shall facilitate installation of payloads late in the integration flow.

Additionally, utility mission requirements are specified to enable navigation and communications relay from Mars. These requirements are part of the SCOPE mission and inform the design of the communications orbiter, PRIME, but for the purpose of this report, emphasis is placed on the design of the science orbiter, OPTIMUS.

3.2. Waivers

Six waivers are obtained from the SCOPE mission stakeholder. The first waiver dismisses the radio communications requirement which states, “All space-to-ground radio communications shall be via ground stations located in the United States.” The second waiver extends the development cycle from 24 months to 36 months. The third waiver

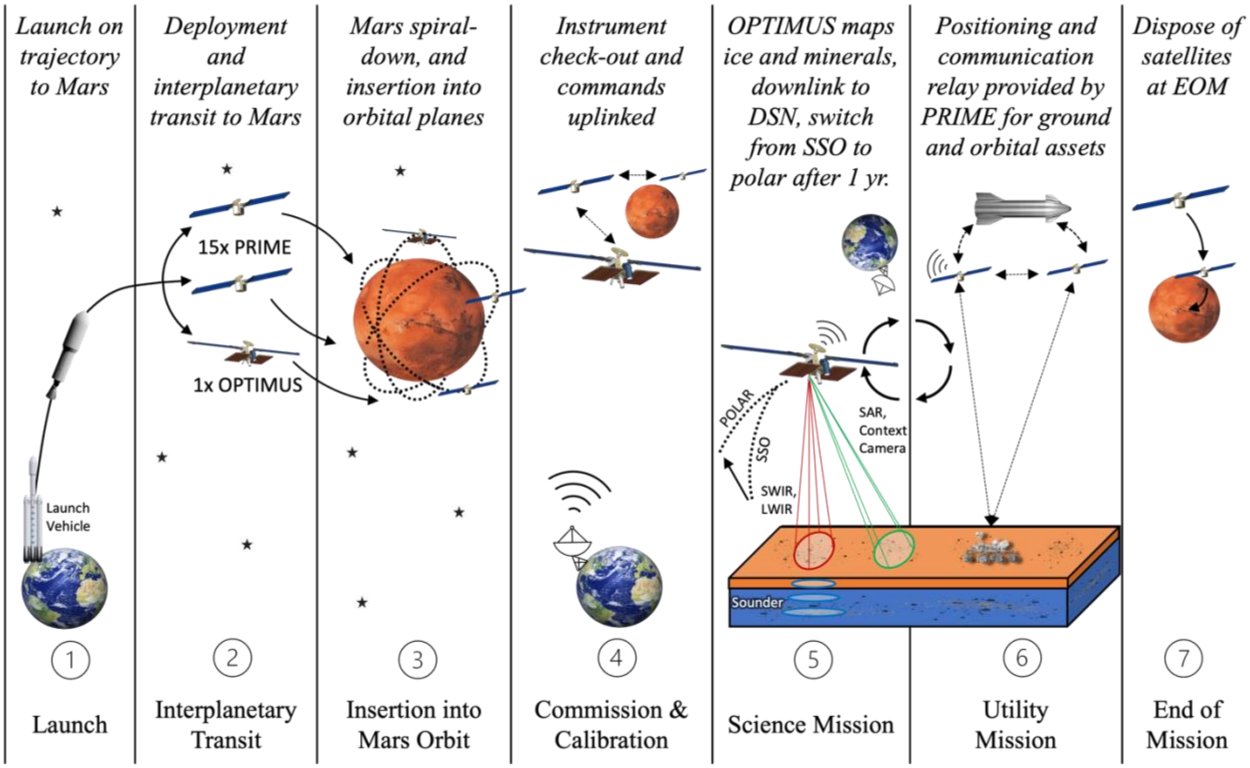


Figure 4: Concept of Operations

dismisses the requirements that acquired payload data must be delivered to the ground within a day of acquisition. This is not possible from Mars orbit without communication relay satellites placed at Lagrange points, which would be extremely costly. The fourth waiver clarifies stakeholder values and dismisses the requirement of 100% Mars coverage. This requirement is not stated in writing in the RFP, but it was imposed by the offeror in the SRR. The fifth waiver modifies the lifetime requirement, which states, "The satellite shall have a 5-year lifetime." This waiver modifies the requirement such that the lifetime is *at least* 5 years. The sixth waiver allows the exclusion of navigation and communications relay mission from the PDR.

3.3. Concept of Operations

SCOPE's concept of operations is shown in Figure 4. SCOPE begins with launch in November 2026, which sends a cluster of satellites on a trajectory to Mars. It is anticipated that more than 1 launch will be necessary to place all satellites in Mars orbit. SCOPE consists of 15 PRIME utility satellites and 1 OPTIMUS science satellite, all of which are deployed from launch vehicle on an interplanetary trajectory toward Mars, which will last approximately 10-15 months.

After this, the satellites spiral down to Mars orbit and are inserted into their designated orbital planes. Once this is achieved, instruments aboard OPTIMUS and PRIME are checked out, with commands uplinked. OPTIMUS uses its instrument payload to map ice and minerals, and PRIME provides radionavigation signals and communications relay for assets on and near Mars. Communications and science data are relayed between the constellation and Earth via the Deep Space Network. At end of mission, satellites are deorbited using onboard propulsion, for which propellant is kept in reserve.

3.4. Mission Timeline

The SCOPE mission adheres to a NASA Robotic Mission Project Lifecycle, with key design points and dates identified in **Error! Reference source not found.**. This includes Pre-Phase A through Phase D of the timeline and major reviews during the mission lifecycle. Also included is the 5-year operations period of the constellation, and end of mission disposal on the Martian surface.

Work continues with preliminary design for the mission until the third quarter of 2024, followed by final design and fabrication until the end of 2025. Before final design and fabrication is completed, assembly, integration, and test is expected to begin for some subsystems at the beginning of 2025. This will conclude with final checkout and launch in late 2026 of all satellites. Once launched, the mission enters the 5-year-minimum operations period. This includes a heliocentric transfer orbit, spiral down and insertion, inclination transfer, and the on-orbit operations phase. During transfer to Mars, satellites are commissioned, and instruments are checked out prior to the commencement of on-orbit operations. The OPTIMUS orbit change occurs during the on-orbit operations phase. At End of Mission (EOM) all satellites in the constellation are retired. Satellites will be decommissioned by deorbit.

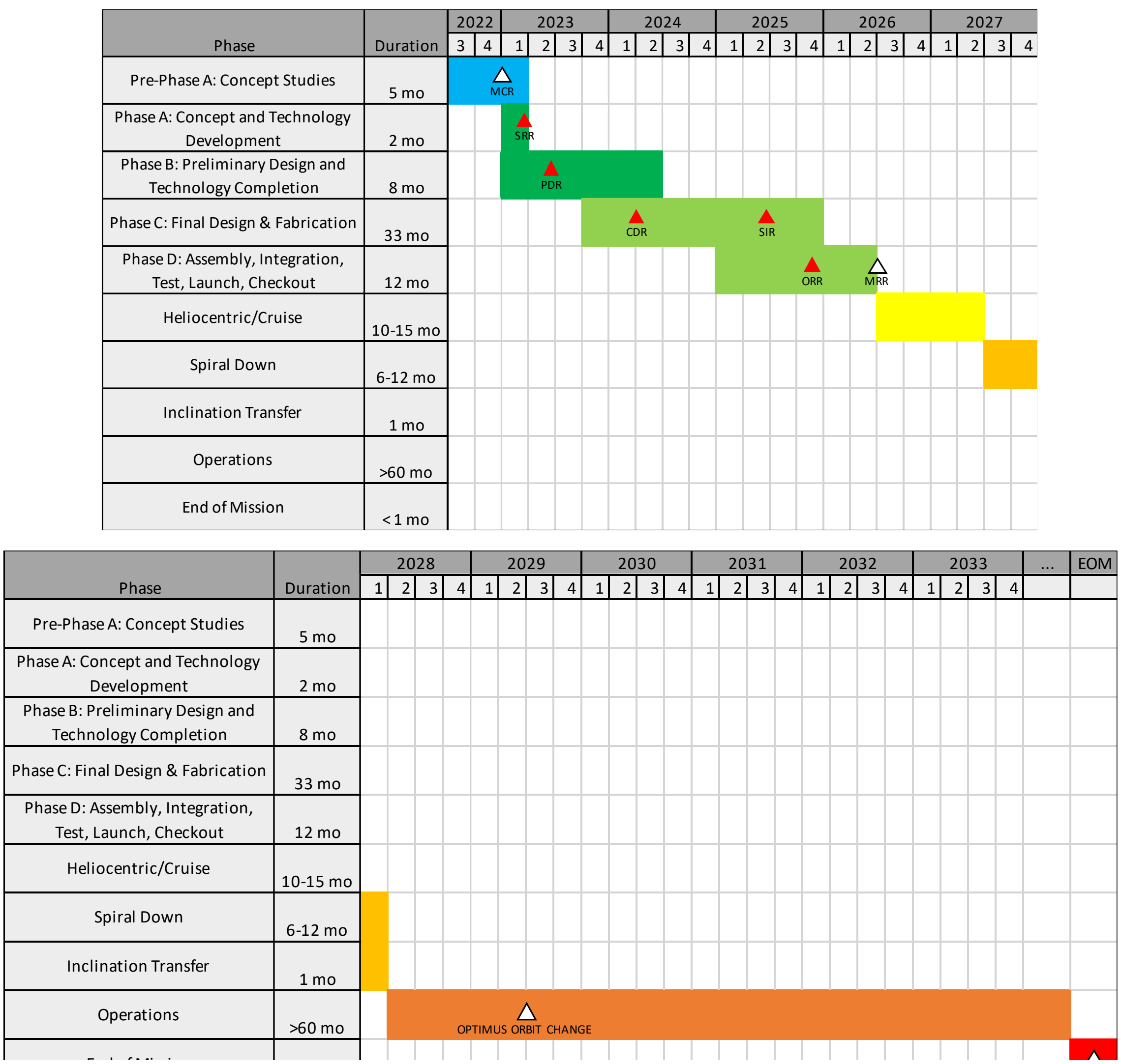


Figure 5: Mission Timeline

3.5. Risk

The following table in Figure 6 contains five major risks which if not addressed could severely affect the success or timeliness of the SCOPE mission.

These risks can be represented in terms of their respective likelihood and consequence in a risk matrix. Mitigations are necessary to address these substantial managerial and technical risks. The associated mitigations are then applied to address each risk. After mitigations are considered, the risk matrix can be used to show that each risk is reduced to tolerable levels.

4. Systems Design

SCOPE minimizes lifecycle costs while meeting all task order objectives. This is achieved through optimization in the design of the SCOPE mission and in the selection of components and systems used to achieve these objectives.

The design of the systems of systems—launch vehicle, ground segment, OPTIMUS satellite, and PRIME satellites—are presented. For OPTIMUS, the design process for subsystems is also specified. An emphasis is placed on the payloads carried by OPTIMUS in meeting the task order objective of Mars mineral mapping.

Figure 6: Table of risks with risk matrix before and after mitigations.

		Consequence				
		1	**2**	**3**	**4**	**5**
Likelihood	**5**	**B**				
	4					
	3					
	2		**C**			
	1	**E**	**D**	**A**		

		Consequence				
		1	**2**	**3**	**4**	**5**
Likelihood	**5**			**B**		
	4					
	3				**A,C**	**E**
	2					
	1					**D**

ID	Risk	Mitigation(s)
A	Given that ASCENT monopropellant for the RCS thrusters has only been flown once, there is a medium probability that the technology will not mature in time for launch, adversely impacting completion of the RCS system, thereby leading to late delivery of hardware.	A date is set by which ASCENT must be ready. If not ready by this date, the system will switch to hydrazine.
B	Given that Mars experiences extreme, large-scale dust storms, there is a high probability that a dust storm will occur, adversely impacting remote sensing of the Martian surface, thereby leading to a failure to achieve science objectives.	A waiver is obtained to permit operation beyond 5 years such that more time is allotted for collecting science data.
C	Given that testing facilities have limited capacity, there is a medium probability that instrumentation testing may not be available as planned, adversely impacting the time needed to complete environmental testing, thereby leading to late integration of the payload.	Time is built into the development and testing schedule to account for delays in testing.
D	Given that the Falcon Heavy extended fairing has not yet been flown, there is a low probability that the fairing will not be available by the planned integration time, adversely impacting the completion of payload integration, thereby leading to delayed launch.	A date is set by which the fairing must be ready. If not ready by this date, the mission will switch launch configurations.
E	Given that attitude control sensors are exposed to extreme environments, there is a medium probability of premature failure, adversely impacting the ability to determine orientation, thereby leading to loss of control.	Redundant star tracker and sun sensors are installed.

4.1. Interface Control Diagram

The interface control diagram in Figure 7 provides an overview of the interfaces between the major systems–satellite, ground segment, launch vehicle–as well as subsystems, such as bus subsystems and payload.

Interfaces detailed in the diagram include thermal, structural, data, and power interfaces. Subsystems include structure, C&DH, propulsion, ACS, power, thermal, communications, and specific instruments as well as those subsystems associated with the ground segment and launch vehicle.

4.2. Launch Vehicle

By virtue of SCOPE's mission design of placing a constellation in Mars orbit, heavy lift launch vehicles are considered for the mission. These include the Blue Origin New Glenn; SpaceX Falcon 9, Falcon Heavy, and Starship; and the ULA Vulcan Centaur are primarily considered. While the Space Launch System is also suitable for this mission, it is not considered, as it will not be possible to acquire a launch within the timeline. Basic data for these vehicles are shown in Figure 8.

At the time of this report, only the SpaceX Falcon 9 and Falcon Heavy have successfully flown, and only the SpaceX Falcon Heavy can complete the mission in 2 launches–all other vehicles require more launches based on payload mass or volume constraints. New Glenn is unlikely to be able to beat this number of launches as it is designed for reuse, and Starship will require additional refueling missions. As a result, based on payload capability and cost per launch SpaceX Falcon

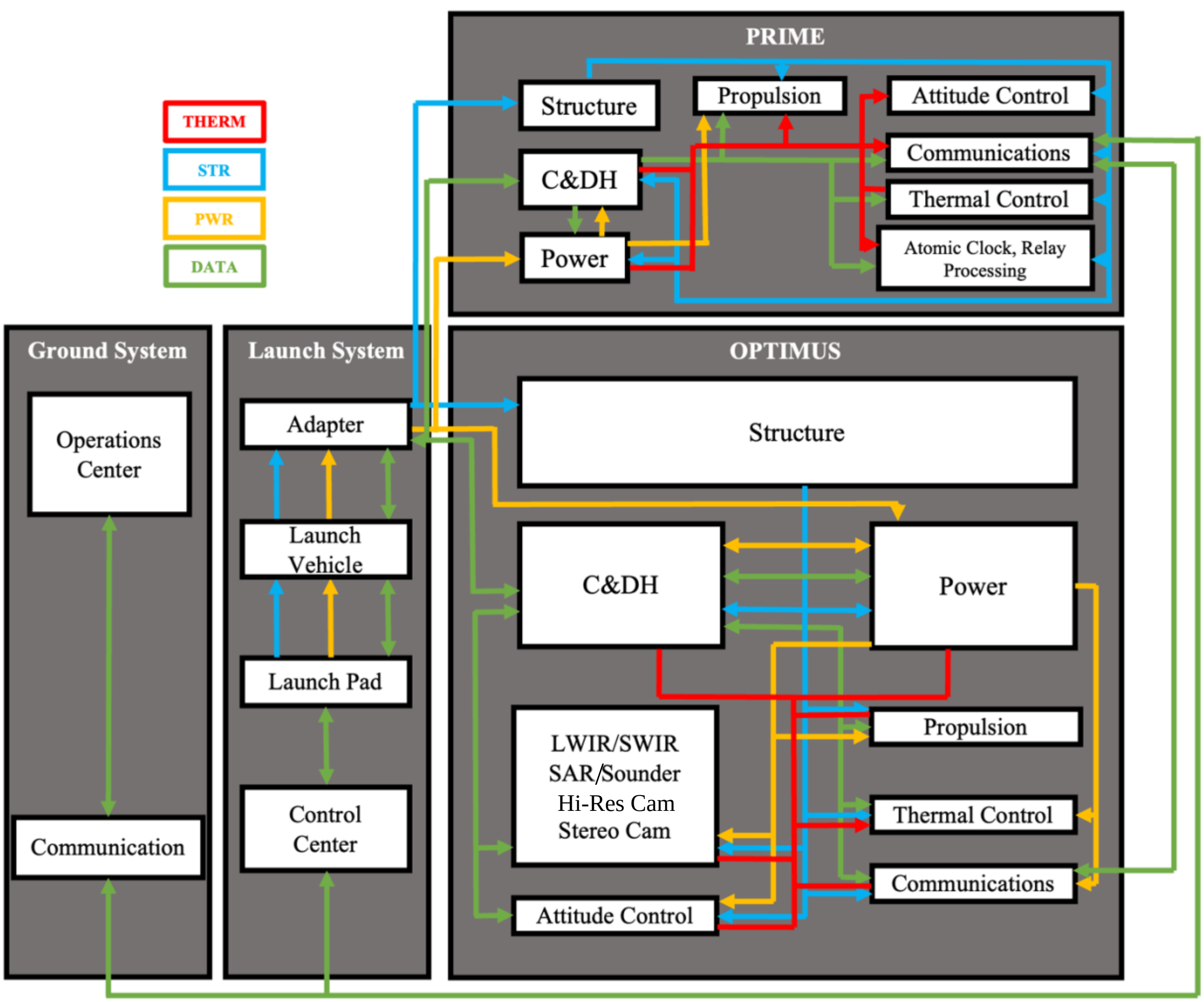


Figure 7: Interface Control Diagram

Figure 8: Table of basic data for launch vehicles under consideration for SCOPE.

Vehicle	New Glenn	Falcon 9	Falcon Heavy	Starship	Vulcan
Mass to Mars	?	4,020 kg	**16,800 kg**	100,000 kg*	8,810 kg
Fairing L x D	21.9x7 m	16.6x4.6 m	**16.6x4.6 m**	22x8 m	21.3x5.4 m
Quantity	2+	8	**2**	1*	5
Flights	0	217	**5**	1	0

Heavy appears to be one of the most cost-effective launch solutions. For these reasons, the mission is designed around the capabilities of the SpaceX Falcon Heavy, which provides 16,800 kg of payload capability to Mars in an expendable configuration and has a payload fairing 16.6 m long (12.1 m cylindrical section) and 4.6 m diameter.

4.3. Ground Segment

For OPTIMUS and PRIME to communicate with the Earth, a suitable ground station is required to close the link. After some research regarding interstellar communication, specifically for Mars, the Deep Space Network (DSN) was chosen. We acquired a waiver so that we could use the other 2 DSN ground stations in Canberra, Australia and Madrid, Spain. Using all three DSN stations allows for nominal communication with the satellites. We are utilizing the 70-meter dish so that the link requirements are closed. We found that the best and worst-case margins for uplink are 54.36 dB and 37.65 dB respectively. The downlink best and worst-case margins were 20.07 dB and 3.36 dB. These margins allowed for the link to close when communicating with PRIME satellites.

4.4. Orbital Probe Targeting Ice and Minerals for Utilization Someday (OPTIMUS)

To guide payload requirements definition, we asked ourselves what we could learn from obtaining a comprehensive mineral map of Mars and how that knowledge could be leveraged by future Mars missions. Gaps in coverage and resolution left behind by heritage missions such as the Mars Reconnaissance Orbit and Mars Express also serve to motivate payload selection for OPTIMUS. Considering these objectives and the requirements in the RFP, our team has formulated a set of 5 mission drivers for the OPTIMUS orbiter.

1. **Broaden coverage and refine spatial and spectral resolution of the current Mars ice and mineral data library.**
2. **Constrain the nature of the Martian Cryosphere, and how ice deposits are linked to the climate over time.**
3. **Interpret crustal characteristics to decipher the transformation of the surface environment over time.**
4. **Determine the extent of subsurface ice and other resources critical to in-situ resource utilization efforts.**
5. **Assess the possibility, including past and present, of life on Mars.**

OPTIMUS provides six key payload capabilities to tackle these objectives: Synthetic aperture radar, radar sounding, short and long wave infrared spectroscopy, high-resolution imaging, and stereo imaging. OPTIMUS achieves and maintains a 300 km by 92.7° sun-synchronous orbit, obtaining 75% coverage of Mars within the first year of operation. After year one, OPTIMUS shifts to a 90° polar orbit to survey the poles, achieving 99% coverage of the entire planet.

To support these payloads, solar arrays and battery storage ensure a reliable power supply, and a high gain antenna ensures communications with the PRIME relay orbiters. Control moment gyroscopes and reaction control thrusters maintain attitude control, and a propulsion system performs orbital maneuvers and maintenance. The subsystems are held together by a robust bus architecture, and a radiation hardened Command and Data Handling system manages upkeep, data storage, and command distribution.

4.4.1. Orbital Analysis

For each satellite, the orbital analysis can be broken up into two primary components. The first is the transfer trajectory. This includes the interplanetary transfer between Earth and Mars, as well as the insertion into the target orbit once in the Mars sphere of influence. Factors driving the transfer trajectory are the propellant mass required to complete the transfer, duration of the transfer, and the length and frequency of launch windows.

Each of these factors are primarily dependent on the type of propulsion system used in the transfer. A chemical propulsion system enables high thrust burns and thus quicker maneuvers. It is these high thrust burns which enable Hohmann transfers. On the timescale of the transfer, chemical propulsion can provide drastic changes in velocity and momentum instantaneously. This is a particularly important factor when considering a Mars orbital insertion. Spacecraft using chemical propulsion approach Mars with an excess velocity above 2 km/s. This means spacecraft only have hours to perform the necessary maneuvers to shed this excess velocity and insert into a bound Mars orbit. The high thrust enabled by chemical propulsion makes this possible. The downside to chemical propulsion is its low mass efficiency. The propellant mass required for these orbital maneuvers is much, much higher than other means of propulsion.

An electric propulsion system is the inverse of chemical propulsion in many aspects. Electric propulsion can only provide incredibly low thrust with incredibly high mass efficiency. As a result, electric propulsion systems cannot perform the same maneuvers as chemical propulsion. Hohmann transfers and other transfers which assume instantaneous burns are no longer possible. These burns require the spacecraft to approach Mars with incredibly high excess velocities. Electric propulsion is not capable of shedding excess velocity and entering a bound Mars orbit on the required timescale. Instead, electric propulsion requires a transfer trajectory that approaches Mars with almost no excess velocity. These transfers take much more time to complete, and the propulsion system must be providing thrust for a majority of the transfer. Electric propulsion does have a much higher mass efficiency than chemical propulsion and thus has a much lower propellant mass required for orbital maneuvers.

An electric propulsion system is used on both OPTIMUS and PRIME. Specifically, a Hall Effect Thruster is used. The transfer trajectory analysis is broken up into an interplanetary transfer, where the satellites "spiral out" from Earth to Mars and a capture spiral resulting in insertion into the target orbit of each satellite.

Due to the low thrust provided by the Hall Effect Thruster, the thruster must be firing for most of the transfer. Rather than the elliptical trajectory seen with a Hohmann transfer, this causes the trajectory to look more like a spiral out from Earth to Mars. A sample trajectory is shown in Figure 9. Selecting when the thruster operates in the transfer trajectory is an optimal control problem. The optimal control scheme is also dependent on the launch and arrival dates. Solving the optimization problem for a set of launch and arrival dates required a license tier for FreeFlyer which was not available to the team. Instead, data was taken from runs of JPL's Mission Analysis Low Thrust Optimization tool for an Earth-Mars transfer using a Hall Effect Thruster with similar thrust. These simulations were performed for a Mars science constellation using electric propulsion as a mission concept for a decadal survey.

The interplanetary transfer takes between 10-15 months, with a maximum ΔV demand of 4 km/s. In total, this requires 382.47 kg of propellant. The minimum ΔV is 3.5 km/s. Once again, the transfer time and ΔV depend on the launch date. An optimal launch window occurs in November 2026. The consequence for launching outside an optimal window is much, much lower than that of chemical propulsion. Only a small increase in propellant mass is needed to launch outside this window. Optimal launch windows occur about as frequently as Hohmann transfer launch windows, but the windows themselves are wider.

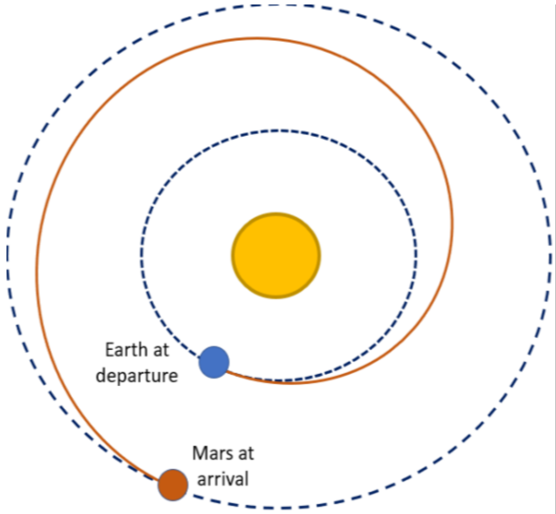


Figure 9: Sample low thrust transfer.

The interplanetary transfer ends when the spacecraft enters Mars's sphere of influence. The spacecraft enters Mars's sphere of influence with no hyperbolic excess velocity. Once within the sphere of influence, the Hall Effect Thruster begins firing to reduce the velocity of the satellite, allowing it to quickly enter a bound orbit.

The thruster fires continuously, except for when eclipsed by Mars, to lower the semimajor axis of the orbit. The maneuver takes about 6-9 months depending on the target orbit. PRIME satellites operate in a higher orbit and enter their target orbit much quicker. OPTIMUS operates at a much lower altitude and takes longer to enter the target orbit. The maximum ΔV demand for insertion into OPTIMUS orbit is 3 km/s, requiring 286.85 kg of propellant. Figure 10 shows a sample trajectory for a Mars science orbit insertion.

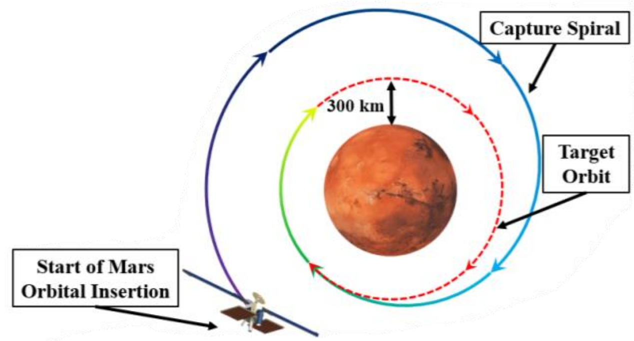


Figure 10: Example Mars capture spiral.

The second component of the orbital analysis is the target orbit the satellite operates in. The driving factors for the target orbit include power generation requirements, restrictions on the orbit from instrumentation, communication windows, and perturbations affecting orbital maintenance. With the orbit configuration about to be discussed, only a single satellite is needed to fulfill the science mission objectives.

The lowest altitude of any satellite is well into Mars's exosphere, where atmospheric particles travel about 11 km on average between collisions. At 300 km altitude, atmospheric drag over the lifetime of the mission can be accounted for with very little orbital maintenance. The dominant perturbation by far is the J2 gravitational perturbation. This is used to design a Sun-Synchronous orbit for the satellite to operate in. To account for the gravitational perturbations, approximately 800 m/s ΔV is budgeted for orbit maintenance. This budget also accounts for the end of mission procedures to deorbit.

The fidelity of some of the scientific data measured by the instrumentation is dependent on the proximity to the Martian surface. These are primarily instruments with fine spectral resolution. Due to the number of channels in some instruments, the altitude is restricted to 300 km above Mars's surface. Increasing the altitude, while maintaining spectral and spatial resolution would require massive increases in data generation. The science orbit is circular to increase the frequency of opportunities to make progress towards scientific objects. If the orbit had a high enough eccentricity to make a noticeable impact on any aspect, the satellite would be too far away from the surface for the instrumentation to work properly for the vast majority of each orbit. When the satellite is close enough, it would spend much less time at the desired altitude and be moving at much higher speeds. For that reason, the orbit is circular. Most of the remote sensing instruments on OPTIMUS are passive, meaning the source of illumination is the Sun. To improve the consistency of the illumination, a Sun-Synchronous orbit is used, so the satellite operates at an inclination of 92.7° with respect to the Mars equator.

OPTIMUS operates in this orbit for 2 Earth years, then moves to a polar, circular orbit. OPTIMUS covers much of this area in just 5-6 months after beginning science operations but stays in this orbit because the orbit has high revisit frequency. There will be a small ~5.5° gap at each pole with this orbit which can't be covered in this configuration.

After 2 years, OPTIMUS begins an inclination transfer to a polar orbit. The inclination transfer takes 3 weeks and requires a delta-V of about 250 m/s.

The motivation for operating in a polar orbit is to obtain measurements of the Martian poles. The poles are of interest due to the presence of ice water and carbon dioxide. In this orbit, there are short periods where there may not be any incident solar radiation on the solar panels. As a result, this orbit is used in the sizing of the power subsystem rather than the first orbit.

Using a.i. solutions, Inc. FreeFlyer to perform orbit simulations, analyses of science orbits were performed over the mission lifetime. Figure 11 displays the simulated coverage of each instrument over the mission lifetime.

Figure 11: Instrument lifetime coverage.

Instrument	Coverage [%]
P-Band SAR	99.58
Sounder	97.81
SWIR payload	11.71
LWIR payload	77.26
High-Res Color Imager	11.71
Dual Stereo Cameras	99.71

These values represent minimum expected coverage due to the nature of the FreeFlyer simulation. Coverage values were determined by whether instrumentation fields of view pass over points on the Martian surface. With the available FreeFlyer license, the number of points is limited to a maximum of 25,000. The fraction of Mars's surface coverage by each instrument increases with the number of points. Decreasing the time step of the simulation would also result in improved simulated coverage. Decreasing the time step allows more opportunities for a point to appear in an instruments field of view. Figure 12 shows a simulation of the coverage for the P-Band SAR.

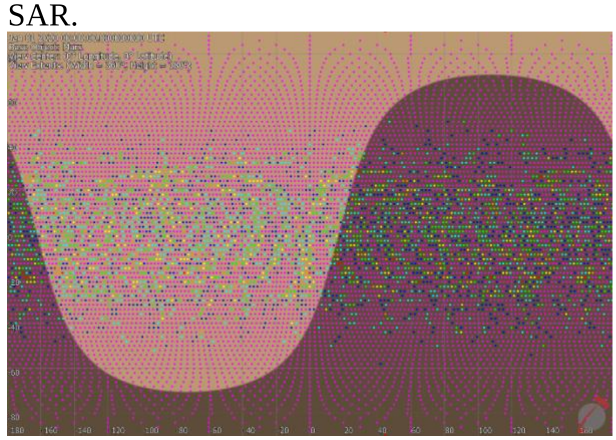


Figure 12: SAR Coverage and Revisit. Pink dots indicate 6+ passes, blue is 5, aqua is 4, green is 3, yellow is 2, orange is 1, and red is 0 passes.

The figure shows how many times each point on Mars's surface is observed by the P-Band SAR over this mission lifetime. The simulation shows high revisits of the Martian poles (pink). There are less revisits of points closer to the equator, but most points still have multiple observations.

Across the polar and sun-synchronous orbits, orbit maintenance requires 800 m/s of delta-V. The orbit maintenance and inclination change require a net propellant mass of 95.62 kg of propellant. The required propellant mass of each satellite for the entire mission lifetime is 764.93 kg.

The PRIME orbits are motivated by the mission requirement to provide full coverage for communication relay and navigational services. To provide navigational resources, PRIME must provide fourfold coverage of the Martian surface. With fourfold coverage of the surface, it is guaranteed that at least four satellites are visible to assets near Mars. Technically, only threefold coverage is required to provide navigational services, but this requires that the asset have a high enough precision clock. By providing fourfold coverage, PRIME removes this requirement on Mars assets.

The orbits for PRIME are primarily motivated by minimizing the number of satellites required to provide fourfold coverage. A secondary factor is the altitude of each satellite. Higher altitudes require more power for communication between Mars assets.

PRIME operates in 5 shells, each with 3 satellites. These shells are circular orbits at an altitude of 15,000 km and 55° inclination.

Using FreeFlyer to perform analyses on orbit configurations, PRIME guarantees fourfold coverage of every point on the Martian surface for 4 years of operation in Mars orbit.

4.4.2. Structures

The OPTIMUS satellite bus is a 1.65 m x 1.4 m x 1.9 m rectangular prism with chamfered edges. OPTIMUS has three solar arrays with an area of 19.151 m^2 each. The bottom of the satellite has six SAR/sounders which attach and can fold up. This design, as shown in **Figure *13*: Transparent view of the OPTIMUS bus, external systems are hidden for better viewability.**Figure 13, is the smallest configuration possible while incorporating all internal subsystem components. The shape of the structure is influenced by the propellant systems (for ACS and propulsion) and the externally mounted components of various subsystems.

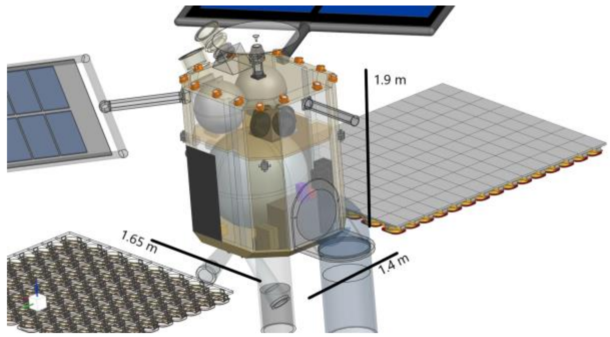


Figure 13: Transparent view of the OPTIMUS bus, external systems are hidden for better viewability.

The structures subsystem is responsible for accommodating and protecting the subsystems and their components both internally and externally to the satellite bus. The structure must consider several factors such as the weight of the payload, radiation, thermal, and mechanical loads during launch and operation, and the environment. In addition to these conditions, the structure and subsequent mass distribution impacts the position and orientation of the satellite, view of the surface, and how effective the solar panels are. Special consideration must be taken for requirements imposed by other subsystems with regards to environmental protection or location.

Figure 14: Table of materials and their property values.

Material	Density $[\frac{kg}{m^3}]$	Elastic Modulus [GPA]	Yield Strength [MPa]
Aluminum	2,640 – 2,810	70-80	455
Carbon Fiber	1750	228	7000

CFAH	20 - 163	560	1315
Kevlar	1380	76	1240

The material composition of the satellite bus is composed of carbon fiber aluminum honeycomb (CFAH) plates with aluminum 6061 hardware to support assembly and mounting for structural components. The selection of CFAH arises from the necessity to select a strong, durable, lightweight structure that can withstand initial launch to end of life conditions. When comparing CFAH plates to kevlar, aluminum, and carbon fiber, as shown in Figure 14, the result shows that the plates introduce the benefits of both materials:

- A high yield strength from carbon fiber
- A high elastic modulus from aluminum

CFAH also provides a low-density option with a high elastic modulus, indicating considerable stiffness and higher resistance to deformation under loading, and a high yield strength, indicating a large upper limit before permanently bending. These factors result in a high strength-to-weight ratio reducing the material required while maintaining structural integrity.

Calculating the buckling force required is defined the critical buckling load of the structure based on the moment of inertia, modulus of elasticity, length of structure, and a factor K which is 1 for non-pinned columns. Utilizing the moment of inertia for the folded-up design (as launch loads are the largest force experienced), the 1.9 m bus height, and modulus from Figure 13. The result is a critical buckling load of **2,495.56 kN** which can withstand the launch loads and the 0.66 N from the prop system.

OPTIMUS' structure must be able to withstand extreme temperatures and radiation levels. Several radiation challenges can appear as galactic cosmic rays, solar particle events, and secondary radiation from the Martian atmosphere. CFAH was also selected due to its high radiation resistance. Radiation in orbit is around 0.3 mSv/day, similar to MRO values. The CFAH paneling brings down the average dosage rate in conjunction to radiation shielding provided to the instruments. This accounts for protection of systems within and outside of the bus.

With regards to thermals, radiation absorption can lead to high heat values which require additional thermal control methods. CFAH has high thermal conductivity which helps dissipate heat generated by radiation absorption. The heat emitted from OPTIMUS is a function of solar heating, albedo, how much OPTIMUS generates, and how much temperature is stored. IR heating from Mars is ignored in this equation as Mars has no significant internal heating and thus the results are negligible.

Hardware and mechanisms are rated to operate without detriment for atleast the same time as the propulsion system, 17520 hours.

Internal components each have various ideal temperatures of operation, a common ground was selected for the internal idle temperature of the satellite. To maintain 25°C, active and passive methods are implemented for thermal control. Thermal louvers are thermally activated shutters that regulate how much heat the louvered surface can dissipate. This passive method is represented by the panels flanking both sides of OPTIMUS and underneath the HAL effect thrusters, see Figure 15.

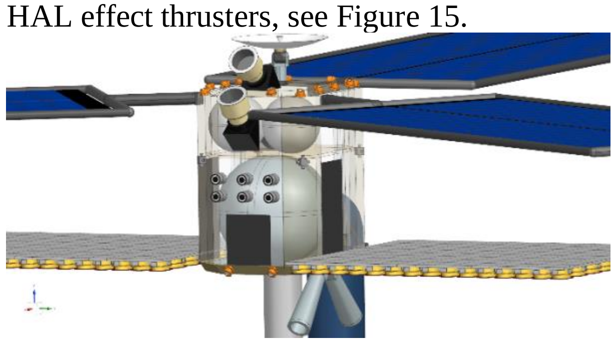


Figure 15: Rear three-quarter view of satellite bus.

Additional methods such as white paint to decrease absorptivity and reflective gold where white paint would be ineffective or shadowed can be seen in Figure 16.

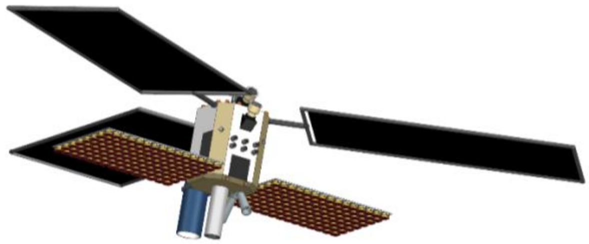


Figure 16: Lower front three-quarters view of the satellite.

The mass of OPTIMUS is listed in eight categories with special distinction for propellant to list the dry mass and wet mass requirements. Figure 17 lists the unit mass, mass margin range (which encompasses the lowest and highest margin rating for each component), and the resulting total mass. The mass margin provides an additional safety factor accounted for all hardware components. This is in addition to each subsystems' safety considerations.

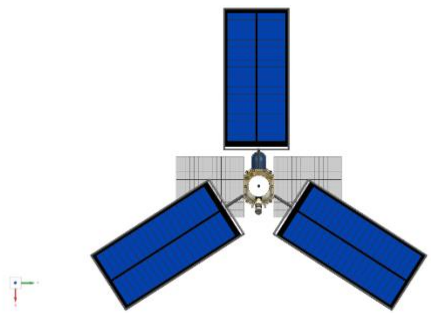
Figure 17: Mass budget table.

Category	Unit Mass [kg]	Margin Range [%]	Total Mass [kg]
Structure, Thermals & Mechanisms	548	18	646.64
Propulsion (Dry Mass)	132.72	3-30	139.75
Attitude Control	132.56	3-15	137.24
Command & Data Handling	6.5	50	9.75
Comms	13.5	8-10	14.79
Power	164.65	10	181.11
Dry Total			1704.11
Propellant	1064.9	10-15	1186.42
Wet Total			2890.54
Payload Ratio		0.58	

The mass margin is selected based on the design maturity of the hardware, the maturity code, and an appropriate Mass Growth Allowance. The result is differing margins for each item factored into the final masses. OPTIMUS' dry mass is **1704.11 kg** and wet mass is **2890.54 kg**.

The layout of OPTIMUS begins with the structure which consists of eight walls of CFAH plates attached with aluminum support brackets at the connection points and internal supports made of aluminum to secure the second level of the bus. The top and bottom faces of the bus are angled to accommodate multi-angle orientation for different subsystems.

Figure 18: Top view of OPTIMUS.



The batteries are internally secured within the bus near the rest of the internal hardware minimizing wiring length to critical components and CDH. There are three 3 m x 6.384 m x 0.1 m roll out solar panels attached to the outer walls via support structures made of aluminum 6061, see Figure 18 for reference. The solar panels are capable of 360-degree roll rotation and 90-degree pitch rotation to allow for various angles to maximize continuous view of the sun and avoid interfering with other components. These actuating points also allow for storage modularity when the arrays are rolled up. The main support beams that extrude the solar panels away from the bus are on hinge joints which allow for the arms to fold neatly near the body. The total structure folded up fits within a 2.92 m x 2.22 m x 3.5 m volume, see Figure 19.

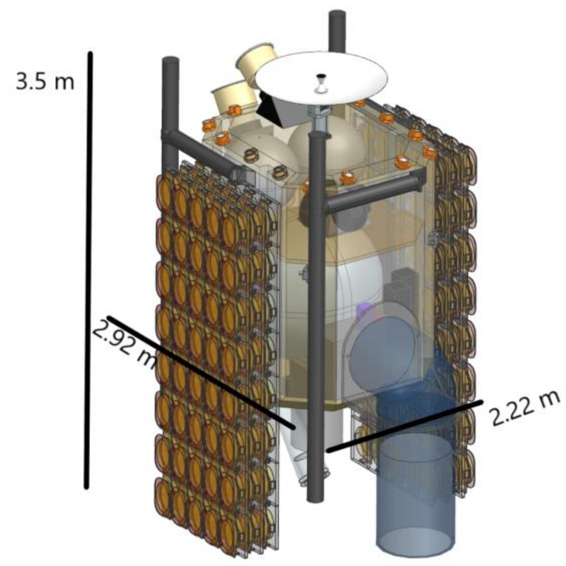


Figure 19: Stored OPTIMUS configuration with dimensions.

The instruments are split into internal data processing components, these are the LWIR and SWIR which are mounted behind the LWIR/SWIR spectrometer telescope, and the external components. The LWIR/SWIR spectrometer telescope is mounted on the lower half of the front face of the satellite pointing nadir to the surface. This prevents the RCS thruster gas from interfering with the telescope. There are two context cameras pointing in the azimuthal directions angled and offset from the center to avoid interference from the high-resolution imager and the spectrometer telescope as shown in Figure 20. The high-resolution imager is pointing directly nadir located in the center of the bottom of the satellite bus. Flanking either side of the satellite bus are three 2.84 m x 1.85 m x 0.127 m SAR/Nadir Sounder panels pointing nadir to the surface. These instruments are secured so that they can fold up and lay against either side of the bus.

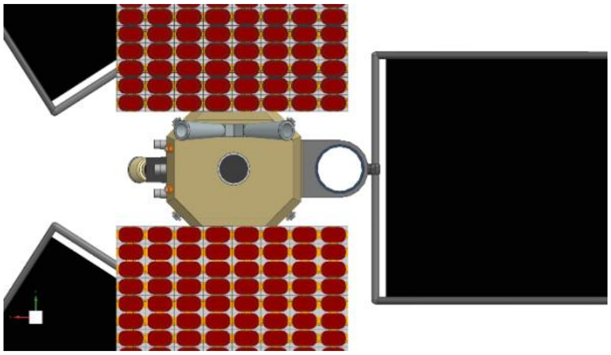


Figure 20: Bottom close view of OPTIMUS.

The propulsion subsystem has six HAL effect thrusters on the rear face of the satellite panel, shown in Figure 15. The internal fuel tank is right behind the face with the HAL thrusters and the walls surrounding the tank all have thermal louvers to help dissipate heat from the tanks and engines. The PPU and XFC hardware are located near the tanks and batteries.

The communications hardware for OPTIMUS is a 1 m Ka-band array that is attached via a gimbal arm capable of rotating 360 degrees and angling the dish to face PRIME. The internal supporting hardware is located near the power system as shown in Figure 21.

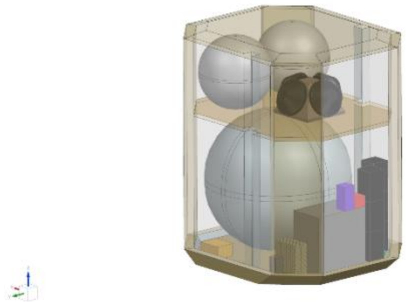


Figure 21: Transparent OPTIMUS bus three-quarters view.

The ACS hardware is split into internal and external components. The IMU is located near the power source while the reaction wheel mechanism is near the center of mass of the satellite. This places the reaction wheel mechanism onto the second level of the bus along with the two internal fuel tanks detailed in Figure 21 and Figure 22. 16 sun sensors are attached to chamfered edges of the bus, 14 on the top which has the most view of the sun while 2 are attached to the rear bottom edge of the bus to detect the sun as the satellite moves further along in orbit. Two star trackers are located on the external of the bus, one angled from the top face away from the bus and the other is located angled away from the rear face of the bus, see Figure 15. The RCS thrusters are placed in the center of the chamfered faces of the satellite bus, this location offers two benefits: isolation from any emissions interfering with other components and allowing for the RCS thrusters to be equally spaced from each other providing maximum influence on the bus.

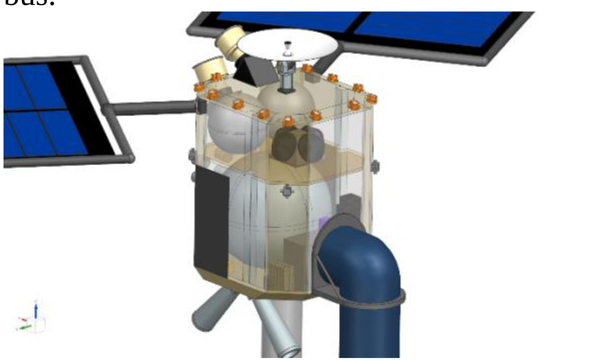


Figure 22: Three quarters transparent OPTIMUS view.

Stored configuration for OPTIMUS employs the use of the folding and actuating hardware. The solar panels are rolled up and utilize the roll actuation joint to twist the solar arrays laterally. All three solar panels' actuating arms fold near the body, further condensing the volume of space occupied by OPTIMUS. The high-resolution imager also retracts a short length (like a regular telescope) to allow for ease of storage. No internal volume or layout is disturbed during the storage configuration. Figure 19 shows the final condensed form. The geometry of OPTIMUS along with its small, folded dimensions allows for 4 rows of 3 OPTIMUS satellites, shown in Figure 23, each fitting easily within a Falcon Heavy extended fairing. This proves that PRIME satellites also fit within Falcon Heavy extended fairing given similar dimensions.

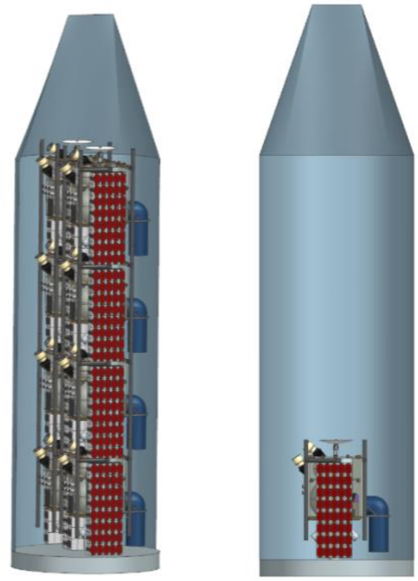


Figure 23: Stored OPTIMUS in Falcon Heavy extended payload fairing. Left indicates 8 satellite stack, right image indicates individual OPTIMUS.

While the location requirements played a crucial role in the layout and sizing of components, the moment of inertia is critical in determining the stability and maneuverability of the satellite. The largest impactor in the moment of inertia determination is the solar panels and mass distribution inside and outside of the bus. This impacts the amount of torque acting on the satellite when it is not in equilibrium. The result, shown in Figure 24, highlights the mass moment of inertia for the structure in XX, YY, and ZZ. With the mass moment of inertia, proper tank sizing for the ACS hardware can occur, ultimately resulting in a stable satellite.

Figure 24: Table of mass Moment of Inertia values

Mass Moment of Inertia	Value $[\mathbf{kg} * \mathbf{m^2}]$
$\boldsymbol{I_{xx}}$	$6.483 * 10^3$
$\boldsymbol{I_{yy}}$	$4.233 * 10^3$
$\boldsymbol{I_{zz}}$	$8.467 * 10^3$

4.4.3. Command and Data Handling

The command and data handling (C&DH) subsystem is composed of three primary components: the on-board computer (OBC), the data storage unit, and the satellite data bus. Each of these components play a vital role in ensuring that the entire subsystem can satisfy its primary purpose of managing where and what data and commands are sent on the satellite.

The factors driving the component specification for the data storage unit are as follows:

- The data storage unit must be capable of storing the maximum amount of data generated during the maximum opposition time.
- The stored data must be quickly accessed when offloading data to the communications satellites.
- The data storage unit must reliably store mission critical data.

To ensure that the data storage unit can store the maximum amount of data generated during the opposition time, a data budget was created to track the rate at which the instrumentation payload and other subsystems generated data. This budget considered the rates at which the instrumentation payload generated data, the rate at which telemetry data was generated, and a safety factor to account for any inaccuracies in the predicted instrumentation and telemetry data rates. More detail into how the rates at which the instrumentation calculated data were calculated is included in Section 4.3.4. These rates were then multiplied over a thirty data opposition time that was found by using Free Flyer to simulate every time the satellite couldn't contact a ground station on Earth through the DSN. Through finding the maximum period that the ground station couldn't make contact over a year, the thirty-day opposition period was found. The total amount of data storage needed was then divided across the fifteen satellites in the constellation to find the data storage needed per satellite. The data storage needed per satellite due to each of the contributions is shown in the table below.

Figure 25: Total Data Storage Needed per Satellite from Data Budget

Contribution	Value
Instrumentation	1.24 TB
Telemetry	0.10 TB
Safety Factor	0.27 TB
Total:	1.61 TB

To further support this, a data storage over time plot was created in FreeFlyer. This was done through taking the time it would take for at least one opposition between Mars and Earth to happen, the average rate at which the instrumentation generated data, the average rate in which the other subsystems generated telemetry data, and the average rate at which data could be downlinked from the communication satellites within the constellation to the ground stations on Earth and plotting them against a sample total storage of 2TB. The plot that was generated is shown below. This plot only showed dips in the data storage that were up to 2.7GB which was very different than what was expected by simply using the thirty-day opposition period. However, it must be noted that this model only accounted for the periods when the sun directly blocked Earth from Mars. Data storage would also need to account for the period where the radiation surrounding the sun would make it difficult to close a communications link. Because of this, the 2.7GB buildup was scaled over a thirty-day period to get the final amount of data storage needed per Satellite.

Figure 26: Total Data Storage Needed per Satellite from FreeFlyer

	Value
Build-Up Rate	2.7 GB/Day
Total Build-Up	81 GB
Number of Sats	15
Total Data Storage:	5.4 GB

Through using FreeFlyer to simulate our system, we were able to decrease the data

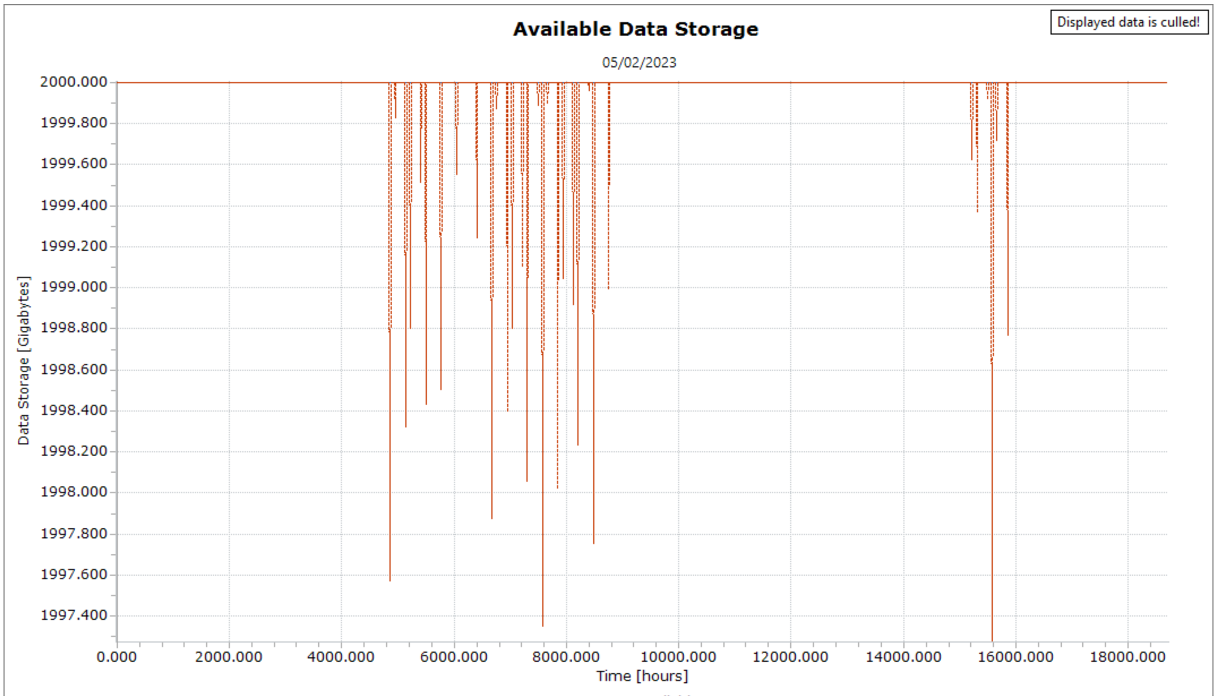


Figure 27: Total Data Storage Available Over One Opposition.

budget drastically when compared to what we found with theoretical calculations.

Once the data storage unit was sized, different types of storage needed to be considered. These options included Solid State Drives (SSDs) that were notable due to being easily accessed for data transfers, Hard Disk Drives (HDDs), Tape Storage, Optical Storage that was notable due to its high level of durability, Radiation-Hardened Memory that was notable due to it withstanding high levels of radiation, and Electrically Erasable Programmable Read-Only Memory (EEPROM) that was notable to it retaining data in the event of a power loss. Out of these options, a combination of SSD's and Radiation-Hardened Memory were chosen due to the data being easily transferrable between satellites and withstanding the high levels of radiation the satellites will need to endure throughout the mission.

The factors driving the component specification of the OBC were as follows:

- Processing Capability - The OBC must have the processing capacity needed to operate the instruments and other subsystems.
- Power Requirements – The OBC must be capable of querying the power subsystem for available power and turning off noncritical systems if needed. Additionally, it must fit into the power budget.
- Reliability of Software – The OBC must have reliable software running on it to handle event sequencing, monitor health and performance of all subsystems, handle any problems on orbit, and interpret and execute commands sent from ground control.
- Interfacing Capability – The OBC must be able to work effectively with the required interfaces. This includes having enough ports to connect to the necessary external subsystems.
- Size and Weight – The OBC must fit into the mass and volume budget without requiring a redesign.
- Resilience - The OBC must be suitably hardened to survive the necessary levels of radiation it will encounter in space.

Additionally, it must be able to survive the extreme temperatures that happen in solar flares.

In order to meet the processing power needed to meet mission requirements, the processor must have enough RAM and a sufficient clock speed. Separate from the storage sizing talked about previously, RAM is needed for the processor to execute tasks and process data quickly and efficiently through data getting transferred to RAM while an operation is being completed and then getting transferred back to SSD once it's done. Through using the data rates to calculate the maximum amount of processing that will be done at any point of the mission, it was found that a RAM size of roughly 2GB is needed to process the scientific data as it's sent from the instrumentation payload to the OBC and then to data storage. A similar process was used to find that a clock speed of 150 MHz is needed for the processor to complete the tasks it needs to in a sufficient manner. Three processor configurations were considered that varied in the amounts of radiation they could absorb, the amount of power they required, their clock speeds, and how much RAM they could contain. Out of these configuration, they all had similar power requirements, and they could all used external dynamic random-access memory (DRAM) that could be sized appropriately to meet mission requirements. However, some configurations could absorb higher levels of radiation and had higher clock speeds than others. Because of this, a final configuration was chosen that had the best combination of radiation hardening and a high clock speed. This configuration could absorb 200,000 to 1,000,000 rads and had a clock speed of 110-200 MHz. In addition to this, the processor configurations had a legacy of running software that's able to decode commands from the ground station and format telemetry for transmission according to standard CCSDS protocols. Additionally, all the configurations contained an on-board clock that can be used for planning commands to execute in the future.

The factors driving the component specification of the satellite data bus and links were as follows:

- It must be capable of interfacing with every subsystem on the satellite.
- It must be capable of transferring data at an acceptable rate.

The options that were considered to meet these specifications included:

- MIL-STD-1553**:** Uses a protocol that is made to be reliable in harsh conditions and transfers data at 1Mbps. It also has built-in error detection and correction capabilities.
- Controller Area Network (CAN): Allows multiple devices to communicate with each other in a network without a central computer. It can transfer data at a maximum rate of 1 Mbps.
- Serial Peripheral Interface (SPI): Allows for a high-speed data transfer with a simple and efficient protocol but it doesn't include error checking. Because of this, data integrity can be questionable.
- SpaceWire: Uses a high-speed, serial communication protocol that's designed to handle large amounts of data with low power consumption. It transfers data at rates up to 200 Mbps but can be expensive and complicated.

Although SpaceWire could transfer data at the highest rates out of all the options, it wasn't chosen due to the potential issues that could arise from using it. Because of this, MIL-STD-1553 was selected due to its reliability and low effort implementation. In addition to this, it transferred data at a rate that was comparable to all the other data bus configurations while also having built-in error detection and correction capabilities. The rate at which it transfers data allows for the average data rate between C&DH and all of the instruments and subsystems to be between 0.1 and 0.8 Mbps with a maximum of 1 Mbps which is suitable for mission standards. However, due to MIL-STD-1553 transferring data at a rate that is slower than some of the instructions that are only on for a short period of time but generate data at an extremely high rate, data storage is included within those instruments to store the data as it's being transferred.

4.4.4. Instrumentation

Of the six science capabilities, special attention was paid to eliminating redundant components that that can be shared by two payloads. This not only reduces the mass of the orbiter, but drastically cuts down on the cost of developing extraneous parts at minimal increase in complexity.

The first hybrid instrument is the P-Band SAR and Radar Sounder, which share a common antenna and electronics architecture. Consisting of six 2.84 m x 1.05 m panels, each with five subarrays of eight antenna elements, the radar is built with flexibility in mind. Entire panels, subarrays, or elements can be added or subtracted with ease to meet the evolving needs of the stakeholder or adjust to any changes to the orbiter design during the development cycle. Figure 12 illustrates this architecture in detail. At the element level, each antenna is paired with a transceiver for conversions between electrical signals and radio waves. From there a feed network passes signals between the radar control and transceivers using Frequency Domain Multiplexing (FDM). The FDM approach allows each antenna signal to be transmitted simultaneously to and from the controller, bringing the number of Digital-to-Analog (DAC) and Analog-to-Digital (ADC) converters down to one per panel. FDM also enables centralized waveform generation at the radar controller in a single Field Programable Gate Array (FPGA), as well as the consolidation of data in one location.

In SAR mode, the 400 MHz beam is steered to a 30° look angle, capturing a 50 km swath at 30 m spatial resolution. With full polarimetry on transmit and receive, the SAR payload collects data on received power, time delay from transmission, phase and frequency of the return, and beam incident angle, providing the capability to detect scatterers consistent with the top shelf of subsurface ice within 3 m of the surface. When in sounding mode, the radar operates nadir at two frequencies, 200 MHz and 400 MHz, detecting changes in the real part of dielectric constant to inform on the presence and purity of subsurface ice. The higher frequency provides a vertical resolution between 0.5 and 1 m at shallow depths of up to 15 m, while the lower 200 MHz sacrifices resolution to return data from depths of up to 40 m. As shown in Figure 13 the vertical resolution is an order of magnitude finer than MRO SHARAD, and two orders of

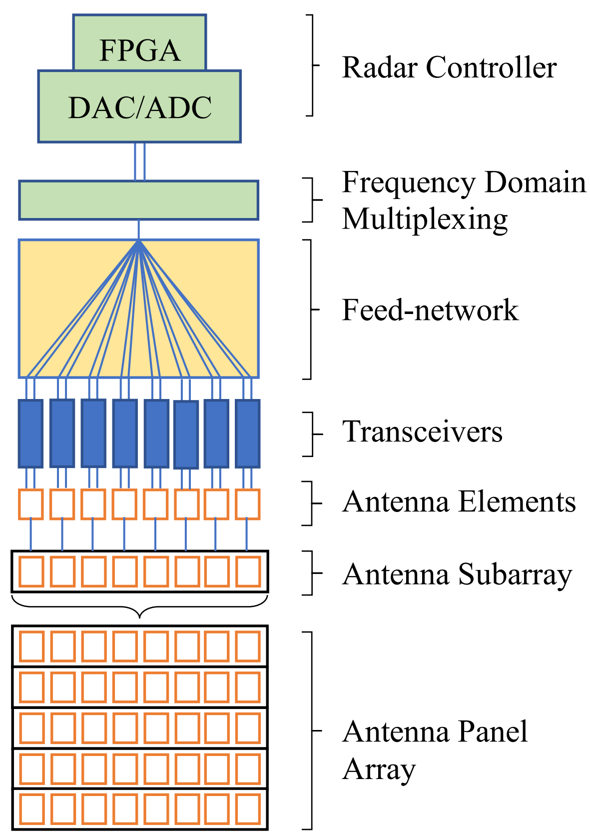


Figure 28: Antenna architecture for the SAR/Sounder hybrid instrument. Each antenna array consists of 40 antenna elements, where a set of time delays control the phase of each signal for beamforming.

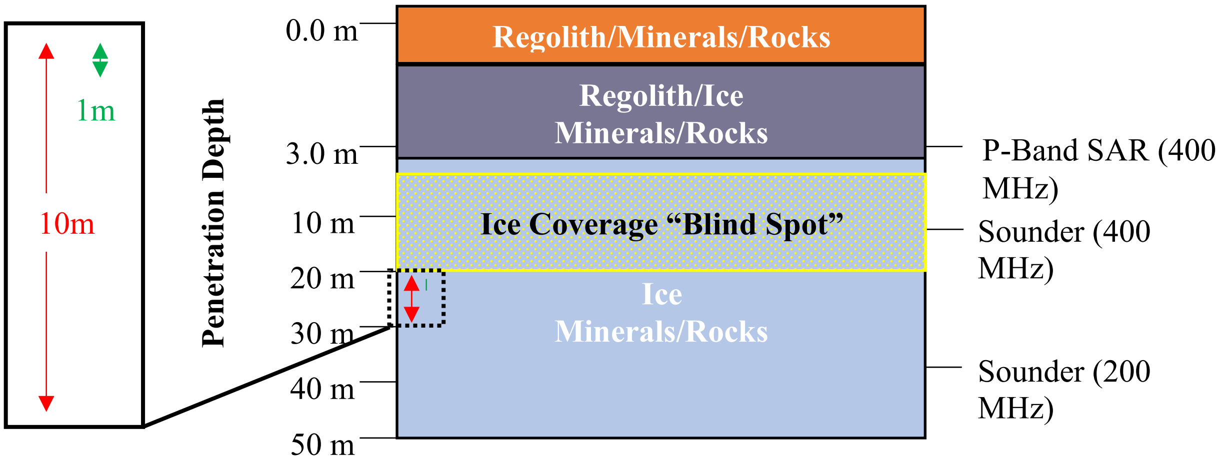


Figure 29: Penetration depths of radar displayed over inferred subsurface cross-section. The sounder achieves a vertical resolution of 1m (green), an order of magnitude improvement over the current resolution provided by MRO SHARAD.

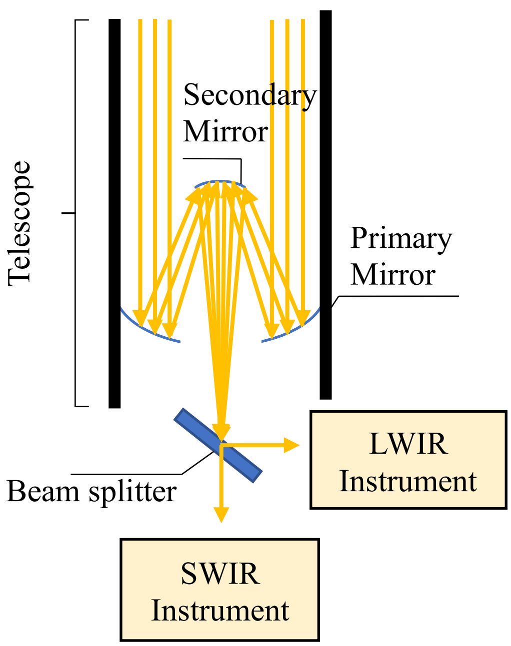


Figure 30: Schematic of the shared 50 cm telescope and beam splitter for the SWIR/LWIR hybrid instrument.

magnitude finer than that provided by Mars Express MARSIS. An L-Band system was considered for even higher vertical resolution, but ultimately rejected due to its failure to close a key gap in coverage between 5 and 20 m depths that remains a blind spot to this day. The lateral resolution of the sounding data is 1.5 km, but the SAR imagery captures the necessary resolution to provide context to the data collected by the sounder. Both operate at a 100 MHz bandwidth to enable the collection of the high-resolution data. Since SAR and sounding are active remote sensing applications, this instrument only operates during an eclipse, although it may operate in daylight if desired.

Figure 31: Tabulated data sizing and design of SAR/Sounder hybrid instrument.

Characteristic	Value
Swath	50 [km]
Resolution	30 [m]
Pixels	1658
Metrics (5 total)	Received power, delay from sending, phase, frequency, incident angle
Data Type	Double [64 bits]
Bit/pixel	320 [bit]
Bit/frame	530560 [bit]
Sample Rate	700
SAR Duty Cycle per Orbit	10 [min]
Sounder Duty Cycle per Orbit	10 [min]
Data per Orbit	55.7 [GB]

The second hybrid instrument is the short-wave infrared and long-wave infrared

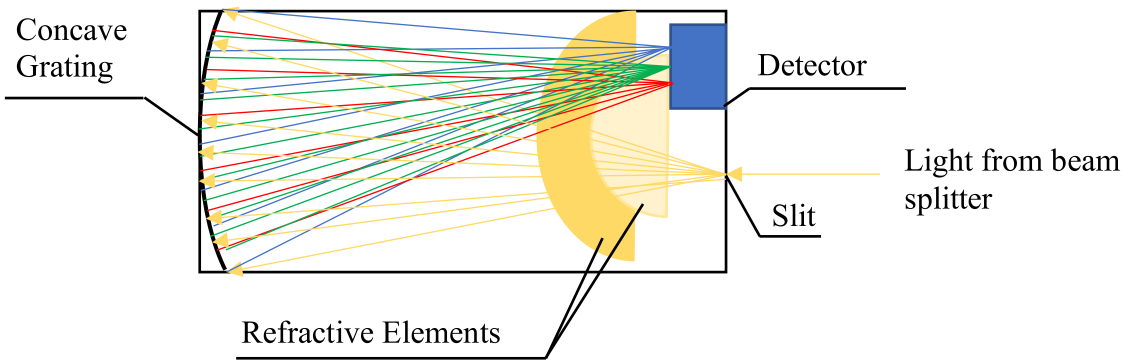


Figure 32: Schematic of the hyperspectral SWIR payload. Light enters through a slit from the beam splitter, reflects off of the concave grating into constituent wavelengths, and then is channeled by the refractive elements towards the detector.

spectrometer telescope (SWIR/LWIR). Each spectrometer remains separate but shares a common 50-cm telescope. A beam splitter divides the collected light in two equal halves, with one directed towards the SWIR sensor, and the other to the LWIR sensor. The SWIR instrument uses a 1.3-4.2 µm Dyson spectrometer with a 1152 x 1152-pixel mercury cadmium telluride Teledyne CHROMA (Configurable Hyperspectral Readout for Multiple Applications) detector. Operating during the day, the CHROMA detector allows the SWIR spectrometer to collect hyperspectral data at 10 nm spectral resolution, and 5 m spatial resolution. To complement the hyperspectral data collected by the SWIR spectrometer, the LWIR instrument records multi-spectral data at a bandpass of 1 µm using a 20 channel 6-25 µm grating spectrometer with a 576 x 576-pixel uncooled microbolometer array detector.

Sharing common optical specifications such as design focal length, diffraction, and signal to noise ratio, the common telescope shared between each spectrometer drastically reduces mass of the instrumentation, as it eliminates the need for two different telescopes. The set of mirrors in the telescope direct the light towards a beam splitter, which is a flat glass plate with a semi reflective coating that divides the light evenly between the two instruments.

The design of the hyperspectral SWIR spectrometer is driven by the need to collect spectral data from minerals at unprecedented spatial and spectral resolution that are at least an order of magnitude greater than MRO CRISM. Those resolutions (10 nm spectral and 5 m spatial) result in a very high data collection rate. A balance was struck between the operational duty cycle of the instrument, which is limited by the communications and data storage capabilities of the spacecraft, and the coverage achieved by the instrument, which is a driving parameter to meeting mission requirements, in order to determine the optimal design of the payload. The final data design characteristics are shown in Figure 33, where the payload operates in a "postage stamp" fashion, taking high fidelity data at targeted locations of interest. Figure 34 illustrates the physical design of the instrument.

Figure 33: Tabulated data sizing and design for SWIR hyperspectral Spectrometer.

Characteristic	Value
FOV	1.1°
Focal Length	12 [m]
Total Bandpass	2900 [nm]
Spectral Resolution	10 [nm]
Channels per Pixel	290
Spatial Resolution	5 [m/pixel]
Pixels	1327104
Bits/pixel	14

Bits/frame	5.39E09
Sample Rate	1.7
Duty Cycle per Orbit	1 min
Data per Orbit	68.32 GB

The design of the multispectral LWIR spectrometer is driven by the need to collect spectral data over a broader bandwidth than achieved by Mars Odyssey THEMIS at an order of magnitude increase in spatial resolution. Since this instrument shares a telescope with the hyperspectral instrument but does not have the same level of spectral resolution or spatial resolution, the data produced during operation is not a limiting factor for the instrument duty cycle. As such, it collects data continuously during the day side of the orbit and can be activated for thermal spot-checking during eclipse for around two minutes per orbit. Figure 35 reports the final data design characteristics of the LWIR spectrometer.

Figure 35: Tabulated data sizing and design for LWIR multispectral spectrometer.

Characteristic	Value
FOV	1.1°
Focal Length	12 [m]
Channel Bandpass	1 [μm]
Channels per Pixel	20
Spatial Resolution	10 [m/pixel]
Pixels	331776
Bits/pixel	14
Bits/frame	92897280
Sample Rate	1.7
Duty Cycle per Orbit	62 [min]
Data per Orbit	73.0 [GB]

Both payloads search for distinct spectral bands associated with ionic compounds, minerals, and rocks critical to the understanding of Martian crustal modification over time, while the two spectral ranges allow for complimentary data sets that synergize to survey the entire spectral response curve of minerals and rocks. Hyperspectral capabilities in the SWIR are of particular interest due to high concentration of ionic absorption bands in this spectral region, while multi-spectral LWIR data reveals clay silicates and other minerals that are not present in SWIR but demonstrate wider spectral responses in this region. Mapping these mineral groups lends insight into potential biological signatures from Mars' ancient history. A detailed, but not all inclusive, accounting of potential constituents is provided in Figures 40 and 41.

The high-resolution imaging capability of OPTIMUS is provided by a 5760 x 5760-pixel

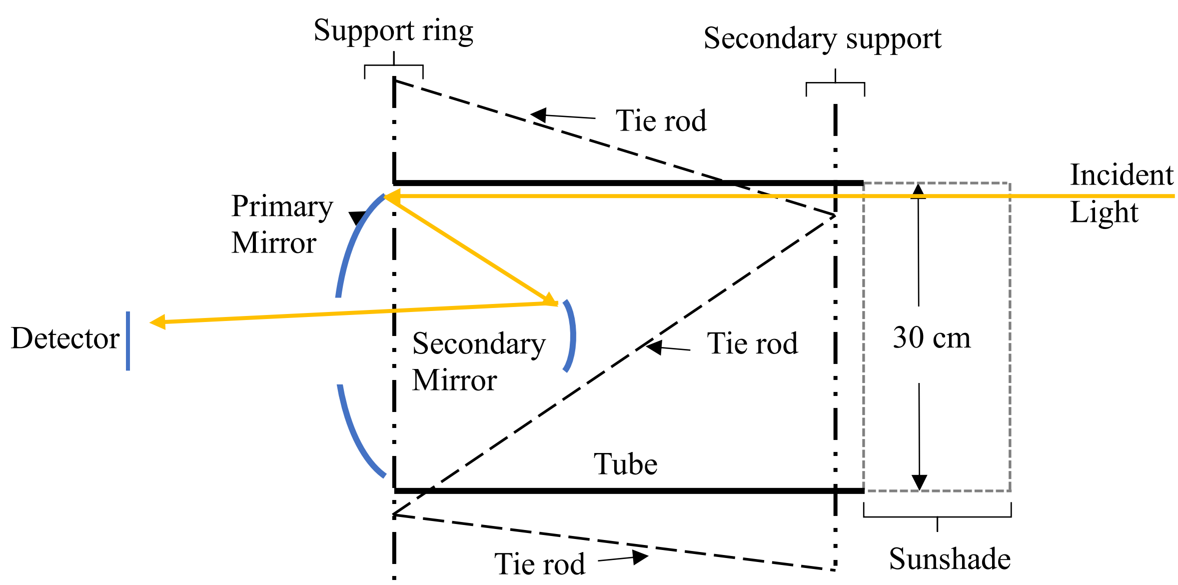


Figure 34: Schematic of high-resolution imager telescope design. A similar architecture is used for the SWIR/LWIR hybrid telescope to reduce design complexity.

mercury cadmium telluride detector paired with a 30 cm, 5.4 m telescope. Surveying targets at nadir during the day with 20 channels between 0.4 and 1.7 µm, this imager achieves 1 m resolution using time delay integration and can return data in visible and very near infrared wavelengths. Six channels included between 1.2 and 1.7 µm specifically target major absorption features of H_20 and CO_2, critical to tracking the cycling of volatiles between the surface and the atmosphere, a capability that is especially important in the second polar science orbit. This instrument also overlaps with data collected by the hyperspectral SWIR instrument, creating complementary data sets that further constrain the nature of surface minerals. Should a meteorite impact occur, the instrument takes images to determine if any subsurface ice is revealed. OPTIMUS can also leverage this imager to discover new ice scarps or glacial features on the surface such as lobate debris aprons.

The design of the high-resolution imager is driven by the need to take super resolution "postage stamp" snapshots of scientific regions of interest like the ones mentioned in the previous paragraph, while also providing spectral data in VNIR to compliment the SWIR instrument. Like the SWIR instrument, the duty cycle is 1 minute and the FOV is small to limit the amount of data produced, as the extreme resolution translates into copious amounts of data that is not storable on the satellite bus, or reasonable to downlink to Earth in an ordinate amount of time. The telescope used by the imager is similar in design to the telescope used by the hybrid spectrometers for reduced design complexity, with the key difference being the shorter focal length, and narrower aperture. Figure 34 illustrates the physical design as well as supporting structure. The final data sizing characteristics of the high-resolution camera are reported in Figure 37, where a final metric of note is the 60 nm bandpass achieved by the camera.

Figure 37: Tabulated data sizing and design of high-resolution camera.

Characteristic	Value
FOV	1.1°
Focal Length	5.4 [m]
Channel Bandpass	60 [nm]
Channels per Pixel	20
Spatial Resolution	1 [m]
Pixels	33177600
Bits/pixel	14
Bits/frame	9.29E09
Sample Rate	1.7
Duty Cycle per Orbit	1 [min]
Data per Orbit	117.8 [GB]

The final instrument carried by OPTIMUS are the dual stereo context cameras, two panchromatic imagers facing 22.5 degrees forward and aft of nadir. Two cameras are required to synthesize a 3-D model of the surface. This instrument features a 1467 x

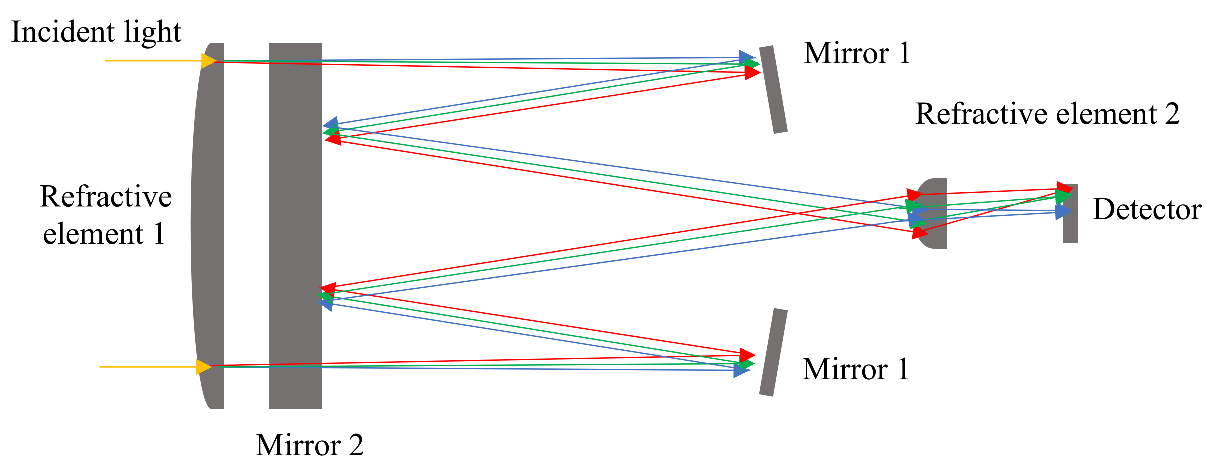


Figure 36: Schematic of context camera design. A telescope directs light towards a set of refractive elements that coalesce all the light to the detector.

1467-pixel sensor, meeting high-resolution topography requirements of 15 m. The stereo cameras provide radar clutter mitigation for the SAR and sounder and enable the generation of digital elevation models (DEMs) which mitigate geometric image or reflection distortion. The final design characteristics of the cameras are shown in Figure 38, where we note that due to the presence of two cameras and continuous daytime operation, the data produced is the greatest among the payloads.

Figure 38: Data sizing and design of dual context stereo cameras.

Characteristic	Value
FOV	4.2°
Focal Length	0.35 [m]
Channels per Pixel	1
Spatial Resolution	15 [m]
Pixels	2152089
Bits/pixel	14
Bits/frame	30129246
Sample Rate	6.5
Duty Cycle per Orbit	60 [min]
Number of Cameras	2
Data per Orbit	175.1 [GB]

To inform OPTIMUS subsystem design, specifically command and data handling, power, and structures, a mass, power, and average data rate budget is provided in Figure 39. Both mass and power are reported without margin, and the data rates are considering the raw uncompressed data collected by each instrument. Thermal accommodations are discussed in the Section 4.4.2.

When added to the overall mass and power budgets appropriate margins are included, and the data ultimately downlinked to the DSN has a 10:1 lossless compression applied. In eclipse, the SAR and Sounder operate continuously over ten-minute intervals, while the LWIR spectrometer only surveys selected targets. During the day, the stereo cameras and LWIR spectrometer operate continuously, while the high-resolution imager and SWIR spectrometer only take pictures of select targets. The distinction between continuous and select operations reduce the power required and the data rates to be accommodated.

The synergy of these payloads produces a holistic data set of surface and subsurface constituents on the Red Planet, permitting an unprecedented dive into the past and present nature of Mars. This data includes the first SAR imagery of Mars, an order of magnitude increases in subsurface sounding resolution, and unprecedented spatial and spectral resolution of minerals on Mars. Through OPTIMUS, SCOPE answers questions about the Martian cryosphere and the evolution of the surface environment over time, revealing the potential for life on Mars, and mapping resources critical to ISRU efforts in future missions.

Figure 39: Tabulated mass, power, and data sizing for instrumentation suite. Coverage over 2-year duration reported, but mission extensions can further increase these numbers.

Payload	Mass [kg]	Power [W]	Data [MBps]	Coverage[%]
P-Band SAR / Sounder	480	366	92.9	99.58
IR Telescope	39.7	0	0	97.81
SWIR payload	1.5	16	1138.6	11.81
LWIR payload	4.5	6	19.6	77.26
High-Res Color Imager	19	30.0	1963.1	11.81
Dual Stereo Cameras	13.4	11.6	48.6	99.74

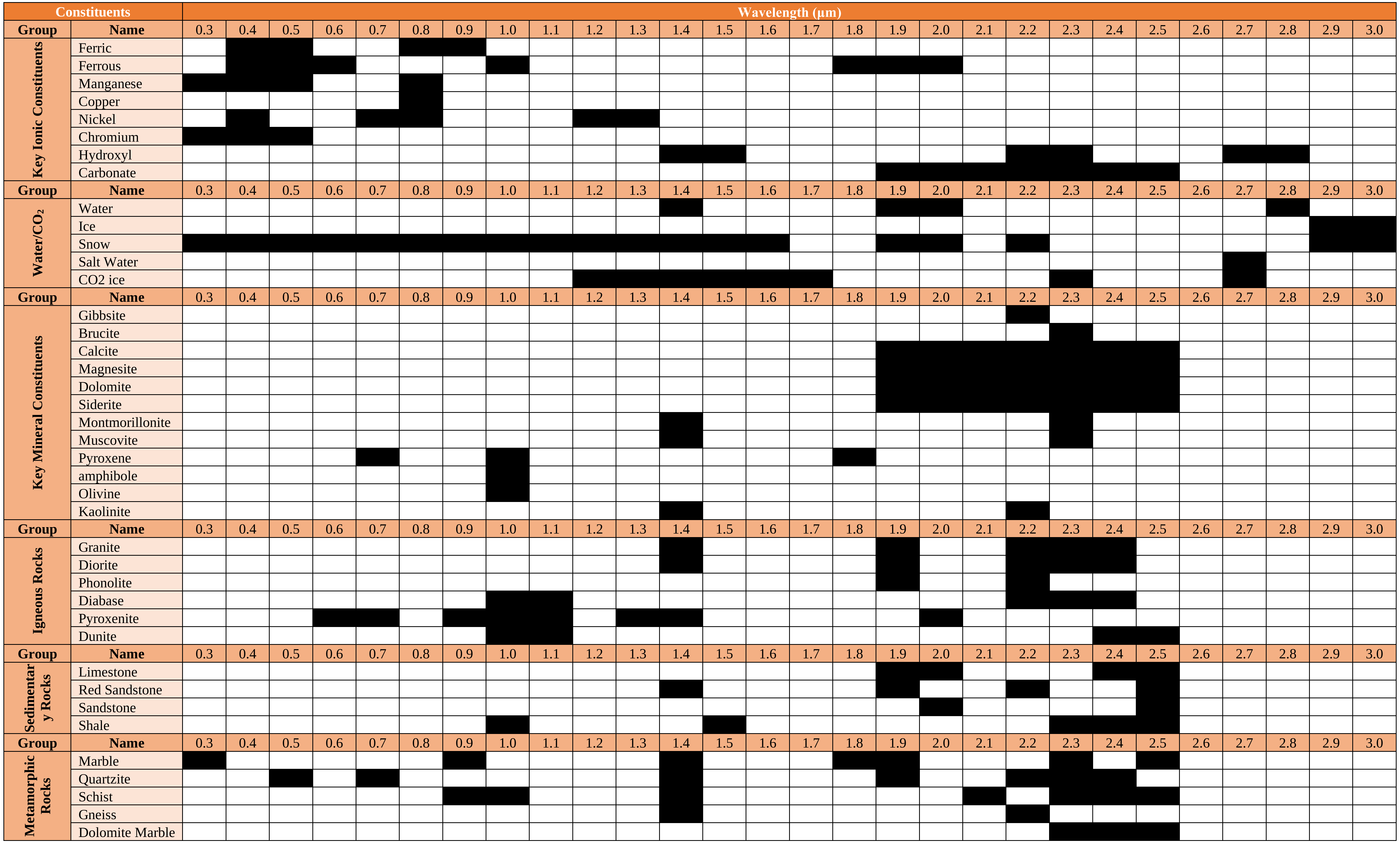


Figure 40: Visible and short-wave infrared absorption bands of select constituents of interest. Black bands indicate the presence of an absorption band near the indicated wavelength, not exactly on it. Many of the bands peak sharply, such as the Ferric ion at 0.87 µm, compelling the need for hyperspectral capabilities in SWIR. This table is not inclusive of all potential minerals in SWIR.

Contributors: Alec. References: Meerdink, S. K., Hook, S. J., Roberts, D. A., & Abbott, E. A. (2019). The ECOSTRESS spectral library version 1.0. Remote Sensing of Environment, 230(111196), 1–8. Baldridge, A. M., S.J. Hook, C.I. Grove and G. Rivera, 2009.. The ASTER Spectral Library Version 2.0. Remote Sensing of Environment, vol 113, pp. 711-715.

Constituents		Wavelength (µm)																						
Group	**Name**	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0	13.0	14.0	15.0	16.0	17.0	18.0	19.0	20.0	21.0	22.0	23.0	24.0	25.0
Key Ionic Constituents	Carbonate					■				■														
	Sulphate							■						■	■									
	Phosphate							■	■															
	Oxide						■	■	■	■	■													
	Nitrate					■																		
	Nitrite						■				■													
	Hydroxyl									■														
	Silicate					■	■	■	■	■	■													
Group	**Name**	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0	13.0	14.0	15.0	16.0	17.0	18.0	19.0	20.0	21.0	22.0	23.0	24.0	25.0
Water/CO_2	Water	■			■						■	■	■											
	Ice	■							■	■	■	■	■											
	Snow	■								■	■	■	■											
	Salt Water				■						■	■	■											
	CO2 ice																							
Group	**Name**	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0	13.0	14.0	15.0	16.0	17.0	18.0	19.0	20.0	21.0	22.0	23.0	24.0	25.0
Key Mineral Constituents	Quartz							■													■			■
	Brucite		■																					
	Calcite	■	■																					
	Augite							■	■		■					■				■		■		
	Dolomite				■	■				■			■										■	
	Siderite		■		■						■	■	■											
	Montmorillonite																							
	Muscovite								■	■								■	■			■		
	Pyroxene																							
	Amphibole																							
	Olivine								■	■						■			■					■
Group	**Name**	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0	13.0	14.0	15.0	16.0	17.0	18.0	19.0	20.0	21.0	22.0	23.0	24.0	25.0
Igneous Rocks	Granite	■	■		■		■	■			■													
	Diorite	■					■		■															
	Phonolite	■						■																
	Diabase	■					■																	
	Pyroxenite	■						■	■	■														
	Dunite		■							■														
Group	**Name**	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0	13.0	14.0	15.0	16.0	17.0	18.0	19.0	20.0	21.0	22.0	23.0	24.0	25.0
Sedimentary Rocks	Limestone				■	■				■														
	Red Sandstone	■	■				■	■				■												
	Sandstone		■				■	■				■												
	Shale		■			■		■																
Group	**Name**	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0	13.0	14.0	15.0	16.0	17.0	18.0	19.0	20.0	21.0	22.0	23.0	24.0	25.0
Metamorphic Rocks	Marble	■	■	■		■				■														
	Quartzite	■	■				■	■				■												
	Schist		■					■			■													
	Gneiss	■	■				■	■																
	Dolomite Marble		■		■	■				■	■		■											

Figure 41: Long-wave infrared absorption bands of select constituents. Many of the spectral bands in this regime are wider than in SWIR, especially greater than 3 µm, but still reveal critical information necessary to distinguish between minerals. This necessitates the need for a broader, multispectral LWIR spectrometer to survey the entire spectral response curve of each constituent.

Contributors: Alec. References: Meerdink, S. K., Hook, S. J., Roberts, D. A., & Abbott, E. A. (2019). The ECOSTRESS spectral library version 1.0. Remote Sensing of Environment, 230(111196), 1–8. Baldridge, A. M., S.J. Hook, C.I. Grove and G. Rivera, 2009.. The ASTER Spectral Library Version 2.0. Remote Sensing of Environment, vol 113, pp. 711-715.

4.4.5. Attitude Control

The attitude control system serves two major purposes: attitude determination and attitude control. Specification of components used in the attitude control system are driven by the following factors:

- The pointing requirements from the science instrument which has the most restrictive field of view (high resolution camera).
- The accuracy of the attitude determination to point the satellite at ground targets of interest.
- The amount of disturbance torques acting on the spacecraft.
- Redundancy of the overall system, allowing the spacecraft to meet science objectives.

The spatial requirements of the high-resolution camera, the angular pointing requirements and attitude determination accuracy required to achieve the desired pointing accuracy are shown in Figure 42.

Figure 42: ACS Accuracy

Accuracy	Quantity
Spatial [km]	1
Angular Pointing [deg]	.5
Attitude Determination [deg]	.05

Owing to the extremely low tolerance on the determination of the angular position, multiple sources of attitude knowledge are required to be used by a state estimator to produce a highly accurate estimate. A backup source of attitude determination is also required due to the power system relying on ACS for solar panel pointing information. A set of two star trackers, which provide a high accuracy attitude estimate, are used in conjunction with a set of two Inertial Measurement Units (IMUs), which provide high resolution angular position and velocity estimates to produce the accuracy and resolution the orientation estimate in all three rotational degrees of freedom. A set of 16 sun sensors, placed strategically on the body of the spacecraft are used as backup sensors to produce a low-resolution attitude estimate which can be used to point the solar panels if the flight software must revert to a safe mode, allowing solar panel pointing control to be decoupled from the rest of the system.

Due to OPTIMUS being at a higher altitude than any measurable atmosphere, the effects of atmospheric torque are neglected when the sizing of the attitude control system. The effects of magnetic fields can also be neglected owing to Mars having an extremely weak magnetic field. Driving factors of the sizing of the attitude control system are the torque due to gravity gradient when the spacecraft is not at an equilibrium attitude as well as torque due to solar pressure. We were not able to determine a way to perform simulations in FreeFlyer to determine torque due to gravity gradient, so we used an extremely conservative estimate for the total angular momentum required over the mission lifetime by integrating the expected worst-case disturbance torques over the mission lifetime. A table of the worst-case disturbance torques as well as angular momentum needed per orbit and total angular momentum required over the mission lifetime assuming a safety factor of 1.2 are shown in Figure 43.

Figure 43: Disturbance Torques and Momentum Requirements

Parameter	Value
Gravity Gradient [Nm]	. 00072
Solar Pressure [Nm]	9.81E-7
Angular Momentum Per Orbit [Nms]	5.852
Total Angular Momentum Over Mission Lifetime [Nms]	166700

Due to the low instantaneous torque acting on the spacecraft, a set of four reaction wheels are used, placed in a pyramidal configuration, owing to the high momentum storage and relative simplicity offered when compared to alternatives such as control moment gyroscopes, allowing for pointing control in all three rotational degrees of freedom. Because reaction wheels need to be desaturated periodically, four sets of four monopropellant reaction control thrusters are used. The reaction control system uses ASCENT (AF-M315E) propellant owing to its higher specific impulse, relaxed storage requirements, and lower toxicity when compared to the conventional hydrazine which it is designed as a direct replacement for. A table of key metrics of the attitude control system is shown in Figure 44.

Figure 44: ACS Key Metrics

Parameter	Value
Angular Momentum per Reaction Wheel [Nms]	100
Thrust per RCS Thruster [N]	26
Propellant Required [kg]	300
Orbits Between Desaturation Events [kg]	68.5
Power Max/Min [W]	170.8/50.8
Total Hardware Mass [kg]	50.48
MTBF [Hrs]	53000

All telemetry and attitude data generated by the components that comprise the attitude control are be provided to C&DH at an average between the maximum and minimum values shown for each mode of operation shown in Figure 45. All data required by ACS from C&DH to perform functions are provided between the maximum and minimum values shown for each mode of operation shown in Figure 45. The amount of raw data provided by ACS to C&DH shall not exceed 1000 kbps.

Figure 45: ACS Data Metrics

Parameter (Mode)	Min (kbps)	Min (kbps)
Attitude Estimate (All Modes)	2.56	3.00
Telemetry of Driving Components (All Modes)	.512	1.00
Accepting Orientation Commands (Pointing Mode)	5.76	8.32
Desaturation Commands (Desaturation Mode)	8.58	11.14
Passing Raw Data Generated from Sensors (As Requested)	980	985

4.4.6. Power

The power subsystem onboard OPTIMUS is responsible for generating, storing, and distributing power to support payload and bus operations throughout the mission lifetime.

The design and sizing of the power subsystem is driven primarily by the high loads of the Solar Electric Propulsion (SEP) system, eclipse operation of the bus and SAR/Sounder hybrid instrument, and the mission lifetime. The SEP system dominates the power demand by more than an order of magnitude relative to other systems, and the peak power in thrusting modes is greater than in normal science operation. Battery-powered operation of the SEP is unfeasible, and the solar arrays are sized to accommodate the SEP system. Eclipse periods necessitate battery storage which is sized and driven by the operation time of the high-load SAR/Sounder hybrid instrument and the bus power. Additionally, a five-year design lifetime necessitates a low Depth of Discharge (DoD) to prolong battery health, which further drives battery sizing.

The satellite bus power budget given by Figure ***47*** summarizes the power loads and duty cycles for required satellite subsystems, while the payloads power budget given by Figure ***46*** summarizes the power loads and duty cycles for instrumentation used to meet scientific mission requirements. A duty cycle of zero percent was used to indicate operational modes that are

Figure 47: Bus power budget.

Subsystem	Operation	Power (W)	Duty Cycle (%)	Total Power (W)
ACS	*Normal*	118.8	100%	118.8
	Standby	50.8	0%	0
	Desaturation	158.8	0%	0
C&DH		47	100%	47
COMMS		150	66%	98.68
POWER		104	100%	104
PROP	*Near Earth*	9900	0%	0
	Near Mars	4600	0%	0
THERMAL		250	100%	250
Totals	**Max Bus Instantaneous**	**10351.8**	**Nominal Bus Total**	**618.5**

Figure 46: Payload power budget.

<table>
<tr><th>Subsystem</th><th>Operation</th><th>Power (W)</th><th colspan="2">Duty Cycle (%)</th><th colspan="2">Total Power (W)</th></tr>
<tr><td>LWIR Telescope</td><td>2 min – eclipse</td><td>6</td><td colspan="2">2%</td><td colspan="2">0.11</td></tr>
<tr><td>SAR/ Sounder</td><td>20 min – eclipse</td><td>366</td><td colspan="2">18%</td><td colspan="2">64.21</td></tr>
<tr><td></td><td></td><td></td><td colspan="2"></td><td colspan="2"></td></tr>
<tr><td>Context Cameras</td><td>60 min – sunlight</td><td>11.6</td><td colspan="2">53%</td><td colspan="2">6.11</td></tr>
<tr><td>High-Res Camera</td><td>1 min – sunlight</td><td>30</td><td colspan="2">1%</td><td colspan="2">0.26</td></tr>
<tr><td>LWIR Telescope</td><td>60 min – sunlight</td><td>6</td><td colspan="2">53%</td><td colspan="2">3.16</td></tr>
<tr><td>SWIR Telescope</td><td>1 min – sunlight</td><td>16</td><td colspan="2">1%</td><td colspan="2">0.14</td></tr>
<tr><td>Totals</td><td>Sunlight Payload Power (W)</td><td>63.6</td><td>Eclipse Payload Power (W)</td><td>366</td><td>OAP (W)</td><td>73.98</td></tr>
</table>

seldom or never active when OPTIMUS is in its science operational mode, and instrument operation periods were indicated as operating in eclipse or sunlight.

The peak power loads are given by Figure 48 and summarize the peak loads that the power subsystem must be sized to accommodate. A 20% contingency is incorporated into these values to account for uncertainties and variations in component operation.

The solar arrays are sized to accommodate both the peak totals at Earth and at Mars, with consideration given to the solar constant which varies by distance from the sun. The batteries are sized to accommodate the peak power and

Figure 48: Peak power requirements.

Power Mode	Load [W]	% Cont.	Total [W]
Bus + PROP (near Earth)	10351.8	20	**12422.16**
Bus + PROP (near Mars)	5051.8	20	**6062.16**
Bus + eclipse payloads	984.5	20	**1181.38**

total energy usage in eclipse with consideration for the desired DoD.

The possible solar array configurations were limited by the large array area and the need for articulation which permits sun tracking as OPTIMUS is thrusting or operating in science mode with nadir-pointing instruments to maximize power generation. Configurations of two and three deployable solar arrays were considered. Two arrays provide a standard configuration ensuring optimal power generation with sun tracking, but the extended array mass increases the MOI of OPTIMUS, necessitating increased mass of the ACS subsystem. A three-array configuration was selected because it reduces the array length required to attain the array area to accommodate desired power generation levels but reduces the MOI by over a factor of 1,000 km/m^3 relative to the two-array configuration, reducing the required mass of the ACS subsystem. Rotating mounts provide 360° of roll and 180° of pitch, enabling compact storage and sun tracking when combined with rotation of OPTIMUS about the nadir-pointing axis.

Consideration for array architecture was given to rigid panels, UltraFlex arrays, and Roll Out Solar Arrays (ROSA). Rigid panels have heritage and are the cheapest option but feature relatively low specific power of 58.5 W/kg due to lower efficiency and heavy mechanical hinges. UltraFlex arrays unfold from a compact form offering specific power of 115 W/kg at improved efficiencies of 28% but have failed to open and latch in past missions. ROSA is a newer design with TRL 9 given by heritage on the ISS and DART. They feature a specific power of 100-120 W/kg and improved efficiency of 30-34% at lower mass and stowed volume while requiring no motors to deploy. ROSA was selected for its improved performance, which offsets higher costs.

Consideration for battery storage was given to Ni-Cd, Ni-MH, and Li-Ion. Li-Ion technology was selected due to higher specific energy of 100-200 Wh/kg compared to alternatives, lower mass due to use of lithium, lower auto-discharge rate, and capability for short and high discharge peaks.

Solar array sizing is given by Figure 49 and factored in peak power load when thrusting near Mars, the solar constant for a given distance from the Sun, and ROSA performance parameters. A total array area of 57.45 m^2 accommodates the peak power loads both when thrusting near Mars and near Earth. The power generated in sunlight exceeds peak power loads of the instruments and bus when operating in science mode, meaning the solar arrays are capable of powering full operation in sunlight should the battery storage fail.

Figure 49: Solar array sizing

Parameter	Value
Peak power load	6062.16 W
Solar constant (Mars)	586.2 W/m^2
Solar constant (Earth)	1350 W/m^2
Concentration ratio	0.6
Cell efficiency	30%
Specific power (Earth)	100 W/kg
Stowed power density	40 W/m^3
BOL Power (Mars)	**6062.16 W**
BOL Power (Earth)	**13960.96 W**
Total mass	**126.92 kg**
Total array area	**57.45 m^2**
Total stowed volume	**0.152 m^3**

The battery sizing is given by Figure 50 and factored in energy usage in eclipse, the desired DoD, and 18650 Li-Ion cell performance parameters. A 20% DoD is

Figure 50: Battery sizing

Parameter	Value
Eclipse power load	1181.38 W
Max time in eclipse	0.67 hr
Desired DoD	20%
Concentration ratio	0.6
18650 Cell specific energy	10 Wh/cell
Cell mass	0.045 kg/cell
Cell volume (rectangular)	2.11×10^{-5} m^3
Number of cells	**394 cells**
BOL stored energy	**3940 Wh**
Total mass	**17.73 kg**
Total volume	**8.30×10^{-3} m^3**

prescribed to prolong battery health and capacity for a mission duration of five years resulting in upwards of 23,000 battery cycles. Analysis in FreeFlyer simulation verifies DoD does not exceed this value of 20% over the course of a full year. This data is given by Figure 51. Data was exported for plotting in Python to display un-culled data. The red line represents the charge/discharge cycle of the batteries, and the period with zero discharge indicates a period when OPTIMUS is in sunlight for the full period. Cells are arranged in series and parallel to provide power at adequate voltage and current for instruments and bus components. Battery storage ensures that a constant power level is supplied to the payload through the sunlight-eclipse cycle of the orbit.

The power conversion, management, and distribution unit (PDU) interfaces between the solar arrays, batteries, and instruments and satellite bus. The PDU regulates and distributes power to instruments and subsystems and controls the state of charge of the batteries. The PDU also interfaces with the C&DH subsystem to accept commands and report bus voltage and the state of charge for inclusion in telemetry. Prior to deployment of OPTIMUS from the launch vehicle, power is inhibited by two independent inhibit switches actuated by physical deployment switches. Power is inhibited until the physical switches are activated by deployment from the launch vehicle.

4.4.7. Propulsion

The primary purpose of the propulsion system is to maneuver OPTIMUS from Earth orbit to Mars orbit. In addition, the propulsion system provides a means to deorbit OPTIMUS at end-of-mission (EOM). OPTIMUS is deorbited rather than raised to a parking orbit to keep orbital shells clear for future missions.

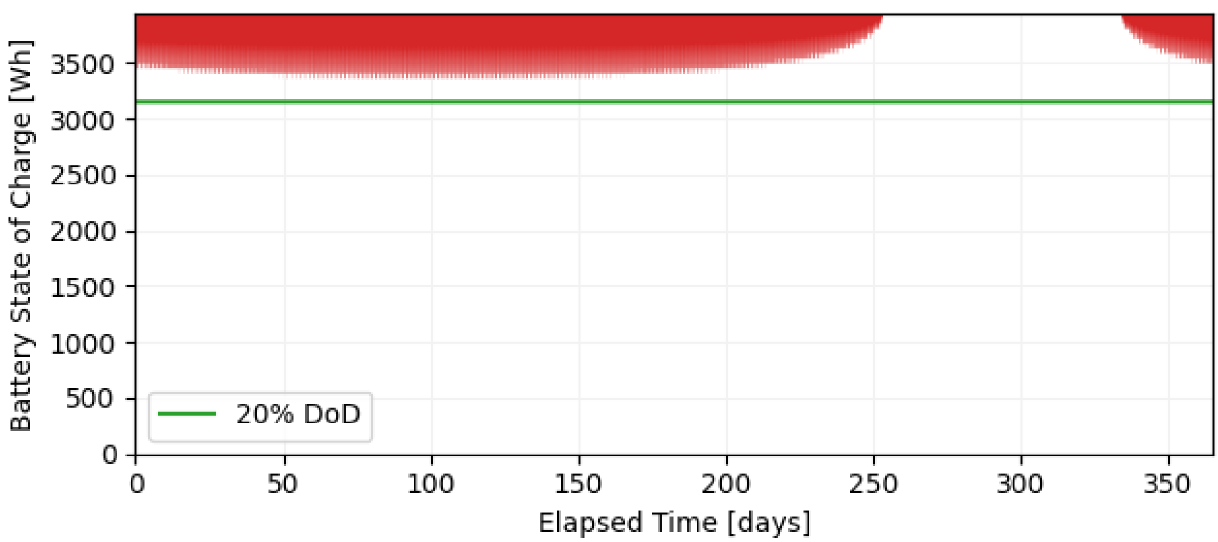


Figure 51: Battery charge/discharge model obtained from FreeFlyer.

OPTIMUS is constructed in a clean-room environment to mitigate the risk of microorganism contamination from deorbiting.

The Earth-Mars transfer necessitates a mass-efficient propulsions system that can deliver required ΔV with sufficient I_{sp} to maximize payload mass while simultaneously providing appropriately high thrust to minimize spacecraft lifetime allocated to interplanetary transfer. Payload ratio and transfer time are the primary drivers of propulsion system design.

The team evaluated the performance of conventional chemical propulsion systems alongside solar electric propulsion (SEP) systems. Based on orbital mechanics simulations as well as analysis of past, present, and planned missions using chemical and SEP, preliminary sizing for a 1500 kg dry-mass satellite using a hydrazine mono-prop system and SEP was computed and compared. The initial sizing estimates are shown below in Figure 52. The system specifications used represent the near-cutting-edge specifications of either architecture at the time of writing.

Due to the nature of high-impulse burns, the ΔV requirements of chemical mono-prop were found to be 4 times than if an SEP system is used. However, due to the higher Isp of SEP, the propellant mass needed by chemical mono-prop systems was approximately 3 times greater. It should be noted that the primary disadvantage of SEP compared to chemical mono-prop is the increase in transfer time associated with a low-thrust architecture. However, a waiver was received to extend the mission past its 5-year lifetime to the extent needed to meet mission requirements. For the aforementioned reasons, OPTIMUS uses SEP thus allowing for higher spacecraft payload ratios.

The team evaluated three candidate SEP architectures to be used on OPTIMUS, namely, Arcjets, Gridded Ion Thrusters (GIT), Hall Effect Thrusters (HET). The team chose to evaluate the aforementioned architectures as these thrusters have the highest TRL and total

Figure 52: Chemical Propulsion vs. SEP Trade

Chemical Propulsion vs SEP Comparison of 1500 kg Spacecraft, Earth-Mars Transit			
Chemical		**SEP**	
ΔV [km/s]	2	ΔV [km/s]	8
Isp [s]	250	Isp [s]	2200
Thrust [N]	1020	Thrust [mN]	660
Prop Mass [kg]	1892	Prop. Mass [kg]	673
Transfer Time [Months]	7	Transfer Time [Months]	24

amount of flight hours and thus highest reliability among the various SEP architectures.

The specifications of the most cutting-edge systems of the three architectures are tabulated above in Figure 53. For the same ΔV of 4 km/s the HETs provide considerable mass saving over Arjects while having a smaller power envelope than GITs. For the aforementioned reasons the team evaluated three HET options with the highest specific-power, namely the Aerojet Rocketdyne XR-5, the SPT-140, and the Exoterra HALO-12 to be used on OPTIMUS. The team chose HALO-12 over the XR-5 and SPT-140 to mitigate the risk of engine-out scenarios. The total rated throughput of a single HALO-12 engine is greater than 500 kg and operates at 1.65 kW

Figure 53: SEP Architecture Trade

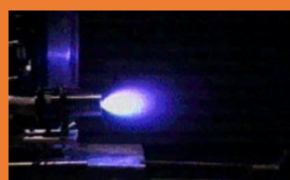
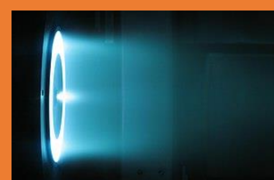
Comparison of 3 Candidate Space Electric Propulsion Architectures for 1500 kg Spacecraft, Earth-Mars Transit			
Parameter	**Arc Jet Thruster**	**HET**	**GIT**
Power [kW]	**1.8**	**1. 5**	2.3
ΔV [km/s]	4		
Peak Thrust [mN]	254	110	92
Isp [s]	500	2200	3100
Propellant Mass [kg]	1880.38	**305.56**	**210.94**
			

Figure 54: Propulsion Sizing Chart

Hall Effect Thruster (Earth/Mars)		**Xenon Propellant Tank**	
Max Power/Unit [W]	1650	Dimensions	1.17 m x 1.14 m (diameter)
Total Peak Power[kW]	9.9 / 4.6	Pressure [MPa]	18.7
Isp [s]	2200	Prop. Volume [Liters]	848
Thrust/Unit [mN]	110	**Item**	**Mass [kg]**
Total Peak Thrust [mN]	606 / 306	HET	21.01
ΔV [km/s]	8	Power Processing Unit	23.48
Item	**Quantity**	Propellant Tank	81.68
HALO-12 HET	6	Tank Heater	0.650
Power Processing Unit	6	Feed System	7.150
Propellant Tank	1	Hardware Total	133.9
Tank Heater	1	Propellant	841.4
Feed System	1	Subsystem Total	981.2

Figure 55: Propulsion System Telemetry Data Rates

Sensor	Qty	Bits/read	Read/s	Total bps	Resolution	Range
Thermocouple	6	12	1	72	.16 [°C]	-270 – 400 [°C]
Pressure Transducer	2	16	60	1920	.1 [psi]	0 – 3000 [psi]
Xenon-flow Controller	6	28	60	10080	N/A	N/A
Voltage	6	16	60	5760	12.2 [mV]	0 – 800 [V]
Current	6	16	60	5760	.09 [mA]	0 – 6 [A]
Power	6	16	60	5760	.03 [W]	0 – 1800 [W]
Total	**0.03 Mbps**					

with an estimated firing time of greater than 20000 hours. In comparison, the XR-5 and SPT-140 have rated throughput of less than 400 kg. Coupled with the larger power envelope of these two thrusters at 4.5 kW, OPTIMUS would only be able to accommodate two of these engines within a sub 10 kW power budget meaning that any engine-out scenario would catastrophically jeopardize the mission. The final propulsion system of OPTIMUS consists of a cluster of six HALO-12 HETs that allow up to two inoperable engines with no risk to the mission. Additionally, the HALO-12 HET is anticipated to achieve a TRL of 7 by Q4 2023, further insuring reliability.

The propellant of choice is Xenon. The team also considered using Krypton as a cheaper alternative, however, due to the 11-17% decrease in thrust and 8-11% decrease in specific impulse leading to a longer transfer time with a higher propellant mass penalty, Xenon is ultimately used in the propulsion system of OPTIMUS. To provide a 8 km/s ΔV OPTIMUS carries 841 kg of Xenon pressurized to 187 bar. The Xenon is stored in a high specific strength carbon-fiber composite

The propellant of choice is Xenon. The team also considered using Krypton as a cheaper alternative, however, due to the 11-17% decrease in thrust and 8-11% decrease in specific impulse leading to a longer transfer time with a higher propellant mass penalty, Xenon is ultimately used in the propulsion system of OPTIMUS. To provide a 8 km/s ΔV OPTIMUS carries 841 kg of Xenon pressurized to 187 bar. The Xenon is stored in a 900-liter high specific strength carbon-fiber composite Large Xenon Storage Tank (L-XTA). The final specifications of OPTIMUS' propulsion system is tabulated in Figure 54.

To monitor propulsion system health, the subsystems collect and sends telemetry data to CD&H. Sensor include thermocouples to monitor tank temperature, pressure transducers to monitor tank pressure, xenon-flow controllers to control and monitor propellant mass flow rate, and lastly voltage, current, and power monitoring capabilities included in the power processing unit to monitor and throttle the thrusters. The bitrates of propulsion system telemetry data are tabulated in Figure 55.

4.4.8. Communications

The primary purpose of the communications subsystem is to close the link between our OPTIMUS and PRIME, and that between PRIME and the Deep Space Network (DSN). The encrypted data received is sent to the Command and Data Handling (C&DH) subsystem to be processed. Similarly, the science and telemetry data generated by the satellite is received from the C&DH subsystem to then be transmitted back to the DSN.

The main design drivers for the communications subsystem of OPTIMUS are the maximum distance between OPTIMUS and the closest PRIME satellite, the power available to the subsystem, the desired uplink data rate, and the frequency desired for those. The subsystem is designed so that the link margin for both uplink and downlink modes do not go below 3 dB at the worst-case scenario, as is conventional. The worst-case scenario values for the OPTIMUS-PRIME link budget can be seen below in Figure 56.

Figure 56: OPTIMUS to PRIME downlink.

Parameter	Value
Data Rate	129 Mbps
RX Sensitivity	-83 dBm
FSPL	197.9 dB
Efficiency	60 %
Antenna Diameter	1 m
TX Frequency	7200 MHz
RX Frequency	7200 MHz
Antenna Gain	36.7 dBi
TX Power	75 W
Cable Loss	2 dB
Link Margin	**3.21 dB**

Note that the values are only given once as the radios for OPTIMUS and PRIME-relay are identical.

The main design drivers for the communications subsystem of PRIME are the maximum distance between PRIME and the DSN, the power available to the subsystem, the desired uplink data rate, and the frequency desired for those. Like OPTIMUS, the system also requires a minimum link margin of 3 dB. The worst-case scenario values for the PRIME-DSN link budget, only mentioning space radio specifications, can be seen below in Figure 57.

Figure 57: PRIME to DSN downlink.

Parameter	Value
Data Rate	8.6 Mbps
DSN RX Sensitivity	-100 dBm
FSPL	281.1 dB
Efficiency	70 %
Antenna Diameter	6 m
TX Frequency	8420 MHz
RX Frequency	7250 MHz
Antenna Gain	52.9 dBi
TX Power	2 kW
Cable Loss	2 dB
Link Margin	**3.36 dB**

As we have a waiver to not fully design PRIME, the numbers given in the figure above are not final and are only meant to show that closing the link is possible.

The communications subsystem for OPTIMUS is composed of three components: the antenna, the power amplifier, and the transceiver. The antenna was chosen to minimize our power requirements and mass for the subsystem, while still closing the link with PRIME. The signal amplifier is necessary to amplify the encrypted data received from the command and data handling subsystem to a desired amplitude that is compatible with the capacity of the antenna. The transceiver allows for our system to do both transmitting and receiving signals. It serves to convert received signals into electrical form so that it can be analyzed by our command and data handling system, and it converts electrical signals into suitable form for transmission.

The mass, power, and estimated dimensions for the communications subsystem can be seen below.

Figure 58: OPTIMUS radio sizing.

Component	Dimensions	Mass [g]	Power [W]
X-Band Parabolic Dish Antenna	1 m diameter	600	75
X-band Amplifier	5.3×2.9×1.1 cm^3	52.6	30
X-band Transceiver	9.6×7.1×3.2 cm^3	200	16
	Total	**852.6**	**121**

5. Life Cycle Cost Drivers

Major cost drivers for the SCOPE program are identified. Potential opportunities to avoid unnecessary costs are included.

An overall cost driver for the program is the power required by the instrument suite onboard OPTIMUS and the TT&C hardware onboard PRIME, respectively. To maximize temporal resolution and science coverage for sensors aboard OPTIMUS as well as increase performance for PRIME radionavigation and communications relay systems, satellites in each constellation require more power. This in turn leads to increase volume and mass which may manifest as larger batteries and solar panels, which directly lead to higher component costs and indirectly lead to higher costs in other mission segments including but not limited to launch, attitude control, and propulsion. Consideration should be given to the operation and design of these systems such that power needs are balanced with other requirements.

Launch costs vary greatly depending on the number of launches needed and the type of vehicle used. Not all launch vehicles considered for the SCOPE mission have flown, and it is unclear whether launch providers' claimed launch costs hold true at time of launch. Additionally, development and testing of bus and payload systems generally represents a majority or large minority of satellite development costs. With the large number of satellites that will be launched as part of the SCOPE mission, uniformity in design saves money and time in development as well as testing.

Fabrication, acquisition, and/or integration of the instrument payloads aboard OPTIMUS and the DSAC units aboard PRIME. The scientific instruments for OPTIMUS are purpose-built instruments which are not COTS systems and are expensive to fabricate and acquire as custom hardware. The DSAC units for PRIME are also custom hardware which are be expensive to acquire, handle, and integrate.

6. Compliance Matrix

Identifier	Requirement	Page Number
*MR-01**	*The mission architecture shall provide navigational resources for assets transiting between orbit and ground.*	*Waived*
*MR-02**	*The mission architecture shall provide navigational resources for assets on the ground.*	*Waived*
*MR-03**	*The mission architecture shall provide a relay for communications between Earth and assets on Mars.*	*Waived*
*MR-04**	*The mission architecture shall provide a relay for communications between Earth and assets near Mars.*	*Waived*
MR-05	**The mission architecture shall determine the mineral make-up of Mars.**	pp. 6, 7, 8, 12, 15, 24, 25, 26, 27, 28, 29, 30, 31
MR-06	**The mission architecture development timeline shall be 36 months.**	p. 9
MR-07	**The mission architecture shall enable a minimum design lifetime of 5 years, including ground test and operations.**	pp. 7, 9
MR-08	**The mission architecture shall include a bus design capable of orbiting Mars.**	pp. 13, 14, 15
MR-09	**The mission architecture shall include an imaging payload.**	pp. 24, 25, 26, 27, 28, 29
MR-10	**The mission architecture shall include a spectroscopic payload.**	pp. 24, 25, 26, 27, 28, 29
MR-11	**The mission architecture shall minimize propulsion requirements for achieving orbit at Mars.**	pp. 13, 14, 15, 36, 37, 38
MR-12	**The mission architecture shall achieve coverage requirements at minimum cost.**	pp. 13, 14, 15, 24, 25, 26, 27, 28, 29, 30, 31

**Navigation & relay requirements are waived*

Identifier	Requirement	Page Number
MR-13	**The mission architecture structure shall facilitate installation of payloads late in the integration flow.**	pp. 9, 10
LV-01	**The launch vehicle(s) shall deliver the satellite(s) on a trajectory to Mars.**	pp. 11, 12, 13, 14, 15
LV-03	**The launch vehicle(s) shall release the satellite(s) after reaching the desired trajectory.**	pp. 13, 14, 15
LV-04	**The launch vehicle(s) shall accommodate the physical characteristics of the satellite(s).**	pp. 11, 12, 19, 20
GS-01	**The Ground Control System shall send uplink data to the satellite.**	p. 12
GS-02	**The Ground Control System shall receive downlinked data from the satellite.**	p. 12
GS-03	**The Ground Control System shall encrypt data uplink.**	p. 12
GS-04	**The Ground Control System shall decrypt data downlink.**	p. 12
GS-05	**Ground Control shall have the ability to operate when human operation is not available.**	p. 12
SAT-01	**The satellite(s) shall generate power.**	p. 35
SAT-02	**The satellite(s) shall store energy.**	p. 36

Identifier	Requirement	Page Number
SAT-03	**The satellite(s) shall communicate with Earth.**	p. 40
SAT-04	**The satellite(s) shall maintain orbit around Mars to achieve mission objectives.**	pp. 13, 14, 15
SAT-05	**The satellite(s) shall include an imaging payload to determine the mineral makeup of Mars.**	pp. 15, 24, 25, 26, 27, 28, 29, 30, 31
SAT-06	**The satellite(s) shall include a spectroscopic payload to determine the mineral makeup of Mars.**	pp. 24, 25, 26, 27, 28, 29, 30, 31
SAT-07	**The satellite(s) shall store data.**	pp. 21, 22, 23
SAT-08	**The satellite(s) shall maintain an operational thermal environment.**	p. 17
SAT-09	**The satellite(s) shall provide a structural interface for all subsystems.**	p. 16, 17, 18, 19, 20
SAT-10	**The satellite(s) shall accommodate payload instrumentation.**	p. 16, 17, 18, 19, 20
*SAT-11**	*The satellite(s) shall provide navigational resources for assets transiting between orbit and ground.*	*Waived*
*SAT-12**	*The satellite(s) shall provide navigational resources for assets on the ground.*	*Waived*
*SAT-13**	*The satellite(s) shall provide a relay for communications between Earth and assets on Mars.*	*Waived*
*SAT-14**	*The satellite(s) shall provide a relay for communications between Earth and assets near Mars.*	*Waived*

**Navigation & relay requirements are waived*

Identifier	Requirement	Page Number
SAT-15	**The satellite(s) shall conform to the payload mass limit of the launch vehicle(s).**	pp. 11, 12, 18, 19, 20
SAT-16	**The satellite(s) shall have a minimum design lifetime including ground test and operations.**	pp. 9, 10
SAT-17	**The satellite(s) shall determine orientation in three axes.**	pp. 32, 33
SAT-18	**The satellite shall provide control for orientation in three axes.**	pp. 32, 33
INS-01	**The Instrumentation Subsystem shall collect SAR images of Mars.**	pp. 15, 24, 25, 26, 27, 28, 29
INS-02	**The Instrumentation Subsystem shall collect radar sounding data of Mars.**	pp. 15, 24, 25, 26, 27, 28, 29
INS-03	**The Instrumentation Subsystem shall collect hyperspectral spectroscopic data of minerals on Mars.**	pp. 15, 24, 25, 26, 27, 28, 29, 30, 31
INS-04	**The Instrumentation Subsystem collect multispectral spectroscopic data of minerals on Mars.**	pp. 15, 24, 25, 26, 27, 28, 29, 30, 31
INS-05	**The Instrumentation Subsystem shall collect stereoscopic images of Martian topography.**	pp. 15, 24, 25, 26, 27, 28, 29, 30, 31
INS-06	**The Instrumentation Subsystem shall collect visual images of the Martian surface.**	pp. 15, 24, 25, 26, 27, 28, 29, 30, 31
*INS-07**	*The Instrumentation Subsystem shall provide timing data to assets utilizing the Instrumentation Subsystem.*	*Waived*
*INS-08**	*The Instrumentation Subsystem shall provide satellite navigation data to assets utilizing the Instrumentation Subsystem.*	*Waived*

**Navigation & relay requirements are waived*

Identifier	Requirement	Page Number
INS-09	**The Instrumentation Subsystem shall pass telemetry to the C&DH subsystem.**	pp. 25, 26, 27, 28, 29
INS-10	**The Instrumentation Subsystem shall accept commands from the C&DH subsystem.**	pp. 25, 26, 27, 28, 29
INS-11	**The Instrumentation Subsystem shall accept electrical power from the power subsystem.**	pp. 25, 26, 27, 28, 29
INS-12	**The Instrumentation Subsystem shall collect calibration images.**	pp. 25, 26, 27, 28, 29
INS-01.01	**The SAR shall achieve at least 80 [%] global coverage of Mars over the mission operation duration.**	pp. 15, 24, 25
INS-01.02	**The SAR shall detect near surface ice within 50 [m] of the surface.**	pp. 24, 25
INS-01.03	**The SAR shall achieve a spatial resolution of < 50 [m].**	pp. 24, 25
INS-01.04	**The SAR shall collect images of Mars in full polarimetry.**	pp. 15, 24, 25
INS-02.01	**The Sounder shall achieve at least 80 [%] global coverage of Mars over the mission operation duration.**	pp. 15, 24, 25, 29
INS-02.02	**The Sounder shall survey the subsurface between a range of 0 [m] < DEPTH < 40 [m].**	pp. 24, 25
INS-02.03	**The Sounder shall achieve a vertical resolution of up to 1 [m] of the subsurface.**	p. 25

Identifier	Requirement	Page Number
INS-02.04	**The Sounder shall achieve a spatial resolution of < 1 [km] of the subsurface.**	pp. 24, 25
INS-03.01	**The SWIR Spectrometer shall achieve at least 1 [%] global coverage of minerals over the mission operation duration.**	pp. 15, 24, 25, 26
INS-03.02	**The SWIR Spectrometer shall collect hyperspectral data of wavelength 1.3 [μm] < wavelength < 4.2 [μm].**	pp. 24, 25, 26
INS-03.03	**The SWIR Spectrometer shall achieve a spatial resolution of < 10 [m] for hyperspectral data.**	pp. 24, 25, 26
INS-03.04	**The SWIR Spectrometer shall achieve a spectral resolution of < 20 [nm] for hyperspectral data.**	pp. 24, 25, 26
INS-04.01	**The LWIR Spectrometer shall achieve at least 40 [%] global multi-spectral coverage of minerals over the mission operation duration.**	pp. 15, 24, 25, 26, 27
INS-04.02	**The LWIR Spectrometer shall collect multi-spectral spectroscopic data of wavelengths 6 [μm] < wavelength < 25 [μm] TBR.**	pp. 24, 25, 26, 27
INS-04.03	**The LWIR Spectrometer shall achieve a spatial resolution of < 15 [m] for multi-spectral data.**	pp. 24, 25, 26, 27
INS-04.04	**The LWIR Spectrometer shall collect multi-spectral spectroscopic data in 20 [channels].**	pp. 24, 25, 26, 27
INS-04.05	**The LWIR Spectrometer shall achieve a bandpass of < 2 [μm].**	pp. 26, 27
INS-05.01	**The Context Cameras shall achieve at least 80 [%] global coverage of topography over the mission operation duration.**	pp. 15, 28, 29
INS-05.02	**The Context Cameras shall achieve a spatial resolution of < 20 [m] for topographic mapping.**	pp. 28, 29

Identifier	Requirement	Page Number
INS-05.03	**The Context Cameras shall collect images of the same location twice to create stereoscopic image pairs.**	pp. 28, 29
INS-05.04	**The Context Cameras shall collect the secondary image at a different angle than the first for topographic mapping.**	pp. 28, 29
INS-05.05	**The Context Cameras shall collect panchromatic images.**	pp. 28, 29
INS-06.01	**The High-Resolution Camera shall achieve at least 1 [%] global coverage of the surface over the mission operation duration.**	pp. 15, 27, 28, 29
INS-06.02	**The High-Resolution Camera shall achieve a spatial resolution of < 5 [m] when imaging the surface.**	pp. 27, 28
INS-06.03	**The High-Resolution Camera shall collect images of the surface between wavelengths 0.4 [µm] < wavelength < 1.7 [µm].**	pp. 27, 28
INS-06.04	**The High-Resolution Camera shall have 20 [channels].**	p. 28
INS-06.05	**The High-Resolution Camera shall achieve a bandpass of < 60 [nm].**	p. 28
*INS-07.01**	*The Instrumentation Subsystem shall achieve accurate timekeeping of TBD [ns].*	*Waived*
*INS-07.02**	*The Instrumentation Subsystem shall achieve timekeeping stability of TBD [ns] over TBD [years].*	*Waived*
*INS-08.01**	*The Instrumentation Subsystem shall store orbit ephemeris in TBD [kb] of non-volatile modifiable memory.*	*Waived*
*INS-08.02**	*The Instrumentation Subsystem shall transmit satellite navigation data to users at TBD [MHz].*	*Waived*

**Navigation & relay requirements are waived*

Identifier	Requirement	Page Number
*INS-08.03**	*The Instrumentation Subsystem shall transmit satellite navigation data to users at TBD [MHz].*	*Waived*
INS-09.01	**The Instrumentation Subsystem shall maintain a 0.1 [Mbps] < average data of rate < 0.8 [Mbps] when passing data to C&DH.**	pp. 25, 26, 27, 28, 29
INS-09.02	**The Instrumentation Subsystem shall maintain a data of rate that is less than 1 [Mbps] when passing data to C&DH.**	pp. 25, 26, 27, 28, 29
INS-10.01	**The Instrumentation Subsystem shall accept a 0.1 [Mbps] < average data of rate < 0.8 [Mbps] when receiving commands from C&DH.**	pp. 25, 26, 27, 28, 29
INS-10.02	**The Instrumentation Subsystem shall receive a data of rate that is less than 1 [Mbps] from C&DH.**	pp. 25, 26, 27, 28, 29
INS-11.01	**The Instrumentation Subsystem shall accept 5 < power [W] < 500 from the power subsystem.**	p. 29
COMM-01	**The communications subsystem shall accept uplinked data from ground control.**	p. 12
COMM-02	**The communications subsystem shall downlink data to ground control.**	p. 40
*COMM-03**	*The communications subsystem shall accept uplinked data from Mars surface assets.*	*Waived*
*COMM-04**	*The communications subsystem shall downlink data to Mars surface assets.*	*Waived*

**Navigation & relay requirements are waived*

Identifier	Requirement	Page Number
*COMM-05**	*The communications subsystem shall uplink data from Mars orbital assets.*	*Waived*
*COMM-06**	*The communications subsystem shall downlink data to Mars orbital assets.*	*Waived*
COMM-07	**The communications subsystem shall accept data from the C&DH subsystem.**	p. 39
COMM-08	**The communications subsystem shall pass data to the C&DH subsystem.**	p. 39
COMM-01.01	**The communications subsystem shall maintain 8.4 [Mbps] < average data of rate < 8.8 [Mbps] when uplinking from ground.**	p. 40
COMM-01.02	**The communications subsystem shall close the link with the ground station at no less than 3 [dBm] above RX sensitivity at any point during the mission.**	p. 40
COMM-02.01	**The communications subsystem shall maintain 8.4 [Mbps] < average data of rate < 8.8 [Mbps] when downlinking to ground.**	p. 40
COMM-02.02	**The communications subsystem shall close the link with the ground station at no less than 3 [dBm] above RX sensitivity at any point during the mission.**	p. 40
*COMM-03.01**	*The communications subsystem shall maintain TBD [kbps] < average data of rate < TBD [kbps] when uplinking data from Mars surface assets.*	p. 40
*COMM-04.01**	*The communications subsystem shall maintain TBD [kbps] < average data of rate < TBD [kbps] when downlinking data to Mars surface assets.*	*Waived*
*COMM-05.01**	*The communications subsystem shall maintain TBD [kbps] < average data of rate < TBD [kbps] when uplinking data from Mars orbital assets.*	*Waived*

**Navigation & relay requirements are waived*

Identifier	Requirement	Page Number
*COMM-06.01**	*The communications subsystem shall maintain TBD [kbps] < average data of rate < TBD [kbps] when downlinking data to Mars orbital assets.*	*Waived*
COMM-07.01	**The communications subsystem shall maintain 0.1 [Mbps] < average data of rate < 0.8 [Mbps] when accepting data from C&DH.**	p. 39
COMM-08.01	**The communications subsystem shall maintain 0.1 [Mbps] < average data of rate < 0.8 [Mbps] when passing data to C&DH.**	p. 39
PROP-01	**The propulsion subsystem shall maneuver spacecraft from Earth orbit to Mars orbit.**	pp. 13, 14, 38
PROP-02	**The propulsion system shall deorbit the spacecraft into Mars at EOM.**	pp. 8, 9, 36, 37, 38
PROP-03	**The propulsion subsystem shall pass telemetry to C&DH.**	pp. 21, 23, 39
PROP-04	**The propulsion subsystem shall receive commands from C&DH.**	pp. 21, 23
PROP-05	**The propulsion subsystem shall receive electrical power from the power subsystem.**	pp. 34, 38
PROP-06	**The propulsion subsystem shall conform to mission power budget.**	pp. 34, 38
PROP-07	**The propulsion subsystem shall be reliably operable throughout the service life of the mission.**	p. 39
PROP-08	**The propulsion subsystem shall enable an appropriate payload ratio to accommodate all instrumentation without unnecessarily driving up launch cost.**	p. 38

Navigation & relay requirements are waived

Identifier	Requirement	Page Number
PROP-01.01	**The propulsion subsystem shall provide a total ΔV of 8 km/s for the mission.**	p. 38
PROP-01.02	**The propulsion subsystem shall allocate 4 km/s of the of 8 km/s ΔV budge to the Earth-Mars transfer maneuver, including trajectory corrections.**	pp. 13, 14, 38
PROP-01.03	**The propulsion subsystem shall allocate 3 km/s of the of 8 km/s ΔV budget to the low-thrust spiral maneuver for Mars orbit-insertion.**	pp. 13, 14, 38
PROP-01.04	**The propulsion subsystem shall allow spacecraft to reach Mars orbit at most 730 days (about 2 years) after escaping Earth orbit.**	pp. 13, 14, 38
PROP-02.01	**The propulsion subsystem shall allocate 1 km/s of the 8 km/s ΔV budget for de-orbiting at EOM, orbit maintenance, and miscellaneous burns.**	pp. 13, 14, 38
PROP-03.01	**The propulsion subsystem shall measure propellant pressure with at least 0.1% full-scale resolution.**	p. 39
PROP-03.02	**The propulsion subsystem shall measure propellant temperature with at least 0.1% full-scale resolution.**	p. 39
PROP-03.03	**The propulsion subsystem shall measure anode voltage with at least 0.1% full-scale resolution.**	p. 39
PROP-03.04	**The propulsion subsystem shall measure anode current with at least 0.1% full-scale resolution.**	p. 39
PROP-03.05	**The propulsion subsystem shall measure total subsystem power draw with at least 0.1% full-scale resolution.**	p. 39
PROP-03.06	**The propulsion subsystem shall maintain an average data of rate 30 kbps when passing system telemetry to C&DH.**	pp. 21, 23, 39
PROP-04.01	**The propulsion subsystem shall maintain an average data rate of 30 kbps when accepting commands from C&DH.**	pp. 21, 23, 39

Identifier	Requirement	Page Number
PROP-05.01	**The propulsion subsystem shall receive no less than 9.9 kW of peak power from the power subsystem at 1 AU at 100% duty cycle.**	pp. 34, 38
PROP-06.01	**The propulsion subsystem shall require no more than 9.9 kW from the power subsystem at 1 AU at 100% duty cycle.**	pp. 34, 38
PROP-07.01	**The propulsion subsystem shall have an MTBF > 17520 hrs. (100% duty-cycle for 2 years)**	p. 39
PROP-08.01	**The propulsion subsystem shall enable a payload ratio of at least 0.55.**	pp. 18, 38
ACS-01	**The attitude control subsystem shall determine orientation of the satellite in 3 degrees of freedom.**	p. 32
ACS-02	**The attitude control subsystem shall provide pointing control in 3 degrees of freedom.**	p. 32
ACS-03	**The attitude control subsystem shall provide telemetry to C&DH.**	p. 33
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ACS-06	**The attitude control subsystem shall be reliably operable through the service life of the mission.**	p. 33
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ACS-01.02	**The attitude control subsystem shall determine the pitch angle of the spacecraft to a tolerance of ≤ .05 [deg].**	p. 32
ACS-01.03	**The attitude control subsystem shall determine the yaw angle of the spacecraft to a tolerance of ≤ .05 [deg].**	p. 32
ACS-02.01	**The attitude control subsystem shall point the spacecraft in the roll axis to a tolerance of ≤ .5 [deg] from the commanded roll angle while in normal operation.**	p. 32
ACS-02.02	**The attitude control subsystem shall point the spacecraft in the pitch axis to a tolerance of ≤ .5 [deg] from the commanded pitch angle while in normal operation.**	p. 32
ACS-02.03	**The attitude control subsystem shall point the spacecraft in the yaw axis to a tolerance of ≤ .5 [deg] from the commanded yaw angle while in normal operation.**	p. 32
ACS-03.01	**The attitude control subsystem shall maintain an 2.56 [kbps] < average data of rate < 3.00 [kbps] while passing spacecraft orientation to C&DH.**	p. 33
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PWR-02.03	**The power subsystem storage device shall supply power at TBD [V] < volts < TBD [V].**	p. 36
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PWR-03.01	**The power subsystem shall distribute 0 [W] < power < 366 [W] to the payload.**	p. 34
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CDH-11.01	**The C&DH subsystem shall maintain a 0.1 [Mbps]<average data of rate< 0.8 [Mbps] when accepting telemetry data from the power subsystem.**	p. 23
CDH-11.02	**The C&DH subsystem shall receive a data at rate < 1 [Mbps] from the power subsystem.**	p. 23
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