Phase I Study Report

For The

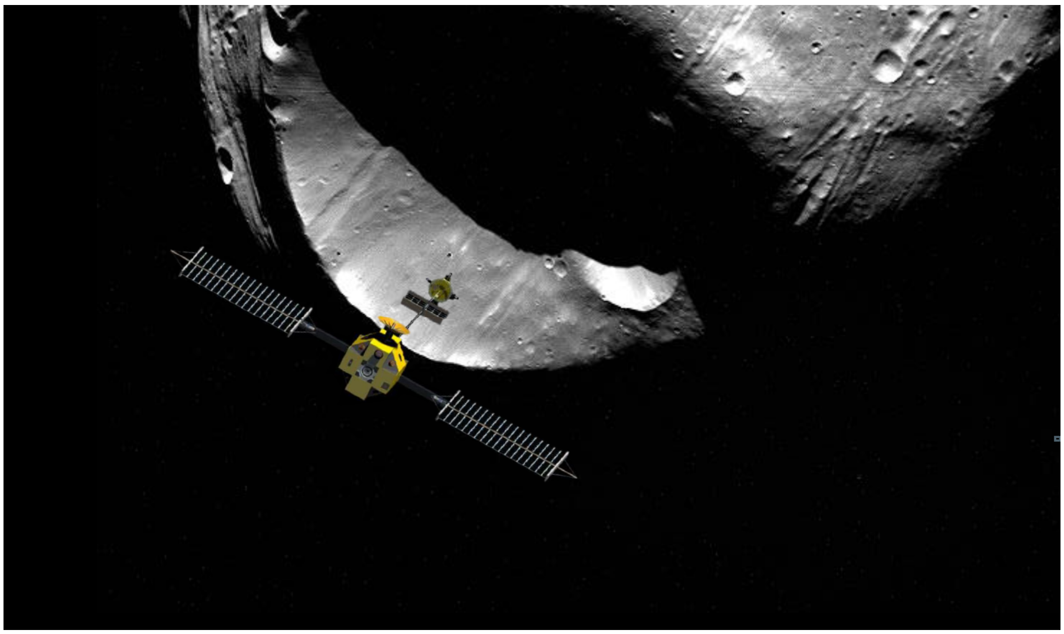
Phobos L1 Operational Tether Experiment (PHLOTE)

NASA Advanced Innovative Concepts
Phase 1 Study

FINAL REPORT

NASA Innovative Advanced Concepts (NIAC)

PHLOTE: Phobos Lagrange-1 Operational Tether Experiment



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Introduction

NASA's Innovative Advanced Concepts (NIAC) program selected the PHLOTE mission proposal for a 2017 Phase 1 study. This PHLOTE study provides a credible example of an innovative mission architecture that can be used to enable many future missions throughout the solar system.

One of the key Phase 1 deliverables identified in the PHLOTE proposal is this Study Report which is derived from the PHLOTE Concept of Operations (ConOps) Document developed during the study. The PHLOTE ConOps describes the PHLOTE mission and also provides a key systems engineering document to support future mission development.

Since this report was produced as part of a NIAC feasibility study, it is intended to be publicly released at the completion of the NIAC study. Significant support was provided through the collaboration of NASA and PHLOTE team members from Space Technology And Research (STAR) Inc. and from the Clouds Architecture Office (Clouds AO). The NASA team was supported by summer and fall interns from five separate universities.

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PHLOTE Mission Overview

A PHLOTE spacecraft would hover at or very near the Mars-Phobos Lagrange 1 (L1) point and suspend a tethered sensor package that would "float" just above the moon's surface. Adjusting PHLOTE's center of gravity by adjusting the tether length would allow the spacecraft to change the altitude of the sensor package over Phobos without expending propellant and also support station keeping operations. If desired, the PHLOTE system could set instruments directly on the surface for in situ science measurements.

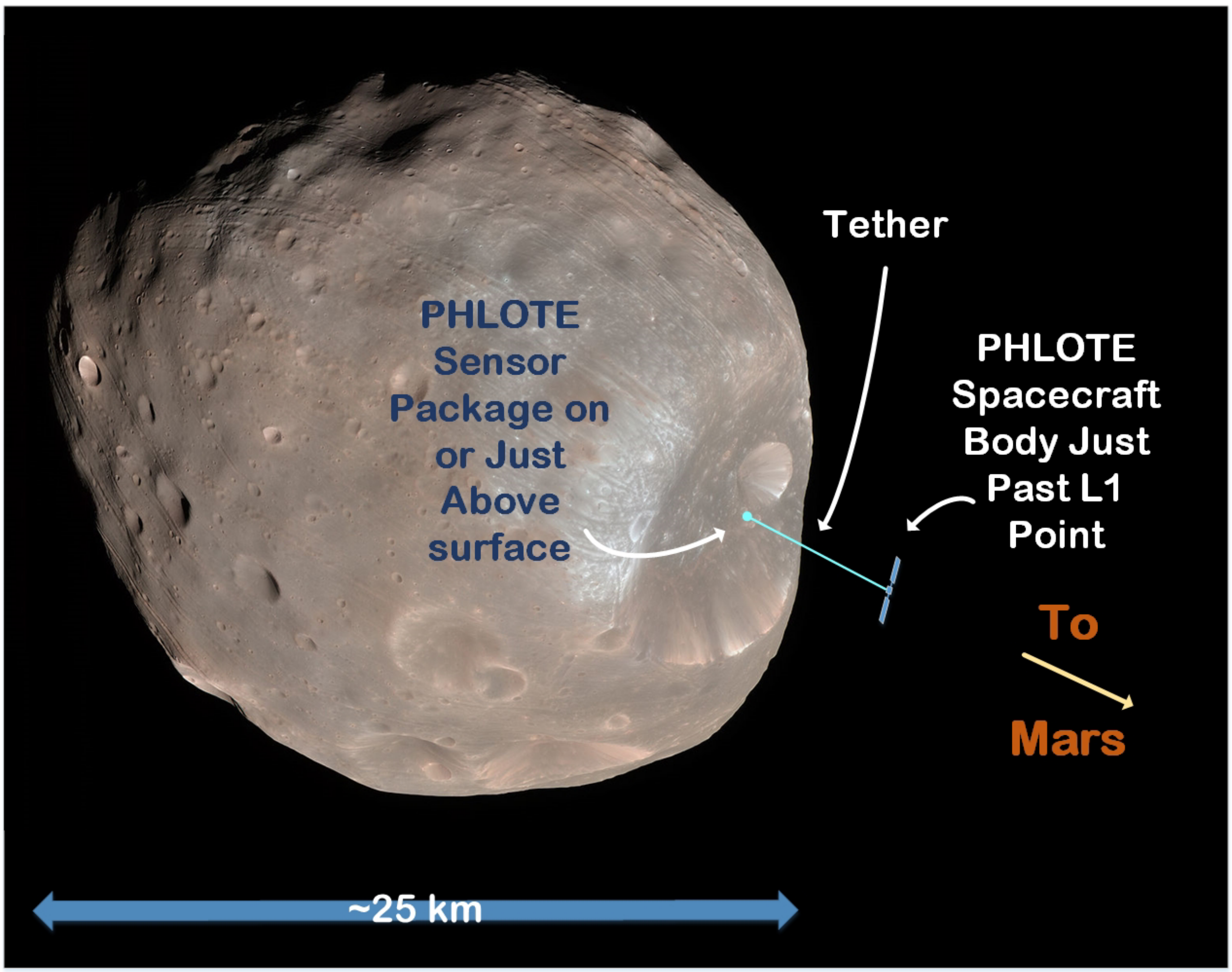


Figure 1: Length of the Deployed PHLOTE Tether Relative to the Size of Phobos

The Mars-Phobos L1 point is unusually close to the surface of Phobos due to Phobos' weak gravity and its close proximity to Mars. A Lagrange point is interchangeably called a libration point because this is where the gravitational and dynamic forces acting on a third body (i.e. the PHLOTE spacecraft) would balance. Note that the Mars-Phobos L1 location is not a fixed-point relative to the moon's surface. Since the orbit of Phobos is slightly elliptical (9234.42 km x 9517.58 km), this causes the L1 location to have a periodic motion of a few hundred meters relative to the moon's surface during each orbit. For hover mode station, keeping the motion of the L1 location must be accounted for in the overall mission design.

Phobos is a small irregularly shaped moon (~26 × 22.8 × 18.1 km) that orbits Mars every 7 hours and 39 minutes. The moon's orbit is synchronous to its rotation so that its long axis is always directed toward Mars. The large impact feature called Stickney Crater in Figure 1 would always face Mars and would be located directly below the L1 location where the primary PHLOTE mission would operate.

The average gravity for Phobos is commonly listed as 0.0057 m/s^2. However, due to its irregular shape and the pull of Mars, the gravitational acceleration at a surface location below the L1 point is closer to 0.0031 m/s^2. This very low gravitational force means that the tension on the tether is very low and thus tensile strength is not a concern. However, the very low tether tension is an engineering challenge for deployment and for adjusting the tether length during operation.

The attached tether will help stabilize the attitude of the spacecraft and the sensor package in the radial direction toward Phobos. Rotation around the tether's axis will still need to be controlled. The PHLOTE tether would be pointed toward Mars so that a fixed camera at the top of the primary spacecraft would have Mars centered in the Field Of View (FOV) and a fixed camera under the spacecraft would always view the Mars facing side of Phobos.

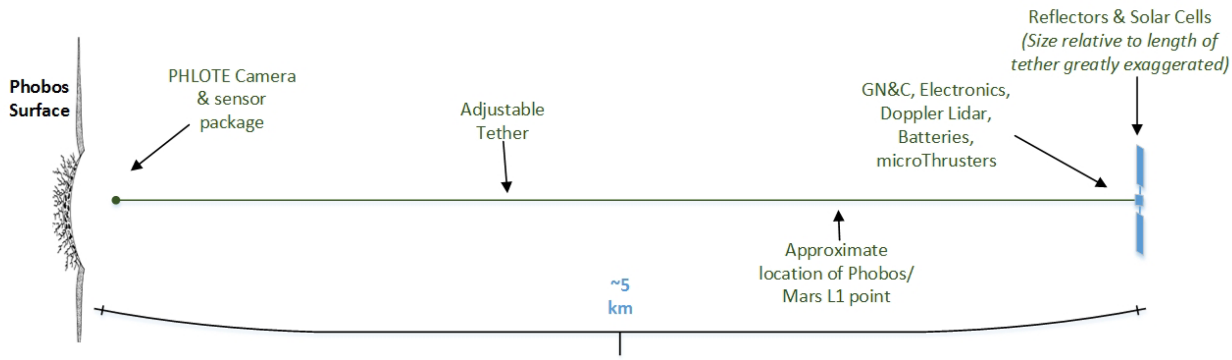


Figure 2: Simplified Diagram of the PHLOTE Spacecraft Operating at the Mars-Phobos L1 Location

L1 and L2 points are unstable orbital locations: once a spacecraft drifts away from these locations it will not return. To remain near these locations, a spacecraft must actively maintain its position. The more accurately it can maintain its position, the less fuel it will need to stay there. This requires very accurate relative position knowledge to at least one of the bodies. Without external systems such as GPS, a typical spacecraft would not have the position knowledge needed to maintain its position at a Lagrange point.

For the PHLOTE mission, the needed relative position knowledge can be obtained with a recently developed sensor called the Navigation Doppler Lidar (NDL). The NDL was developed for precision landing under the Autonomous Landing and Hazard Avoidance (ALHAT) project[1, 2] and can provide very accurate ranges (~23 cm) and rates (~0.2 cm/s) relative to the surface of Phobos along multiple beam paths. These measurements can be used to compensate for spacecraft drift away from the libration point before the error becomes too large and its low thrust high efficiency station-keeping thrusters do not have the control authority needed for it to return.

PHLOTE System Needs Goals and Objectives

Science Need: Many questions remain unanswered about the origin and composition of Phobos. There is much debate as to whether Phobos is a captured asteroid or whether it formed from debris ejected by a large impact during the planets formation[3, 4].

Phobos' spectrum suggest that its composition is similar to carbonaceous chondrite asteroids. If it is a captured asteroid (or composed of the pieces of one), as its mineral makeup suggests, then how did Phobos get into its close circular orbit? If Phobos is a carbonaceous chondrite asteroid, does it retain volatiles such as water, which are typically found in this class of asteroid? Since Phobos is very close to the Roche limit, where tidal forces would pull it apart, how is this affecting its internal structure?

Exploration Need: Phobos offers a potential waypoint and source of resources for future human missions to Mars[5]. To help mission planners, NASA needs an economical and effective method for measuring the radiation environments. Having accurate models of the radiation environment is critical for future human mission development. The microgravity and limited knowledge of the surface composition at Phobos makes both landing and long-term orbital operations challenging. A key question is whether there are resources that can be used to support human missions. If plans include landing a crewed vehicle on the moon's surface, mission planners will want to understand how compact the soil is and what its adhesion properties are like. PHLOTE will expand our knowledge about Phobos with a focus on measuring the specific environmental conditions that will impact future human mission designs.

Technology Development Needs: There are many potential benefits to long-term operations at planet-moon Lagrange Points for science platforms and also as part of a human exploration infrastructure. In addition, systems utilizing tethers have a broad range of potential mission applications. NASA needs operational experience with tethers to enable many potential future missions. This includes understanding the dynamic properties and developing autonomous control of tethered systems where high latencies preclude ground control. A successful PHLOTE mission will raise the Technology Readiness Level (TRL) of these tether system technologies as well as the TRL of key enabling components such as the Navigation Doppler Lidar (NDL). An exciting demonstration at the end of the mission would be to land on and perhaps anchor to the surface and then further extend the tether. This would allow PHLOTE to become the first operational space elevator. Intentional libration could repeatedly lift the mass off the surface and land it elsewhere, allowing multi-site in-situ sampling of Stickney crater. Another interesting demonstration would be to attach to a rock and lift it off the surface by extending the tether. This would be the first demonstration of an operational "space crane."

PHLOTE Mission Goals:

Technical Goals: Develop a credible spacecraft design that:

a) **Performs hover mode operations at the Mars-Phobos L1 Point**
   i) Provides long duration science measurements (goal is one Martian year)
b) **Deploys a tethered sensor platform to study the environments and surface of Phobos**
   i) Provides selectable altitudes over surface from direct contact to L1 (~4250 m)
c) **Utilizes the latest technology being developed by industry to reduce mass and cost**

i) The Goal is to keep spacecraft dry mass around 100 kg and mission cost under 50 million dollars

Scientific Goals: Carry an array of scientific instruments that:

a) **Gather data on the physical, structural, and chemical composition of Phobos**
   i) Enhances our understanding of the origin of Phobos and the Martian system.
b) **Find potential landing sites, samples, and resources for future missions**
   i) Reduces risks and allows more effective planning for human missions to Mars

PHLOTE System Description

System Context

The PHLOTE System will have three segments: The Ground Segment, the Launch Segment, and the Space Segment. The primary focus of the NIAC Phase I study will be the PHLOTE Space Segment since it has the highest technical risk.

The Ground Segment will include a PHLOTE Operations Center and utilize NASA's Deep Space Communications Network to send and receive commands and data. Part of this segment will include the communications relay satellite(s) expected to be operating in Mars orbit during PHLOTE operations.

The Launch Segment will include the Launch Operations Center and the Launch Vehicle. The Launch Vehicle will provide mechanical and electrical interfaces to the PHLOTE spacecraft through separation.

The PHLOTE space segment will be composed of the Spacecraft (near the Lagrange Point), a Tethered Sensor Package near the surface of Phobos, and potentially deployable surface nodes if mass allows.

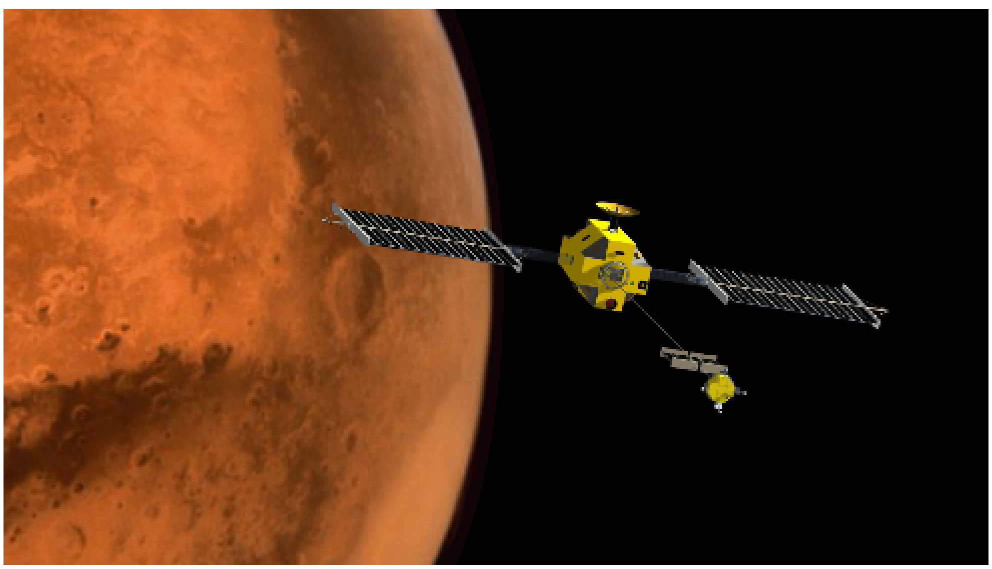


Figure 3: PHLOTE Spacecraft beginning deployment of the sensor package (MLI Blankets not shown)

Basic Assumptions for the PHLOTE Mission

Trade studies were performed to help constrain the mission architecture. Additional information can be found in the Trade Study section. Some of the key decisions are:

1. PHLOTE Launch Date – Summer 2026 (Mars arrival mid-2027) mission complete 2029
    a. Based on need to inform mission designers early in the development of a human precursor mission to Phobos in the mid 2030s[6].

   b. A 2026 Launch Date implies that the PHLOTE Mission Preliminary Design Review (PDR) will be held in the 2023 timeframe. This is important since there must be a high confidence that all components selected in this PHLOTE mission study will be at a TRL of 6 (and preferably higher) by 2023.

2. PHLOTE Launch Vehicle – Virgin Orbit LauncherOne[7]
   a. Based on advertised performance to deliver up to 300 kg to a 500 km Sun Synchronous Orbit (SSO). A high inclination is preferred to minimize radiation dose on the spacecraft components. Also, a near-sun synchronous orbit can allow continuous power for the solar-electric propulsion system as it spirals up to a Mars transfer orbit. Virgin Orbits flexible air launch capability is ideal for a PHLOTE launch.
   b. Payload fairing provides 1.26 Meter diameter. Similar to USAF rideshare volume constraints and provides credible dimensional limitations for the mission concept.
   c. $10M advertised cost. This provides a credible launch cost for the PHLOTE mission study.
   d. Note, the Rocket Labs Electron launcher, was originally baselined but current mass estimates indicate there is inadequate mass margin at this early stage of design.
3. PHLOTE Communications to Earth - A communications relay satellite will be available at Mars
   a. Based on Government RFI for orbiter concepts and a $10M line item in the FY18 budget for advanced studies it is expected an observation/relay orbiter to be in place at the time of the PHLOTE mission (2027 through 2029). PHLOTE will use this satellite as a communications relay to Earth to support communications during operation.
   b. The Mars Atmosphere and Volatile EvolutioN (MAVEN) mission is expected to be operational until 2030 and can provide a communication relay option.
   c. Direct communications to Earth will also be required during Mars transit and for Phobos gravity field mapping but this communication will be at a much lower data rate.
4. PHLOTE Main Propulsion - The PHLOTE Spacecraft will utilize Iodine Hall Thrusters (IHTs)
   a. IHTs are currently at a TRL between 5 and 6 and are expected to be commercially available by the PHLOTE PDR. There are technical challenges with IHTs that are being actively worked. This includes cathode erosion, and duty cycle limits on the power electronics. NASA's Space Technology Mission Directorate (STMD) and the Department of Defense are actively funding IHT technology development. Other electric propulsion fuels such as Xenon can be used but would require additional mass and volume for fuel storage. See: https://gameon.nasa.gov/gcd/files/2017/07/AISP_Hall-Thruster.pdf
   b. A 600 Watt thruster offered by Busek Co. Inc. (http://busek.com/index_htm_files/70000701%20BHT-600%20Data%20Sheet%20Rev-.pdf) provides credible performance and resource requirements for the study.
   c. If additional launch mass is available, a hybrid propulsion system will be considered to provide a faster transit through the Van Allen belts to a higher Earth orbit to minimize the radiation dose on PHLOTE components.

5. PHLOTE Station Keeping Propulsion - The PHLOTE spacecraft will utilize Electrospray Thrusters for Station Keeping
    a. Electrospray Thrusters offer low mass, low thrust, and high efficiency making them ideal for long term hover mode station keeping at the Mars-Phobos L1 point.
    b. This thruster technology is currently at a TRL of 5 (https://spinoff.nasa.gov/Spinoff2016/ip_9.html)
6. Operational Life – One Martian Year (~687 Earth days)
    a. To provide scientists and future mission developers a complete, long-term record of radiation and dust environments at Phobos, it would be beneficial to take measurements for one full Martian orbit.

PHLOTE Mission Phases

Integration with the Launch Vehicle: The PHLOTE spacecraft will be packaged in a stowed configuration that can be accommodated within the LauncherOne payload module[7].

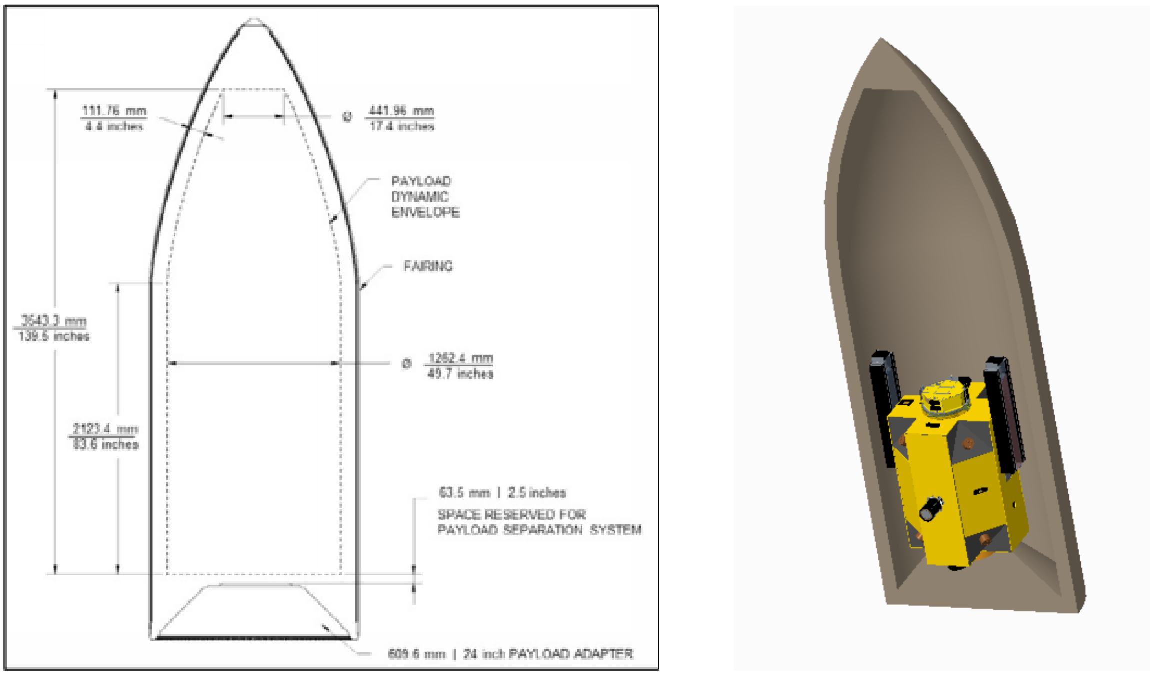


Figure 4: The stowed PHLOTE Spacecraft inside the LauncherOne Payload Fairing

The PHLOTE stowed configuration must fit within the dimensional constraints of the LauncherOne Payload Fairing. This will encourage PHLOTE designers to utilize the wide variety of components being developed commercially for CubeSat and SmallSat missions, which will help drive down the overall mission costs.

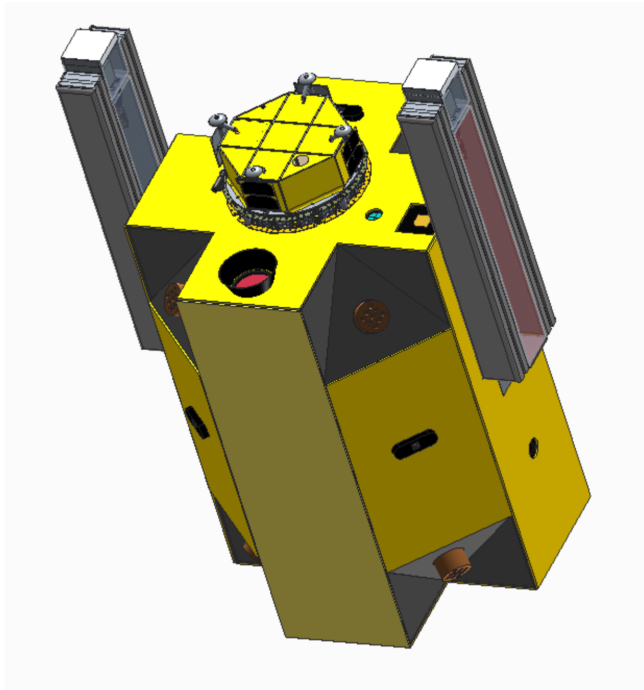


Figure 5: The PHLOTE Spacecraft in a Stowed Configuration (Without MLI Blankets to show components)

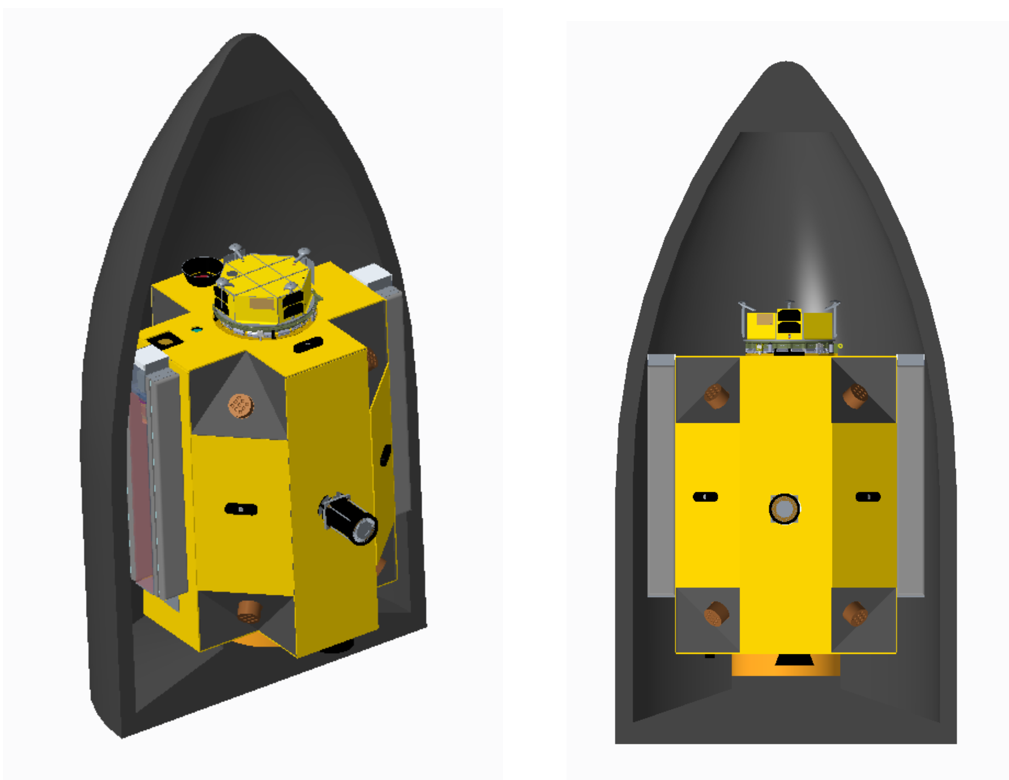


Figure 6: PHLOTE inside the smaller Electron Payload Fairing (The Launch Vehicle Adapter is included)

Launch and Cruise: The PHLOTE spacecraft will be placed into the highest possible (> 500 km) SSO with a high inclination. The high inclination will minimize the radiation dose as the electric propulsion system raises the orbit, but at a small penalty of increased solar proton dose during the climb. After separation from the payload fairing the solar array will be deployed and the electric propulsion system will begin raising the orbit to a Mars transfer orbit.

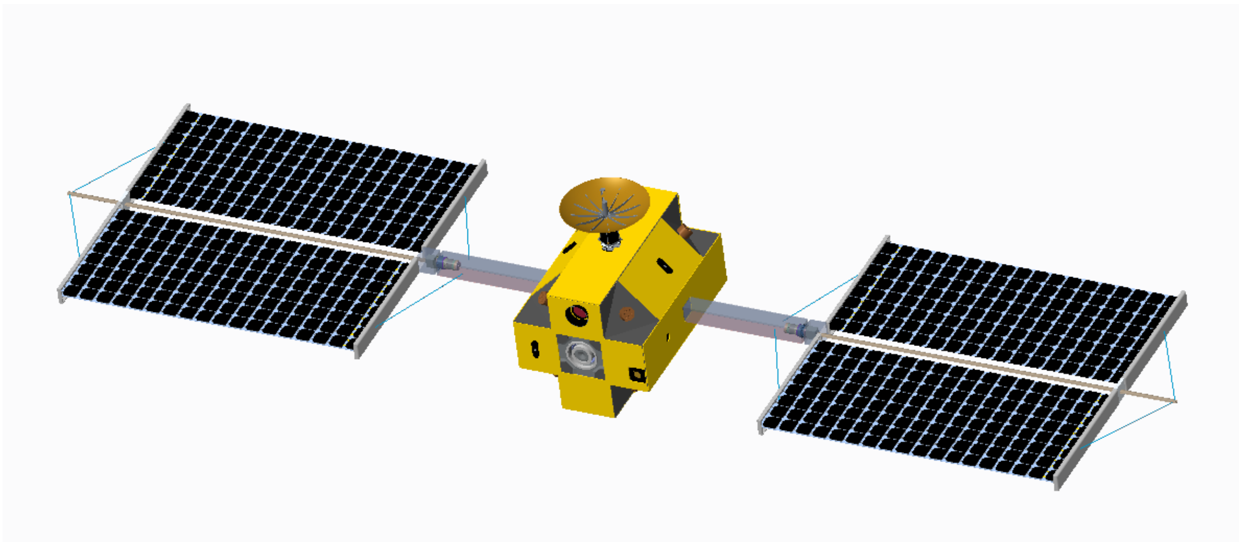


Figure 7: PHLOTE Flight Configuration (Without MLI blankets)

Using electric propulsion from Low Earth Orbit provides mass efficiencies at the expense of longer duration and higher total delta-V to get to Mars. The chief concern is the extended time traveling through Earth's Van Allen Radiation belts, which will cause an increase in the likelihood of component failure and some degradation to the electronics and solar arrays. This longer exposure will require additional radiation hardening, which will drive up mass, component costs, and resource margins.

A hybrid propulsion system that uses a small bipropellant engine could be an option to reduce travel time through the areas of highest radiation. New designs having specific impulses approaching 330 seconds would allow a PHLOTE spacecraft to get to a Mars transfer orbit using a launcher such as the LauncherOne if spacecraft mass was carefully managed. Examples of these engines can be found at: *http://www.northropgrumman.com/Capabilities/PropulsionProductsandServices/Pages/BipropellantEnginesAndThrusters.aspx*.

Notes on the Radiation Environment from Low Earth Orbit (LEO) to Geostationary Transfer Orbit (GTO):
Using the Space Environments, Effects, and Education System (SPENVIS) analysis tool to assess radiation doses when raising the orbit from LEO using electric propulsion, it was found that doses in near-polar planes are almost five times lower than in near-equatorial plane and 1.5 times lower than at a 45-deg inclination. There is little change within +/- 10-15 degree of the 90-degree inclination. This means that besides LauncherOne, PHLOTE can take any ride to a Sun Synchronous Orbit (SSO) or an Iridium type orbit.

The fastest near-polar ascent with electric propulsion could result in an estimated 12 krad dose with 3 mm Al shielding, 6 krad with 4 mm, and 3 krad with 5 mm. So, a few kilograms of shielding around critical electric components will be needed, especially if using COTS components developed for the SmallSat market.

Degradation to the solar arrays due to the high levels of radiation in the Van Allen Belts must also be considered. It is encouraging that the SMART-1 mission lost only 8% of solar array performance during the 2.5 months spent crossing the proton belt in a near-equatorial plane. Interestingly, the degradation stopped above 5,900 km (a much lower altitude than expected), and the electron belt didn't affect the solar arrays:

> http://erps.spacegrant.org/uploads/images/images/iepc_articledownload_1988-2007/2005index/119.pdf

SMART-1 had GaInP2/GaAs/Ge 3J cells, with a 200-micron thick cover glass and a Beginning of Life efficiency of 25%. The radiation dose in a PHLOTE polar ascent could be ~8 times less, and the cover glass on the PHLOTE arrays could be thinner, perhaps 80 micron. The multi-layer insulation (MLI) blankets baselined for PHLOTE must be designed to be suitable for slow ascent from LEO.

Continuous thrusting from the electric propulsion system may provide a beneficial side effect: SMART-1 experienced no space charging - the spacecraft potential was extremely neutral, stable and fully independent of the surrounding natural plasma.

Notes on the sizing of the Iodine Hall Thrusters: If starting from LEO, 600W may be low for Electric Propulsion only. We are considering using either a cluster of up to four BHT-600 thrusters or a tandem of BHT-1000 thrusters (9.7 kg with a power processing unit, cathode, and cables).
Both options are very similar and can give up to 200 mN in LEO. These thrusters work in a wide range of voltages and power (up to 30-50% over nominal). PHLOTE can traverse the radiation belts at lower voltages and higher thrust with a small loss of efficiency, and then switch to higher voltages and lower thrust. Table 1 gives typical performance numbers. Traversing the radiation belts can be done at lower voltages, yielding higher thrust but lower I_{sp}. The rest of the transfer can be completed with higher voltages and higher I_{sp}, but somewhat lower thrust.

Voltage	***I_{sp}***	***Thrust/Power***
200 V	*1370 sec*	*76 mN/kW*
300 V	*1730 sec*	*67 mN/kW*
400 V	*2000 sec*	*60 mN/kW*
500 V	*2350 sec*	*50 mN/kW*

Table 1: Ion thrust versus voltage

The change in velocity needed to get from low Earth orbit to Phobos orbit with impulsive thrust is around 6 km/s, however, when using a low thrust trajectory, it is closer to 15 km/s. The exhaust velocity of the 600 Watt Iodine Hall thruster is expected to be greater than 15,000 km/sec. The expected "dry" spacecraft mass of 120 Kg with 20 Kg of margin is up to 140 Kg. This leaves about 170 kg available for fuel/hybrid propulsion if launched on the Virgin Orbital LauncherOne vehicle to a 400 km Sun Synchronous Orbit. Any extra fuel can be used for long-term station keeping during the operational phase of the PHLOTE mission or potentially to move the spacecraft to the Mars-Phobos L2 location for an extended mission with additional science return.

During the cruise phase, low rate communications will be performed with an X-Band communications link to NASA's Deep Space Network. There may be potential science opportunities during the cruise phase.

Arrival and spacecraft checkout at Phobos: Once the PHLOTE spacecraft arrives at Phobos it will go into an Initial Parking Orbit as shown below:

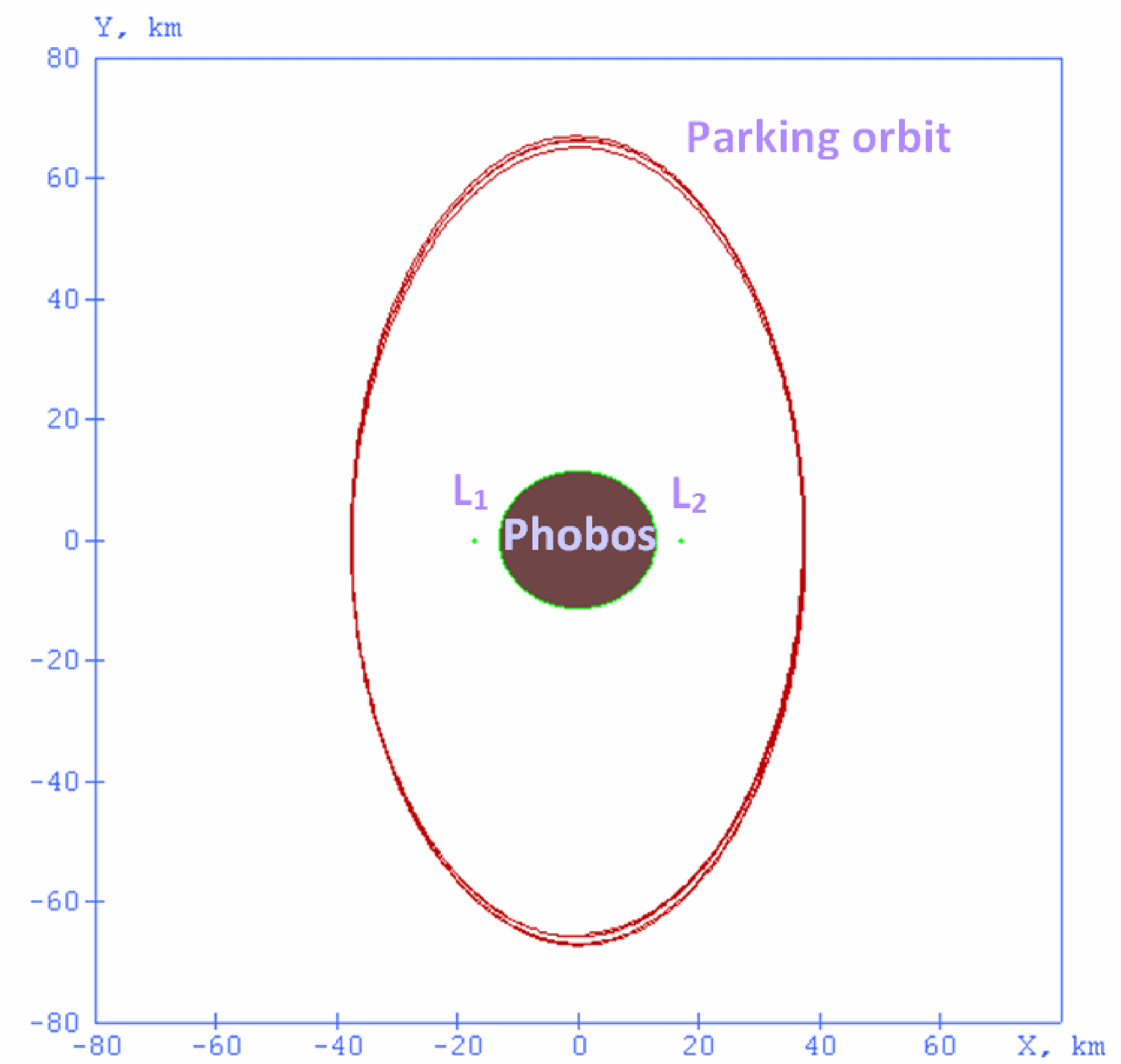


Figure 8: Initial Parking Orbit at Phobos

- The PHLOTE Spacecraft can stay in this parking orbit for some time during which it will perform gravitational and visual mapping of Phobos. During this period, controllers will refine estimates of the Mars-Phobos L1 and L2 locations. A much higher accuracy will be needed for the PHLOTE mission than is currently known. Gravity mapping requirements will likely drive the transmitter stability requirements and mission operations during this period. During Phase 2 the team will investigate the accuracy of the gravity maps needed for the initial deployment and also develop the operational methods needed to achieve it.

Communications will be established with Mars Relay obiters so that high rate engineering data, imagery, and science measurements can be sent back to the Earth. Periodic ephemeris updates for the relay satellites will be sent to PHLOTE to optimize pointing of the high gain antenna. Since the PHLOTE spacecraft will be only a few kilometers above the surface of Phobos, valuable science can be obtained during this period. This includes gravity field mapping, imagery, and creating an accurate 3D elevation map of the surface below the L1 point using the NDL sensor. This Phase is expected to take about 5 months.

After mapping is complete, the PHLOTE spacecraft will move to the vicinity of L1 using its Electric Propulsion System. Because Phobos is in an elliptic orbit, there is no exact equilibrium near L1, but there is a periodic motion around mean L1 point. In this motion, the spacecraft will move mostly in the radial direction around a center point located approximately at R = 17247 m (from the centroid of Phobos), based on the available data (Note that Phobos is not a spherical body). The spacecraft will be ~244 m below the center point when Phobos is at the periapsis and ~244 m above the center point when Phobos is at the apoapsis.

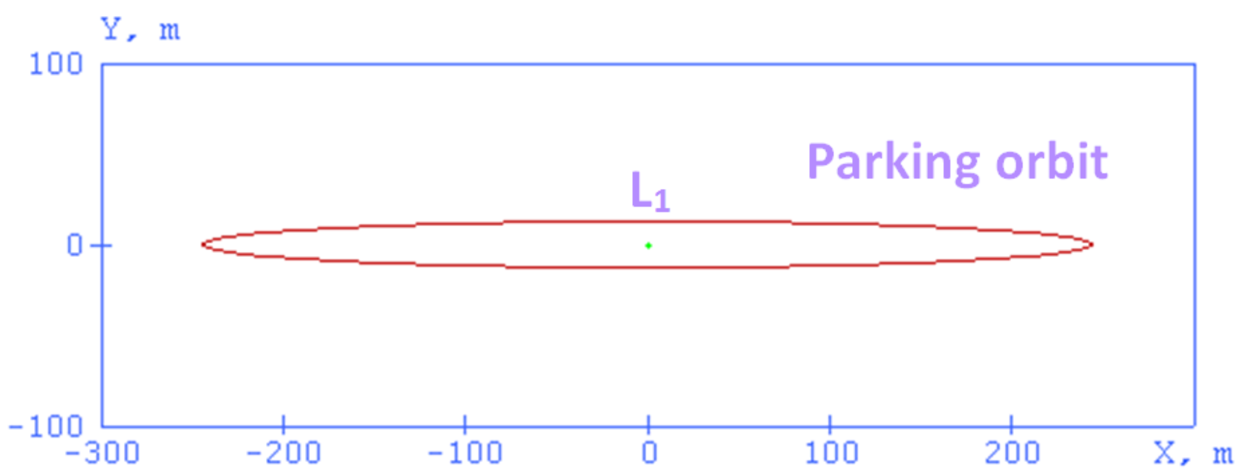


Figure 9: Parking Orbit near the Mars-Phobos L1 Location

This is related to a semi-stable Lissajous orbit. See: https://en.wikipedia.org/wiki/Lissajous_orbit

A non-spherical Phobos field model was used in simulations:

$$\mathbf{g} = \nabla U = \left[-\frac{GM_c}{R^3} - \frac{3G}{2R^5}\,\mathrm{Tr}\Lambda + \frac{15G}{2R^7}\left(\Lambda\mathbf{R},\mathbf{R}\right) \right]\mathbf{R} - \frac{3G}{R^5}\Lambda\mathbf{R}$$

Where G is the universal gravitational constant, Mc is the mass and Λ is the tensor of inertia of Phobos.

Instability of Motion near L1: Trajectories of motion near the Phobos L1 point are very unstable. The area below the Hill Surface represent locations where an object not moving relative to Phobos is gravitationally bound.

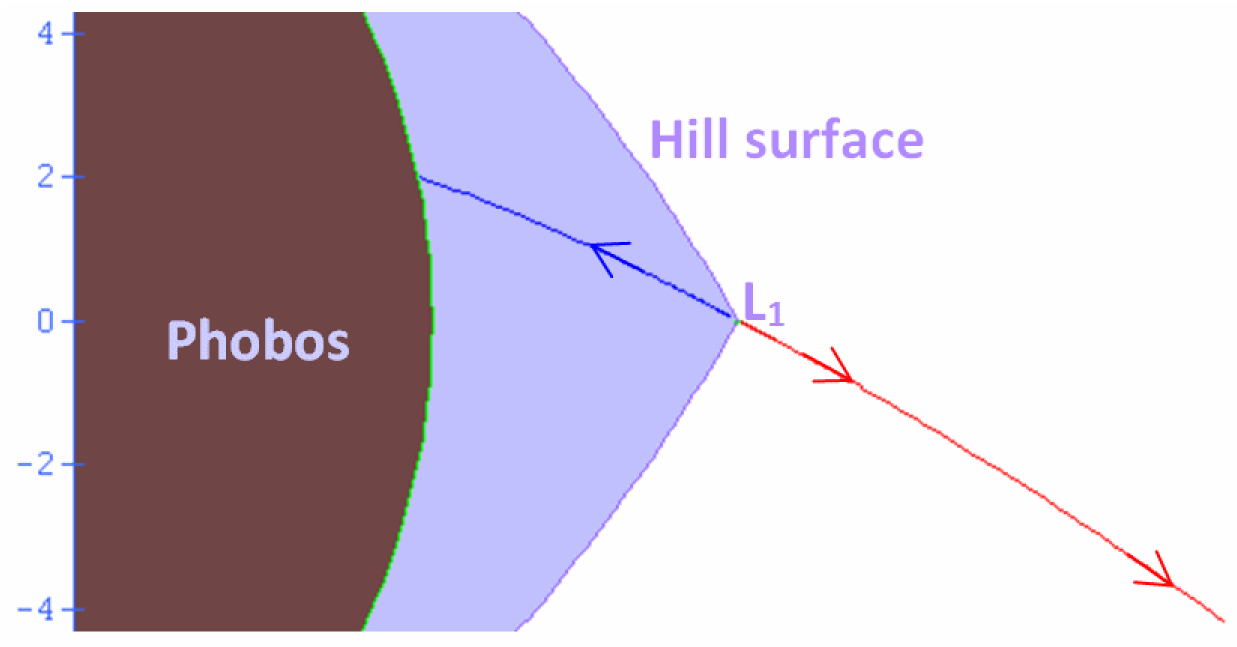


Figure 10: Instability of motion near the Mars-Phobos L1 Location

Without active control, a spacecraft placed initially only 10 m away from L1 will drift more than 20 km away in less than 4 hours (red arrows in Fig. below). This is because of the radial force gradient pulling the spacecraft away from L1. However, if PHLOTE shifts just 1 m from its equilibrium toward Phobos, the sensor package will end up on the surface in less than three hours (as shown by the purple arrows in the Figure below).

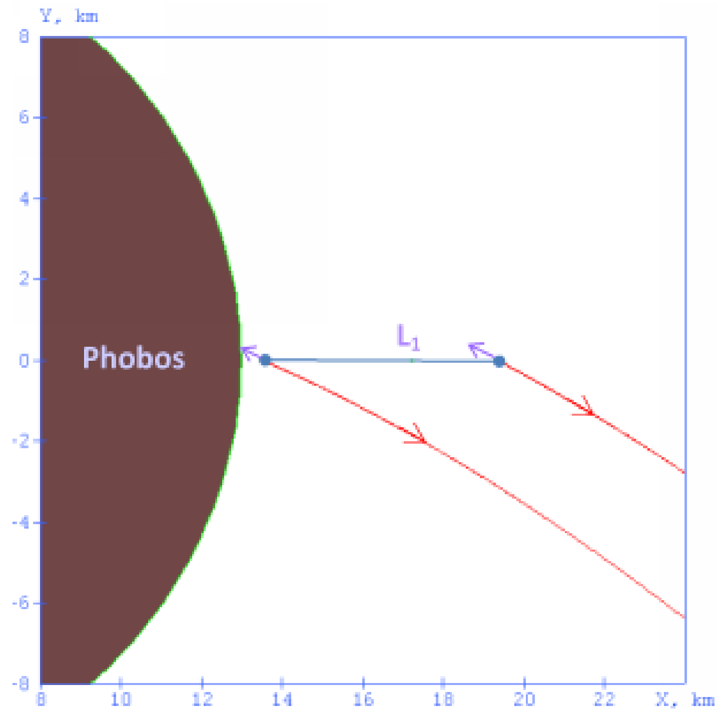


Figure 11: PHLOTE Motion without Active Control

The station-keeping strategy near L1 is to stabilize the motion near the periodic trajectory of the L1 point and not to fight the periodic shifts caused by the eccentricity of the orbit of Phobos. Since Phobos orbits Mars every 7 hours and 39 minutes this periodic motion near the L1 location would have the same duration.

A 120-kg spacecraft will experience a 60-µN/m radial force gradient near L1. Three 1-mN thrusters at 45° to the radial direction could be adequate up to a maximum displacement of 35 m in the radial direction from the libration point. In the lateral direction, the maximum displacement can be somewhat larger before control authority with the Electrospray thrusters is exceeded. The tethered sensor platform will not be deployed until the operations team is confident the spacecraft can maintain its position at the L1 location. This will allow the spacecraft to use its much more powerful Iodine Hall Thruster(s) if needed to return to a safe orbit so that its control parameters can be updated and it can try again.

Due to high communication latencies to Mars, the PHLOTE Guidance, Navigation, and Control (GN&C) software must be autonomous and the control algorithms must make the appropriate corrections as early as possible to minimize propellant use or return to a safe orbit if needed. During this time, the Navigation Doppler Lidar sensor will be providing highly accurate range and rate data relative to the surface of Phobos so that the spacecraft can maintain the desired motion near the L1 Point. During the mission, the actual spacecraft performance will be compared to high fidelity GN&C simulations so that the control algorithms and parameter values can be refined to maximize efficiencies in L1 station keeping.

Tether Deployment Phase: At this point, the PHLOTE spacecraft will be operating at the L1 location and the sensor platform, designated the Phobos Observation Platform (POP), will be separated from the main spacecraft by a spring mechanism. This may be augmented by cold gas thrusters on the POP to reduce the impulse imparted on the spacecraft if using a spring only. It is recommended that the initial separation velocity be ~2 m/s to assist tether system deployment. This action will keep the center of mass near the L1 location and allow enough separation so that the gravity gradient will begin generating some tension on the tether to continue the separation.

The initial tether deployment must be far enough to realize a minimum tether tension that will allow the tether to remain taut. It is estimated that the tension on the tether will be very low. For a sensor package with a mass of 15 Kg and a tether length of 3 km, the deployed tension when at equilibrium will be 2.5 grams of force or about the weight of a US dime. This means that the tether itself must be very limp to prevent it from coiling up when released and acting as a spring. This also requires that the breaking mechanism must be very sensitive so that the two parts can slow down and settle out without rebounding. Measurements from accelerometers on the spacecraft and sensor platform will be used to dampen out the oscillations using motors/mechanisms on the tether reel or in combination with thruster impulses from the spacecraft. Since the tether will be periodically reeled in and out for station keeping as well as for positioning the sensor platform, tensioners will be required so that it can be compactly rewound on a spool or reel.

<u>Options being considered for the initial tether deployment:</u> The currently planned deployment method would be an impulsive operation using springs to separate the sensor platform from the main spacecraft as used on flight experiments such as the Space Expendable-tether Deployment System (SEDS), the Plasma Motor Generator (PMG), The Tether Physics and Survivability Experiment (TiPS), or the Tether Electrodynamics Propulsion CubeSat Experiment (TEPCE). It should be noted that the gradually growing gravity-gradient force as the sensor package moves toward Phobos will provide ~80 joules of energy towards overcoming deployment friction by the end of a 5-km deployment. The initial ejection must provide enough energy to get the gravity gradient large enough to take over and continue deployment.

An impulsive ejection velocity of 2-3 m/s would be needed to ensure full deployment. Given the expected mass ratio this would mean a velocity change on the main spacecraft of between 0.3 and 0.45 m/s. The solar array and antenna structures must be designed to handle this load. The spring impulse places a dynamic load on the deployed solar arrays, even if a long-stroke low-peak-force stacer spring is used to drive deployment, as on the upcoming NRL TEPCE experiment. The deploying end may require healthy attitude stabilization to keep it pointed close enough to the right direction after ejection, if there is any CG offset from the center of action of the springs.

There are several ways to arrange PHLOTE tether deployment from the vicinity of L1. One of the options is to place a derivative of the mini-SEDS deployer with tether winding stowing most of the tether length on the POP and a reel with a relatively short tether segment on the main spacecraft which is refered to as the Mars Opservation Monitor (MOM), as illustrated in Fig. 12.

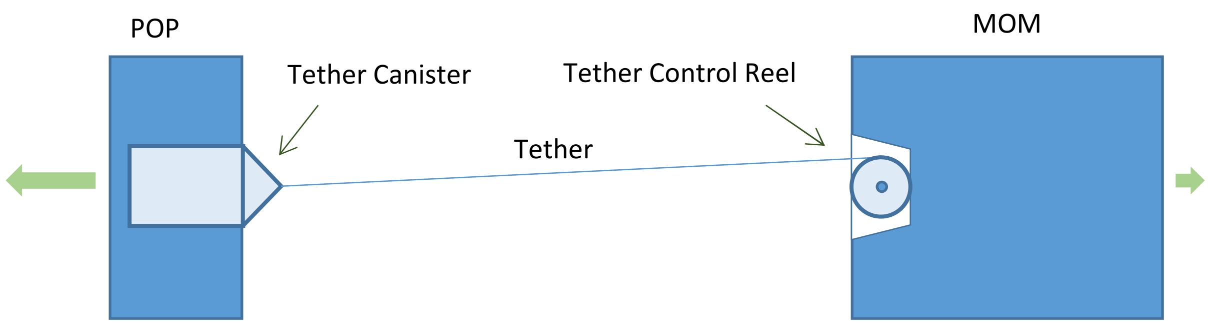


Figure 12: Deployment Using a Mini-SEDS Deployer and a Reel

Placing the mini-SEDS tether canister on the POP, along with the separation springs, will add mass to the POP, which is beneficial for the stability of the entire system. In the deployment scheme shown above, the tether control reel remains on the MOM and is not activated until the tether pays off the winding inside the mini-SEDS canister. The main advantage of placing the reel on the other end of the tether is the availability of stabilizing tether reel control regardless of how much tether has paid off the mini-SEDS winding. If the tether deployment stops mid-way because of an unexpectedly high friction inside the mini-SEDS canister, the tether control reel can be used to stabilize the whole system and then pull the tether off the winding to restart deployment. The downside of using a mini-SEDS deployer is in a non-deterministic tether friction in the canister that would create system stabilization difficulties in the vicinity of L1. Another disadvantage of utilizing a single-use mini-SEDS deployer is that the tether cannot be retrieved for repositioning.

Another option is to use only a reel on the main spacecraft (MOM), as shown in Fig. 13.

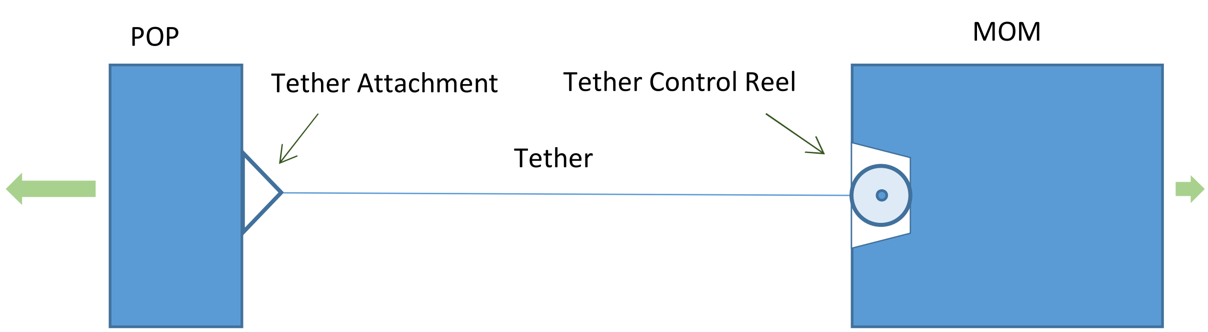


Figure 13: Deployment Using a Reel on the Main Spacecraft

This deployment scheme offers an advantage of applying a desired torque to the reel and creating well-defined dynamic conditions favorable for the overall system stabilization near L1. Another advantage is the ability to retrieve the tether for repositioning. It can be an important part of recovery from a loss of stabilizing control and can also allow relocation of the entire system to L2 for further exploration of Phobos.

A third option is to mount both a SEDS-like deployer and a reel-type deployer on MOM, with the deploy-only SEDS tether deploying first, followed by a shorter length of reeled tether deploying through the now-empty SEDS deployer. This reduces the required size and mass of the reel-type deployer (like the first option), but allows both tether lengths to be deployed from MOM, which can more easily provide positive attitude control during deployment.

PHLOTE deployment dynamics has been simulated in the first approximation, and a set of suitable deployment trajectories has been identified. A typical deployment trajectory is shown in Fig. 16.

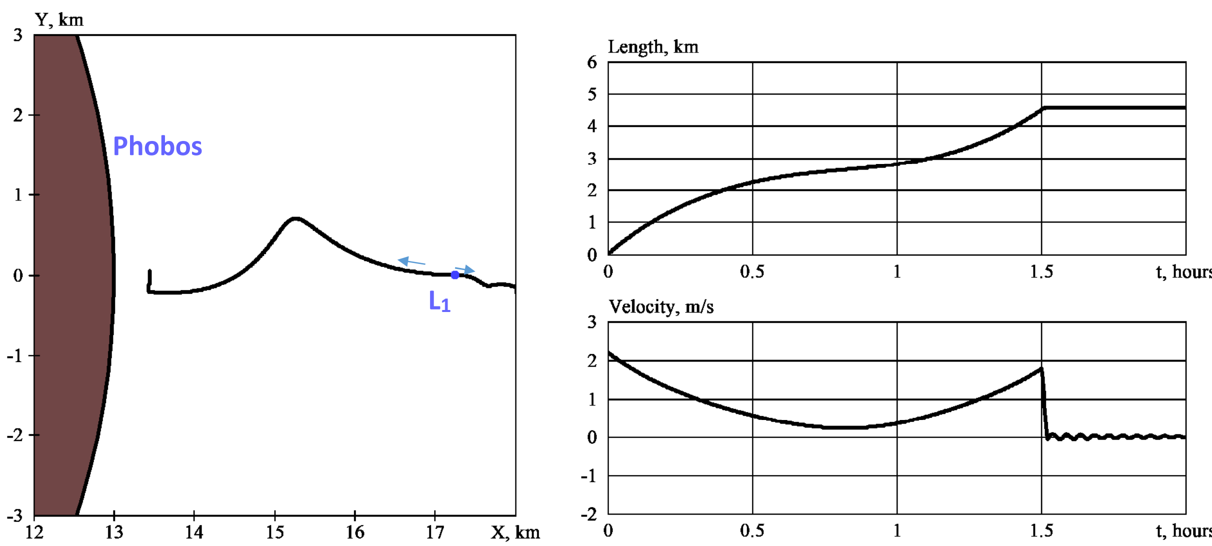


Figure 14: Deployment into a Libration Mode

In this simulation, the POP mass was 15 kg, and the total mass of the system was 120 kg. The initial separation velocity was 2.2 m/s, and the tether tension was maintained by the reel control mechanism at a 20-mN level until the tether length reached 4.6 km, and a braking tension of 0.4 N was applied. When the deployment stopped, the system was found in a small-amplitude libration mode. The entire deployment process took 1.5 hours. In the middle of deployment, we see a "knee" in the deployment trajectory. This is where the deployment velocity dropped, and the gravity gradient took over, capturing PHLOTE into a synchronous rotation with Phobos. After deployment, the tether reel control will damp residual longitudinal and lateral oscillations and engage in a nominal stabilization control to keep PHLOTE hovering near L1. This is achieved by periodically reeling a relatively short segment of the tether in and out, based on the LiDAR measurements.

Further analysis of the deployment trajectories has shown that there is a possibility of deploying PHLOTE into a parked state, with POP resting on the surface of Phobos. An obvious advantage of this operation is to put the system into a stable position, where no stabilizing control is required, and to allow as much time as necessary for an all-system check before lifting the system off the surface of Phobos and bringing it in a hovering dynamically-stabilized state. Fig. 17 illustrates a deployment trajectory of this type.

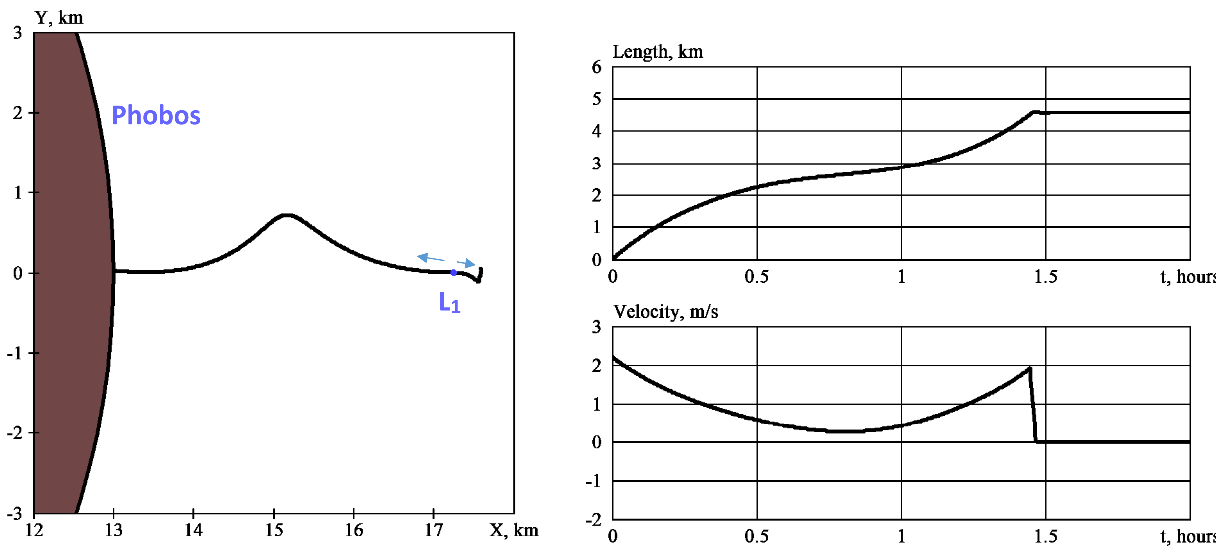


Figure 15: Deployment into a Parking Mode

In this simulation, like in the previous case, the POP mass was 15 kg, and the total mass of the system was 120 kg. The initial separation velocity was 2.2 m/s, and the tether tension was maintained by the reel control mechanism at a 20-mN level until the POP started approaching the surface of Phobos and a braking tension of 0.4 N was applied to bring POP to a full stop, parked on the surface of Phobos. When the deployment stopped, the system was found in a small-amplitude libration mode, with its lower end anchored at the surface by the gravitational pull of Phobos. The entire deployment process took slightly less than 1.5 hours. Again, in the middle of deployment, we see a "knee" in the deployment trajectory, where the deployment velocity dropped, and the gravity gradient captured PHLOTE into a synchronous rotation with Phobos. After deployment, the system can remain the parked state until all checks are complete. To lift off the Phobos surface, the tether reel will pay off some extra length of the tether, and the increased gravity gradient acting on the main spacecraft will gently pull PHLOTE away from the surface. Then, the tether reel control will start a nominal stabilization control to keep PHLOTE hovering near L1 by periodically reeling a relatively short segment of the tether in and out, based on the LiDAR measurements.

A real concern for landing a craft in microgravity is not the landing itself, but the resultant bounces off the surface. The POP will be designed to tolerate a a worst-case-scenario tether failure: a freefall from 788m above the surface of Phobos. Even in freefall, this descent would take approximately 713 seconds, or close to 12 minutes. From that height, the POP would impact the surface at 2.99 m/s (This is the same impact speed as after a 0.46m or 18 inch drop on earth). The Pop will have three apparatuses to minimize bouncing: damping mechanism on the legs, cold gas thrusters pointing upwards, and potentially Microspine grippers on the landing pads: See (https://www-robotics.jpl.nasa.gov/tasks/taskVideo.cfm?TaskID=206&tdaID=700015&Video=147)

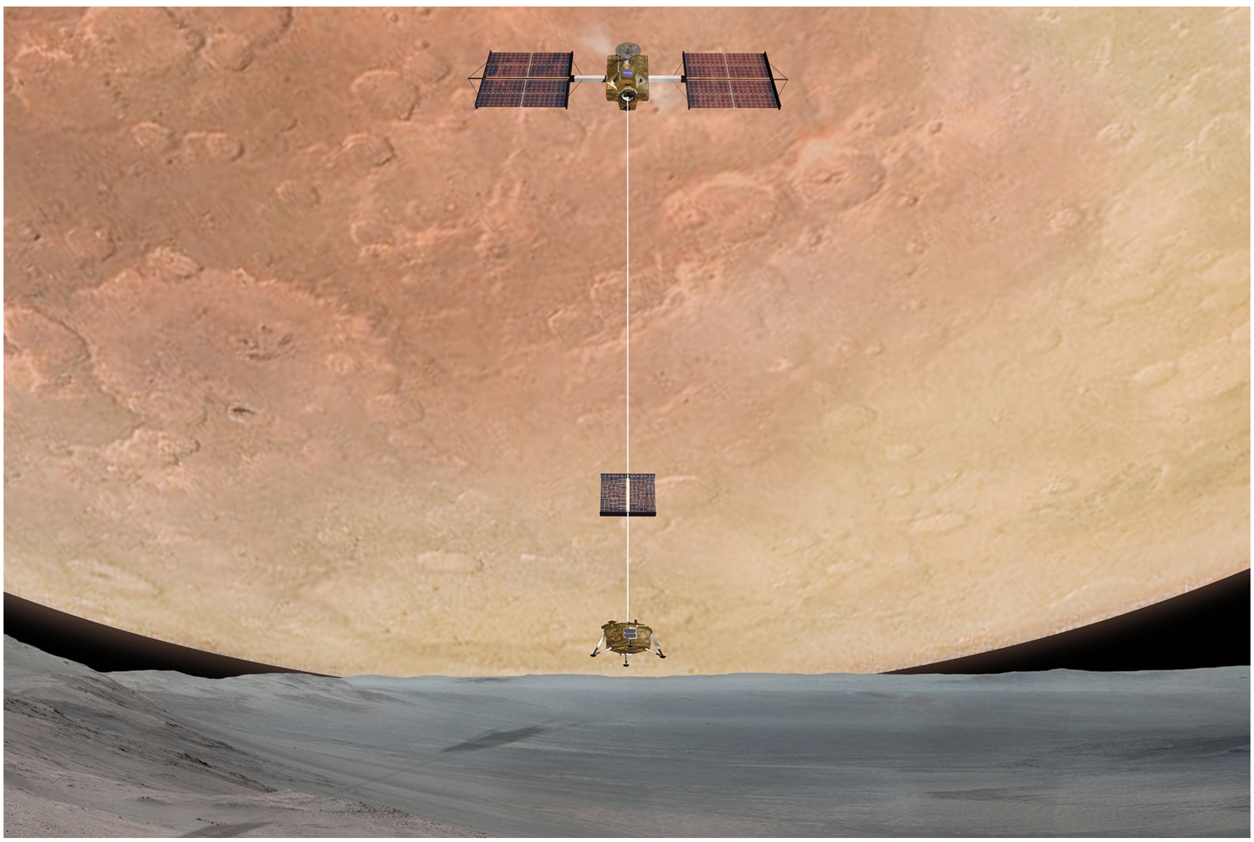


Figure 16: PHLOTE POP near the surface of Phobos (Tether length greatly reduced for illustration)

An alternate deployment method would be to rotate the spacecraft and release tethered sensor platform allowing the rotation to pull the two sections apart. This method has control challenges but has some advantages for the tether deployment mechanisms and reduces the impulsive load on the spacecraft from the separation spring(s). It may also be moderately lighter due to elimination of springs and appendage reinforcements.

This deployment option involves slowly spinning the combined spacecraft up, releasing the sensor package while keeping a high deployment tension, and providing persistent low tangential thrust. The thrust must be from the main spacecraft on the heavier propulsive end, but it can be at low thrust levels that are achievable by electric propulsion thrusters. The tangential thrust adds spin angular momentum while the tether brake retards deployment. Eventually, when there is enough tangential relative velocity (~5 m/s), the brake tension is quickly reduced on the “barber pole” brake system to essentially throw the rest of the tether out while de-spinning the assembly, "yoyo-despin" style. Up to 95% of the tether length would be deployed in this fashion. This can be analytically challenging, but reliable controls can be developed as part of the PHLOTE Phase II study effort.

<u>Notes on the Separation System:</u> Planetary Systems Corporation’s Mark II Motorized Lightband separation system operates without the use of pyrotechnics, an advantage to this mission due to the satellite sensitivity to shock because of the instability at the L1 point. Pyrotechnics also create debris whereas this system does not. The separation system needs to fit as close as possible to the perimeter of the surface of the deploying object to reduce unintended stack dynamics. The band we would need for the exact geometry of the POP’s octagon surface falls between the two smallest COTS sizes for this system, 8” and 11.732”. There are no custom sizes available for this size system. The 8” Lightband leaves about 1.6” (4cm) between the edge of the Lightband and the POP, so a small circular ledge was added to the top of the POP frame to accommodate the 11.732” band.

Spring	Spring Constant [N/mm]	Stroke [mm]	Force Before Separation [N]	Force After Separation [N]	Stored Energy [J]	Remark	Graphic
Separation Spring	4.08	21.64	88.29	0.00	1.02 (±10%)[11]	Used to create the separation velocity. Has telescoping features. PSC PNs 2001071 and 2001065	

Figure 17: Spring specifications on the Lightband separation system

Maximum Resultant Rotation Rate: 0 ± 5.0 deg/s
Nominal Rotation Rate: 0 ± 1.0 deg/s
Separation Velocity (depending on mass) : 0.25-2.0 ft/s
If the spring force does not act through the center of mass of the payload (POP), there will be rotation rates. For this reason there will be small cold gas thrusters on the POP for all three degrees of motion.

<u>Additional Notes on Tether Deployment:</u> A concern for deployment is that we may not be able to predict the tension of the tether exiting the deployer accurately enough to control and stabilize the motion of the center of mass during the deployment. Remember that we are operating near in a very unstable location at the Mars-Phobos L1 location. At the separation rates predicted, the deployment could take between half an hour and hour and a half which is long enough for the CM to drift very far from the desired location near L1, with a possibility of crashing into Phobos.
It is preferable to have a more predictable system, such as with just one reel with a well-defined control torque.
If we consider a born-spinning deployment, we should realize how fast and precise this maneuvering should be in such an unstable location, and how little control we actually have over the motion of the center of mass (~1 mN from electrospray thrusters on a 100+kg system), and how much uncertainty we

have from tether friction. There may be a narrow set of ideal trajectories, but we may not be able to reliably execute any of them in practice.

To reduce risks from a loss of controllability, the tether deployment period should be minimized. Due to high communications latencies to Mars, the deployment must be managed autonomously. Since ground based testing of the deployment would be difficult, the verification would need to be accomplished through simulation and analysis. As a last resort, the tether will be cut to drop the sensor platform if the main spacecraft is in jeopardy. The main spacecraft could then perform alternative mission operations, for instance, at the Mars-Phobos L2 or at Deimos.

The mass ratio of the main spacecraft and the sensor platform is important for minimizing the tether length as well as for controllability. If the sensor platform has a higher mass relative to the spacecraft it will require a longer tether to balance the system so that its center of mass is kept near the L1 location. In addition, the planned maximum range of the NDL is about 6 km with its current optics, so the spacecraft should not greatly exceed this limit or overall mass will increase as optical apertures for the sensor increases. For controllability, it appears that a mass ratio ~12-15% is preferable (e.g. a 16 kg Sensor Platform (POP) in a 120-kg system with a tether mass of ~2 kg).

Figure 18: PHLOTE with tether Deployed

Operations Phase: During this stage, the main spacecraft must maintain its center of mass near the L1 location. This will be done using electrospray micro thrusters and tether length adjustments. Some of the compensation for orbital eccentricities can be done by periodically adjusting the tether length. This implies that the tether and the adjustment mechanism must be designed to operate for many cycles. In addition, the spacecraft must maintain its rotation so that the solar panels are oriented toward the sun.

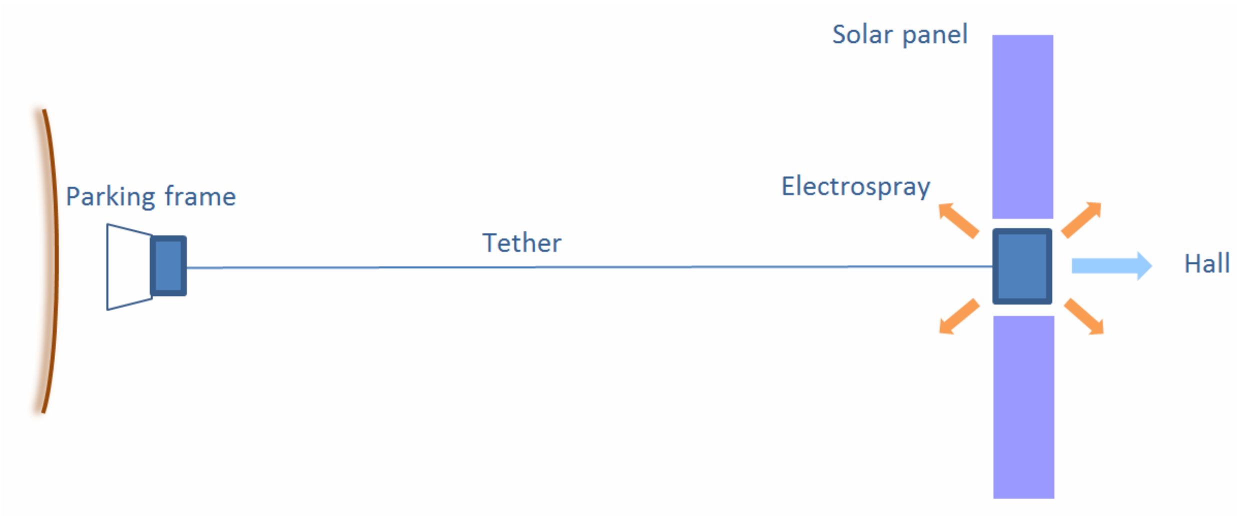


Figure 19: PHLOTE configuration shown in the plane normal to the Phobos orbital plane

In the configuration shown above, two solar panels normal to the orbital plane can track the sun by simple 1-axis rotation without disturbing the tether configuration. This greatly simplifies the deployment and tracking mechanisms for the solar arrays. Electrospray thrusters oriented at 45° will not spray on the tether or the solar panels (Since these high velocity exhaust particles can erode the tether or degrade the solar panels over time). The Hall thruster will not be obstructed either, and can operate with a deployed tether. If software updates are required or when communications with Earth are restricted, such as during a solar conjunction, the spacecraft can park the sensor package on the surface so that the system will remain stable indefinitely without active control.
This system configuration also provides multi-layer control authority to handle the key failure modes below:

Condition	Control
Normal Operation	Considering reeling in and out the tether
Reeling cannot be performed	Use the electrospray thrusters
Reeling cannot be performed and electrospray thrust is insufficient (too far from L1)	Use Hall thruster, if drifting away from PhobosPark sensor package on surface if drifting toward Phobos

Table 2: PHLOTE Control Modes for L1 Station Keeping

A few notes on the Electrospray Thrusters: *Based on the near-L1 dynamics, hovering with the electrospray thrusters could require a delta-V of ~100 m/s per year. With a Specific Impulse (I_{sp}) of 800 sec and thrusting angles ~45 deg., approximately 0.7 kg of the ionic liquid is needed per year. This is a ROM estimate that assumes less than 10 m deviations from the periodic motion. To get a more accurate estimate, a model the control system with the NDL in the loop is needed (Planned for the PHLOTE Phase 2 study).*

A few notes on position sensitivity for surface area coverage: *There will be a desire to sample as much of the surface as possible during normal operations at the Mars-Phobos L1 point. The sensor package could be moving in a box ~200 m laterally and ~700 m radially (between the highest and the lowest point). Additional libration could be induced to increase the lateral size of the box, both in-plane and out-of-plane. It may be also possible to induce lateral oscillatory motion of the center of mass, while staying within the control authority of the electrospray thrusters and the tether length control. While the sensor package is parked on the surface taking measurements, the system is stable and does not need active control.*

For the purpose of large area sampling, quasi-periodic orbits around Phobos can be used. The orbits selected would approach the surface to within the LiDAR range of 6 km at least periodically to determine the length of the tether that could be safely deployed.

PHLOTE Stability and Control

Balancing PHLOTE near L1: A peculiar feature of Phobos is that its Hill surface is close to its physical surface, and there is only a small region of Phobos-bound motions. One end-mass of PHLOTE (the POP) will be inside this region, while the other will be outside (the MOM). For simple analysis, an exact equilibrium of PHLOTE can be found if we disregard the eccentricity of the orbit of Phobos.

Because of the elongated shape of Phobos (26 × 22.8 × 18.1 km), the L1 radius in the non-spherical field of Phobos will be 17.25 km, compared to 16.6 km in a spherical field. With a 15-kg lower mass located 3.75 km below L1 (~0.5 km above the surface), a 103-kg upper mass, and a 2-kg tether, the overall tether length required to maintain equilibrium will be 4.6 km. In this configuration, the average tether tension will be 40 mN.

If the lower mass is 30 kg, and the upper mass is 88 kg, the tether length should be around 5.8 km to maintain equilibrium, which places the mother satellite ~6.3 km away from the Phobos surface and out of the range of the current Generation 3 Navigation Doppler Lidar. Therefore, the lower mass should be much less than the mass of the mother satellite.

To reduce tether bending, it is preferable for the lower mass to be at least 3-5 times more massive than the tether itself. Also, for the stability of PHLOTE in a parked state, when the lower mass is resting on the surface of Phobos, the tether length should be at least 4.5 km, and preferably, 4.7 km or more to ensure the main spacecraft is well beyond the L1 point.

Consequently, there is a minimum mass for the lower end-body. For a total mass of 120 kg, the lower mass should be at least 14 kg. For example, with an 18-kg lower mass located 3.75 km below L1 (~0.5 km above the surface), a 100-kg upper mass, and a 2-kg tether, the overall tether length required to maintain equilibrium will be about 4.8 km. The average tether tension will be 48 mN. In a parked state, the average tension will be 25 mN. These values are close to the mass projections for the latest spacecraft design and provide some payload opportunities on the lower body.

In all these cases, the center of mass of PHLOTE is located outside the Hill surface, and if the tether is retrieved in a short period of time, then the spacecraft will start moving away from Phobos.

PHLOTE Orbital Stabilization at the Mars-Phobos L1 Location: One way to stabilize PHLOTE near the L1 location would be to use electrospray thrusters on the upper end-body. This has to be done frequently to keep PHLOTE very close to the unstable periodic motion of the L1 point.

Another way would be to control the length of the tether. To stabilize PHLOTE in a hover mode near L1, we have designed a novel tether length control that will utilize the NDL's measurements of the position and velocity of the upper end-body. The basic idea is to vary the tether length L proportionally to the radial deviation ΔX of the end-body from the periodic motion around L1, so that $\Delta L = k \cdot \Delta X$, where the control coefficient k varies from 3 to 5 depending on the ratio of the end-masses. Another term depending on the radial velocity deviation adds damping of tether oscillations on top of the natural damping caused by the friction of the fibers in the tether.

The existence of this control is a rare and exciting find, and it is not immediately obvious. When the main spacecraft starts moving away from Phobos, instead of pulling on the tether, an extra tether segment is unreeled off the spool. This places the sensor package closer to Phobos, the total gravitational force acting on the system increases, and the tether pulls the main spacecraft back toward Phobos.

We have conducted a linear analysis of the tether system motion around L1 under the tether tension control and derived first approximation conditions of its stability. The analysis has shown that the stabilization effect is achieved in a very particular region in the parameter space and has allowed better understanding of the way this control works. It also provided the necessary guidance for the parameter selection in simulations.

A typical trajectory of the lower end-body under the stabilizing tether control is shown in Fig. 16 below. In this example, the center point of the sensor platform's trajectory is located 0.5 km above the surface of Phobos.

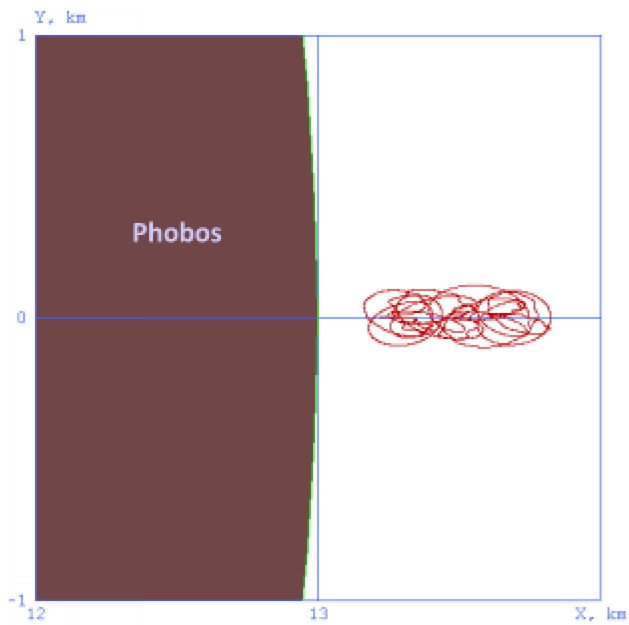


Figure 20: Motion of the sensor platform with tether stabilization

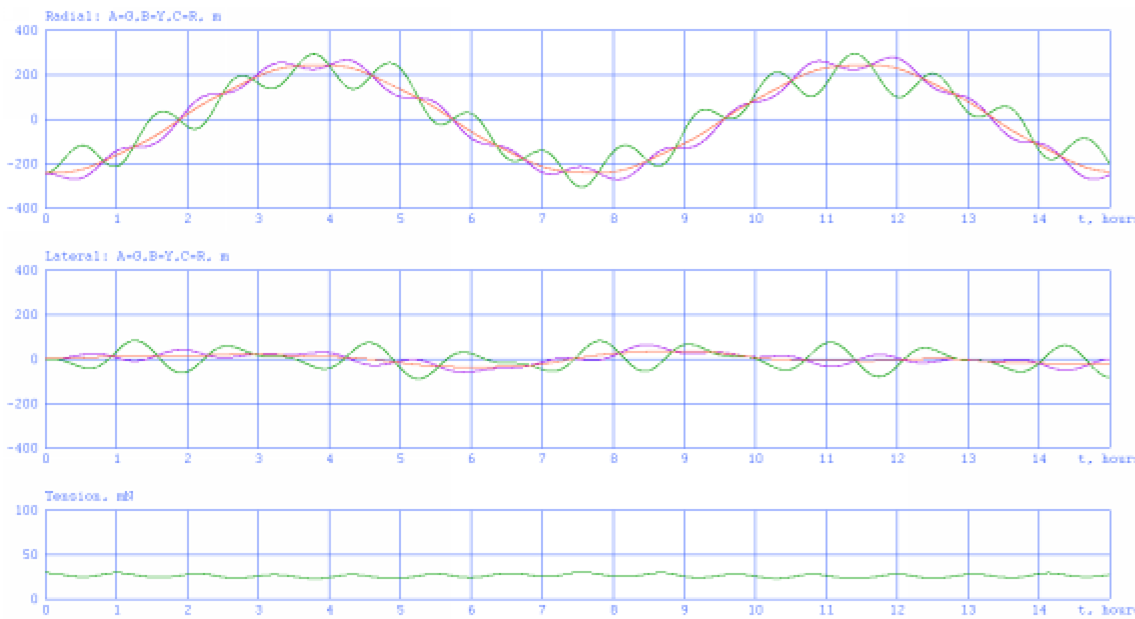


Figure 21: Radial and lateral components of motion with tether stabilization

In this example, we see some residual coupled longitudinal-transverse oscillations superimposed on the slow periodic variations caused by the eccentricity of the orbit of Phobos. They can be reduced but it should be noted that the sensors platform at the end of the tether will not remain in a fixed location relative to the surface unless it “parks” directly. This motion also means that the sensor package must maintain an average distance above the surface that minimizes the risk of unintended contact. The timing of this periodic motion should also be considered when performing a parking maneuver or when sampling the surface.

To stabilize its motion near L1, PHLOTE can switch between using the electrospray thrusters and adjusting the tether length as needed. The ability to hover in a very unstable location near L1 for long periods of time without using thrusters will result in fuel and mass savings and will provide a necessary redundancy in the motion control system of PHLOTE.

Recovering from a Temporary Loss of Control: If both systems for stabilizing control become temporarily unavailable, PHLOTE will drift away from L1, as shown in Fig. 11. There are two cases to be distinguished and two forms of recovery.

Case 1: If PHLOTE drifts toward Phobos, it will stop when the sensor package touches the surface and remain stable for the duration of any wait period that may be necessary, if the condition of the minimum length of a parked tether discussed earlier is satisfied. To get back to its periodic motion around L1, the thrust of the electrospray thrusters will not be sufficient. The thrust of the Hall thruster may be sufficient, but it would need to fire downwards close to the tether line. A simpler solution would be to reel an extra segment of the tether out. A minimum of 665 m would need to be reeled out with a 14-kg lower mass, 680 m with a 15-kg lower mass, and 694 m with a 16-kg lower mass, assuming a total mass of 120 kg. Then, PHLOTE will lift off and start drifting away from Phobos. Once the sensor package is a few hundred meters away from the surface, the extra tether segment can be retrieved, and the periodic motion near L1 can be restored. In order to have this recovery option, the tether system should have an extra 1 km stored on the reel. To have this recovery option available, the full tether length should exceed 5.18 km for a 14-kg lower mass, 5.26 km for a 15-kg lower mass, and 5.33 km with a 16-kg lower mass, assuming a total mass of 120 kg.

Case 2: If PHLOTE drifts away from Phobos, this drift could be contained by the Hall thruster provided it remains operational. However, in the case of a computer reboot or a power outage, drift will continue until the power and/or control is restored. Fortunately, the tether will remain under tension for a few hours after the departure and will retain its shape, but it raises a question of how to return to L1 with a deployed tether without having it tangled.

One option is to induce a slow spin about the center of mass, which will keep the tether under tension for any period of time necessary for recovery, and then use the Hall thruster to steer the rotating system toward L1 and stop rotation once at L1. The downside is that the Hall thruster will be rotating with the rest of the system, and its thrusting cannot be continuous and would need to be timed with a favorable orientation of the system. It may be doable, but it will be a rather complex maneuver.

Another option would be to retrieve the tether so that the spacecraft can repeat the L1 approach and the tether deployment sequence. Tether retrieval will typically cause rotation about the center of mass, which will help to keep the tether under tension during retrieval. The residual angular momentum and rotation rate can be kept in check with the electrospray thrusters. In order to have this recovery option, the tether should be fully retrievable.

If recovering control of both the main spacecraft and the tether is not an option, and the tether segment becomes too costly to readjust or salvage, the tether will be cut as a last-resort effort to re-stabilize the spacecraft. The POP will be prepared for a freefall impact with the surface of Phobos, and can continue low-power operations from the surface and communications with the main satellite. The tether will be cut at the MOM end so that it falls to the surface of Phobos and does not present an orbital debris hazard at Mars for future missions.

PHLOTE is targeting the High-Performance Space Computer (HPSC)[8] being developed by NASA's Space Technology Mission Directorate for the avionics processor that will control the spacecraft. The ARINC 653 software specification for space and time partitioning for safety critical applications will be used to partition the station keeping activities from the other C&DH activities that are not as time critical. This fault tolerant processor combined with the use of application partitioning architecture will greatly reduce the chances of a computer reset that impacts L1 station keeping operations.

Instrumentation for a PHLOTE Mission

Science Payload

The PHLOTE Mission provides significant opportunities for science. The spacecraft can provide evidence regarding both the origin of the small moon Phobos and the formation of the entire Martian system. Collecting data about the physical and chemical compositions of Phobos is crucial to understanding its past. An important question is how Phobos ended up in its current location. The chemical composition implies it is a captured asteroid, but its low density and near circular orbit implies it formed from material that coalesced after a large impact during the formation of Mars. Detailed spectrographic and gravitational measurements would greatly help answer this question.

The Phobos Operational Platform's (POP) purpose will be to carry sensors to elevations above and at Phobos. Most other missions are restricted to using scientific instruments that are only used either on the surface or in orbit, but PHLOTE's mission allows for opportunities at varying elevations including contact with the surface. A major advantage of the POP is that so little of its available mass and volume need to be committed to GN&C due to the utilization of the tether. Most of this mass and volume on the POP will be dedicated to the scientific payload. A PHLOTE science working group will prioritize the final instrument selections. Below are primary and secondary instrument candidates for the PHLOTE mission:

- **Visible Imagers:** A visible imager will be located on the bottom and top of the main spacecraft as well as the bottom, top, and side of the POP. The cameras are critical to the mission for not only examining physical characteristics of Phobos and Mars, but to also monitor the tether mechanism and the craft itself to make sure there are no difficulties, damages, or hazards.

- **Infrared Spectrometer:** Both the main spacecraft and the sensor platform will be equipped with infrared spectrometers, which measures the absorption of infrared wavelengths and produces a spectrum showing which elements are present in the material. Phobos is considered a possible waystation for human exploration of Mars, but little is known about the surface composition and Phobos' history. There may be utilizable resources in the surface of Phobos. An infrared spectrometer would be able to reliably identify elements in the surface, answering questions to both the origin of Phobos and to whether we can make use of its resources.

- **Infrared Camera:** This imager would capture changes in Phobos' infrared radiation over time. This will show the heating and cooling rates of the surface, which is closely related to its fluffiness and will provide insights on regolith characteristics and the density of the moon's outer layers.

- **Dosimeter:** The sensor platform will carry a dosimeter to track the radiation environment around Phobos. Besides monitoring radiation that affects instrumention it will provide a critical data set for future manned missions to the surface of Phobos. This radiation data is needed to prepare and protect astronauts from harmful radiation doses.

- **Gamma Ray Spectrometer:** Galactic Cosmic Rays are everywhere in the universe and originate from far outside our own solar system. When they collide with other particles, each element releases different gamma rays. These bursts of energy are measured passively using a Gamma Ray Spectrometer. The elements present in any given material can be determined by examining the observed gamma ray spectrum, and the density or abundance of these elements can be determined by the intensity of the spectrum. This instrument is low mass, low power, and CubeSat compatible.

  Neutron Detector: Neutrons are an additional source of radiation that crews and equipment will experience, therefore this mission needs to measure and monitor these particles as well. A 1.5" CLYC scintillator crystal coupled with a silicone photomultiplier, though not yet commercially available, will provide the sensitivity to thermal neutrons and desired output while requiring much less power and volume than traditional photomultiplier tubes or solid-state photomultipliers.

- **Ground Penetrating Radar (such as RIMFAX planned for Mars 2020):** Phobos is not a uniform mass. The interior may have pockets of different densities or materials altogether. A ground penetrating radar on the surface of Phobos will be able to map out the structure of the subsurface. This will also add to determining safe and viable locations for future landing and sampling.

- **Dust Detector:** Much of the dust around Phobos is sub-millimeter in size, and is therefore too small to track from Earth. Dust detection would provide better understanding Phobos' active dust and particle environment near the surface or on orbit. A Piezo Dust Detector uses a piezo sensor grid to detect collisions of dust or debris particles and records mass, velocity, and impact energy of each particle.

- **Radio Science Instrument (onboard MOM):** The main spacecraft's high gain communication system will be used in conjunction with receiving stations on Earth to precisely measure the velocity of the spacecraft. This will allow scientists to produce a detailed gravity map of Phobos. This gravity map will identify mass concentrations (and locations with low mass). This will help scientists determine the internal composition and structure of the moon.

Further scientific instrumentation, experiments, and opportunities have also been considered. While the aforementioned instruments have already been integrated into the mass/power calculations and the models later in this document, a list of additional Optional Payload Opportunities follows:

- **Phobos Surface Properties Experiment (PHOSPER)** – This experiment will consist of accelerometers to determine soil compactness, materials on the landing pads to look at adhesion and electronics to determine surface charge.

- **Phobos Hydrated Mineral Experiment (PHAME):** Water would be an incredibly valuable resource to find on Phobos because the location would be convenient to provide supplementary fuel for rockets traveling between Earth and Mars. Phobos' spectrum is similar to a D-type asteroid (carbonaceous and having phyllosilicates/clay minerals.) The questions is whether Phobos has any hydrated minerals (water or hydroxyl bound into the phyllosilicate clay minerals.) D-type asteroids frequently have these hydrated minerals. However, several studies have looked carefully at the diagnostic part of the spectrum (3-micron band) for evidence of water from Earth, and they find that it is either absent or in the noise. This tells us that at least the surface of Phobos must be desiccated. These measurements are an upper limit; there still might be several weight percent water in the surface material, but it may be too small to definitively measure from this distance. Another possibility could be that the surface was warmed from being in Mars orbit for so long, closer to the sun than most asteroids, but below the surface regolith there could be more hydrated minerals. On the other hand, the surface material could be a dusting of carbonaceous material while the deeper core is Martian ejecta and not carbonaceous or hydrated at all. To definitively determine whether this is the case, planetary scientists recommend performing measurements for the presence of water or hydroxyl from material extracted just below the surface. This can be done by looking even more carefully at the spectrum or by taking samples and heating them to about 600-degree C or higher to see if water comes out. The latter is preferable because it is diagnostic. You could heat the soil by hitting it with a laser or by putting it into a sealed container and using any heating method such as resistive heaters. Then you need instrumentation to detect the released water vapor.

- **Surface Observing Nano-Lander (SON)**: There may be enough mass and power deliverable on the POP to allow a small surface module to be attached to the POP and deploy upon touchdown on the surface. The SON would be roughly 2-3 Units in size, and could deliver Pop-Up Flat-Folding Explorer Rovers (PUFFERs)[9] and a small variety of science instruments to remain on the surface full-time. It would be equipped with a patch antenna to send data from the sensor packages and to receive commands from the MOM.

- **Microscopic imager:** A microscopic imager located at the bottom of the sensor platform would provide a detailed look at the surface composition and structure while parked on the surface.

- **Soil Probe:** The POP will also be equipped with a way to test soil and regolith properties such as adhesion, including electrostatic effects. Manned missions to the moon overlooked this quality, and so they were unprepared for the substantial damage that was done to their equipment and suits, especially any vacuum seals the dust encountered. The soil will be examined to avoid the same circumstances on future missions to Phobos and Mars.

Since PHLOTE acts as a precursor mission for a manned mission to Phobos, it will have to collect useful information for accurate planning in order for us to prepare properly for such an endeavor. The spacecraft can provide data on radiation and the surrounding environment, land in enough places on Stickney to evaluate surface property statistics, evaluate potential landing spots, and determine the available resources on Phobos that could be used in future missions. PHLOTE will also be able to identify possible sampling sites and confirm whether Phobos is made up of Martian debris. Determining this key fact opens up many more opportunities to more easily access Martian material and learn more about the planet. Since a sample return mission to Phobos would be much less costly and complex than one to Mars, this would open up more opportunities to more easily reach and examine Martian soil.

> *"The big picture for the human mission we care the most about is understanding the chemical and mechanical characteristics of the regolith on Phobos. This is to understand if the soil is toxic, how much of an adhesion issue we would have with space suits and machinery, and also to understand the geotechnical characteristics of the surface as it would relate to using a "hopper" style excursion vehicle." – Michael L. Gernhardt, Astronaut JSC*

PHLOTE Technology Development

Several technologies will be demonstrated on the PHLOTE mission. This includes the control software, the NDL, and the tether system. Engineering data from multiple sensors will be acquired to assess the performance of these technologies. Engineering sensors will include a POP deployment camera, a Tether strain gauge, and accelerometers.

Navigation Doppler Lidar Development

The NDL is a key component for enabling the PHLOTE mission. This sensor is currently at a TRL of 5 and has flown successfully on rocket powered test beds (Morpheus and Xodiac) during precision landing tests[11-13]. Components in the latest NDL all have a path to a flight version. Since the NDL has strong pull for use in precision landing it is expected that a version of the NDL will be at a TRL of 6 in 2019 long before the proposed PHLOTE mission PDR in 2023.

Parameter		GEN 3	PHLOTE NDL	PHLOTE NDL w/beam steering
Velocity Error		0.2 cm/sec	0.2 cm/sec	0.2 cm/sec
Altitude Error		23 cm	23 cm	23 cm
Attitude Error		0.5 deg.	0.5 deg.	0.5 deg.
Maximum LOS Range		3500 m	4500 m	6000 m
Maximum LOS Velocity		200 m/sec	10 m/sec	10 m/sec
Data Rate		20 Hz	10 Hz	0.1 Hz
Dimensions	Electronic Chassis	29 x 23 x 20 cm	25 x 20 x 18 cm	25 x 18 x 18 cm
	Optical Head	34 x 33 x 21 cm	34 x 25 x 17 cm	30 x 7 x 6 cm
Mass	Electronic Chassis	8.7 kg	6.2 kg	5.0 kg
	Optical Head	5.0 kg	3.5 kg	1.5 kg
Power (28 VDC)		80 W	60 W	55 W

Table 3: Performance of Navigation Doppler Lidar

The PHLOTE version of the NDL will not need the high update rates needed for precision landing on Mars. It is expected that the update rate can be 1/10th Hz or less. It will also not need to detect the high velocities needed during a Mars landing. It is expected the maximum relative velocities encountered on the PHLOTE mission will be ~3m/s. This will greatly reduce the Doppler Lidar signal processing requirements and this function may be offloaded to the spacecraft processor which is expected to be based on the High-Performance Space Computer currently being developed by NASA’s STMD.

Figure 22: Generation 4 Navigation Doppler Lidar

Another modification that will further reduce the size and mass of the NDL is replacing three optical lenses with a single lens. In current NDL, the laser power is divided to three beams transmitted simultaneously. Slower PHLOTE velocity allows the use of single beam that is directed into multiple fixed pointing angles sampled sequentially in order to extract the three components of the vector velocity. The ability to concentrate all the laser power into a single beam reduces the required power for achieving the required operational range and consequently results in further reduction in mass and power consumption of the sensor. The pointing is achieved using a new Non-Mechanical Beam Steering (NMBS) technology utilizing thin film Liquid Crystal Polarization Gratings (LCPG) [14]. This new NMBS technology offers a drastic mass and power reduction compared with mechanical motor/encoder devices. Under a different project, two devices were acquired to assess their operation as an integrated component of the NDL instrument. The preliminary results are very promising indicating a total loss of less than 20% in the lidar signal power.

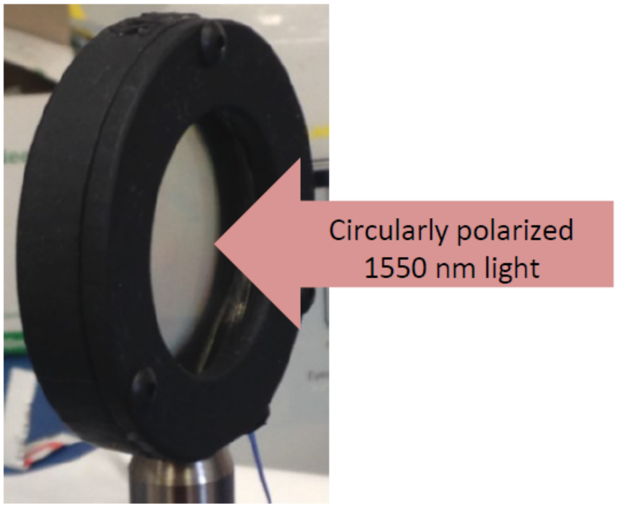


Figure 23: NMBS based on Liquid Crystal Polarization Grating being tested for integration into the NDL

If a TRL 6 version of the NDL exists to support precision landing, a version tailored for the PHLOTE mission is expected to cost 5 to 7 million to design, build, and qualify. This will be assessed in detail as part of the Phase 2 PHLOTE study effort based on ongoing NDL development as Part of the GCD SPLICE project.

Phase II Planning: During the Phase II study, a design concept of NDL per PHLOTE operational and physical requirements will be developed and high fidelity estimates of its mass, size, and power consumption will be provided. The NDL design concept will be based on the Environmental Test Unit (ETU) that is currently under development for landing missions and a series of performance and operational range measurements. A part of the phase II effort will be focused on the integration of the NMBS into the existing GEN 3 NDL and characterizing the NDL long range (> 4 km) operation. These tests include the NDL full system characterization with the NMBS from the LaRC gantry. For this test, the NDL is placed in an instrumented basket and raised to over 60 m height. The basket will be moved both vertically and horizontally to measure the vector velocity and range to the ground. This presents a full system test of the NDL with the NMBS demonstrating its capabilities as main PHLOTE position and range sensor. Additional tests include operational range measurements along the length of the Langley Air Force Base runway of about 3.8 km.

Figure 24. NDL system dynamic testing at NASA LaRC Gantry

Tether System Development

Tether: The primary candidate for a low-risk tether design for PHLOTE is a flat non-conductive braid of Spectra or Dyneema oriented polyethylene fiber, roughly 0.15 x 3.0mm. Flat braids have odd numbers of strands. To get to ~3mm width for a decent Mean Time Before Failure (MTBF) from micrometeoroid impact, about 31 strands of 100 denier strands are needed, or 35 strands of 75 denier, or 43 of 50 denier (75 is the finest Spectra strand available, and 50 is the finest Dyneema). Western Filament has braided narrower but otherwise similar flat braids, such as the 13x100 "leader tether" for ProSEDS. They can produce flat braids with roughly 31, 35, or 43 strands. A 31x100 denier braid would weigh 0.35 kg/km, or 2.1 kg for a 6-km tether. It may be feasible to reduce mass and/or increase width by adding several pre-tensioned elastic strands to the braid, to make it buckle and get wider when under the expected deployed tension of ~28 mN (which is ~14% more than the weight of a US penny). This should reduce vulnerability to micrometeoroids.

Dyneema is the European equivalent of the Spectra fiber used in the 4 km x ~2mm round TiPS tether that survived 10 years in LEO. In most cases, a flat braid should be intermediate in life between an equal-width round tether and an equal-mass round tether, but the exact probable life (which seems likely to be most limited by vulnerability to transverse near-grazing impact) is difficult to estimate.

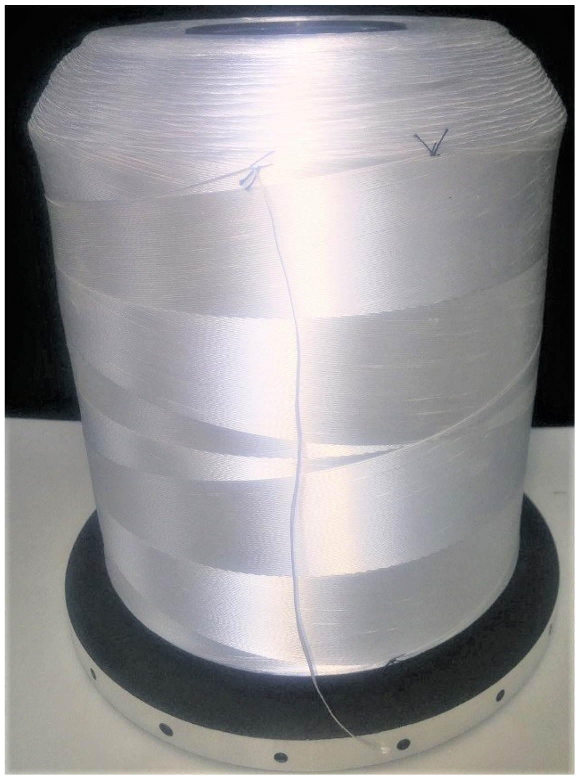


Figure 25: Flat Dyneema Braid on Mini-SEDS deployment reel

Spectra and Dyneema are not CNT fibers, but they have the highest strength/weight and impact tolerance of any fiber you can buy in kilogram quantities and handle on ordinary braiding machinery. One way of describing the fibers is 85% oriented carbon chains in 15% ductile hydrogen matrix. These fibers are stronger below room temperature, and have even higher strength/weight than new Zylon below ~200K, and don't lose strength over time like Zylon. And they have very low outgassing and generate much less free particulates than most other high-strength fibers. Their main shortcomings are a low melting point of 420K, and a tendency towards creep under sustained loads exceeding 5-10% of the short-term breaking strength at temperatures substantially above half the absolute melting point, or ~210K. But the fiber is very white and will run <200K in Mars orbit, and the sustained tension will be <1% of breaking strength. The real issue may be keeping the tether clean, because anything that discolors it will raise its local temperature and increase susceptibility to creep.

Note that the tether need not be uniform in design along its length. The few percent of the length intended for re-reeling can use a moderately narrower and thicker and more tightly braided flat braid to *make it easier to handle repeatedly. One can reliably splice even quite different flat braids together by stitching them through each other for a conservatively safe distance.*

Notes on the Coefficient of Thermal Expansion of tether Materials

Spectra itself has a substantial negative axial thermal expansion coefficient (perhaps as large as -28ppm/K), as well as a substantially positive radial thermal expansion coefficient that may exceed +50ppm/K. The radial increase interacts with the braid angle to make the axial coefficient even more negative. So, the final axial thermal expansion coefficient depends on the braid angle (and probably on temperature and tension as well).

Clean Spectra has an extremely low absorption of sunlight, so it will not heat up much in the sun. In informal testing, it was found that it was easier to sever a Kevlar tether using sunlight focused by a foot-square Fresnel lens, than to melt a Spectra braid (which melts at only 420K or 296F). Polyethylene is even melted and used to dissolve chemicals for UV spectroscopy because it absorbs so little UV. On the other hand, a flat Spectra braid also has relatively modest longwave thermal emittance, not because it is reflective like metals, but because its simple chemistry (only C-C and C-H bonds) make it transparent through much of the thermal IR regime. The combination of very low absorptance and low emittance will tend to make heating and cooling take more time than might be expected. This quality reduces thermal shock (and unwanted length changes) when going in and out of eclipse.

If it is assumed that the final braid has a -28 ppm/K axial thermal expansion, and changes temperature by 35-70K, then its periodic length change will be 6-12m in 6 km. This is small compared to the ~30m or more length changes that are needed for stability control. So, thermal length changes need not be an issue, particularly if we measure, model, and account for them. This is a detail that can be evaluated in the Phase 2 study.

Note that NRL's ATEx experiment had an anomaly that triggered experiment termination that occurred about a minute or two after the tether came into the sun. But that tether involved Spectra fibers laminated between two layers of polyethylene film. It is quite possible that solar heating shortened the fiber and lengthened the film enough to buckle the tape and trigger the tape departure angle anomaly sensor. PHLOTE will not be vulnerable to anything like that.

It would be worth estimating min and max equilibrium temperatures, for tethers with a/e ratios of 0.1 and 0.2, at roughly the mid-point on the tether (to get a representative view factor of Phobos).
It is assumed that the tether has a random twist, so all clock angles to the sun are sampled. Once this is done, we can estimate the thermal mass and time constant near the expected min and max temperatures. Note that if Theta is Arcsine(body-radius/orbit-radius), in radians, then the view factor of a planet or moon from a vertical tether is F = (Theta - Sin(Theta)*Cos(Theta)) / Pi. This is from page 9 of the Guidebook for Analysis of Tether Applications[20]. You can use this to calculate view factors for both Mars and Phobos. Small departures from the vertical should not change the results much, since they have camparable opposing effects on the two sides of the tether.*

Notes on the tether tension

Note that if we use a 31x100 tether weighing 0.35 g/m, 28mN is the weight of 8 meters of tether. The ability of this tension to overcome tether bending stiffness is easily checked by cantilevering tether *samples off the end of a table. A 5cm length does not bend much, but 10cm or more of flat Spectra braid does. So 28mn is more than enough tension to keep such a tether nearly straight.*

For SEDS-2, the braiding done by Cortland Cable, with a moderately tighter braid than was used for SEDS-1. They did not specify that Cortland add no lubricant when braiding, and they found later that roughly a cup of coconut oil was added to each tether during braiding, as a braiding lubricant. This increased deployment tension in ground tests, particularly at low temperature, since coconut oil is solid below about 76F. Hughes Spacecraft had to clean the flight and backup tethers in supercritical CO2, using equipment they developed to clean critical spacecraft battery components. The "BeadSim" deployment model fit the SEDS-2 flight data best with a fixed braking tension term of about 13mN, only about 37% as high as for SEDS-1. PHLOTE will definitely require cleaning the tether. (It was found that if no oil is added, we can reduce the fixed deployment tension by cleaning the tether in water.)

Note also that when the tether sticks at all during deployment, that raises the time-averaged deployment tension far more in flight than in ground tests, because there is little compliance in the length of tether between the deployer and the takeup reel on the ground, while in flight there is an increasing length of tether with some imposed transverse dynamics that add compliance. So, something that makes a bit of tether stick during flight will result in a much longer-lasting tension ramp (and a higher average tension) than in ground tests, where the sticking point is freed up by a very brief tension spike. That is my main candidate for explaining the difference between ground-test and flight tensions for SEDS-1.

A flat braid may have somewhat more variable and perhaps higher average deployment tension than a round SEDS-flight-like tether of the same fixed total mass per unit length, due to the higher local stiffness in the plane of the flat braid. A concern is about having deployment tension margin strengthens the preference for a yoyo-despin deployment over a SEDS-like deployment driven by ejecting the two masses apart.

Adding several pre-tensioned elastic strands to the tether to cause it to buckle out in width is an option for micrometeorite resistance but would likely increase deployment tension. If the elastic strength is enough to widen the tether at deployment tensions of order 28mN, then it is likely to puff it out even as it exits through the deployer and brake. That could raise the fixed term in deployment tension. On the other hand, if we do commit to a yoyo-despin style deployment, then we can afford substantially higher tension during deployment.

Tether Reel: *The tether reel can be based on the "Mini-SEDS" tether deployer. This was a scaled-down version of the SEDS deployer and brake that flew successfully in 1993 and 1994, and without brake as part of the 1996 NRL TiPS experiment. The full-size SEDS deployer holds up to 7.5 kg of Spectra tether, and the Mini-SEDS deployer holds up to 2.1 kg--such as 6 km of a 3mm wide 31x100 flat braid. Protoflight Mini-SEDS deployer and brake hardware exist as SBIR deliverables which MSFC may still have on hand. Even if they don't have that anymore, Tether Applications has several sets of that hardware. One is shown in the figures below. The deployer weighs 2 kg empty including the separate "barberpole" brake. Little effort went into minimizing deployer mass for the Mini-SEDS protoflight hardware. The empty hardware could probably be reduced to 1.2-1.4 kg with little difficulty, if necessary.*

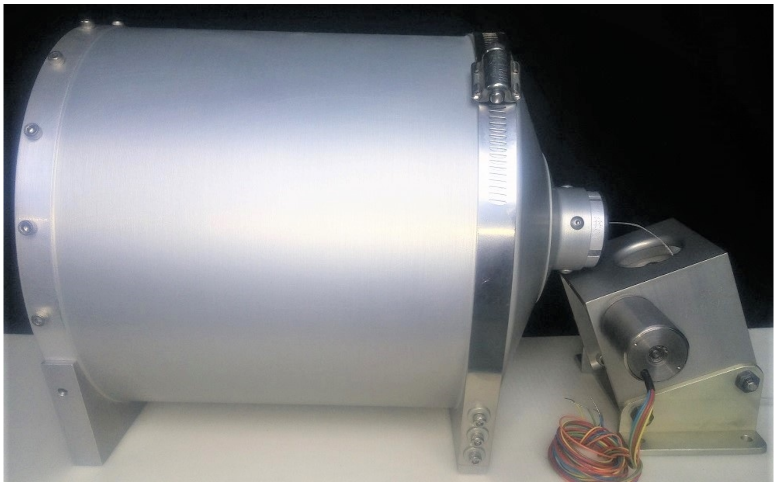


Figure 26: Mini-SEDS deployer and brake side view

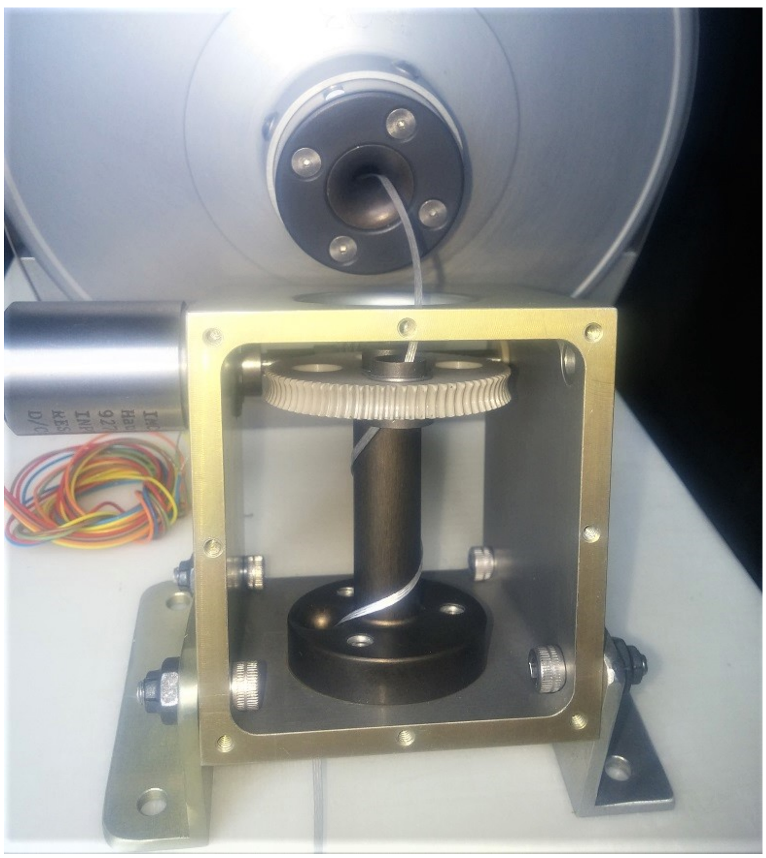


Figure 27: Tether Path Exiting the Deployer an Adjustable "Barberpole" Tension-Amplifying Brake

This brake raises the dynamically-driven deployment tension from the deployer (which varies linearly with the square of the velocity, plus a small constant term) by roughly a factor of 3 per spiral wrap around the barberpole. Usually one can wrap at least 4 turns around the barberpole without problems.

This allows a ~80:1 tension control ratio during deployment, multiplied by whatever the upstream deployer tension is.
Brake control is by driving a worm on the space-rated stepper motor that protrudes from the brake frame. Most of the tether brake heating actually leaves with the tether, and the rest goes into the barberpole post, mostly near where it attaches to the frame. Very little goes into the floating guide in the worm gear, since that is the low-tension inlet end.

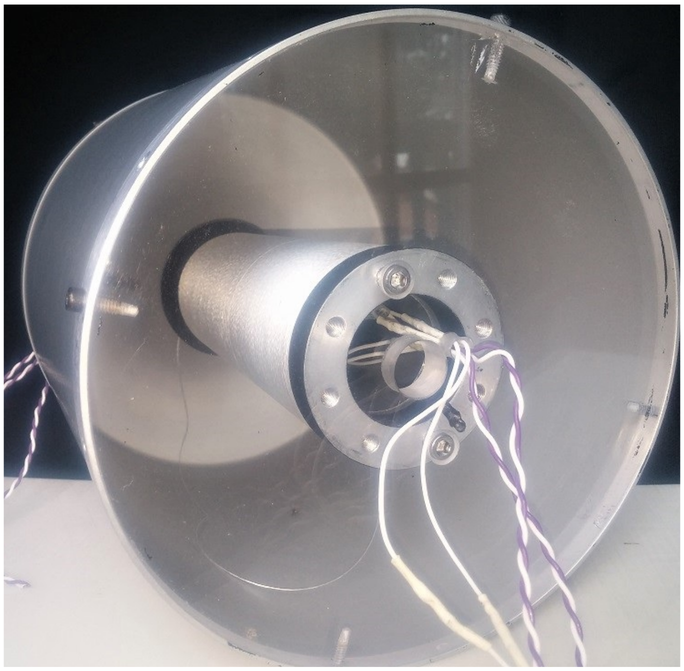


Figure 28: Inside of the Mini-SEDS Deployer through a Clear Plastic Base

This configuration was used to videotape tether dynamics during deployment tests. Deployment tests are usually inside a vacuum chamber since air drag totally dominates gravity effects as a perturbation, especially for deployment velocities above ~2 m/s, which are dominated by centrifugal force rather than gravity.

Note that the Mini-SEDS deployer is a deploy-only device, but the inner end of the tether can exit through a hole in the base of the deployer. There it can connect to a length of tether stowed on a much smaller reel-type deployer that can hold a few percent additional tether length that can be reeled in and out, through the Mini-SEDS deployer and barberpole brake. This add-on reel concept was originally proposed by a group at Auburn University. If the "re-reelable" tether length is kept short enough, there is no need for a level-wind mechanism, which adds most of the complexity and failure modes of a rotating-reel-type deployer. This added re-reelable tether length is what would be used in the control laws to stabilize the tether dynamics and orbit. Note that buildup of an orderly winding may require more winding tension than is available externally. But the barberpole brake can amplify tension in both directions, so the rewinding tension can greatly exceed the external tension if necessary.

The TEPCE (Tether Electrodynamics Propulsion CubeSat Experiment) was developed by the Naval Research Laboratory (NRL). The TEPCE tether was a 1.5 x 0.2mm flat braid, with 3 flexible conductive strands; typical tension 3-4 mN. This shows a compact design of tether stowed around a stacer-type separation spring. This may be applicable to a PHLOTE mission.

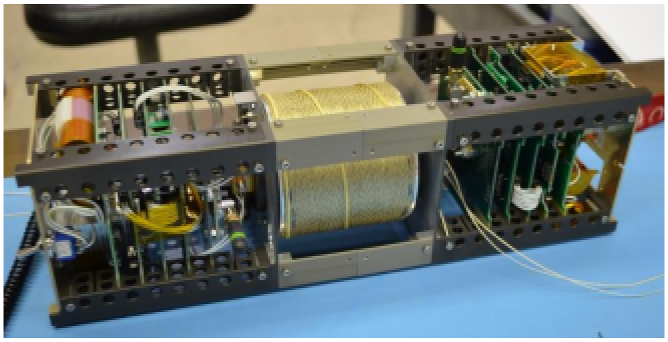


Figure 29: TEPCE with the Tether Spool

PHLOTE System Architecture and Mission Design

PHLOTE System Decomposition

The PHLOTE System will have three elements: The Ground Segment, the Launch Segment and the Space Segment. The primary focus for the NIAC Phase I Study will be the Space Segment.

PHLOTE Space Segment (PHLOTE Spacecraft)

The PHLOTE spacecraft will be decomposed into the following Elements: Ion Propulsion Element (IPE), Electrical Power Element (EPE), Attitude Control Element (ACE), Mechanical-Structural-Thermal Element (MSTE), Communications Element (Comm), Tether Element, Command and Data Handling Element (C&DH), and the Science Element.

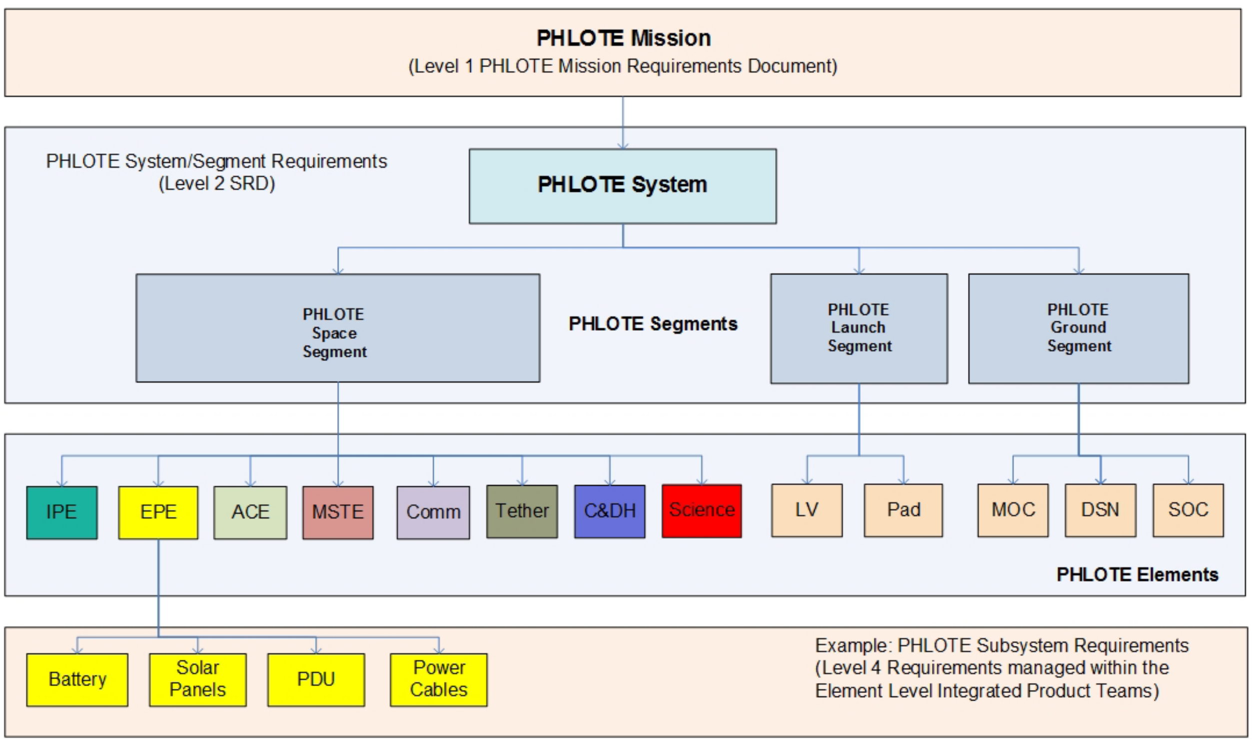
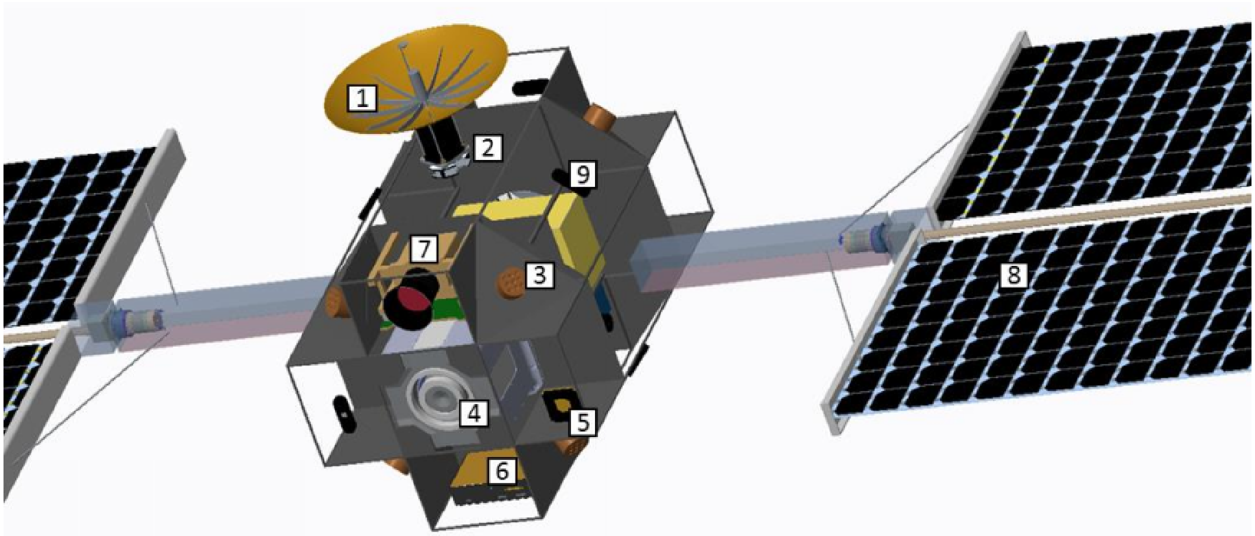


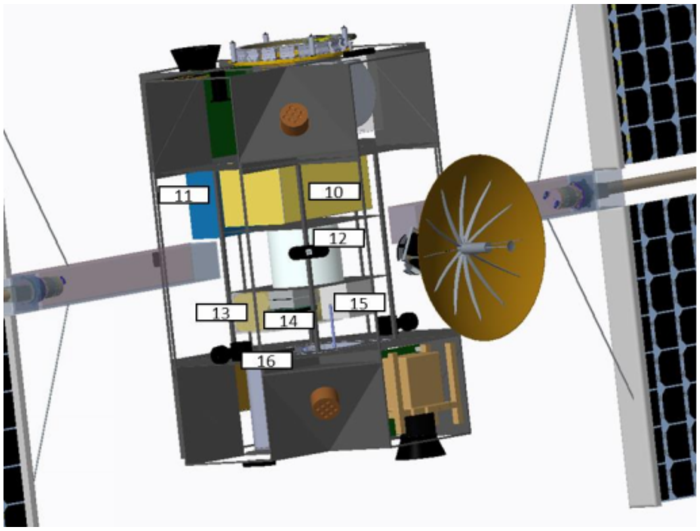
Figure 30: PHLOTE System Decomposition

The PHLOTE Space Segment shall have a separable sensor platform connected to the main spacecraft by a tether subsystem. The main spacecraft will be referred to as the Mars Observation Monitor (MOM). The sensor platform will be referred to as the Phobos Observation Platform (POP). The POP will be part of the Science Element.



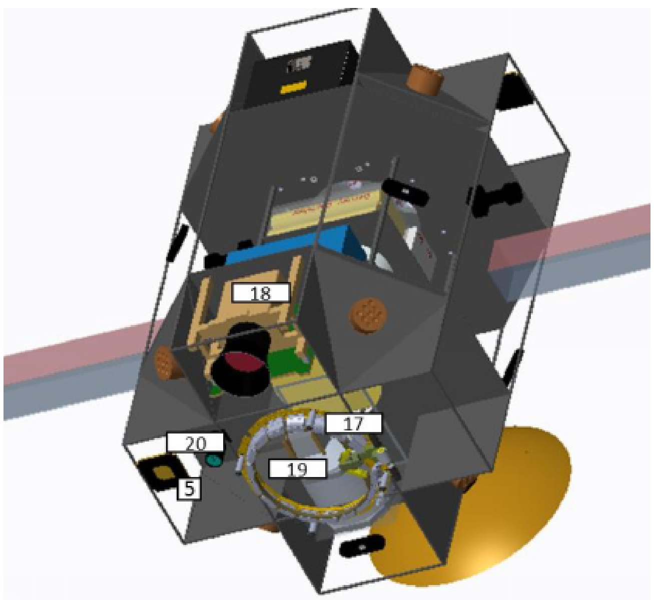
#	Elements/Subsystem
1	Deployable X-Band High Gain Antenna
2	Antenna Gimbal System
3	Electrospray Thruster
4	Iodine Hall Thruster
5	S-Band Patch Antenna
6	Iodine Hall Thruster Control
7	Mars Imaging Camera
8	Deployable Solar Array System
9	Sun Sensors

Figure 31: Main Spacecraft View 1 (without MLI to show components)



#	Elements/Subsystem
10	Power Distribution and Control System
11	Navigation Doppler LIDAR
12	Iodine Tank
13	Battery Array
14	Communication Transmitters
15	High Performance Spaceflight Computer
16	Star Tracker

Figure 32: Main Spacecraft View 2 (without MLI to show components)



#	Elements/Subsystem
17	MOM POP Separation System
18	Phobos Imaging Camera
19	Tether Reel and Deployment System
20	Navigation Doppler LIDAR Optical System

Figure 33: Main Spacecraft View 3 (without MLI to show components)

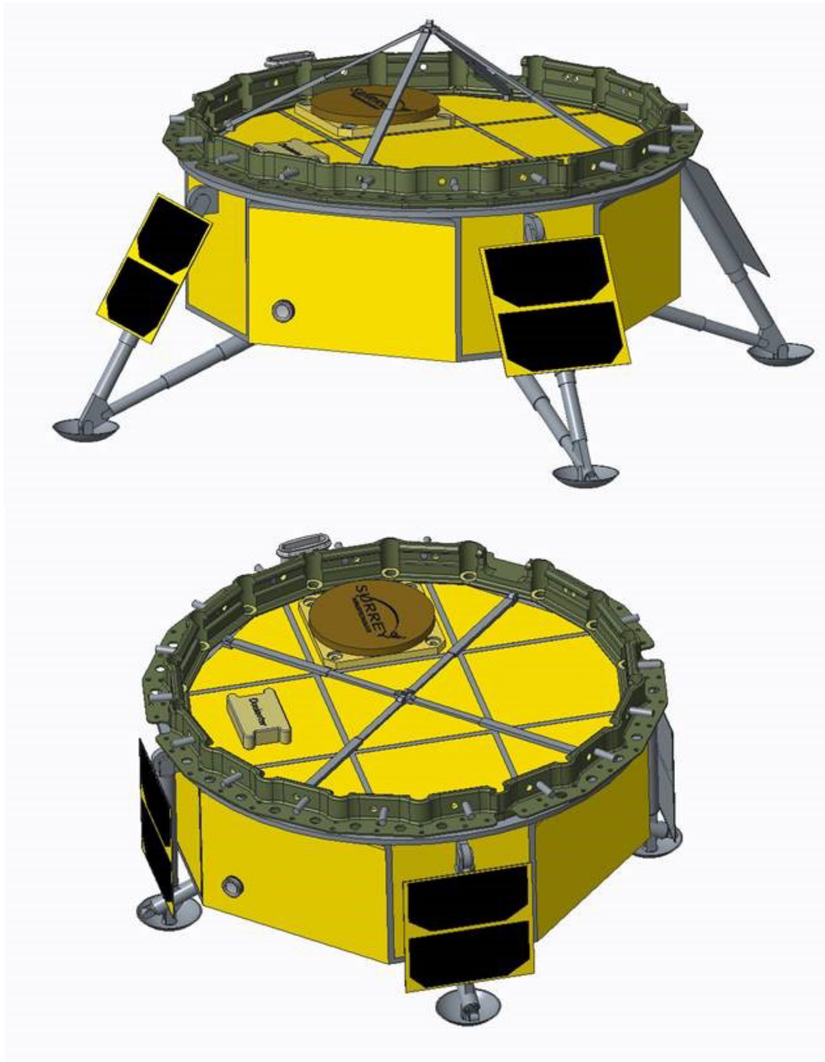
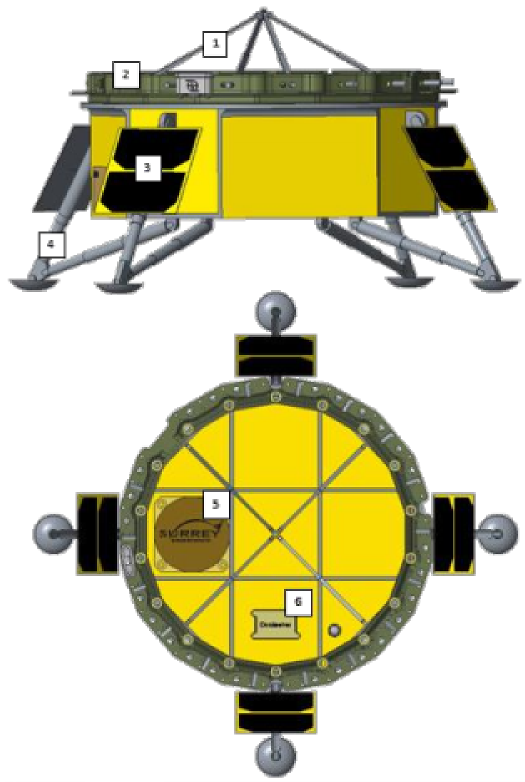


Figure 34: Components in the Phobos Observation Platform (POP) (Deployed and Stowed)

#	POP Components
1	Tether Struts
2	MOM/POP Separation System
3	Solar Cells
4	Landing Legs
5	S-Band Patch Antenna
6	Dosimeter
7	Integrated Battery and Electrical Power Subsystem
8	Gamma Ray Spectrometer
9	ADCS Control Board
10	Command and Data Handling Board
11	Infrared Spectrometer
12	Dust Detector
13	Reaction Wheels (x3)
14	Transceiver
15	High Res Phobos Imager
16	Tether and Phobos Imagers



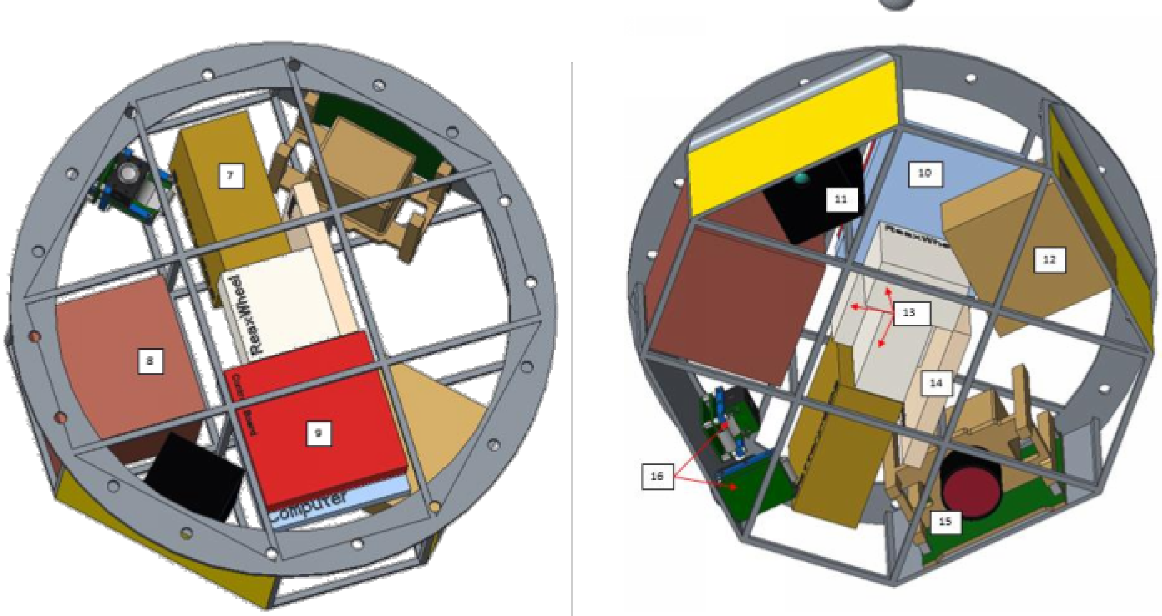


Figure 35: Upper side of the Phobos Observation Package (Top View)

Notes on PHLOTE Mission and Spacecraft Design Trades

Options for Delivery to Phobos

There are several options for getting the PHLOTE spacecraft to Phobos. One method is to obtain a ride as a secondary payload. NASA is also actively investigating the potential for commercial payload services to Mars (Mars Payload Services Sought[10]). Potential launch vehicles as a primary or secondary payload include:

1. GSLV Mk III: http://www.isro.gov.in/launchers/gslv-mk-iii
2. Ariane 6: http://m.esa.int/Our_Activities/Space_Transportation/Launch_vehicles/Ariane_6
3. Falcon 9/Heavy: http://www.spacex.com/falcon9
4. Minotaur V: http://www.orbitalatk.com/flight-systems/space-launch-vehicles/minotaur/docs/BR06006_3862%20MinotaurV_030117.pdf
5. Delta IV: http://www.ulalaunch.com/products_deltaiv.aspx
6. Vulcan: http://www.ulalaunch.com/Products_Vulcan.aspx
7. Electron: https://www.rocketlabusa.com/electron/
8. Vector-H: https://vectorspacesystems.com/vector-h/
9. LauncherOne: https://virginorbit.com/

PHLOTE Thruster Calculations if a dedicated payload placed into a low Earth orbit:

Starting at higher altitudes is always preferred. Starting inclination does not matter much for low thrust transfers. A downside to starting at LEO is that the spacecraft will spend substantial time in the radiation belts on its way up and will therefore have to carry extra radiation shielding. From GTO, the spacecraft will cross the hot places in the radiation belts only for relatively short periods during the first two weeks, and it will be in the sun almost constantly. This may justify the extra expense of a GTO ride. The PHLOTE Mass Estimator generated the tables below which compare different rides to show the big picture. There are no rides to Geosynchronous Equatorial Orbit (GEO) at this time, but it illustrates the range. There are three Moon launches announced so far (Falcon Heavy, EM-1, and EM-2)and this may start some traffic. Rides to escape can dramatically reduce time and delta-V. Table 4 provides estimates based on simulations (not fully optimized), assuming a dry mass of 85 kg without the propulsion elements and a 2,000 sec I_{sp}.

Destination of the primary payload	Mars	Moon	GEO	GPS	GTO	LEO
ΔV to Phobos, km/s	4.2	7.2	10.5	11.3	11.8	15
exp=	1.24	1.44	1.71	1.78	1.83	2.15
Tank mass, kg	4.5	8.8	14.9	16.6	17.8	26.9
Spacecraft dry mass, kg	95	99	105	107	108	117
Fuel, kg	23	44	74	83	89	134
Fuel volume, liters	4.6	8.9	15.1	16.9	18.1	27.3
Launch mass, kg	**117**	**143**	**179**	**190**	**197**	**251**

Rideshare rate for the mass range, $K/kg	55	30
Spaceflight Ind. launch cost, $M	10.8	7.5
Virgin launch cost, $M		10

Table 4: Transfers with Iodine Thrusters

Transfers from LEO would be the most challenging. On the upside, there are many launch opportunities to LEO. There are some advantages of having a dedicated launcher (a rocket where PHLOTE is the primary payload), such as the Virgin Orbit's LauncherOne ($10M, 310 kg to 400 km SSO). A flight test is expected in 2018 and it would provide margin for mass and volume or to reduce time spent traversing the radiation belts.

Another key consideration is the need to start in a sun synchronous orbit that is fully in the sun to allow PHLOTE's IHTs to thrust continuously and achieve the fastest possible ascent and lowest radiation dose. In this respect, the Air Launched LauncherOne has an advantage since is can be launched from an optimal location. Launching to a high inclination orbit will reduce the radiation dose on the payload as it spirals up with an electric propulsion system.

Power needs at Phobos may require up to 5.7 m2 of solar cell area with a 32% efficiency. This solar array would produce close to 2.4kW near Earth. We can therefore use a cluster of four 600W thrusters (BHT-600) capable of producing up to 0.2N of thrust near Earth. It will add 19 kg to the dry mass of PHLOTE, but would dramatically reduce the time to escape and the time spent in the radiation belts and will result in a much lower radiation dose. Table 6 shows how system characteristics depend on the number of BHT-600i thrusters.

Number of thrusters	1	2	3	4
Power required, kW	0.6	1.2	1.8	2.4
Mass of thruster assembly, kg	5	10	15	20
Tank mass, kg	26.9	28.4	29.9	31.4
Spacecraft dry mass, kg	117	123	130	136
Fuel, kg	134	142	149	157
Fuel volume, liters	27.3	28.8	30.3	31.8
Fuel per thruster, kg	134	71	50	39
Fuel per thruster, liters	27.3	14.4	10.1	7.9
Launch mass, kg	**251**	**265**	**279**	**293**
Time to Mars-bound insertion, months	20	10	7	5
Radiation dose with 4 mm Al shielding, krad	24	12	8	6
Spaceflight Ind. launch cost, $M	7.5	8.0	8.4	8.8

Table 5: PHLOTE Main Thruster Configuration Options

Alternatively, a tandem of 1000W thrusters (BHT-1000) could be used instead of four BHT-600 thrusters. A BHT-1000 thruster with a power processing unit, cathode, and cables weighs a bit less than two BHT-600 thrusters and can run at 1200W, making these two options very similar. Multiple IHTs also provide redundancy but will add to the mission cost.

Transfers with Hybrid Propulsion: Some hybrid propulsion options combining chemical and electric propulsion are being considered, see Table 6. Hybrid propulsion systems are getting more interest now, and they would allow PHLOTE to go from LEO or GTO to escape after a perigee burn of a chemical thruster and then continue with a Hall thruster. This will significantly reduce the transfer time and the radiation dose.

PHLOTE Launch Vehicle Calculations: The baselined option is for PHLOTE to be a dedicated payload on a small launch vehicle and have it placed in a low Earth orbit where it would use an electric propulsion system to travel to Phobos. Another option is launching as a secondary payload where the PHLOTE spacecraft would be ejected from a larger rocket, for example the SpaceX Dragon vehicle during its cruise phase to Mars. Once ejected it would deploy its solar arrays, establish communications, and begin slowing down using its electric propulsion system to enter a Martian orbit. The table below lists several options.

	From a Mars-bound trajectory			From GTO		
Fuel	Hydrazine	Water	Iodine	Bi-prop*	Water*	Iodine
ΔV, km/s	2.2	2.2	4.2	3.7	3.7	11.8
S/C w/o thrusters & tanks, kg	85	85	85	85	85	85
Number of thrusters	4	4	1	1	4	1
Mass of thrusters, kg	9.6	6	5	2	6	5
Isp, sec	220	300	2000	280	300	2000
exp=	2.77	2.11	1.24	3.85	3.52	1.83
Fuel/tank+feed mass ratio	5	5	5	5	5	5
Tank mass, kg	52.0	26.1	4.5	115.4	92.5	17.8
Spacecraft dry mass, kg	147	117	95	202	183	108
Fuel, kg	260	130	23	577	462	89
Fuel density, kg/liter	1.02	1.00	4.93		1.00	4.93
Fuel volume, liters	255	130	4.6		462	18.1
Launch mass, kg	**407**	**247**	**117**	**780**	**646**	**197**

*Impulsive ΔV depends on the GTO parameters

Table 6: Impact of Different Main Propulsion Options for PHLOTE Delivery to Phobos

The current dry mass estimate of the MOM is 69 kg (which includes contingency based on the TRL of the components). With a 25% margin the total mass is 86.25 kg. The current POP is 7.5 kg with margin is 9.375 kg. All together that is 76.5 kg with a margin of 95.625 kg. The mass of the fuel tank(s) is extra. Detailed information on specific component selections is provided in the PHLOTE Component Selections section.

PHLOTE Thruster Calculations as a Secondary Payload on a Mars Bound Vehicle: After separation from a Mars bound spacecraft on a trajectory for direct entry the PHLOTE spacecraft will need to match Phobos' orbital velocity of 2.1 km/s. To do that, PHLOTE will need to be capable of adjusting its own velocity from the velocity of the spacecraft carrying it from Earth. However, the speed of the spacecraft (and therefore PHLOTE) varies depending on the mission trajectory so that the specific time during cruise that it needs to eject from the carrier spacecraft in order to slow down with an electric propulsion system is also variable. The time to eject can be determined using the following equation:

$$t = \frac{\left(v_{entry} - v_{phobos}\right) * m_{satellite}}{(Thrust)}$$

Another option is to start from GTO. Hydrazine is not suitable for this kind of transfer, but bi-propellant systems are. Even though there are many ride opportunities, particular GTO parameters may not be favorable for impulsive departure to Mars. The required delta-V for insertion on a Mars-bound trajectory could range from 1.5 km/s to over 3 km/s, depending on the GTO parameters, resulting in a total delta-V ranging from 3.7 to over 5.2 km/s. Table 5 on the previous page shows typical parameters.

For the PHLOTE study, a dedicated launch to low Earth orbit was baselined to provide tangible constraints and cost based on advertised launch vehicle requirements. Since minimizing overall PHLOTE mission costs is a key design goal, the cost of the launch vehicle is a significant factor in the selection criteria. The Virgin Orbit's LauncherOne launch vehicle was selected as the best option given current mass estimates.

Main thrusters

Launch mass is one of the most important factors in enabling a low-cost mission to Phobos. An electric propulsion system would be the most effective system since other chemical propulsion systems would require significantly higher launch costs.

Iodine Hall Thrusters from Busek are currently undergoing lifetime testing at Marshall Space Flight Center as part of NASA's Game Changing Technology Development Program[15].

A key advantage of the Iodine Hall Thruster over the Xenon Hall Thruster is fuel storage. Unlike xenon, which requires large, heavy, high pressure fuel tanks, iodine fuel is very dense and can be stored as a solid at low pressure. Although iodine is very corrosive, the high density and low-pressure storage provides a great deal of flexibility in the overall design of the fuel tank so that the shape can be tailored to the spacecraft.

Climbing out of the LEO gravity well will require higher thrust than the iodine propulsion system currently being tested can provide. A larger thruster or clustered smaller thrusters, as shown in Fig. 36, may achieve necessary thrust. Alternatively, the 600W thruster could be run at higher power levels, possibly up to 1000W.This may require an additional power processing unit.

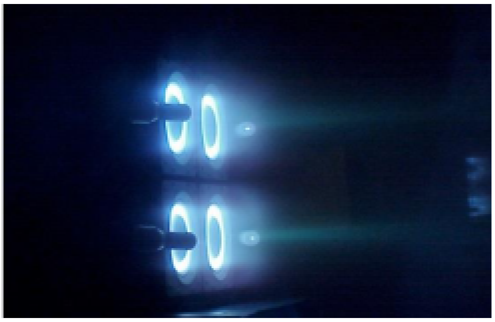


Figure 36: A Cluster of 600W Hall Thrusters in Operation

Power needs at Phobos may require up to 5.7 m2of solar cell area with a 32% efficiency. This solar array would produce close to 2.4kW near Earth. We can therefore use a cluster of four 600W thrusters (BHT-600) capable of producing up to 0.2N of thrust near Earth. It will add 19 kg to the mass of PHLOTE, but would substantially reduce the time to escape and the time spent in the radiation belts which will result in a much lower radiation dose. Alternatively, a tandem of 1000W thrusters (BHT-1000) could be used. A BHT-1000 thruster with a power processing unit, cathode, and cables weighs a bit less than two BHT-600 thrusters and can run at 1200W, making these two options very similar.

Figure 37: PHLOTE Arriving at the Mars System with the Iodine Hall Thruster

Iodine Tank

To minimize the size of the selected reaction wheels, the center of gravity (CG) of the spacecraft should be as close to the geometric center of the spacecraft as possible to minimize inertia. Coordinating its CG with the geometric center will maximize efficiency of the reaction wheels. This will decrease the mass and power needed for reaction wheel control, which is essential for the spacecraft's strict mass and power constraints. At launch, the iodine fuel and tank for the spacecraft's main propulsion system will be about the same mass as the entire craft itself. This makes it especially important for the tank to be placed at the geometric center so that the whole craft's CG maintains the same position as fuel is consumed throughout the mission. To accommodate both the reaction wheel and tank requirements, a custom tank configuration is recommended. Iodine fuel is a solid, so the tank will not be under high pressure. This allows for considerable flexibility in its design; the tank will be a cylinder with a cylindrical cut-out through the middle. Placing the reaction wheel assembly inside the cut-out at the geometric center of the spacecraft protects the components from the space environment. At 4.93 g/cc, solid iodine has a much higher density than propellants such as xenon. The dimensions of the tank are only restrained by volume requirements and can take many shapes. (NOTE: multiple small tanks with independent valves and/or thermal control may allow better CG control. Over time, iodine will tend to sublime from the warm side of each tank, and condense in empty spaces on the cool side. This has to be weighed against the likely increase in mass over a single tank.) It is important that the iodine fuel be consumed uniformly across the volume of the tank so that the CG does not greatly vary as fuel is consumed. The tank itself can have a thin interior and a large cut-out diameter to make room for other electronic components to fit inside. This configuration has the added benefit of doubling as radiation shielding for key components. The tank can also be made with a smaller diameter depending on the spacecraft's width or dimension constraints from the launch vehicle. In the current configuration, the radius of the inner cylinder needs to be at least 14.4 cm to accommodate the currently planned reaction wheel assembly at the center.

Given the advantages and current state of development, it is recommended that the mission baseline an Iodine Hall Thruster for the primary propulsion to get to Mars.

Station keeping thrusters

There are several different thrusters being developed to support the growing CubeSat/SmallSat industry. Several were looked at to support the PHLOTE mission during its primary mission phase.

	Isp (s)	Thrust (mN)	Mass (kg)	Power (W)	TRL	Notes
IMF Nano thruster	2000-5000	0.01-0.5	0.64	8-40	6	See CubeSatShop.com for product info
Busek 1 mN electrospray	800	1	1.15	15	6	
TILE-5000	1,500	1.5	1.4	1.5-25	6	Flight qualification expected Q1 2018

Table 7: Performance of Electrospray Thrusters

Based on analysis of control authority, it was recommended that the station-keeping thrusters have a minimum thrust of 1 mN to provide enough control authority so that the possible deviation from the L1 point is reasonable (10s of meters).

The current design has 8 station keeping thrusters.

The current baseline is the Busek 1 mN electrospray thruster.

Reaction Wheels

The mass (and inertia) of the PHLOTE satellite is much higher than that of a standard CubeSat. COTS reaction wheels developed for CubeSats provide torque an order of magnitude smaller than what the PHLOTE spacecraft will require, making them an impractical choice for attitude control.

Sources of disturbance generally taken into account when sizing reaction wheels are gravity gradient, solar radiation, magnetic field, and aerodynamic effects. However, PHLOTE will be made of mostly composites so the assumption is made that Mars' and Earth's magnetic fields will have negligible effects on the spacecraft.

Prior to tether deployment, the maximum torque due to the gravity gradient (T_g) on the craft at both Earth and Phobos can be estimated by

$$T_g = \left(\frac{3\mu}{2R^3}\right)\left|I_z - I_y\right|\sin(2\theta)$$

Where μ is the gravity constant of the celestial body, θ is the maximum angular deviation of the spacecraft's z-axis from the local vertical in radians, R is the orbital radius of the spacecraft, and I_z and I_y are the moments of inertia around the y- and z- axes of the craft. Assuming a maximum angular deviation of 45° for a worst-case, a simplified spacecraft geometry of a cylinder with a radius of 0.41m and height of 1.04m, and a mass of 150 kg at Earth (including fuel for travel to Mars) at a minimum orbital height of 180 km from the Earth's surface, the gravity gradient maximum at Earth is approximately T_g = 15.3 µNm. At Phobos, we use the same assumptions for angle deviation and geometry, but use a mass of 95 kg for the spacecraft (fuel required for Mars travel depleted) at a minimum orbital height of 3.5 km from the moon's surface to calculate a maximum gravity gradient torque of T_g = 1.18 µNm.

The maximum disturbance due to gravity gradient accumulates over ¼ of an orbit as roughly

$$h = T_g\left(\frac{0.707t}{4}\right)$$

Where h is the estimated wheel momentum and t is the orbital period of the satellite.

At Earth, the orbital period is approximated by a circular orbit of an average height of 440, giving an orbital period of about 93 minutes. Based on this, the maximum disturbance at Earth is h = 0.0151 Nms. PHLOTE's orbital period at Phobos will be the same as Phobos' orbital period around Mars, 7 hours and 39 minutes, and the maximum disturbance due to gravity gradient will be h = 0.00576 Nms.

While the symmetrical design of PHLOTE's solar panels is intended to minimize torque on the craft due to solar pressure (T_{sp}), there may be times when the craft goes into or out of eclipse where the panels are briefly asymmetrically lit but this will be very brief and is not considered a driving factor.

Phobos has no atmosphere to create aerodynamic drag; however, preliminary calculations for aerodynamic drag at the lowest point of PHLOTE's initial low-Earth orbit (~400 km for LauncherOne) will be further investigated in a Phase 2 study.

Although the calculated torque and momentum storage demands are low enough for a smaller standard CubeSat COTS reaction wheel, larger torques may be required for tether deployment and attitude adjustments throughout the mission. The Sinclair Interplanetary 0.06 Nms reaction wheel was chosen for its higher, SmallSat-appropriate torque. This wheel has about ten times the maximum torque and twice the momentum storage capacity of a standard wheel for large CubeSats while being only 13% more massive, as shown in the table below. The SmallSat wheel does require significantly more power to operate at its maximum torque, but can be throttled down for lower torque applications. Four such wheels in a pyramid orientation (see image below) relative to each other are used in the main craft for attitude maintenance. A small COTS reaction wheel produced by CubeSpace will be used in the POP to stabilize any rotation once deployed.

Reaction wheel	Max Torque (mNm)	Momentum storage (mNms)	Mass (g)	Max Power (W)
Medium COTS CubeSat	1.0	10	130	0.180
Large COTS CubeSat	2.3	30	200	2.2
Sinclair SmallSat	20	60	226	23.4

Table 8. Performance of selected Reaction Wheels

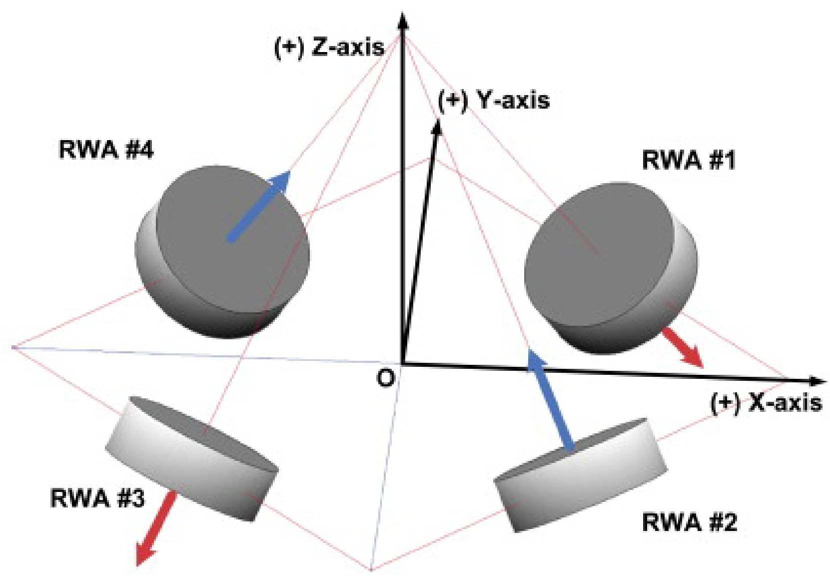


Figure 38. Pyramid Configuration of Reactions Wheels in MOM
(ref: http://www.sciencedirect.com/science/article/pii/S1270963814001734)[16]

The reaction wheels will be placed in a four-wheel pyramid configuration with one wheel providing redundancy. The pyramid can be flattened to provide additional roll authority in the direction of the tether axis since once the tether is deployed the spacecraft will be gravitationally stabilized along 2 axes.

Tether Material

The PHLOTE tether has several key requirements:

1. The primary tether section must be ~5000m long
2. The tether must be resistant to failure by micrometeoroids
3. The tether must be resistant to degradation from UV and ionizing radiation
4. The tether must low enough thermal expansion when going into and out of eclipse
5. The tether must be wear resistant from being reeled in and out to support station keeping
6. The tether must be limp enough so that it remains straight under the low tension expected

The PHLOTE tether will have three sections. A wear resistant section where the tether will be reeled in and out almost continuously to provide tether length control for station keeping. This section may have a protective cladding to minimize fraying and fuzzing and will be about 1000 meters long. The primary tether section about 5000 meters long. A much shorter (~10m?) section between the POP and the solar panels on the tether can include 2 insulated conductors to power the POP from solar arrays held above the POP by the tether.

The original concept of incorporating CNT yarn for an electrically conductive tether does not appear to be feasible primarily due to an assessment of the micrometeoroid environment and the probability of incurring damage that could short out two conductors to each other. This is exacerbated by the need for a full return path due to the lack of an adequate plasma environment to carry charge at Mars. Since a backup power and communications system would be needed on the sensor package anyway for redundancy; it was not considered practical to initially baseline a conductive tether. A CNT based tether section is still being investigated for the much shorter section between the POP and the solar panels as seen in Fig. 16.

A braid of spectra yarn is a good choice due to its ability to handle the space environment, and also its low Coefficient of Thermal Expansion (CTE). Like CNT yarn, Spectra materials also have slightly negative CTE in the radial direction. Several other previous tether design and material combinations that have been documented were considered:

SEDS-2 (Small Expendable-tether Deployer System)**:** NASA, 1994: L = 19.7 km, d = 0.78 mm, braided of Spectra-1000 strands, altitude = 350 km, cut 3.7 days after deployment, possibly by a piece of debris from the same launch, but more likely by a micrometeoroid; the lower segment reentered in hours, the upper segment (7.2 km) survived for two months until reentry.

TiPS (Tether Physics and Survivability) **Experiment:** NRL, 1996: L = 4 km, d = 2.2 mm, mass = 5.44 kg, altitude = 1000 km, survived without cuts for more than 10 years; the outer layer was made of braided Spectra-1000, with a core of acrylic yarn to puff it up to improve resistance to damage from small micrometeoroids and debris.

EDDE: Conductive tapes 30 mm wide, practically immune to micrometeoroid cuts.

Figure 39: Examples of Tether Configurations from Earlier Missions

Micrometeoroid Environment near Phobos: The near-Mars micrometeoroid environment is described in the, "Estimation of micrometeoroids and satellite dust flux surrounding Mars in the light of MAVEN results," Icarus, 2017, http://www.sciencedirect.com/science/article/pii/S0019103516303311.

Mass (g)	Flux ($\frac{1}{m^2 s}$)
0.00001	2.17E-09
0.0001	1.54E-10
0.001	8.86E-12
0.01	4.56E-13
0.1	2.21E-14

Table 9: Interplanetary Meteoroids (Grun et al)

Impacts of interplanetary micrometeoroids produce secondary ejecta from Phobos, which increases the particle population crossing the orbit of Phobos (in private communication, the author of the paper stated that the increase is relatively small).

Micrometeoroid cuts			
Cut probability per year	**10%**	**3%**	**1%**
Min. tether diameter(mm)	1.5	2.4	3.6
Dangerous particle mass (mg)	0.1	0.5	1.8
1-year mission reliability	90%	97%	99%

Table 10: Tether failure probabilities due to micrometeoroids

Reference tether: a flat or round braid or a thin tape ~3 mm wide. Stowage on a reel has advantages for deployment, control, and mission operations. The TiPS tether could serve as an early approximation, possibly in a flat braid version, like the TEPCE tether. The tether tension in the PHLOTE configuration is very low, and the tether strength is not a concern, but the tether must have enough braids far enough apart to survive micrometeoroid impacts. A "mostly hollow" round braid, a loose flat braid, or a thin tape could be suitable. The mass will depend on how "hollow" the braid may be. The TiPS tether at 1.36 kg/km sets the upper linear density limit, but it could be reduced dramatically by a clever braid design. The braid should retain its width or expand to its nominal width after long storage on a tightly-wound reel. For example, a nylon strand in the braid would get shorter while Spectra would get longer when they both cool off after deployment. But a <1% change in relative length would not buckle and hence widen the Spectra braid very much.

The current baseline is Spectra based weave. The marginal tether width and mass required to reduce cut risk is higher than for typical reliability-driven mass penalties, so accepting a somewhat higher than usual risk here appears justified.

Command and Data Handling (C&DH)/Processing

There are many C&DH packages becoming available to support the CubeSat industry. A new radiation-hardened processor called the High-Performance Space Processor (HPSC) is currently being developed under NASA's Game Changing Technology Development Program may be a good option. This multi-core processor uses Advanced RISC Machines (ARM) based architectures so that power usage can be greatly reduced over what is required for current radiation-hardened space systems.

	MOps/W	Mass (kg)	Power (W)	TRL	Notes
HPSC	1000	0.64	selectable	-	Multi-Core, selectable fault tolerance, trade performance for power, Available 2021
CubeSat COTS	TBD	0.05-0.1	0.2-0.5	9	Intended for LEO

Table 11: Performance for Selected Processors

It is expected that the HPSC will be available prior to a PHLOTE PDR. Key advantages are the selectable performance, fault tolerance, and power settings.

The PHLOTE flight software is expected to be developed using the core Flight System (https://cfs.gsfc.nasa.gov/) framework. This has become a standard for many mission developers and a great deal of resources are available to support the developers.

The current baseline is the HPSC currently being developed with software developed under the core Flight System framework.

Power

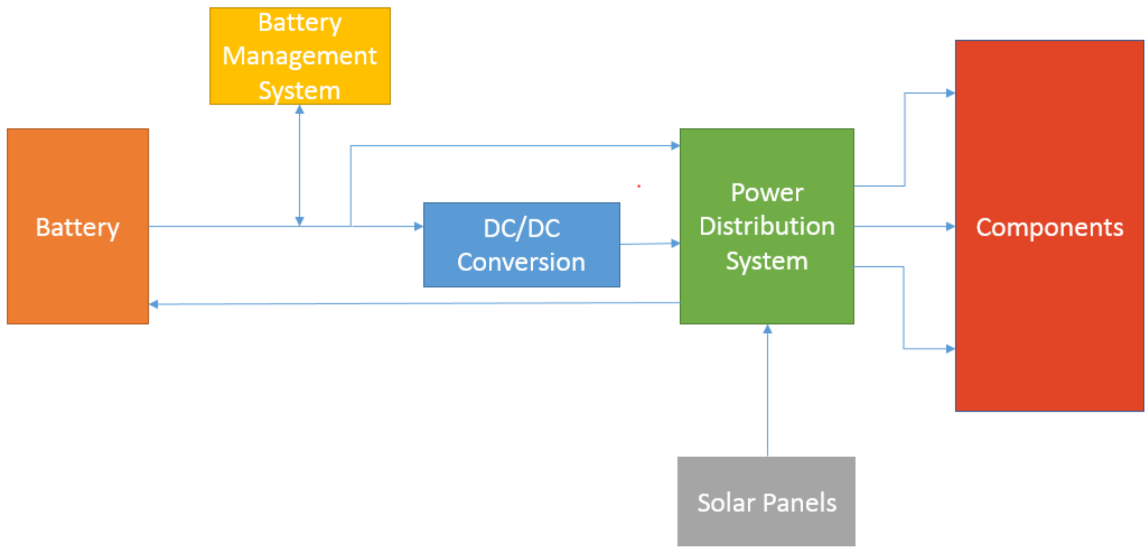


Figure 40: A simplified break out of the PHLOTE power system

The power system must be able to meet all requirements of the PHLOTE spacecraft. The average power consumption of the spacecraft is 412 W during L1 station keeping operations including a 30% margin, but will be close to 500W during traveling operations (with a single IHT) including a 30% margin. The peak power consumption can be as high as 1 kW. The power system must generate enough power to handle these different loads. The method of power generation chosen for PHLOTE is photovoltaic cells, which must transfer enough energy to power the spacecraft as well as produce and store enough excess for operations during periods where the sun is eclipsed by Mars or Phobos. The batteries must have the capacity to store this power, even with efficiency loss from duty cycling and operating is a space environment. Power conversion will be required and the power distribution system must efficiently transfer energy at correct voltages to the components. The voltage levels for most of the chosen components are either 28V, 5V, or 9V.

PHLOTE's Solar Exposure: Due to PHLOTE's operational location at the Mars-facing side of Phobos, both Mars and Phobos will eclipse the spacecraft at some point during each 7 hour and 39-minute orbit of Mars. The following calculations assume that spacecraft's distance from the surface of a spherical Phobos is 4.5 km. To determine how much time the spacecraft spends in shadow, the umbra for both Mars and Phobos were calculated to determine the percentage of its orbit is spent in the respective umbrae. The length of the umbra was calculated using the following equation:

$$L = \frac{r * R_0}{R_s - R_0}$$

Where L is the length of the umbra, r is the distance from the sun to the object casting the shadow, R_0 is the object's radius, and R_s is the radius of the sun. After the umbra's length was calculated, the objects, orbits, and umbra were modeled to scale in the PTC CAD tool Creo. The angle of the orbit that is in the umbra is then calculated, and the percentage of the orbit obstructed by the umbra, and that percentage is applied to the total orbital period to find the time spent in the umbra. This process is used for both the time that Phobos spends in the shadow of Mars and the time that the satellite spends in the shadow of Phobos. The Creo models appears as followed:

Event Descriptor

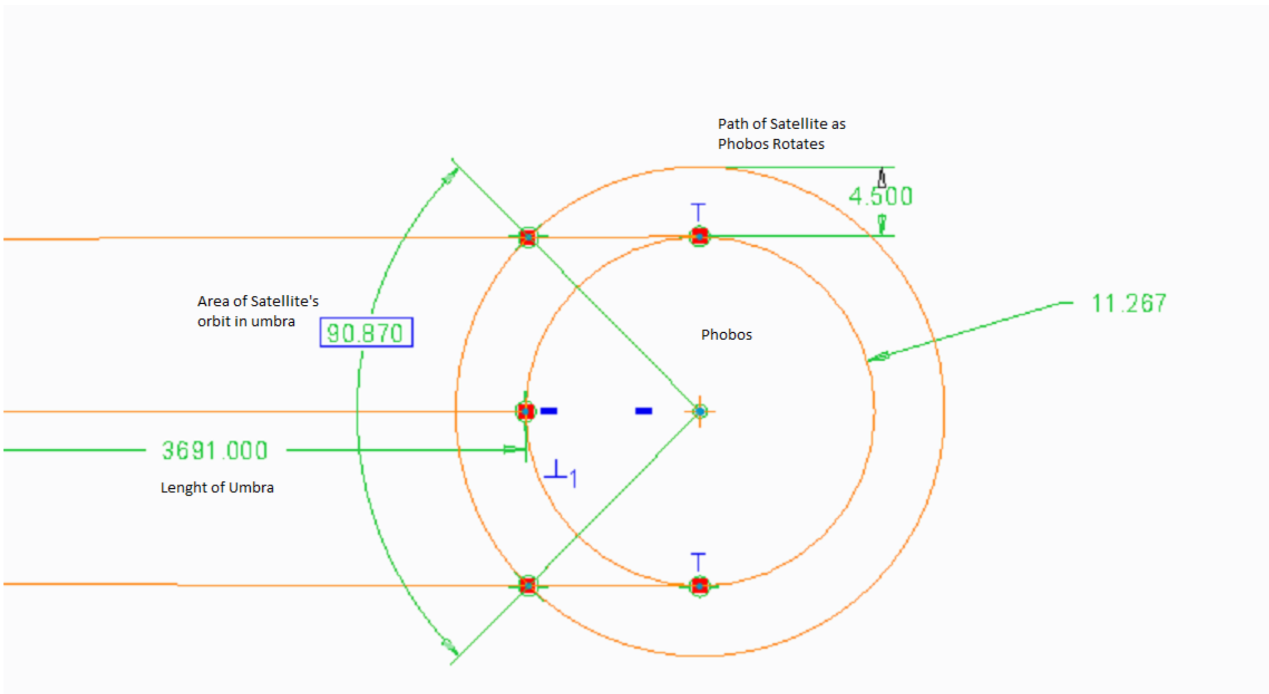


Figure 41: Phobos Umbra Model

Note: Mars model looks similar, but scaled to the size of Mars and Phobos' orbit. The area of Phobos' orbit that is covered by the umbra sweeps out an area of 44 degrees.

According to this model, Phobos spends 56 minutes of its orbit in the shadow of Mars and the satellite is in Phobos' shadow for 1 hour and 54 minutes.

Using the Systems Tool Kit (STK) application, it was found that the disparity between the inclinations of the previously stated orbits was great enough to create large fluctuations in sunlight exposure throughout a Martian year. Over the course of the mission, PHLOTE's exposure to sunlight goes from periods where it is eclipsed by Mars-Phobos for the maximum durations calculated by hand above to periods where PHLOTE will not go into eclipse for up to a month. Due to the nature of STK, PHLOTE could not be placed at the proper distance above Phobos and remain in a synchronous orbit around Phobos and was therefore left in an orbit 10 km above Phobos instead of 4.5 km. While this throws off the calculations for eclipse by Phobos, it still models a clear trend and, since the eclipse by Mars matches the calculations made by hand earlier, it is likely that the eclipse by Phobos calculations are also relatively accurate. There are two periods during the mission where the time spent in eclipse peaks, and two periods in time where eclipse does not occur for an extended interval of time. An image of the system in STK as well as the point of interest in the data follow:

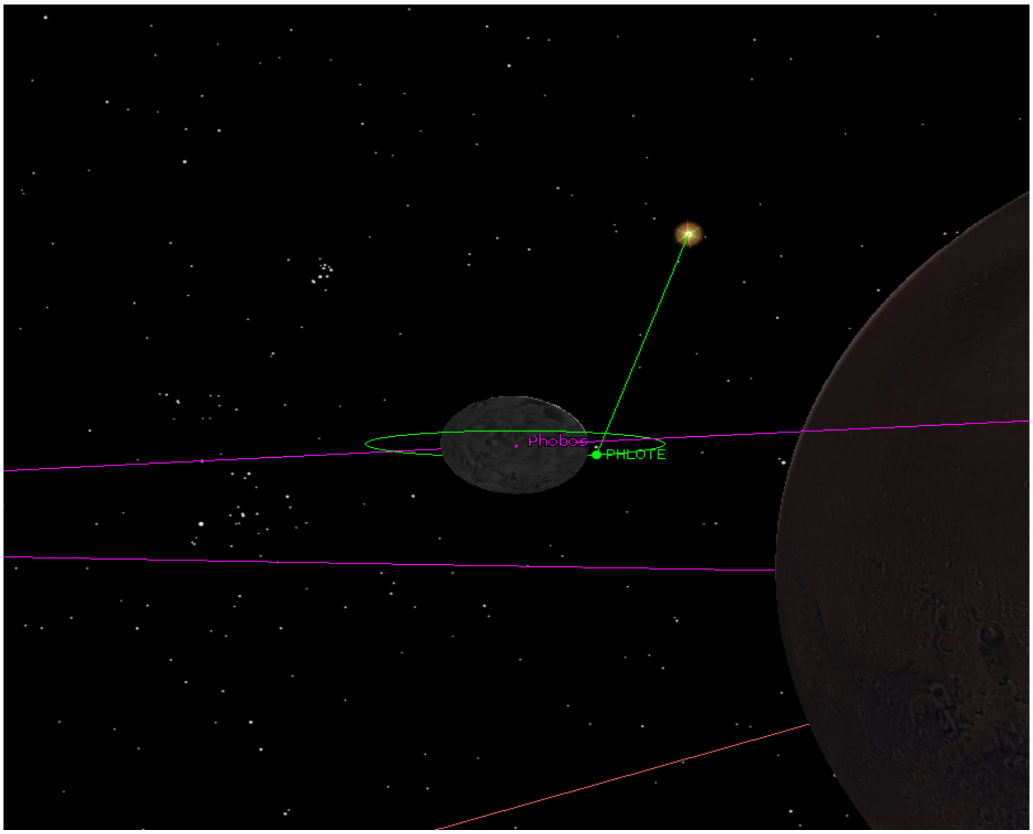


Figure 42: STK Model of PHLOTE in Orbit and Access Line to Sun

PHLOTE Sun Access Interrupted by Phobos			
Event Descriptor	**Beginning of event (date and time)**	**End of event (date and time)**	**Duration**
1st peak eclipse duration	27 Oct 2027 01:46:49.755	27 Oct 2027 03:05:39.171	1 hr. 18 min 49 sec
2nd peak eclipse duration	31 Aug 2028 02:29:02.219	31 Aug 2028 03:47:27.504	1 hr. 18 min 25 sec
1st period of extended sunlight exposure	22 Feb 2028 10:11:33.126	3 Apr 2028 05:57:20.482	40 days, 19 hr. 45 min 47 sec
2nd period of extended sunlight exposure	5 Feb 2029 22:26:25.194	28 Mar 2029 07:36:52.539	50 days, 9 hr. 10 min 27 sec

Table 12: Key Data Points for Solar Eclipse of PHLOTE Caused by Phobos

PHLOTE Sun Access Interrupted by Mars			
Event Descriptor	**Beginning of event (date and time)**	**End of event (date & time)**	**Duration**
1st peak eclipse duration	13 Oct 2027 16:07:21.615	13 Oct 2027 17:02:08.765	54 min 47 sec
2nd peak eclipse duration	14 Aug 2028 15:23:34.257	14 Aug 2028 16:17:09.564	53 min 35 sec
1st period of extended sunlight exposure	14 Jan 2028 23:28:45.795	2 May 2028 20:55:55.587	108 days, 21 hr. 27 min 10 sec
2nd period of extended sunlight exposure	10 Dec 2028 18:24:04.872	17 May 2029 23:46:27.891	158 days, 5 hr. 22 min 23 sec

Table 13: Key Data Points for Solar Eclipse of PHLOTE Caused by Mars

Solar Arrays: There is not a noticeable difference in the amount of power consumed during eclipse or during daylight other than battery charging which only occurs when the solar arrays are illuminated. More power may be needed during eclipse for the heaters but good thermal management will minimize this. Using the latent heat generated, only minimal heating will be required on the MOM. We can assume that similar amounts of power will be needed during the two different scenarios. The average power consumption of the spacecraft is 380W. Using the assumptions of the typical amount of time PHLOTE is in eclipse, the spacecraft is not receiving solar flux during 3 hours of its orbit. With a typical orbital period of 459 minutes, the spacecraft will spend at least 279 minutes in daylight. To estimate the power needed to be generated by the solar panels during daylight to account for eclipse, this formula is used:

$$P= ((Pe*Te)/Xe + (Pd*Td)/Xd)/Td$$

Where P is the amount of power needed, Pe is the power consumed during eclipse, Te is the period of eclipse, Xe is the efficiency of energy transfer during eclipse, Pd is the power consumed during daylight, Td is the period of daylight, and Xd is the efficiency of energy transfer during daylight. Note that the efficiencies are different during daylight and eclipse as stated above.

The orbital average solar flux at Mars is 590 W/m^2. But Mars has 0.0934 orbit eccentricity, so the flux varies substantially around the orbit, from 493 W/m^2 at aphelion to 717 W/m^2 at perihelion. PHLOTE solar arrays will only track the sun along one axis of rotation. Using this tracking and a spacecraft rotation around the tether line will keep the arrays normal to the sun and maximize energy production. There will be a small loss due to the inclination angle of Phobos' orbit relative to the sun.

To account for the loss of flux from the incident angle change, the flux received is multiplied by the cosine of the typical incident angle. The incident angle of the solar panels is found using the solar panel tool kit in STK. Assuming an efficiency on the solar cells to be 32% as stated in the data sheets of the perspective MOM solar cells, the surface area of the solar cells needs to be > 5.7 m^2 to provide the power requirement margins required for a preliminary design.

The MOM has two solar arrays, each composed of two solar cell blankets. In the arrays' stowed configuration these blankets fold up accordion-style into boxes that then fold down to the sides of the boom deployment mechanism as shown in the Fig. below.

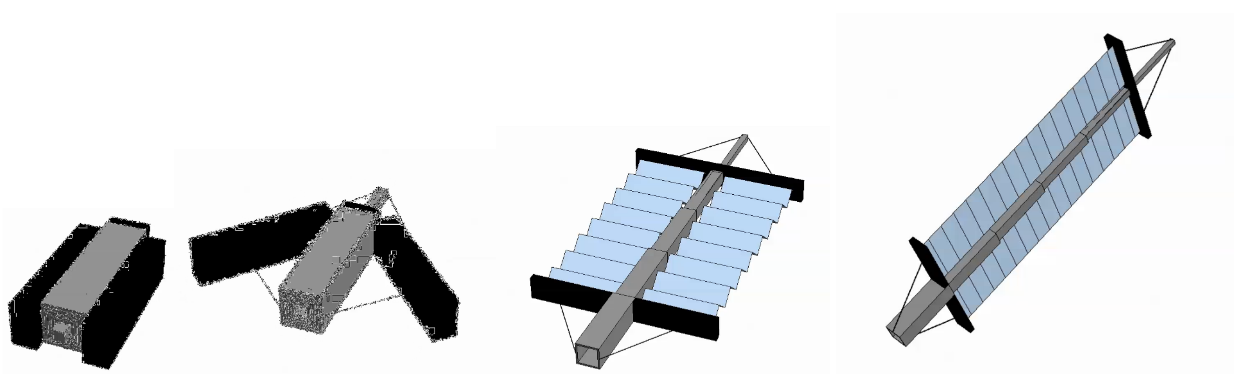


Figure 43: Solar panel deployment

In the current configuration, the deployed arrays extend out from the main body of the spacecraft in such a way that particles from the electrospray micro-thrusters will not impact them. Given the size constraints of the LauncherOne payload fairing versus the originally baselined Electron vehicle and the current geometry of the spacecraft design these arrays could also be widened to reduce the length needed to achieve the required area.

Notes on Solar Array Design to Minimize Spacecraft Vibrations: There are a couple sources of vibrations about the spacecraft. One is the vibration of the solar arrays on the MOM. These vibrations include in-plane and out-of-plane vibrations as well as torsional vibrations along the supporting structural element. All of these vibrations should have a fundamental frequency at or below 0.1 Hertz as to not disrupt the spacecraft's control system. To find the in and out of plane frequencies, use the equation below:

$$f_n \approx \left\{\frac{1}{2\pi}\right\}\left\{\frac{3.664}{L^2}\right\}\sqrt{\frac{EI}{\rho}}$$

From this equation, if we know the material properties (Young's modulus E and linear density p) and the frequency needs to be 0.1 Hertz, rearrange the equation to find the required area moment of inertia needed for the solar panel boom. The torsional frequency equation for a cantilever with uniform mass distribution and no end mass is as given:

$$f = \frac{1}{2\pi}\sqrt{\frac{\frac{N(b/2)^3}{3L} + \frac{GJ}{2L}}{\frac{mL(b/2)^3}{9} + \frac{I}{2}}}$$

This equation can be used to find the tension per unit width (N) if frequency is 0.1 Hertz with b as the array width, m as the blanket mass per area and I is the mass moment of inertia for the cross bar at the end of blanket. Another vibration is the blanket frequency, but as long as the frequency of the blanket is greater than the fundamental frequency of the structural element, the solar array analysis is complete. Blanket frequency can be simplified to a string vibration equation where, T is the tension on the blanket and μ is the linear density of the blanket:

$$f = \frac{1}{2L}\sqrt{\frac{T}{\mu}}$$

Battery Array and Power Distribution and Control System (PDCS): Standard CubeSat electrical power systems are not designed for the higher power demands required on the PHLOTE mission. The largest such commercially available system that could be found, the Clyde Space MicroSat power bundle, based on the company's standard CubeSat EPS and battery stores only 150 W·hr and can handle a throughput of just 100 W. The craft is expected to require a maximum battery capacity of about 500 W and then may have to handle upwards of 1 kW throughput. The battery capacity of the MicroSat power bundle could be scaled up by connecting additional battery modules in parallel. However, such a large array has not been constructed using these modules before, and so no thermal analysis has been done on how appropriate this would be for these power levels. It might also be possible to scale up the EPS to handle the necessary throughput by connecting multiple units, but a custom harnessing solution would be needed as each of the pins on the standard header connector are only rated for up to 4.5 A.

A custom power system also produced by ClydeSpace designed for SmallSats would require far less modification and likely be more appropriate for operation at such high-power levels. This system is designed around a 28V battery bus, with the EPS comprised of a number of 150W solar array regulator modules and the battery comprised of 84Wh (8-Series, 2-Parallel) modules connected in parallel. Thus, the system is also scalable should the demands of our design change, but such modifications will be straightforward.

Like all COTS CubeSat components, both of these systems are designed for a LEO environment. They will need additional qualification (and likely radiation shielding) to meet the PHLOTE mission requirements. Additionally, these systems are normally housed in an aluminium casing that provides more protection from radiation.

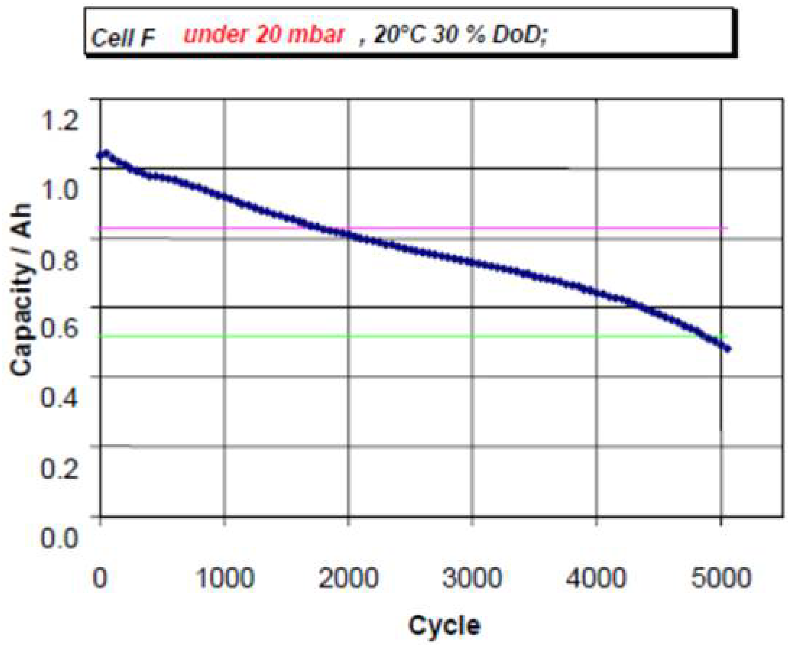
It may be more effective to use a power system designed for larger SmallSats, rather than connecting numerous small modules for CubeSats. The Mini PCDU by AAC Microtec is modular in design and could be scaled depending on power needs. It provides a 28V bus and a peak system power of 900W. It has less overall system mass than the modular SmallSat bundle from ClydeSpace. It does not come with a designated battery, so a suitable battery would need to be selected. A potential battery is the LSE134 from GS Yuasa. These batteries have a very high energy density in comparison to the other power system trades. Less mass is required to provide more energy when compared to the other trades. Two batteries connected could provide over 1000Wh. Both components are manufactured for long lifetimes and reliability. The main concern is the interface between the two systems. Other trades are able to bundle the systems together because they are made by the same manufacturer. A custom interface may need to be made for these batteries to be compatible with the Mini PCDU.

A design trade table provided by ClydeSpace for the three systems based on a slightly older, smaller power estimate is shown below.

Type	Bus Voltages	Item	Unit Dimensions & Mass	Unit Power Capability / Battery Capacity	Estimated Number of Units Required	Estimated Total Dimensions & Mass	Estimated Total Power Capability / Battery Capacity	Modification Required
High-Capacity Microsat Power Bundle	Main: 12.4V Battery Bus (Unregulated) Secondary: 3.3V regulated	EPS	90.17 x 95.89 x 20.82mm 150g	100W	8	190 x 200 x 45mm <1.5kg	800W	Substantial, including power architecture changes
		Battery	90.17 x 95.89 x 21.55mm 270g	30Wh	20	190 x 200 x 130mm <6.5kg	600Wh	Moderate - substantial
Modular SmallSat Power System	Main: 28V Battery Bus (Unregulated) Secondary: customized as required	PCDU		150W	6	240 x 85 x 260mm <7.5kg	900W	Minor, no architecture changes beyond mission-specific configuration of power distribution switches
		Battery		84Wh	8	266 x 185.5 x 296mm <9.5kg	672Wh	Minor
SmallSat Larger Power System	Main:28V	PDCU	340 x 170 x 222 mm 5.9 kg	900W	1	340 x 170 x 222 mm 5.9 kg	900W	Interface Design
		Battery	130 x 50 x 271 mm 3.53 kg	584Wh	2	260 x 100 x 271 mm 7.08 kg	1096Wh	Interface Design

Table 14: Information on COTS batteries

No lifetime information is available for the battery cells currently used in the Modular SmallSat system as they have recently been updated by the manufacturers. However, the ClydeSpace lot acceptance testing has shown similar performance to the older model. A cycling life-test graph for these previous cells, also supplied by the company, is shown below. The GS Yuasa testing is also shown below, for a slightly smaller model of the LSE battery. This information can be used to help size the batteries so they can meet the PHLOTE power requirements at the end of the mission.



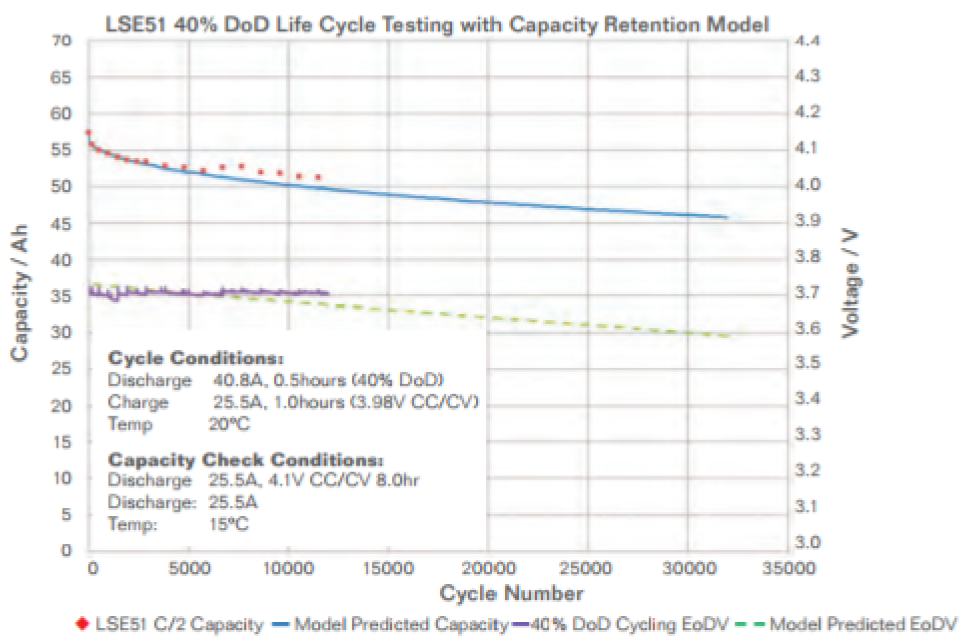


Figure 44: Battery capacity after cycling for ClydeSpace Batteries and GS Yuasa Batteries respectively

Since the PHLOTE spacecraft will be in eclipse by Mars and Phobos during each orbit during most of a Martian year it is expected there will be roughly (668 sols/year * ~3 orbits/sol * 2 eclipse/orbit) 4000 eclipse cycles during a nominal mission. Given the figure above we can (very conservatively) expect about a 40% loss in battery capacity.

POP and Landing Module Power Systems: The POP will not be supplied power from the MOM through the tether, therefore the POP needs to have a self-sustaining power system. The sensor platform needs an average of 20W to operate. Due to eclipsing, the solar panels need to generate 47W of power while in daylight. With an assumed efficiency of 30%, it is estimated that 30 solar cells are needed to sufficiently power the POP. The platform itself does not have enough surface area to accommodate all the solar cells it will need. We are able to put a few solar cells on the Pop integrated onto the landing legs, but cold gas thrusters and sensors will occupy the other sides of the POP. Placing solar cells on the top or bottom of the structure would not be viable due to the rarity of these surfaces getting direct sunlight. Additionally, the POP will be in eclipse more often than the MOM due to its closer proximity to the surface of Phobos. A proposed solution would be to integrate a collapsible solar array suspended along a length of the tether. To increase the amount of sunlight that the solar array would receive, it would be suspended a few tens of meters above the POP. That length of tether between the solar panels and the POP would need to be conductive. In the unlikely event that the relatively short conductive path should be cut by a micrometeoroid, the solar panels on the POP itself may be able to keep the platform running under minimal power requirements from base functions.

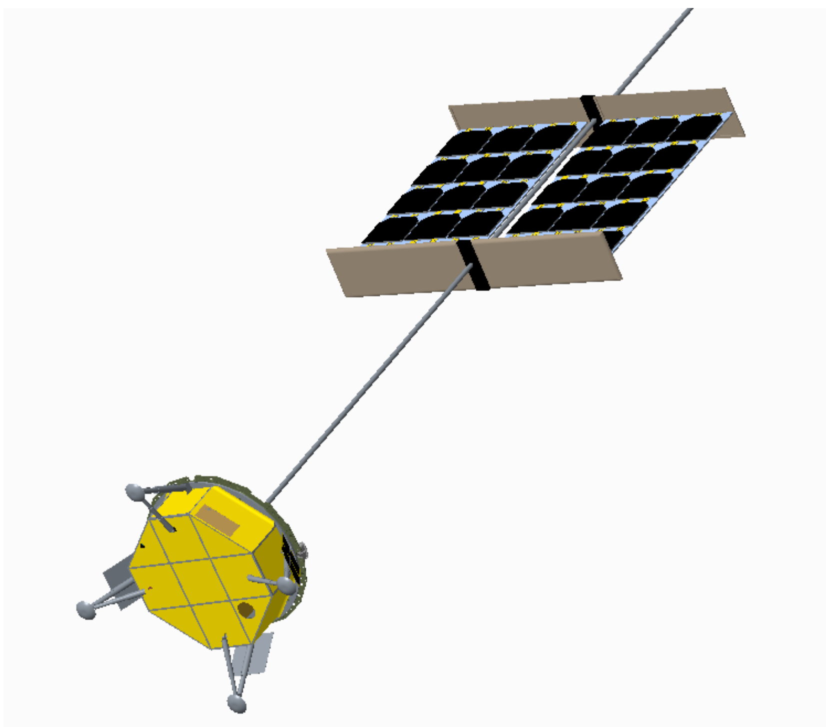


Figure 45: Solar Array Concept to provide power for the POP (separation reduced for illustration)

Communications:

PHLOTE will utilize the Deep Space Network (DSN) for its primary form of communication to Earth. The 34 m Beam Waveguide antenna services that are described in the DSN Telecommunications Link Design Handbook (See: https://deepspace.jpl.nasa.gov/dsndocs/810-005/introinfo.cfm?force_external=0) and the DSN Services Catalog are used as a baseline for PHLOTE's downlink and uplink telecommunication design. The values are documented in Figure x below. X-band communication is typical for deep space communication and will be the primary relay used on the PHLOTE spacecraft.

Station	Parameter	Unit	Value	Remarks
34-m BWG	Uplink gain	dBi	56.3	S-band
			67.1	X-band
			79.5	Ka-band
70-m	Uplink gain	dBi	63.0	S-band
			73.2	X-band
34-m BWG	Uplink 3 dB beamwidth	deg	0.263	S-band, DSS-24
			0.077	X-band
			0.016	Ka-band, DSS-25
70-m	Uplink 3 dB beamwidth	deg	0.128	S-band
			0.038	X-band
34-m BWG	Downlink gain	dBi	56.84	S-band DSS-24
			68.2 38.0	X-band main Ac aid
			78.9	Ka-band, DSS-25
70-m	Downlink gain	dBi	63.6	S-band
			74.6	X-band
34-m BWG	Downlink 3 dB beamwidth	deg	0.242	S-band DSS-24
			0.066 2.1	X-band main Ac aid
			0.017	Ka-band, DSS-25
70-m	Downlink 3 dB beamwidth	deg	0.118	S-band

Table 15: Information on DSN Communication Capabilities

The Iris V2 CubeSat Deep Space Transponder developed at NASA's JPL offers a DSN compatible transponder with low mass and power characteristics. This transponder can operate in the x-band, UHF, and s-band. The Tendeg deployable antenna that is baselined for the PHLOTE HGA is listed as a recommended compatible antenna for this transponder. The modulation type is binary phase shift key (BPSK) which is more resistant to bit errors than other forms of modulation. This is common for DSN services. Communications between PHLOTE and Earth will follow standard communication formats such as those developed by the Consultative Committee for Space Data Systems (CCSDS).

To directly communicate with Earth, PHLOTE would not be able to use the X-band patch antenna if the baselined Tendeg deployable antenna is no longer applicable. Preliminary calculations with no noise loss places the maximum distance that PHLOTE would be able to downlink directly to the DSN with the patch antenna at 109 million km. The distance between Earth and Mars can be as far as 401 million km. With the current configuration selected of the Iris V2 and an X –band modified deployable HGA, it can be predicted that the maximum distance for downlink with the DSN is over 2 billion km. The maximum uplink for DSN to PHLOTE is over 14 billion km. This leaves a very wide margin for considering signal noise and line loss.

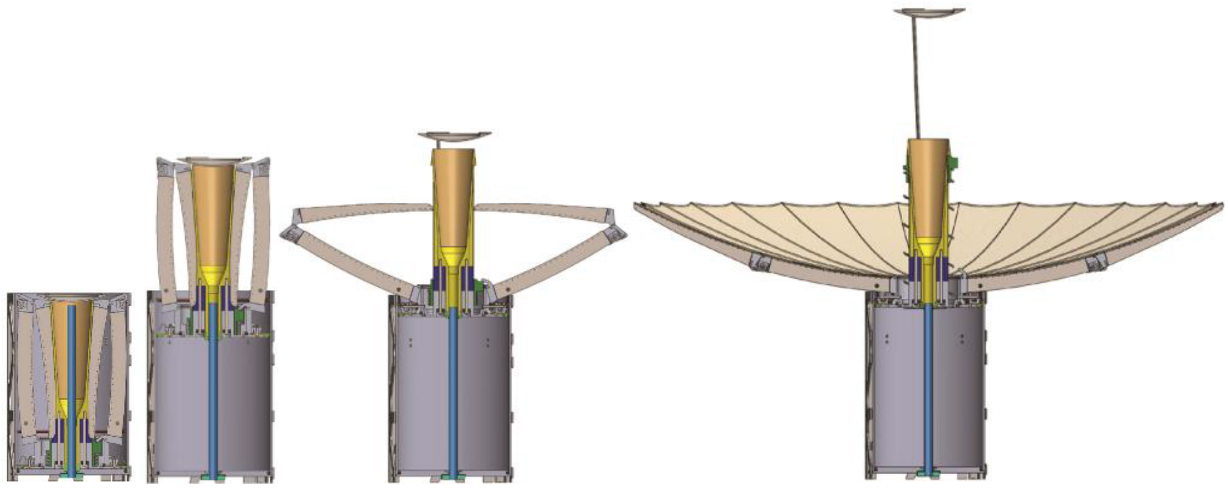


Figure 46: TENDIG High Gain Antenna Packaging and Deployment

For initial x-band communications with Earth, the pointing accuracy and high gain are not required. The spacecraft will have to adjust its attitude and reduce transmit power to keep signal strength below the DSN threshold for the selected antenna. A typical low gain patch antenna with a gain of 5 dBi can downlink with the DSN up to 29 million km from Earth and uplink up to 190 million km disregarding noise loss.

For the capacity of direct communications with DSN, the needed transmit power can be estimated by the equation below:

$$10\log\left(\frac{Pt}{Pr}\right) = 32.44 + 20\log(d) + 20\log(f) - Gt - Gr$$

Where Pt is power transmitted, Pr is power received, or sensitivity. d is distance between the two systems in km, f is frequency in MHz, Gt is transmitting antenna gain, Gr is receiving antenna gain. Both are in dBi.

An Iris V2 patch antenna can also be used for communication between the MOM and POP. Using the transponder for both Earth communications and POP communications would save power consumption and mass of having another transceiver on the MOM. The s-band receiver for the POP should be compatible with the Iris V2 transponder. The S-band transceiver for the POP has a data transmission rate of up to 2 Mbps, which fulfills the current estimate shown in Table 16. A lower data rate is predicted from the Iris V2 transponder, but the uplink to the POP does not require as high of a data transmission rate. The distance traveled without significant noise and losses is over 3000 km, much larger than the needed 6 km.

Using an STK simulation to predict PHLOTE's direct line of access to Earth; the shortest access time that PHLOTE encounters is 403 minutes. The longest direct line of sight access time PHLOTE has to Earth is 91 days. A simulation to predict access to a relay satellite with similar orbital properties of the proposed Mars 2022 Orbiter was run. To a relay satellite with an inclination of 90 degrees and an orbital altitude of 350 km above Mars, PHLOTE has a minimum line of sight access time of 23 minutes but a maximum access time of 121 minutes. PHLOTE has a shorter access time to the relay satellite, but the eclipse times are shorter than the eclipse times of Earth. Assuming a large enough data transmission rate, either option will allow for PHLOTE to adequately transmit data as well as receive data from a ground station. The desired frequency of communication with PHLOTE will determine the best option.

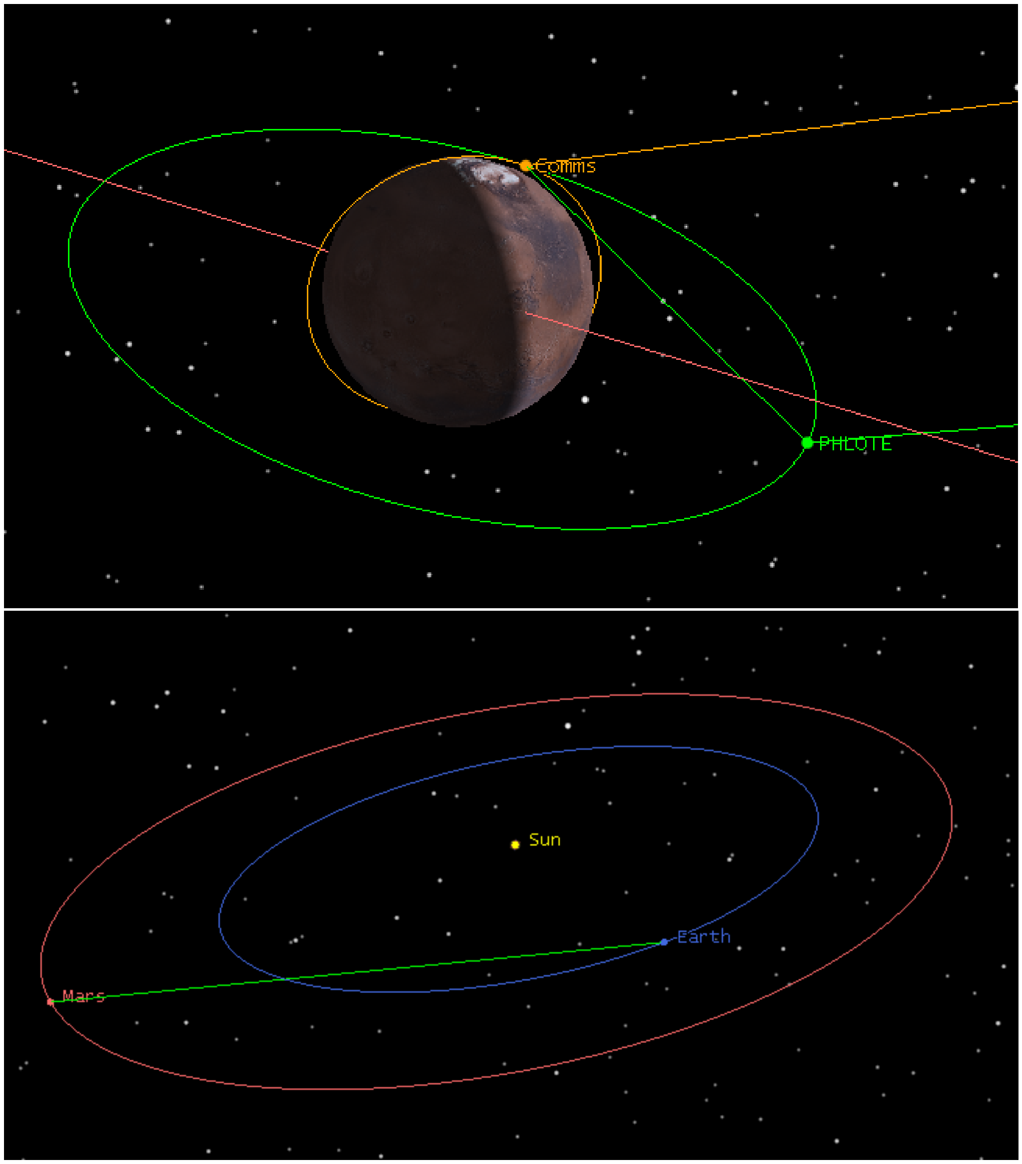


Figure 47: PHLOTE Communication Paths

PHLOTE DATA RATES					
MOM (Mars Observational Monitor)					
Description	Sensor Count	Sampling Rate	Measurements	Total Bit Rate	Notes
Accelerometer	1	100	192	19200	Low sample rate for housekeeping
Gyroscope	1	100	192	19200	Low sample rate for housekeeping
SCS Gecko Imager	2	0.016666667	52800000	1760000	Sample once every minute with 2.2 Megapixel pic
MOM Totals	4	200.0166667	52800384	1798400	

POP (Phobos Observational Platform)					
Description	Sensor Count	Sampling Rate	Measurements	Total Bit Rate	Notes
Accelerometer	1	10000	192	1920000	Higher sampling rate to measure soil characteristics
CAM1U CubeSat Camera	2	0.000277778	9830400	5461.333333	Sample every hour with 640x480 pic
Dosimeter	1	0.016666667	119	1.983333333	Low sample rate for scientific data
Infrared Spectrometer	1	100	64	6400	Low sample rate for scientific data
Gamma Ray Spectrometer	1	100	64	6400	Low sample rate for scientific data
Ground Penetrating Radar	1	100	64	6400	Low sample rate for scientific data
Neutron Detector	1	100	64	6400	Low sample rate for scientific data
Dust Detector	1	100	64	6400	Low sample rate for scientific data
POP Totals	9	10500.01694	9831031	1957463.317	

Total:	3755863.317	bps

Table 16: Data Rates for Selected Instruments and Sensors

Mass distribution on Tether tension

Satellite Positioning and Sensor Mass: To determine how the MOM and POP will be positioned once they reach Phobos and are deployed and at equilibrium straddling the L1 point, it is necessary to understand how the distance between the sensor package and Phobos affects its weight constraints, how the mass of the sensor package affects the tension of the tether, and how the mass of the sensor package affects the position of the satellite above Phobos.

The effect that altitude from Phobos has on the weight of the sensor package was examined. For the calculations, the sensor package is assumed to have a mass of 10 kg and is tested from a range 2.5 km from the surface to 0 km using formulas for the non-spherical gravitational field of Phobos. Figure 24 shows that the force of gravity on the sensor package is not heavily influenced by altitude within the range of interest.

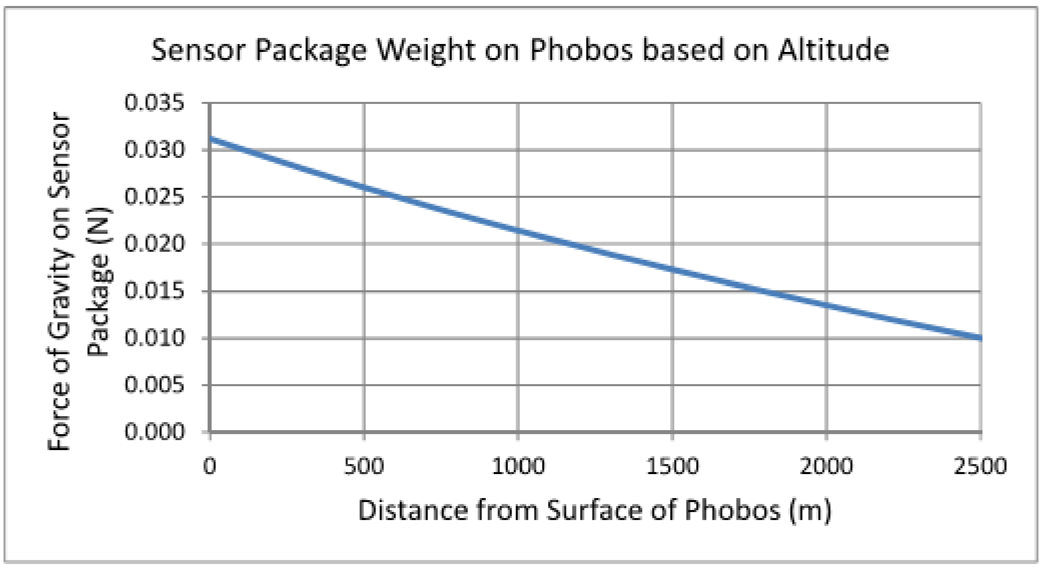


Figure 48: The Effect of Altitude on the Weight of the Sensor Package

For a fixed total mass of 120 kg, the effect that the POP's mass has on the tension applied to the tether was also examined. For the calculations, the sensor package is assumed to be 100 m above the surface of Phobos and was tested in a range of 2 kg to 60 kg using formulas for the non-spherical gravitational field of Phobos.

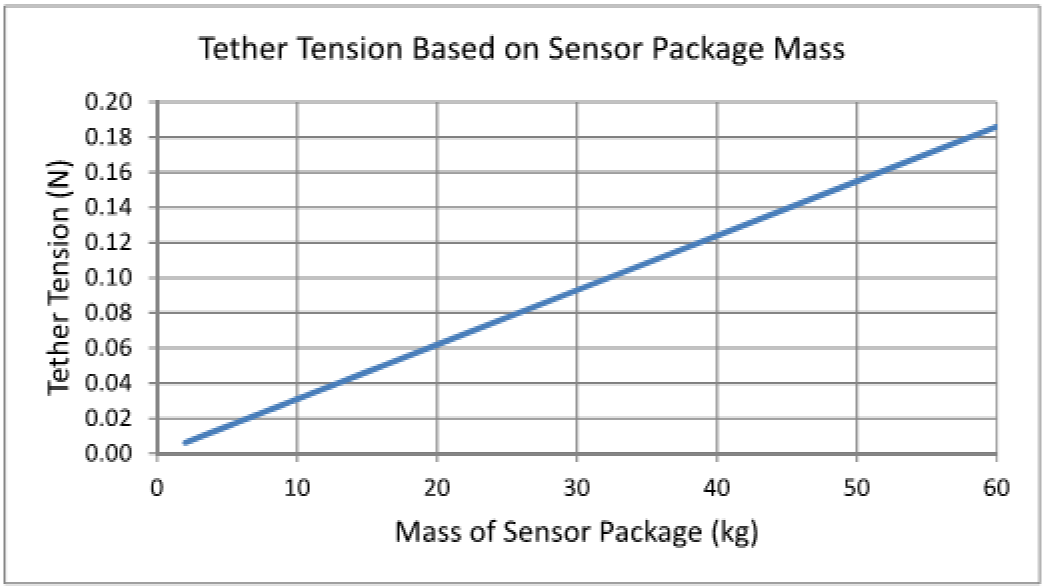


Figure 49: Effect of Sensor Package Mass on Tether Tension

The POP's mass determines the equilibrium distance of the satellite from the surface of Phobos. For the calculations, the L1 point is assumed to be 4.25 km above the surface of Phobos, and the sensor package is assumed to be 500 m above the surface of Phobos.

The mass of the tether was not included, and the mass of the POP was tested in a range of 2 kg to 30 kg assuming a total system mass of 120 kg. Figure 27 shows that the mass of the sensor package can go up to almost 30 kg before the satellite is required to move beyond the LIDAR's maximum operating distance of 6 km. The NDL's operating range can be increased with larger optics and/or higher laser power. This should be avoided if possible.

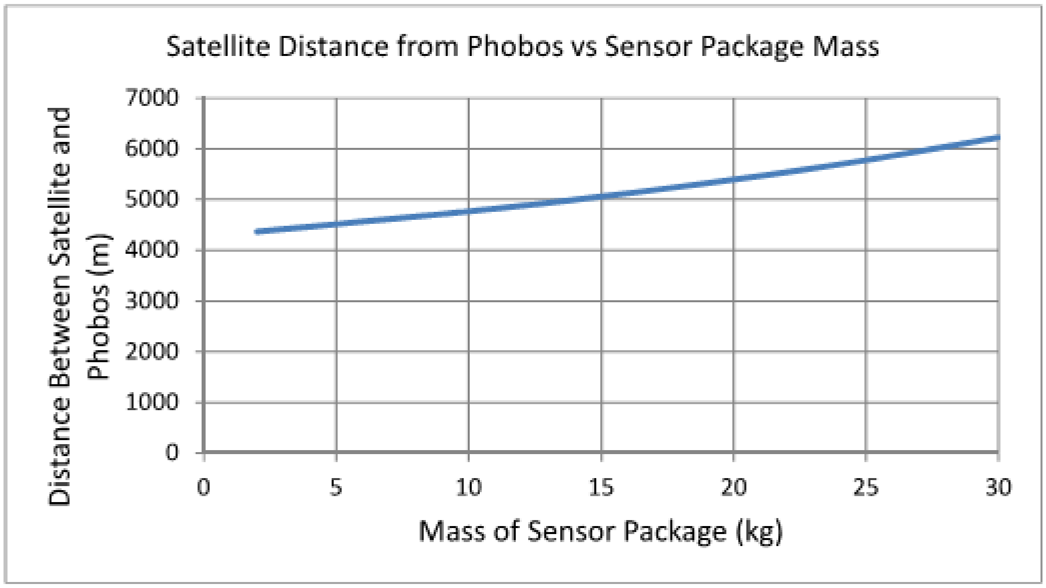


Figure 50: Satellite's Distance from Phobos Effected by Sensor Mass

Conclusions

The PHLOTE Phase 1 study has investigated the key technical challenges to implementing a PHLOTE mission at the Mars-Phobos L1 location. Based on the initial study it appears that the innovative method of using tether length control can help manage the periodic motion of the L1. In addition it appears long term station keeping is possible using the NDL and electrospray thusters without excessive propellant requirements. Simulations of PHLOTE at Phobos will continue to be refined in Phase 2.

The top technical challenge has now become the deployment of the sensor platform at the Mars-Phobos L1 point. Deploying tether systems in space is hard. Deploying a tether system while maintaining position with very limited control authority at an unstable orbital location all under autonomous control is really hard.

A credible PHLOTE spacecraft design has been developed as a foundation for future design iterations. A key focus of future studies will be the design of the tether system which will have some challenging requirements. It will need pretensioners due to the low tension. It will need to be reeled in and out almost continuously to enable tether length control to account for the motion of the L1 point. It will need sensors to help dampen out unwanted oscillations. The tether system design will be developed in Phase 2 once the optimal deployment method is selected.

There is a lot of interest in the NDL for precision landing. Specific work to tailor the NDL for a PHLOTE application must be done. This work will continue in Phase 2 with actual hardware testing of new components that will greatly reduce mass and power.

A PHLOTE mission has a great deal of science potential and feedback from the science, exploration, and technology development community has been very positive. There are many really exciting technology demonstration opportunities that can be done with a PHLOTE platform (i.e. using PHLOTE as the first operational space elevator). In Phase 2, the team will increase advocacy for a PHLOTE mission. This includes bringing on teammembers from JPL and also providing a generous number of internship opportunities to continue work on the PHLOTE mission design.

Technical Resource Estimates

The PHLOTE Master Equipment List (MEL) provides Mass and Power estimates based on selected components.

		Mass (kg)			Avg Power During Flight (W)			Avg Power During Tether Operations (W)		
		CBE (kg)	Contingency	CBE + cont	CBE (W)	Contingency	CBE + cont	CBE (W)	Contingency	CBE + cont
MOM	Attitude Determination & Control System	24.37	6.9%	26.04	341.71	17.12%	400.20	245.71	5%	258.00
	Command & Data Handling	0.64	10.0%	0.70	10.00	10.00%	11.00	10.00	10%	11.00
	Telemetry & Tracking Control	1.89	5.0%	1.99	6.75	5.00%	7.09	16.75	5%	17.59
	Electrical Power System	19.16	9.1%	20.89	0.00	0.00%	0.00	23.50	5%	24.68
	Thermal	1.54	5.0%	1.62	0.00	0.00%	0.00	0.00	0%	0.00
	Structures & Mechanisms	20.25	11.5%	22.56	22.00	5.00%	23.10	46.00	5%	48.30
	Sensors	0.98	5.0%	1.03	0.00	0.00%	0.00	5.00	5%	5.25
	MOM Total	68.82	8.7%	74.83	380.46	16.0%	441.38	346.96	5.1%	364.81
POP	Attitude Determination & Control System	1.89	5.0%	1.98	0.00	0.00%	0.00	0.99	5%	1.04
	Command & Data Handling	0.09	5.0%	0.10	0.00	0.00%	0.00	0.40	10%	0.44
	Telemetry & Tracking Control	0.37	5.0%	0.39	0.00	0.00%	0.00	16.75	5%	17.59
	Electrical Power System	1.31	5.0%	1.38	0.00	0.00%	0.00	0.00	0%	0.00
	Thermal	0.54	5.0%	0.57	0.00	0.00%	0.00	0.00	0%	0.05
	Structures & Mechanisms	1.50	5.0%	1.58	0.00	0.00%	0.00	0.00	0%	0.00
	Sensors	1.80	6.4%	1.92	0.00	0.00%	0.00	10.24	6%	10.90
	POP Total	7.51	5.3%	7.91	0.00	0.0%	0.00	28.38	5.8%	30.02
	Total	76.34	8%	82.75	380.46	16%	441.38	375.34	5%	394.83

Table 18: Power and Mass Estimates for Key PHLOTE Systems

PHLOTE Acronym List

ALHAT: Autonomous Landing Hazard Avoidance Technology

C&DH: Command and Data Handling

CG: Center of Gravity

CTE: Coefficient of Thermal Expansion

CNT:

ConOps: Concept of Operations Document

DSN: Deep Space Network

EDDE: Electro-Dynamic Debris Eliminator

EPS: Electrical Power System

FOV: Field of View

GTO: Geostationary Transfer Orbit

GEO: Geosynchronous Equatorial Orbit

HPSC: High Performance Space Computer

I&T: Integration and Test

I_{sp}: Specific Impulse

L1: Lagrange One

IHT: Iodine Hall Thruster

LCPG: Liquid Crystal Polarization Gratings

LEO: Low Earth Orbit

MOM: Mars Observation Monitor

NDL: Navigation Doppler Lidar

NIAC: NASA Innovative and Advanced Concepts

NMBS: Non-Mechanical Beam Steering

PDCS: Power Distribution and Control System

PHLOTE: Phobos Lagrange One Operational Tether Experiment

PMG: Plasma Motor Experiment

POP: Phobos Operational Platform

SEDS: Space Expendable tether Deployment System

SNR: Signal to Noise Ratio

SSO: Sun Synchronous orbit

STMD: NASA's Space Technology Mission Directorate

TEPCE: Tether Electrodynamics Propulsion CubeSat Experiment

TiPS: Tether Physics and Survivability Experiment