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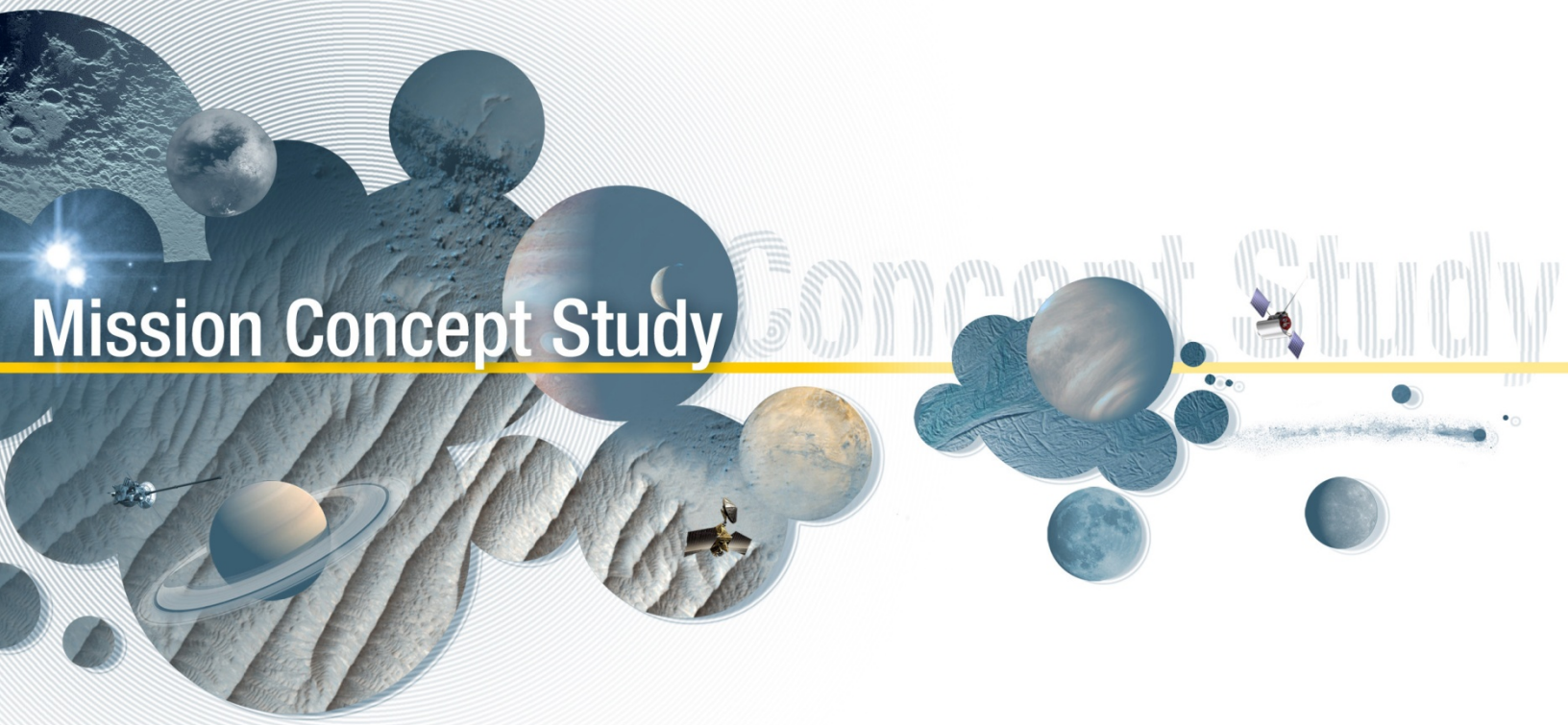

Mercury Lander Mission Concept Study

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Executive Summary

The purpose of this study was to determine the feasibility of a landed mission to Mercury. It was conducted by the JHU/APL Space Department in partnership with Marshall Space Flight Center, Glen Research Center, and Steven Hauck from Case Western Reserve University. This was conducted as a concept maturity level (CML) 3 study focusing on feasibility trades and options for concepts with a goal of determining whether such a mission could be accomplished within a Principal Investigator (PI)-led mission cost cap (See Appendix B for CML definitions). The mission focuses on fundamental science questions that can be best, or only, achieved by surface operations such as determining Mercury's bulk composition, the nature of the magnetic field, surface history, internal structure, and surface-solar wind interactions.

A straw-man instrument suite was selected focusing on a floor mission to try to determine how close the mission could come to the PI-led cost cap. Both a robust and reduced payload set that met the floor were included in the trade space. The complete set of instruments include a panoramic stereo image, a three-axis magnetometer, a miniature thermal emission spectrometer, a descent imager, an alpha proton X-ray spectrometer, a Raman spectrometer, a microscopic imager, and a communication system that supports radio science.

A landed mission to Mercury is extremely challenging from a launch energy and ΔV point of view. Reaching Mercury requires planetary flybys with either a low thrust or ballistic approach. In addition to the challenge of reaching Mercury from Earth, landing on Mercury requires on the order of 4.4–4.7 km/s of onboard ΔV depending on lander performance. No existing or planned deep-space missions have ΔV requirements even close to this magnitude. To address this primary challenge several concepts were explored to determine the trade space of feasible solutions.

To keep this mission close to a PI-led cost cap mission, it was assumed that the spacecraft would need to fit within an Atlas V 551 capability. A three-stage vehicle is likely the most mass-efficient approach to meet this requirement. In addition, the primary braking stage for landing would likely need to be a solid rocket motor to provide the best combination of Isp and propellant mass fraction.

Two trajectory approaches were explored. A ballistic/chemical approach was found to be potentially feasible, but current analysis puts it on the edge of being able to fit within the constraints of an Atlas V 551. Depending on a more optimized mission design, this approach may require a reduction in the instrumentation payload or mass margins.

A low thrust option was also explored using a solar electric propulsion (SEP) cruise stage. This option has the potential of offering more payload to the surface with an Atlas V 551 launch capability. The current concept fits within the Atlas V 541 with the full robust payload. The risk of this concept is its dependency on high-density, high-temperature solar cell technology that has yet to be developed beyond the cell level.

The cost estimates for all options exceeded the PI-led cost cap, including launch vehicle and reserves, of $900M (FY15$) as defined by the study ground rules. A ballistic/chemical option was estimated at $1.2B with a reduced payload, favorable trajectory performance assumptions, and an Atlas V 551 launch vehicle. The SEP option, which includes a robust science payload, is over $1.5B. The same robust payload utilizing the ballistic/chemical option and requiring the Delta IVH launch vehicle is also over $1.5B.

It is important to note that the ground rules required margins on mass, power, and cost that are well above normal practices used on previous APL space missions, making it very difficult to show feasibility within cost constraints. If APL practices of margin calculation were used for mass, most of the ballistic/chemical options would fit within the Atlas V 551 constraints. For these difficult, highly energetic missions, it may not be practical to assign high margins but, instead, a disciplined design to mass and power approach would have to be implemented from the start.

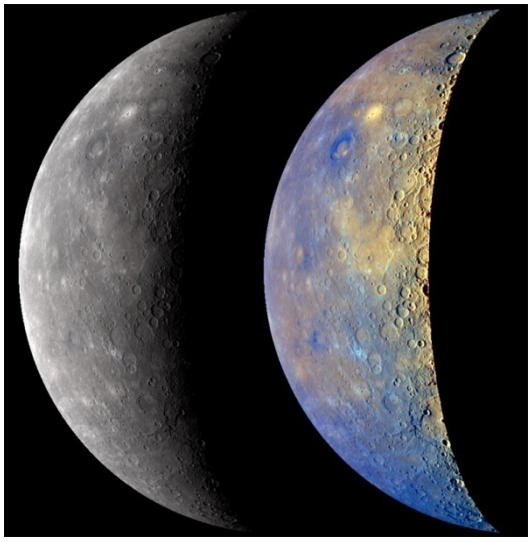
It is also recommended for these high-energy missions that NASA explore the possibility of qualifying higher energy configurations of the Atlas launch vehicle with the 4-m fairing (e.g., 451). This has the potential of greatly improving lift capability and enabling missions like this.

A more detailed study is required to further characterize this challenging mission. Both SEP and ballistic trajectory approaches and concepts should be further explored with a more detailed mission design and concept definition to determine the preferred mission implementation approach. Currently each has benefits and risks that could not be fully characterized at this high level of study.

1. Scientific Objectives

Science Questions and Objectives

Following the completion of NASA's MErcury Surface Space ENvironment GEochemistry and Ranging (MESSENGER) mission to Mercury, significant orbital remote sensing information regarding the planet's history, composition, interior, exosphere, and magnetosphere will be available. The European Space Agency BepiColombo mission that is currently in development is an orbital mission, as is MESSENGER, and will address similar scientific questions from the orbital point of view, though with more instruments on two separate spacecraft.



True and enhanced color views of Mercury from MESSENGER. (NASA/JHUAPL/CIW)

Several fundamental scientific questions about Mercury can be best, or only, achieved with surface operations. In order to properly understand Mercury's place among the terrestrial planets and the formation of planets in the inner solar system it is vital to determine the planet's bulk composition, internal structure, and geological history. Furthermore, Mercury presents a unique environment to understand the interactions between the solar wind, planetary surfaces, and surface-bounded atmospheres as mediated by an internal magnetic field. Following orbital reconnaissance of Mercury, a fundamental leap in understanding of terrestrial planets and their formation through investigations at Mercury is likely best served by surface-located and *in situ* science.

Science Questions

Mercury is the smallest and least explored planet in the inner solar system and will remain so even after MESSENGER's and BepiColombo's orbital reconnaissance of the planet. In addition to holding the place as the innermost planet, Mercury is dramatically unique: its large bulk density implies the largest metal-to-silicate ratio of any planet, it surprisingly hosts an internal magnetic field, and its surface records an intricate history of both the planet's and the solar system's formation and evolution. The scientific rationale for a landed mission to Mercury is based on five fundamental Science Questions aimed at understanding the formation of Mercury and its implications for terrestrial planet formation, evolution of Mercury's surface and interior, and the interaction of Mercury with its dominating neighbor – the Sun.

1. What is the bulk composition of Mercury?
2. What is the nature of Mercury's magnetic field?
3. What is the history of Mercury's surface?
4. What is the internal structure of Mercury?
5. What is the character of surface-solar wind interactions on Mercury?

Science Objectives

A series of five specific Science Objectives have been identified that address the five fundamental Science Questions regarding Mercury. The primary objectives are to constrain the composition and state of materials at the surface, determine the magnitude and variability of the magnetic field, characterize the history of the surface, and constrain the rotational state of the planet.

	Science Objective	Science Questions
A	What is the chemical composition of Mercury's surface (major and minor elements)?	1, 3
B	What is the mineralogy and structural state of the materials at Mercury's surface?	1, 3, 5
C	What is the magnitude and time dependence of Mercury's magnetic field, at least for a point on the surface?	2, 4, 5
D	What is the character of geological activity (e.g., volcanism, tectonism, impact cratering) at scales ranging from regional to local (i.e., lander environment).	3
E	What is the rotational state of Mercury?	4

The Science Objectives for the Mercury Lander mission concept are achieved through a series of measurements made by the scientific payload package. The instruments considered for this concept include the following:

- Panoramic stereo camera (Cam)
- Three-axis fluxgate magnetometer (Mag)
- Miniature thermal emission spectrometer (Mini-TES)
- Descent imager (DI)
- Alpha proton X-ray spectrometer (APXS)
- Raman spectrometer (Raman)
- Microscopic imager (MI)
- Communications subsystem (for radio science)

Science Traceability

Science Objective	Measurement	Instrument(s)	Functional Requirement
Chemical composition of surface (**A**)	Elemental measurement of surface materials	APXS	Surface contact and a calibration target
Mineralogy of surface (**B**)	Mineralogical measurement of surface materials	Raman, Mini-TES, MI	Raman: Clear view to, or near contact with surface Mini-TES: Near- and far-field view of surface, 360º MI: Near surface contact All: Context/targeting imagery
Character of magnetic field (**C**)	Strength and direction of magnetic field	Mag	Magnetically clean, and deployment away from, lander
Character of geological activity (**D**)	Imagery of surface	DI, Cam, Mini-TES	DI: Image collection during descent Cam/Min-TES: Near- and far-field view of surface, 360º
Rotational state (**E**)	Tracking of lander position with time	Communications subsystem	Communication with Earth

In order to fundamentally advance our understanding of Mercury and its place in the solar system a landed mission is vital. Understanding the bulk composition of Mercury (Question 1) will require measurement of both the elemental and mineralogical composition of the surface (Objectives A & B). An APXS will measure *in situ* the elemental composition for at least one point on the surface and the Raman

spectrometer will measure at least one representative mineralogic composition for a point on the surface. A Mini-TES will extend quantitative knowledge of surface materials beyond the reach of the lander. Internal magnetic fields are important indicators of the internal dynamics and structure of a body (Question 4) and also play a crucial role in modulating solar wind interactions with planetary atmospheres and surfaces. Mercury is the only body other than Earth in the inner solar system with a currently operating internal magnetic field; therefore understanding its nature (Question 2) and how it interacts with the solar wind (Question 5) is critical. Measurement of the strength and direction of the magnetic field at the surface as a function of time (Objective C) will substantively improve our understanding of Mercury's magnetic field, and understanding its interaction with the surface will be supplemented by surface characterization (Objective B). Limits on the internal structure of Mercury will come from both magnetic field measurement and constraints on the rotational state (Objective E) of the planet that come from tracking the lander's radio signals. Mercury's geological history is unique among the terrestrial planets (Question 3). Characterizing the geology (Objective D) at a landing site is critical knowledge for properly interpreting compositional information and placing it in the appropriate context.

Two payload cases are considered: (1) the Robust Lander, which contains the full complement of instruments and a robotic arm for placing the APXS, Raman, and MI near the surface and (2) a Reduced Lander, which removes the robotic arm, MI, and Mini-TES; the APXS is extended to the surface with no degrees of freedom and the Raman is operated in a remote, standoff configuration. The Reduced Lander can meet the basic objectives of the mission concept, though with lower fidelity and reduced data return. These technical implementations were chosen in order to understand the lower limit (Science Floor) of what could be achieved with a landed mission to Mercury.

2. High-Level Mission Concept

Concept Maturity Level

This study was conducted as a Concept Maturity Level (CML) 3 study (see Appendix B for Concept Maturity Level Definitions). It is intended to look at the trade space and potential feasible solutions against a floor-level mission. Is there a feasible concept that could be done with in a PI-led mission framework? If not, what would the cost be to do such a floor mission?

Mission Overview

The following constraints and assumptions about the mission were defined early to establish ground rules for conducting the study.

1. Launch in the 2018–2023 timeframe
2. Landing site constrained by lander thermal design, high latitude should be the first-order target
3. Ultimate precision landing is not required
4. Direct-to-Earth communications is the first-order desire; however, it is an open trade as to whether a relay spacecraft would be required
   - If a relay spacecraft is required, the science it might carry is limited, e.g., an additional magnetometer
5. Mission duration is modest; it is estimated that a two-week minimum duration could achieve the science goals.
6. Landing site should be sunlit at landing and initial operations, though a duration that eventually passes into night is scientifically acceptable and would benefit some experiments
7. Assume that radioisotope thermoelectric generators (RTGs) are available if they are necessary to enable the mission

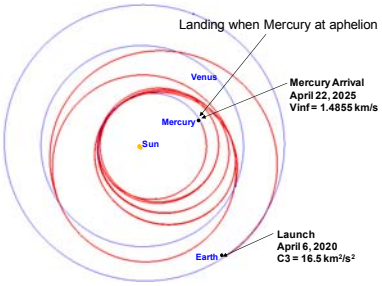
The figure below shows the resulting top-level mission concept developed for this study. Two trajectory options were explored and traded. The first is a ballistic option using all chemical propulsion with a cruise trajectory very similar to that of MESSENGER. The second was a low-thrust SEP trajectory using Earth, Venus, and Mercury flybys to reach Mercury. Both options result in about a 5-year cruise.

At arrival, a direct landing approach is assumed. This reduces mission complexity and allows an option to use a solid rocket motor for the primary descent braking. This is the same approach used by Surveyor on the Moon in the 1960s to maximize the payload mass to the surface, while minimizing total mass. All cruise hardware not necessary for landing is staged, including the heat shield and solar panels. A braking stage will remove the vast majority of the ΔV to land. The lander, itself, will separate and perform a soft landing using onboard propulsion.

Once landed, the lander will deploy a magnetometer boom and a robotic arm and begin science operations. Landing will occur approximately 2 days prior to sunset, allowing imagery with natural sunlight. Earth will be within view for 22 contiguous days in which to perform the primary science mission. An extended mission of additional 68 days may also be possible before the Sun rises high enough to end the mission. Earth will be in contact again 29 days before sunrise.

Ballistic Trajectory

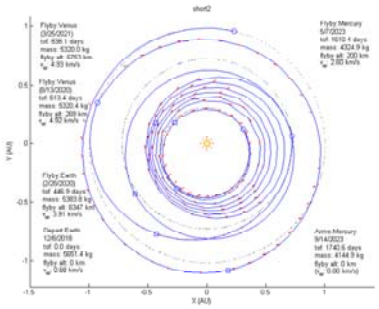
- Launch C_3 = 17.5
- 2 Venus + 4 Mercury Flybys
- 5 years



or

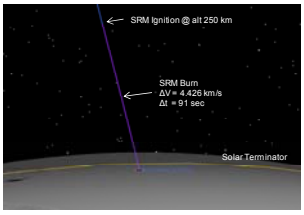
SEP Trajectory

- Launch C_3 = 0.46
- 1 Earth + 2 Venus + 1 Mercury Flybys
- 4.8 years



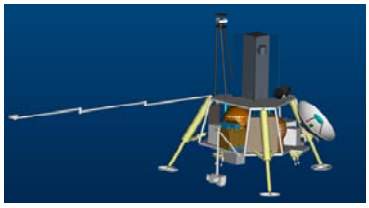
Direct Landing

- Solid Braking Stage
- Soft landing with Hazard Avoidance
- ΔV ~ 4,400 m/s



Landed Operations

- >22 days landed science
- Two days of limited daylight
- 100 Mbits daily science return



Mercury Mission Concept Overview

Several significant challenges drive a landed Mercury mission. The first and most significant challenge is the extremely high ΔV required to land on Mercury. Even if the arrival velocity (V∞) relative to Mercury is 0 km/s, it requires approximately 4.4 km/s to land on the surface. This value goes up significantly as the arrival velocity increases. For this study 4.4–4.7 km/s was required (for two different arrival velocities), which included a finite burn penalty for the propulsion system. Depending on the trajectory approach, significant additional ΔV is required for cruise: 1.4 km/s was budgeted for ballistic/chemical options and 12.4 km/s for a low-thrust SEP approach.

This high ΔV drives the vehicle design considerably, especially in the way of propulsion. Unlike planets with an atmosphere such as Mars, landing on Mercury will require the propulsion system to remove all the velocity. This challenge has been addressed by three stage vehicle, minimizing mass especially in the lander stage, and selecting efficient propulsion systems with high thrust to weight for landing. This mission basically requires the launch of a launch vehicle within a launch vehicle. The long cruise, on the order of 5 years, will also drive qualification of space storage for many propulsion components.

Another driver will be on the guidance, navigation, and control (GN&C) system. Mercury is airless and has more than twice the gravity of the Moon, requiring a responsive GN&C system to land safely. Since the surface is not well characterized and will not be at a high degree of resolution, which is necessary for landing, the vehicle will need to be able to detect and avoid possible hazards not detected with imagery provided by reconnaissance missions.

To address these GN&C challenges, the concept uses a solid rocket motor for braking and liquid propulsion for the lander developed by the Department of Defense. This liquid propulsion system offers fast response time and the high thrust to weight necessary to land in the Mercury environment. The concept includes redundant cameras, and LIDAR for descent, as well as a dedicated co-processor for hazard detection and avoidance computation.

As with MESSENGER, the thermal environment will be a significant challenge. The spacecraft will need to manage multiple flybys of Mercury prior to landing. In addition, to get the science, it is strongly desired to land prior to sunset. Incident sunlight and surface temperatures will challenge the design.

A thermal protection system similar to MESSENGER was assumed to protect the entire stack during cruise. The landing will occur a couple of days before sunset and at high latitudes where the surface temperature will be at manageable levels using an MLI-protected spacecraft. At this level of analysis, it is believed that the direct sunlight can be managed by high-temperature MLI.

The final major mission challenge will be the ability to operate at low power. Most of this mission will be conducted at night, requiring an Advanced Stirling Radioisotope Generator (ASRG). The high ΔV severely limits the mass that can be brought to the surface limiting the solution to be a single ASRG.

Power constraints are addressed by minimizing heater power required on the surface by sealing the lander in MLI to effectively use electronics waste heat. Louvers will be used to maintain the spacecraft core temperature by balancing electronics waste heat and incident solar energy. The coprocessor for landing will be turned off and the main processor clocked down to reduce power. A high-efficiency K-band communications system will be required and will only transmit when instruments are off. Instrument measurements will be performed serially to further reduce power consumption.

Key Trades

The table below summarizes the trades performed in this study. Each of the trades is discussed in more detail in the following subsections along with additional performance sensitivity analysis that was important in conducting the trades.

Summary of Trades Performed

Mission Area	Options	Results
Landed Power Source	• Solar • ASRG	• Mission only feasible in low Sun or no Sun environment • Eternal points of light near poles theoretically possible but not reliable for practical mission concept
Landing Approach (Final Descent)	• Precision Navigation • Hazard Avoidance • Basic landing	• Precision navigation not required. Not adequate resolution of Mercury surface to pinpoint safe landing area a priori • Some level of basic hazard avoidance deemed necessary
Landing Approach (Touchdown)	• Soft landing with propulsion • Air bags	• Airbags are mass prohibitive • Propulsion needed anyways to take out large ΔV
Landing Lighting	• Sunlit • Dark	• Initial thermal analysis indicates the possibility of landing up to three days prior to sunset. Surface temps and incident Sun should be manageable.
Landed Communications	• Direct to Earth • Relay spacecraft	• Direct to Earth communications possible for required mission duration • Relay spacecraft is mass prohibitive and not considered feasible
Landing Location	• Many possibilities	• Mid-latitude and high latitude options may be feasible
Staging	• 2 Stage • 3 Stage	• Three stages necessary to meet mass constraints
Cruise Stage Propulsion (Chemical Trajectory)	• Pump fed + pressure fed bi-propellant system • Pressure fed bi-propellant system	• Pressure fed system provides a more compact design • Pump fed minimizes propulsion mass and has higher Isp providing about 68 kg of system mass savings
Braking Propulsion	• Solid rocket motor • Pressure fed liquid • Pump fed liquid	• Pressure fed liquid system does not provide the Isp or mass performance to meet mission needs • A pump fed liquid system provides significant improvement but is still not competitive with a solid in performance. It also is physically large and this pump fed engine has only been tested at sea-level • Solid meets mission needs but would need to be qualified for long space mission storage
Trajectory Approach/Propulsion	• Ballistic with 2 Venus and 4 Mercury flybys • Low thrust (SEP) with 1 Earth, 2 Venus, and 1 Mercury flybys	• Both options may be possible with differing constraints and risks • Both were defined and costed

Landed Power Source

The availability of an ASRG for a Mercury lander is a critical, enabling system. Mercury's surface has a harsh thermal environment as a result of its proximity to the Sun, which scorches the surface during the daytime. Relatively benign conditions are found at high latitudes, near the terminator and at night. Indeed, approximately two days of twilight is the maximum amount of sunlight that was found to be tolerable during this trade study. However, two to three weeks of science operations are required to achieve minimum science objectives, obviating the use of solar power for a Mercury lander. Several crucial imaging measurements will be made during the twilight hours, but the remaining measurements will be made at night.

Though the thermal design requirements for the mission require operation at low Sun elevation and at night preclude the use of solar panels for power, "peaks of eternal light" (PELs) may exist via analogy with the Moon. However, several issues prevent such PELs from being operationally tenable landing sites: (1) to date no PELs have been identified on Mercury; (2) current and future knowledge of surface topography at the scale of a lander at potential PELs, which will be at very high latitudes, from MESSENGER and BepiColombo will be insufficient to guarantee that even deployment of a solar panel mast could achieve solar power upon landing; and (3) PELs are also likely to be found in regions of rugged topography (e.g., crater rims) that are unfavorable in terms of risk for landing

Landing Approach (Terrain-Based Navigation and Hazard Avoidance)

Several approaches were considered for Mercury landing. The first was to determine whether a safe landing could be made by simply targeting a landing location based on imagery from MESSENGER with no additional terrain based navigation or hazard avoidance. This approach was deemed too risky since the imagery would not provide resolution adequate to distinguish potential landing hazards. It could only provide context of safer areas to land. Precision terrain based navigation was also considered but is not necessary for this mission since there are no precise landing requirements and no precise knowledge of a safe landing area. Inertial navigation should be adequate to land within 100 m of the target. The solution chosen was to include basic hazard avoidance using optical cameras for shadow detection and LIDAR to estimate local slope. This approach was assumed in the concept to best balance cost and mission risk for this level of study. This is an area that would need further definition in a more detailed study.

Landing Approach (Airbag vs. Soft Landing)

Another trade area that is often asked is why not use airbags to land on Mercury. The table below addresses the major factors in such a decision. Basically, since there is no atmosphere, propulsion is required to remove a significant amount of ΔV anyway. An airbag system would add significant mass on a severely mass-constrained system.

Airbag vs. Soft Landing Trade

Considerations	Soft Landing	Airbag Landing
Landing location	Can precisely control landing location	Less precision in landing location
Landing orientation	Lands in controlled orientation	Lands in any orientation & **requires reorientation of lander after impact**
Terrain sensitivity	Sensitive to the nature of the terrain	Sensitive to the nature of the terrain
Mass	Requires DACS engines	Requires DACS engines plus **airbag system mass**
Science	For geochemical payload, may contaminate landing site	No chemical contamination, but may end up in a terrain depression

Landing Lighting

The science strongly benefits from landing prior to sunset to take landing and context imagery. Therefore, this study took a first look at the feasibility of landing under lit conditions. The initial conclusions show that a landing prior to sunset may be possible if the landing occurs at high latitudes and within a few days prior

to sunset where the surface temperatures are in an acceptable range. For the purposes of this study 70°C was used as maximum surface temperature limit. Based on MESSENGER experience, it is believed that the lander could take the direct Mercury Sun with thermal protection that would not add significant mass to the lander (e.g., high-temperature MLI). This is a trade and risk area that would need to be examined much more closely in a more detailed study.

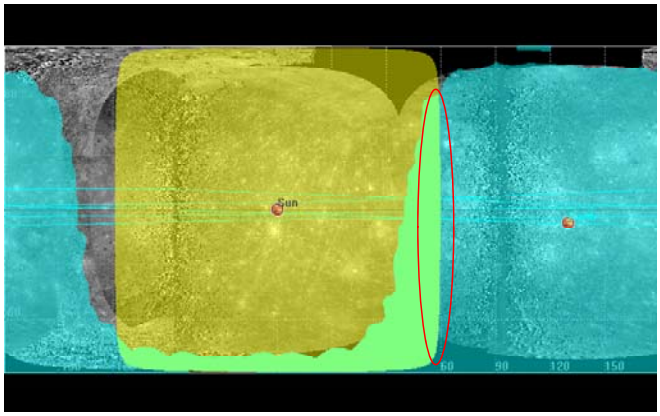
Lander Communications

The mission design developed for this study allows for direct to Earth communications making this trade easy to close. An orbiter is not required, would not provide any significant advantage, and would add significant system mass.

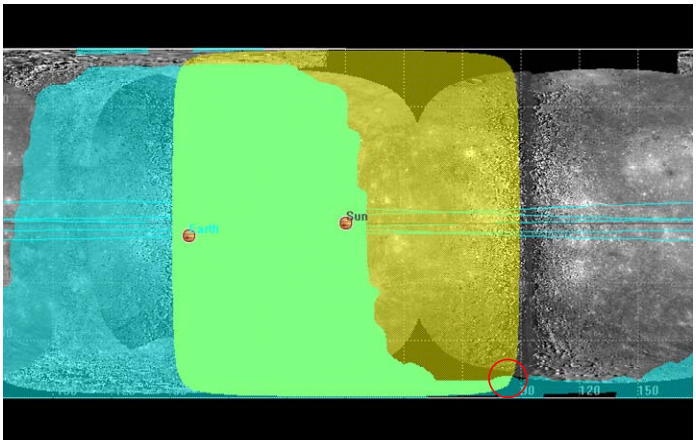
Landing Location

As a part of this study landing site opportunities were considered that met the lighting and Earth access requirements. The first figure below shows ideal cases that are seasonally available and allow a large range of latitude landing sites. The green indicates an overlap of Sun and Earth access. The circled area represents the approximate area where the 2 days of lighting prior to sunset constraint could be met. These opportunities come available every 3 to 4 months with a window of about 50 days each.

The next figure shows additional daily opportunities that are much more limited to the poles. The Earth communication constraint switches poles about every 180 days.



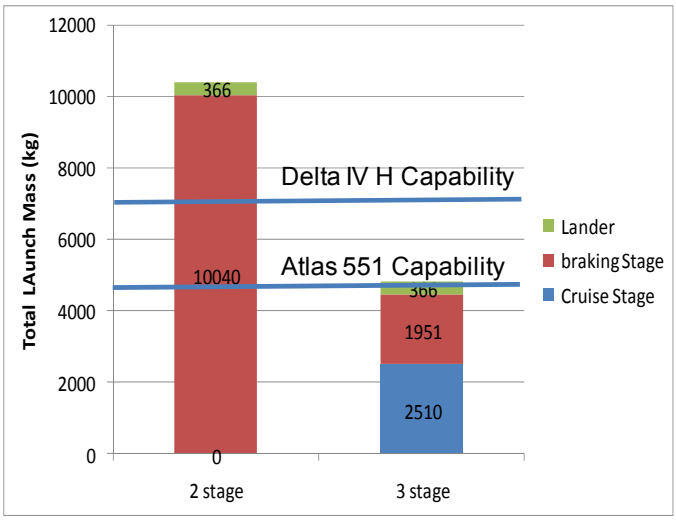
Seasonal Landing Location Opportunities



Daily Landing Opportunities

Number of Stages

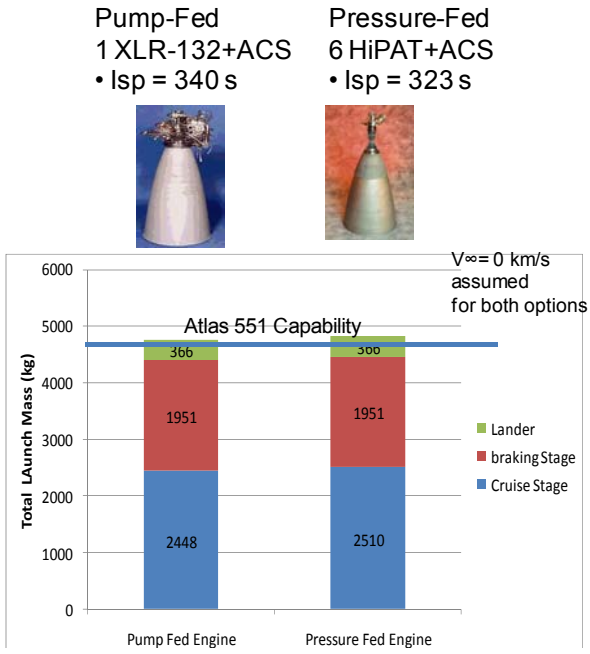
This trade was looked at purely for the ballistic chemical options. To achieve roughly 5.8-km/s total ΔV within a reasonable mass and volume, a multiple-stage spacecraft is required. A trade was performed between a two- and a three-stage vehicle. A two-stage vehicle would have the advantage of reducing the number of propulsion systems developed and reducing staging complexity if mass constraints could be met. The figure below compares the required launch mass of a two- and three-stage vehicle against the launch vehicle lift capability. Launch mass and margin is a primary metric of performance and will be shown in several of the trades. Both assume the same lander with the full robust instrument payload. For this trade, the two-stage vehicle is prohibitively heavy not only for the ATLAS V 551 but for a Delta IV Heavy as well. This even assumed staging of the large sunshield and solar panels prior to the braking burn. Even the three-stage vehicle used in this trade is slightly above the ATLAS V 551 capability with the full payload. Further optimization of the staging split may improve mass performance of both stages. However, based on these results, it is considered unlikely that a two-stage vehicle could fit within an ATLAS V 551 performance capability.



Comparison of a Two- and Three-Stage Concept

Chemical Cruise Propulsion Options

A trade was performed to compare the benefit of using a higher thrust pump-fed engine against using multiple lower thrust pressure-fed engines. For deep-space missions, pressure-fed engines have been the standard approach. However, use of a pump-fed engine has the potential to reduce mass by significantly reducing propulsion tank mass by operating at much lower pressure. In addition, pump-fed engines offer significant Isp advantages. The XLR-132 evaluated in this trade offers an Isp of 340 s against a traditional pressure fed engine operating at an Isp of 323 s. The disadvantage of the pump-fed engine is that it is much larger than the pressure-fed engines. In addition, a second pressure-fed bi-propellant system would be required for an attitude control system (ACS) and small maneuvers, adding complexity. The size of the pump-fed engine would also require a spacecraft structure and thermal sunshade, making the dry mass advantage almost negligible. As shown in the figure below, the pump-fed engine offered some mass advantage in using less propellant (about 68 kg). However, this was not adequate to get the concept within the ATLAS V 551 constraints and adds significant complexity and cost to the design. This option should be looked at more closely in more detailed studies but was not considered further in this study.

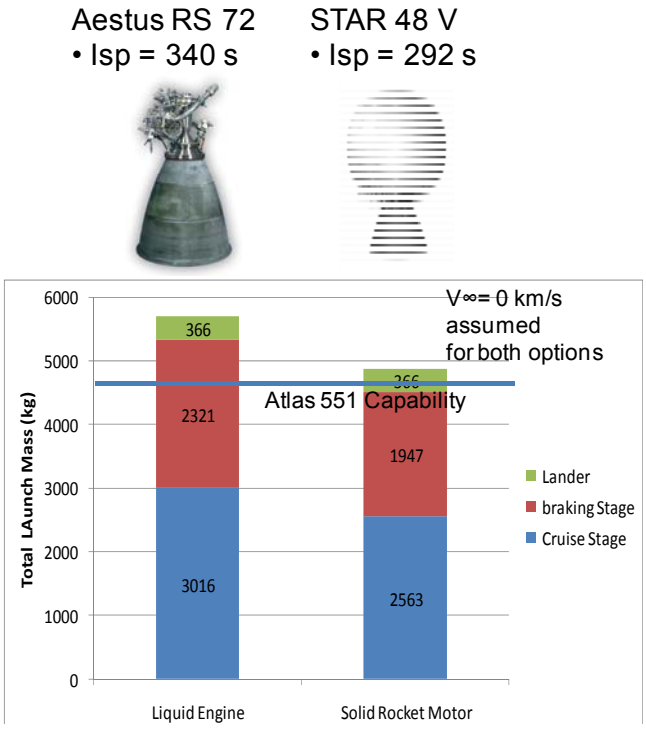


Comparison of Pump-fed and Pressure-fed Cruise Stage

Braking Stage Propulsion Trade

A trade was performed between using a solid and a liquid propulsion for the braking stage. The braking stage is responsible for removing the majority of ΔV for landing. To minimize finite burn penalty, options in the 50- to 70-kN range were explored. The liquid system evaluated is based on the Aestus RS 72 pump-fed bi-propellant engine. As with the cruise propulsion trade, a pump-fed system offers significant propulsion mass and Isp advantages over a pressure-fed system. This engine is designed for launch vehicle applications and would need to be qualified for deep-space missions.

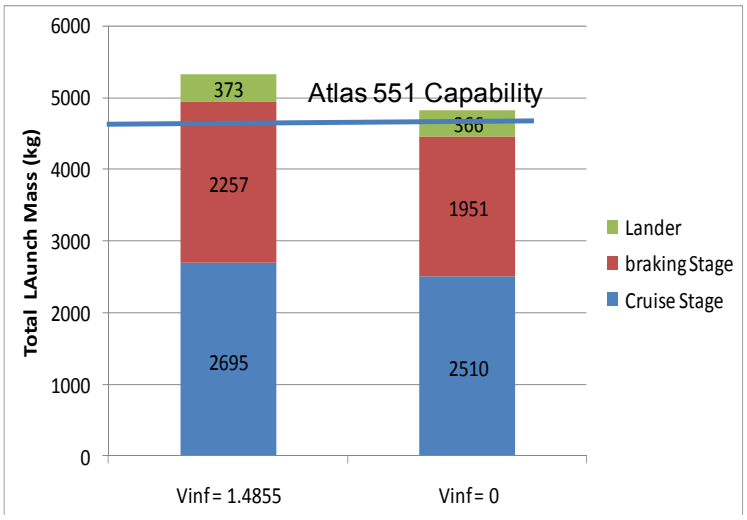
The solid is based on the STAR 48V. The thrust vector control version was considered advantageous for managing attitude during the braking burn without assistance of additional ACS engines or use of the lander engines. As shown in the figure below, the STAR 48V offers significant performance benefit over the liquid stage. A solid is difficult to compete with for this application since its Isp is competitive, it has a very high propellant mass fraction, and it is volumetrically compact. The issue with using a solid is the need to qualify it for an extended storage in space prior to firing. Magellan used a solid for Venus orbital insertion 15 months after launch.



Comparison of a Liquid and Solid Braking Stage

Arrival Excess Velocity (V∞) Sensitivity

The system performance sensitivity to Mercury arrival V∞ is another very important relationship in developing feasible Mercury lander concepts. A higher V∞ results in a higher ΔV required to land and, therefore, in a higher launch mass. For this analysis, two V∞ values were assumed. The first is a V∞ of 1.4855 km/s, resulting in a required ΔV of 4.71 km/s. This is the reference ballistic mission design developed for this study. The second is a V∞ of 0 km/s. This represents the SEP reference trajectory. An optimized ballistic trajectory may also approach a V∞ of 0. The figure below illustrates the impact on system launch mass. The case of V∞ of 0 km/s shows a reduction of 472 kg. Both concepts could be made to fit within the Atlas V 551 capability by reducing payload. The case of V∞ of 0 km/s will close within the ATLAS V 551 with removal of the microscopic imager and robotic arm from the instrument suite. The case of V∞ of 1.4855 km/s will close if the payload set is reduced to the defined minimum and the margin is reduced to 23%. The results of this trade show a significant value to lowering arrival V∞ as low as possible given other constraints such as cruise ΔV. In a future study, other trajectory optimization parameters could also be explored to improve performance, such as trajectories with reduced cruise ΔV or reduced required launch C_3 or a combination.



System Mass Sensitivity as a Function of Arrival V∞

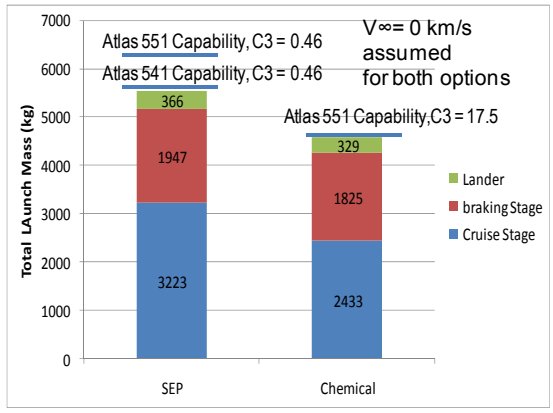
SEP vs. Ballistic Chemical Cruise Trades

Comparing SEP to a ballistic trajectory is one of the primary trades of this study. The figure below illustrates the comparison between the two options. Overall, the SEP option has the potential to offer the highest payload capability and fit within the ATLAS V 541 or 551 with good margins. However, the SEP option is more dependent on newer technology, specifically on development of high density solar arrays that can meet the high temperature requirements, potential first flight demonstration of NEXT engines, and a significant challenge in thermal management.

Based on this preliminary analysis, the ballistic/chemical concept developed does not perform as well as the SEP concept and needs to have the payload reduced to fit within the ATLAS V 551 with the required 30% margin. If the acceptable margin were reduced (~26%), it could potentially carry the full instrument payload set identified in the study. The ballistic option shown also minimizes new technology required by using proven pressure-fed propulsion systems and MESSENGER solar array technology. Cost also favors the ballistic approach.

It is recommended that both of these options be looked at in more detail in the future to further define the cost vs. risk vs. performance trades of these two approaches.

Metric	SEP	Chemical
Instrument Payload (CBE)	37 kg + margin Robust Payload	21 kg Remove robotic arm and micro imager Requires stand-off Raman
Launch C_3	0.46 km2/s2	17.5 km2/s2
Launch Mass (with Margin)	5460 kg	4587 kg
Launch Vehicle	Atlas V 541	Atlas V 551
Cruise Duration	4.8 yrs	5.0 yrs
Primary Risks	• Mass risk on Solar array • Stage heat dissipation	• How well trajectory can be optimized to reduce mass



A Technical Comparison of SEP and Ballistic Chemical Approaches

Mass Growth Sensitivity

Concept configurations were modified to evaluate the sensitivity of dry mass growth on the overall system mass. The results indicate a very high sensitivity of the lander dry mass to the overall system growth. A change of 1 kg of lander dry mass affects the system launch mass by 8 kg for the SEP cruise stage option and 11 kg for the chemical propulsion option. The sensitivity of 1-kg change on the cruise stage results in only 2.5 kg of change in system mass. Therefore, it is important for this mission to minimize the mass in the lander even at the expense of putting more mass in the cruise stage. One of the options discussed but not pursued was to duplicate much of the avionics in the cruise stage as redundant systems and carry only single-string components in the lander to reduce mass. This approach could have further reduced mass but would have added some risk to the lander system

A summary of the primary configuration options considered as part of the trades are shown in the table below. Most of the costing effort focused on Options 2, 5, and 6. Options 5 and 6 were the only options assuming the full 30% margin that could fit within an ATLAS V 551 and had any chance to fit within a PI mission cost cap.

Summary of Primary Concepts Considered in Trades

- All masses include 30% margin

Parameters	Option 1	Option 2	Option3	Option 4	Option 5	Option 6
Instrument Payload	Robust	Robust	Robust	Robust	Reduced	Robust
Trajectory/Prop	Ballistic	Ballistic	Ballistic	Ballistic	Ballistic	Low Thrust
V∞ at Mercury Arrival	1.486	0	0	0	0	0
Cruise Stage Propulsion	Press. Bi-prop	Press. Bi-prop	Press. Bi-prop	Pump Bi-prop	Press. Bi-prop	SEP
Braking Stage Propulsion	Solid	Solid	Pump Bi-prop	Solid	Solid	Solid
Instruments Mass (kg)	53	53	53	53	30	53+
Lander Dry Mass (kg)	331	323	323	323	289	323
Lander Wet Mass (kg)	373	366	366	366	329	366
Braking Stage Dry Mass (kg)	181	181	351	181	181	181
Braking Stage Wet Mass (kg)	2257	1951	2344	1951	1813	1951
Cruise Stage Dry Mass (kg)	726	725	793	722	704	1486
Cruise Stage Wet Mass (kg)	2695	2510	2858	2448	2373	3223
Launch Mass (kg)	5325	4827	5568	4765	4515	5540
Launch Vehicle	Delta IV H	Delta IV H	Delta IV H	Delta IV H	Atlas V 551	Atlas V 541
Launch C3 km^2/s^2	17.5	17.5	17.5	17.5	17.5	0.46
Launch Vehicle Capability (kg)	6915	6915	6915	6915	4630	5770

Recommended Future Trades/Analysis

There were several options not considered because of a lack of time for this study that should be considered in future studies. They are as follows:

- Continue to refine the trajectory estimates to improve performance including arrival V_∞, cruise ΔV, and C_3.
- Explore the viability of liquid propulsion for braking with SEP stage if long-duration space qualification of solid stage becomes an issue.
  - It is less efficient than the solid but may still work with an ATLAS V 551.
- Further explore optimization of ΔV split braking stage and lander.
- Analyze off nominal landing performance and hazard avoidance to develop a more refined estimate for lander ΔV and resulting propellant load.
  - A 10% margin was used for this study to cover these areas.
- Refine the thermal estimate for landed configuration with incident sunlight.
  - Can this be managed by high-temperature multi-layer insulation (MLI) alone?
  - Does this assumption put restrictions on aphelion vs. perihelion landing?
- Explore the feasibility of a qualified Atlas V 451 and its impact to the study

Technology Maturity

For a CML-3 study, technologies that were either enabling or possibly enhancing were identified. The table below summarizes the new technology areas addressed by the Mercury Study. Some of these may be specific to this mission, although most may apply to other missions as well. Each technology area is addressed in a little more detail in Section 3 in each of the subsystem sections. All other components not listed here are assumed to be Technology Readiness Level (TRL)-6 or above.

Technology Table

Technology	Need	TRL	Development Needed
Efficient Ka-band SSPA or TWTA	Mission is severely mass and power constrained. NASA encouraging K-band use.	3-7	Reduce mass and improve efficiency of existing technologies. Invest in low power TWTA and solid state amplifier technologies.
High speed graphics processing	Hazard avoidance algorithms using cameras and LIDARs need significant processing beyond normal space qualified processors.	4	Develop Co-processor based on high density FPGA technology. Demonstrate with algorithms.
Raman Spectrometer	Provide composition information considered necessary for mission success. Close contact version developed by Wash U and JPL would provide the best return, but requires a robotic arm.	3-4	Two options are being developed. Wash U/JPL has a space design prototype. University of Hawaii has done some field testing with non-space hardware. Both need to be advanced to TRL 6
Lightweight/Low Power LIDAR	LIDAR is needed for hazard avoidance, altimetry, and local slope calculations necessary to assure safe landing. Mission is very mass and power constrained	3-7	Different LIDARs are in various stages of development. Improvements need to be made to minimize power and mass for this type of application.
Solid rocket qualification for long space storage	The rocket motor itself is not new technology but it is not currently qualified for long duration missions	5	No technology development needed. Needs to be qualified for this type of mission.
Ultra-light high temperature solar arrays	An SEP concept depends on significant power and cannot afford in mass to use demonstrated MESSENGER SA technology	4	High temperature cells have been demonstrated. Array level design and demonstration needs to be performed.
Bi-Propellant pump fed engine for long term space use	May provide some mass savings if used in cruise stage. May be easier to approve than solid for long space storage.	5	Qualify engine for mission environment
DoD DACS for Lander Use	Provides high thrust to weight needed for mission	5	Qualify thrusters for mission environment

3. Technical Overview

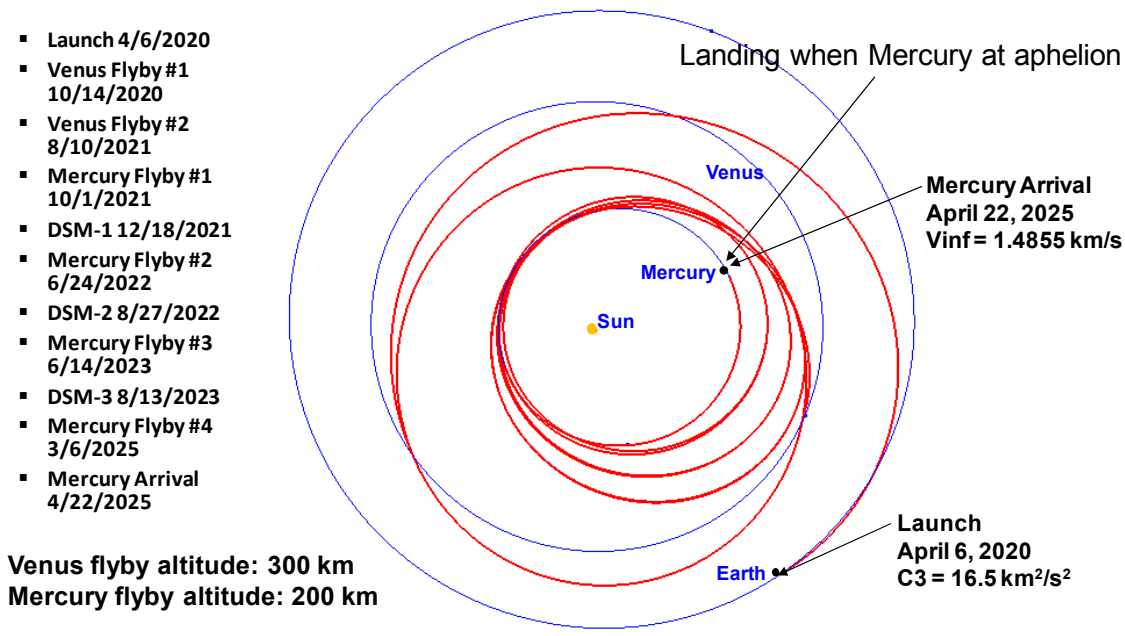
Concept of Operations and Mission Design

As summarized in Section 2, both a ballistic/chemical and SEP mission trajectory options were studied. The following assumptions were made for this study:

- All trajectory options assume direct landing on Mercury
  - Allows a solid braking stage
  - Not adequate time to explore orbital options
- Minimizing V∞ at Mercury arrival provides the lowest ΔV requirement for landing
  - V∞ = 0 km/s achieved for low thrust trajectory
  - V∞ = 0 km/s may be achievable for ballistic case with further optimized trajectory. ΔV= 1.4855 km/s achieved for reference case
  - Ballistic trajectory (reference case)
    - Two Venus + four Mercury flybys
    - 5.0 years
  - Low-thrust trajectory (reference case)
    - One Earth + two Venus + one Mercury flybys
    - 4.8 years

Ballistic Reference Trajectory

The launch date for this reference mission is April 2020, out of Cape Canaveral Air Force Station (CCAFS) in Florida. The nominal C_3 is 16.5 km^2/s^2 with an estimated maximum C_3 of 17.5 km^2/s^2 for a 20-day launch window. The trajectory includes two Venus and four Mercury flybys. Three deep-space maneuvers will be performed with a ΔV of 1238 m/s. The arrival V∞ = 1.4855 km/s. Arrival will be when Mercury is at aphelion, which was desired to lower heating rates on the spacecraft during the first two days of the landed mission when the Sun is not yet set. The total cruise duration is 5.0 years. Landed operations need to exceed 2 weeks to meet science requirements. The following figure illustrates the trajectory approach.



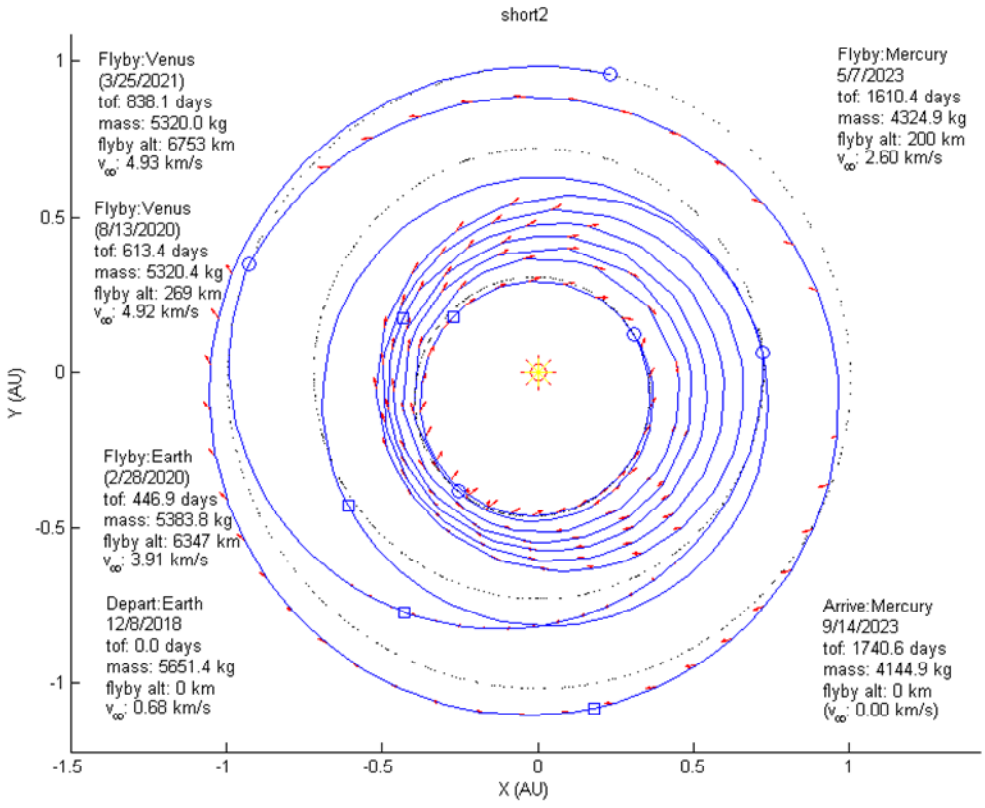
Ballistic/Chemical Reference Trajectory

Ballistic/Chemical ΔV Budget

Segment	V∞ = 0 km/s	V∞ = 1.4855 km/s
Cruise		
Deep Space Maneuvers	1238	1238
Navigation (Statistical)	124	124
Cruise Margin	38	38
Cruise Total	**1400**	**1400**
Landing	4420	4709
Total	**5820**	**6109**

Low-Thrust (SEP) Reference Trajectory

The SEP reference trajectory assumes an Earth departure in December 2018 launched from CCAFS. The C_3 for this trajectory is 0.46 km^2/s^2. The remaining cruise ΔV will be taken out by the SEP system. This trajectory consists of one Earth, two Venus, and four Mercury flybys. The cruise duration is 4.8 years. The low-thrust trajectory is illustrated in below.

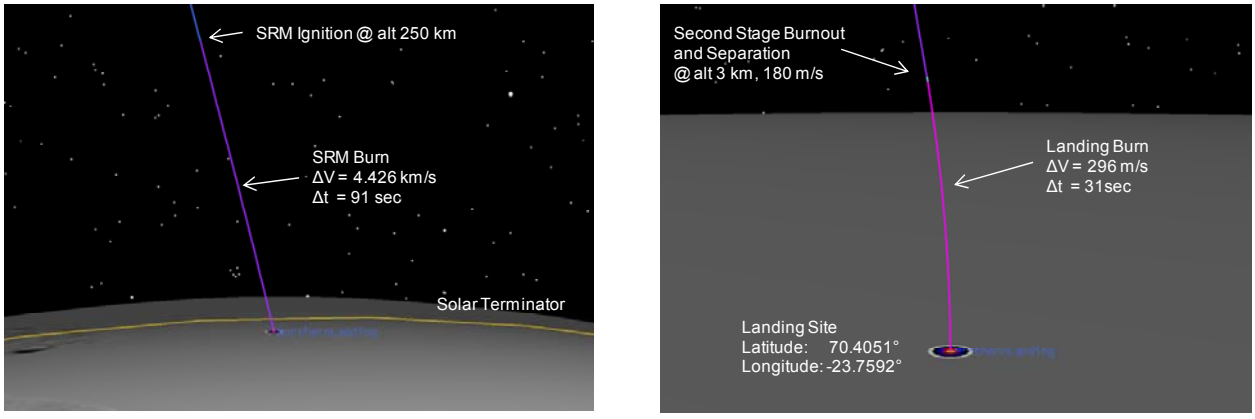


Four flybys (1 Earth, 2 Venus, 1 Mercury)
Total cruise duration: 4.8 years

SEP Reference Trajectory

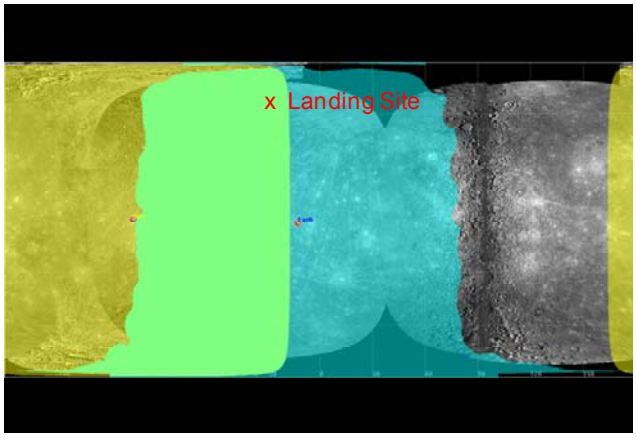
Landing Approach

The landing approach is similar for both trajectory options. Upon arrival, a direct landing on Mercury will be initiated. At this time, the cruise stage, including the Sun shield and solar array, will be separated. Primary braking for landing will be performed with a solid rocket motor with thrust vector control (TVC) following a gravity turn profile. After burnout, the solid rocket motor will separate and the final landing sequence will be performed by the lander's onboard liquid propulsion system. The lander will navigate and guide using optical sensors and lidar using techniques developed for ILN and ALHAT. The lander will touch down, avoiding hazards such as large rocks at a rate of about 1 m/s. The following figure illustrates the landing based on the ballistic reference trajectory. The SEP case would be similar with variations in initial velocity, flight path, and altitude for the start of the braking burn.



Mercury Landing Profile

The reference latitude and longitude for landing was 70.4 deg and –23.8 deg, respectively. This allows 2 days of limited sunlight prior to sunset. The Sun will return 88 days later. Earth contact will be possible for 22 contiguous days and will resume 29 days before the Sun rises again. The figure below shows the reference landing site.



Reference Trajectory Landing Site

Mission Design Table

Parameter	Value	Units
Orbit Parameters (apogee, perigee, inclination, etc.)	Landing	
Mission Lifetime	<6 years	mos
Maximum Eclipse Period	35 min Ballistic 31 hours SEP	min
Launch Site	CCAFS	

Concept of Operations

For the purposes of this study, it was assumed that the cruise portion of the mission would be managed by operations in a manner similar to MESSENGER. For normal cruise, there would be two 4-hour contacts per week. SEP will require a little more caretaking for propulsion and navigation operations. SEP engines will need to be shut down prior to flybys. All critical events, including trajectory correction maneuvers (TCMs), will be monitored in real time.

During landing, real-time communications will be managed by the medium-gain antenna (MGA) on the lander. Once initialized, the landing itself will be performed autonomously by the spacecraft. Because of Mercury's high gravity, there will be no opportunity to abort.

Landed operations assume 8 hours of contact per day using the Deep Space Network (DSN) 34-m dish. This will bring down an average of 100 Mbits of science data per day. The magnetometer will operate continuously throughout the mission. Other instruments will be operated serially starting with imagers while still in daylight. The primary mission ends after 22 days of surface operations. An extended mission is possible while the spacecraft is out of contact but was not costed as part of the mission.

Mission Operations and Ground Data Systems Table

Down link Information	Cruise	Landing	Landed
Number of Contacts per Week	2	Continuous	7
Number of Weeks for Mission Phase, weeks	260	260	260
Downlink Frequency Band, GHz	8.4	32	32
Telemetry Data Rate(s), kbps	0.720	0.016	3.5
Transmitting Antenna Type(s) and Gain(s), dBi	LGA	MGA	HGA
Transmitter peak RF power, watts	12	8	8
Total Daily Data Volume, (Mb/day)	NA	NA	100
Uplink Information			
Number of Uplinks per Day	2/week	1	1
Uplink Frequency Band, GHz	7.2	7.2	7.2
Telecommand Data Rate, kbps	2	0.015	0.031

Instrument Payload Description

For this study being at CML-3, only a limited effort was placed on the instrument besides defining the notional set and priority discussed in Section 1. For this study, basic information such as mass and power were gathered to accommodate them in the lander concept. It is anticipated that a more detailed study will place a greater emphasis on instrument definition. The next table defines the mass and power for the instruments used in this study.

Instrument Mass and Power

Instrument	Mass (kg)	Ave. Power (W)
Descent Imager (2)	0.3	2.2
Stereo Camera (2)	0.3	2.2
Microscopic Imager	0.4	12.9
Mini-TES	2.1	5.6
Camera Actuators	3.0	15 movement
Raman Spectrometer	3.8	15
APXS	1.5	4
Robotic Arm	15	45 movement
Magnetometer	0.2	0.5
Magnetometer Boom	3.0	-
Flash Lamps	3.0	Negligible due to low duty cycle
Instrument Component Electronics (5)	4.0	Included above
Total	37	

The table below breaks down assumptions on science data accumulation showing that 100 Mbits per day should provide adequate science return capability. If necessary, DSN contacts on high-volume days could extend past 8 hours.

Instrument Data Return Calculations

Data Product	Data per "frame" (Mbits)	Frames	Total Data Raw (Mbits)	Compression factor	Total compressed	Total for 8 bit images	Total for 8 bit/3-color Bayer images
3 color stereo pan (1024x1024 pix, 12 bits)	1.26E+01	436	5486.1	4	1371.5	914.4	457.2
2 Hz Mag in 14 days	5.65E-05	2419200	136.7	2	68.3	68.3	68.3
Additional images	1.26E+01	200	2516.6	4	629.1	419.4	419.4
APXS spectra	3.20E-02	10	0.3	1	0.3	0.3	0.3
Raman spectra	1.23E-02	200	2.5	4	0.6	0.6	0.6
100 MI images	1.26E+01	100	1258.3	4	314.6	209.7	209.7
Mini TES spectra	1.00E-03	500	0.5	4	0.1	0.1	0.1
Descent images (1024x1024 pix, 8 bits)	8.39E+00	50	419.4	4	104.9	104.9	104.9
	Total (Mbits)		9820.4		2489.5	1717.8	1260.6
	Days to Downlink				20	20	20
	Downlink Rate (Mbits per 8-hr Pass)				**124.5**	**85.9**	**63.0**

Flight System

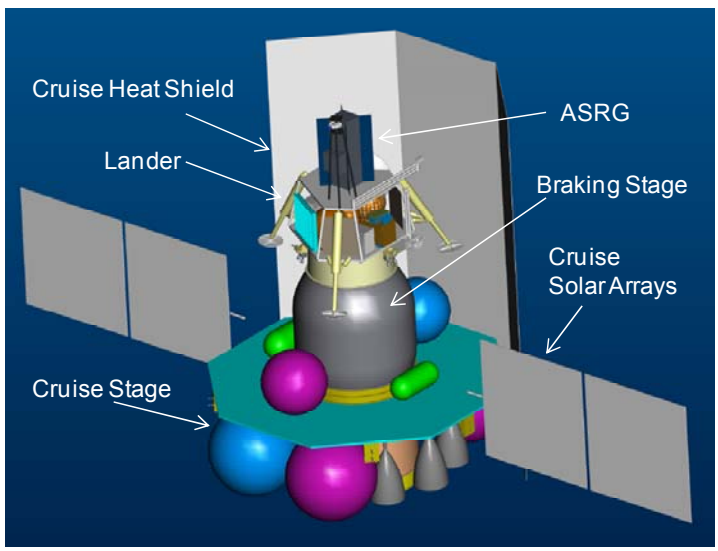
The architecture for the flight system is based on a three-stage concept. Functionally, the cruise stage is responsible for supplying the propulsion during cruise, including TCMs and ACS requiring thrusters. The cruise stage also carries GN&C hardware necessary for cruise but not for landing, such as reaction wheels and sun sensors for safing. The cruise stage is also responsible for thermal management of the cruise stage and braking stage. The solid rocket motor requires careful thermal management for the long cruise, and the cruise stage will need to provide MLI and heat to support the braking stage during cruise. The cruise stage also must provide thermal protection for the entire vehicle during cruise by incorporating a sunshade that protects the entire stack from the intense sunlight as the spacecraft approaches Mercury. The cruise stage also accommodates MGAs and low-gain antennas (LGAs) used during cruise along with X-band amplifiers that are only used during cruise.

The braking stage is responsible for the large braking ΔV maneuver prior to landing on Mercury. It also provides thrust vector control for ACS during descent.

The lander is responsible for final descent and soft landing. Command and data handling (C&DH) and GN&C processing for the entire mission will be housed within the lander. Most of the GN&C hardware will be located on the lander including star trackers and IMU. The lander will handle its own thermal management for the entire mission. The ASRG will provide power to lander hardware for the entire mission. The lander is also responsible for accommodating all the instruments and performing the science measurements on the surface.

Cruise Stage Chemical

The cruise stage shown in the figure below is based on a pressure-fed bipropellant propulsion system with six 445-N main engines and 12 ACS engines. The solar arrays shown are based on MESSENGER technology. The heat shield is also assumed to be based on MESSENGER technology. The stage contains four reaction wheels, sun sensors, X-band amplifier and antennas, a small lithium battery for eclipses, and MLI insulation for the braking stage that separates away with the cruise stage for landing.



Configuration with Chemical Cruise Stage

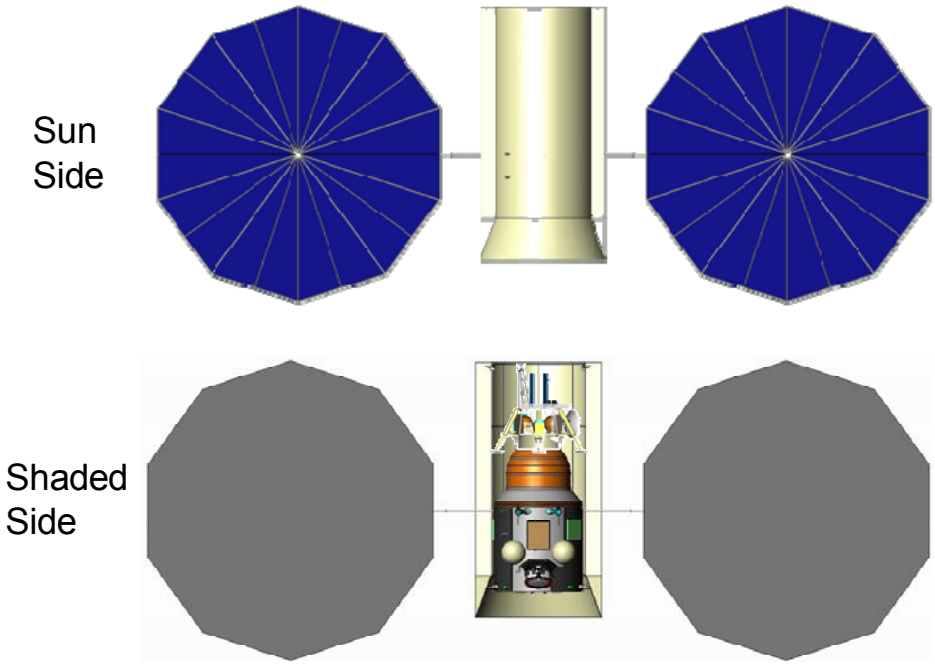
Chemical Cruise Stage Characteristics

Parameter	Summary/Value
Primary Structure	Aluminum, Aluminum-Li (SEP)
Cruise to Braking Stage Separation	4 point pyro separation
RF Hardware	X-band SSPA, 2LGA, 2 MGA
Cruise telemetry w/LGA	X-band, 720 bps
Cruise command w/LGA	X-band, 2000 bps
GN&C Hardware	Reaction Wheels (4), Sun Sensors
Attitude Determination During Cruise	Star Trackers – Inertial attitude, IMU – Rates, Sun sensors – safe-hold
Attitude Control During Cruise	3-Axis using reaction wheels +12 thrusters
TCM Engines	6 445 N thrusters, MMH-NTO, 323 s Isp
ACS Engines	12 thrusters 22 N each, MMH-NTO
Solar Array Power	780 W
Solar Array Type	High temp arrays based on MESSENGER technology
Solar Array Size	8 sq. m
Battery	Li-Ion, 8 A-Hr
Thermal Management	MLI, Heaters, software controlled
Propulsion Stage	+20°C to +40°C
Antennas	-50°C to +250°C
Solar Arrays	-150°C to +200°C

Cruise Stage with SEP

The SEP stage uses four NEXT ion propulsion engines + one spare. These operate at 7 kW each, use about 1600 kg of xenon during cruise, and provide ACS as well as thruster control. The stage is supplemented with a small hydrazine propulsion system and reaction wheels for additional ACS control. The cruise stage requires approximately 30 kW of power at Mercury to drive the ion engines. To keep mass to reasonable levels, this concept assumes the use of new high-temperature solar cell technology configured into two parasol arrays (similar to the Orion spacecraft design). Each of these panels would be 5.5 m in diameter. On approach to Mercury, they could be feathered up to 67 deg.

The heat shield, like the chemical stage, would be based on MESSENGER technology. Since the amount of dissipated power is significantly higher in this stage than the chemical stage, heat will need to be dissipated using variable conductance heat pipes. During final approach to Mercury, the current reference trajectory has an eclipse that lasts for approximately 28 hours before landing. This will have to be managed by a fairly large lithium ion battery.



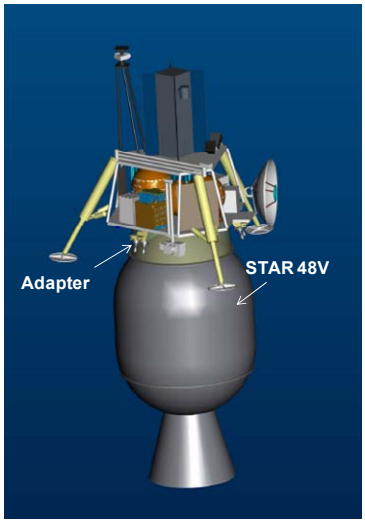
Configuration with SEP Stage

SEP Stage Characteristics

Parameter	Summary/Value
Primary Structure	Aluminum, Aluminum-Li (SEP)
Cruise to Braking Stage Separation	4 point pyro separation
RF Hardware	X-band SSPA, 2LGA, 2 MGA
Cruise telemetry w/LGA	X-band, 720 bps
Cruise command w/LGA	X-band, 2000 bps
GN&C Hardware	Reaction Wheels (4), Sun Sensors
Attitude Determination During Cruise	Star Trackers – Inertial attitude, IMU – Rates, Sun sensors – safe-hold
Attitude Control During Cruise	3-Axis using NEXT engines, reaction wheels +12 thrusters
TCM Engines	NEXT 4+1 ion propulsion, Xenon propellant, 4100 s
ACS Engines	16 thrusters 4 N each, Hydrazine
Solar Array Power	31,350 W
Solar Array Type	High Temp Cells based on GRC tech, Array type based on Orion
Solar Array Size	2 arrays, circular 5.5 m diameter each
Battery	Li-Ion, 130 A-Hr
Thermal Management	MLI, variable conductance heat pipes
Propulsion Stage	+20°C to +40°C
Antennas	-50°C to +250°C
Solar Arrays	-150°C to +230°C

Braking Stage

The lander with the braking stage is shown in the figure below. This would be the configuration for the majority of the landing phase.



Braking Configuration

Braking Stage Characteristics

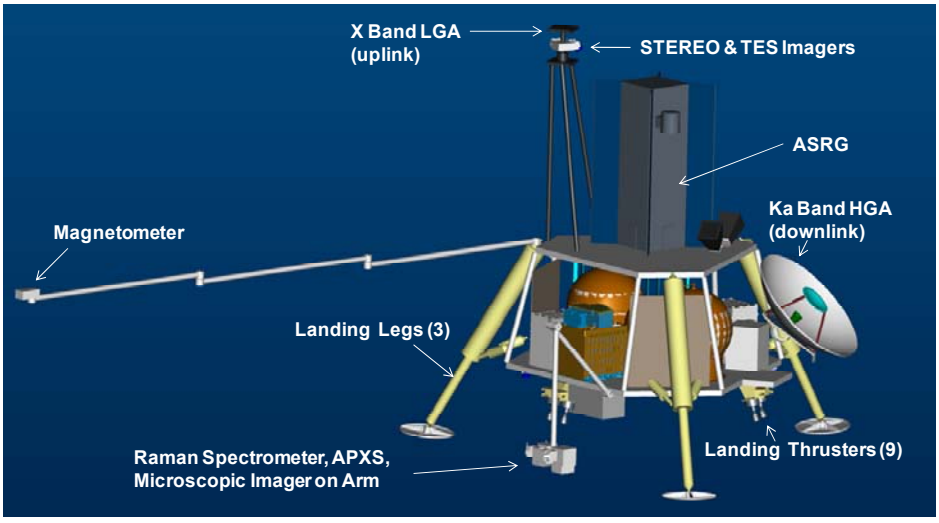
Parameter	Summary/Value
Adapter	Aluminum, 4 point pyro separation
Landing telemetry w/MGA	16 bps, X-band
Landing command rate w/LGA	15.6 bps, X-band
Rocket Motor	STAR 48V (Custom loaded)
Propulsion Stage Thermal Range	+20°C to +40°C

Lander

The following figure illustrates the lander in its deployed configuration on Mercury's surface. The lander is powered by a single ASRG mounted to the top of the deck for proper balance. It has three landing legs with honeycomb shock absorption. The primary structure consists of honeycomb panels. The communication system consists of redundant X/Ka-band transponders and amplifiers using X-band uplink and Ka-band downlink. The HGA is mounted on the side with the lander oriented at landing so that the HGA will have coverage for the entire mission. The LGA and imaging instruments are mounted on a fixed mass to have clearance above the ASRG. The magnetometer is mounted on a multi-segmented boom 3 m away from the lander.

Avionics, including flight processor, recorder, power generation, and distribution electronics, are all redundant. The configuration also carries a small lithium ion battery primarily to supplement power during cruise, but it also could help manage peak loads on the surface as well.

To accommodate thermal loads at landing and communications, the lander will face a preferred direction relative to the Sun. Additional high-temperature MLI may be required on the perpendicular faces to handle the 4 Suns of incident energy for the first two days.



Lander in Surface-Deployed Configuration

Lander Characteristics

Primary Structure	Composite panels
Primary Structure	Composite Honeycomb Panels
Landing Legs	Based on Apollo, Aluminum with honeycomb to absorb energy
Robotic Arm	3 DOF, Instruments mounted side-by-side
Magnetometer Arm	3 m, multi-segmented, composite boom
RF Hardware	X/Ka-band coherent transceiver, SSPA or TWTA, HGA, LGA, MGA
Oscillator	Ultra-stable Oscillator (USO)
RF Power	8 W
HGA Diameter	60 cm
Landed telemetry w/HGA	3.5 kbps, Ka-band
Landed command w/LGA	31 bps, X-band
Processor	RAD 750 (133 MIPS)
Digital Signal Processor	High density FPGA based (>20MFLOPS)
Data Storage Capacity	32 Gb SDRAM
Landing Sensors	Star Trackers- Inertial attitude, IMU-attitude rates, Descent Imagers – surface relative rates, hazard avoidance, LIDAR- relative slope, hazard avoidance, surface relative altitude and altitude rate
ASRG Power	142 W – Cruise, 141 W- Landed
Battery	Li-Ion, 8 A-Hr
Landing Engines	9 Engines based on MDA DACs Technology, 445N each, MMH-MON-3
ACS Engines	9 Engines based on MDA DACs Technology, 22N each, MMH-MON-3
Thermal Management	"Thermos bottle" approach, Louvers, heaters for external instruments, high temp. MLI
Lander Bulk Temperature	+20°C to +40°C
ASRG (interface Temperatures)	0°C to +60°C

Mass and Power Summaries

Mass and power margins were calculated using the Decadal Mission Study Ground Rules

- 30% using the following definition

  Margin = Max Possible Resource Value – Proposed Resource Value

  Margin (%) = (Margin/Max Possible Resource Value) × 100

A margin of 30% was applied to all hardware with the following exception:

- STAR 48V inert mass since it is a known mass with a finite tolerance
- Power use on NEXT power processing units (PPUs); assume 5% uncertainty on efficiency

Note that the margin definitions for this study are different from, and considerably more conservative than, APL Space Department practices, which use 30% growth margin.
Margin (%) = (Margin/Proposed Resource Value) × 100)

The difference between the two methods of calculation in kilograms being Total Dry Mass =1.43 × Estimated Mass required by the decadal guidelines vs. Total Dry Mass = 1.3 × Estimated Mass required by APL practices. This difference can be very significant for high C_3 and ΔV missions.

For missions that demand such high energy and low power, it may be impractical to set margins at these levels. An alternative is to introduce more rigor into the development process early and design to mass and power. This approach may have risks and may increase cost risk, but it also has merits to consider.

The Mass Summary for the chemical propulsion option with an assumed V∞ of 0 km/s and reduced payload is shown in the table below. The next table shows the mass for the SEP option.

Mass Summary of Ballistic Chemical Option 5

Lander Stage	Est. Mass (kg)
Instruments	21
Mechanical	50
Propulsion	23
Avionics	11
Power	41
GN&C	11
Thermal	14
RF Communications	22
Harness	10
Total Dry (Estimated)	203
Total Dry (30% Reserve Margin)	289
Consumables (Propellant, Helium)	39
Total Wet (30% Margin)	329

Braking Stage	Est. Mass (kg)
Motor Case and Nozzle	154
Adapter, S&A, and break-up	19
Total Dry (Estimated)	173
Total Dry (30% margin –not motor)	181
Propellant	1632
Total Wet (30% margin)	1813

- **Chemical Propulsion Cruise Stage**
- **Reduced Payload – No robotic arm or microscopic imager**

Cruise Stage	Est. Mass (kg)
Mechanical	60
Propulsion	215
Avionics	0
Power	81
GN&C	28
Thermal	69
RF Communications	16
Harness	24
Total Dry (Estimated)	493
Total Dry (30% Reserve Margin)	704
Consumables (Propellant, Helium)	1669
Total Wet (30% Margin)	2373

Stack Mass	Est. Mass (kg)
Total Stack (30% Margin)	4515
Maximum Launch Mass ATLAS V 551	4630

Mass Summary of SEP Option 6

Lander Stage	Est. Mass (kg)
Instruments	37
Mechanical	55
Propulsion	23
Avionics	11
Power	41
GN&C	11
Thermal	14
RF Communications	22
Harness	12
Total Dry (Estimated)	226
Total Dry (30% Reserve Margin)	323
Consumables (Propellant, Helium)	43
Total Wet (30% Margin)	366

Braking Stage	Est. Mass (kg)
Motor Case and Nozzle	154
Adapter, S&A, and break-up	19
Total Dry (Estimated)	173
Total Dry (30% margin –not motor)	181
Propellant	1770
Total Wet (30% margin)	1951

- **SEP Cruise Stage**
- **Robust Payload**

Cruise Stage	Est. Mass (kg)
Mechanical	140
Propulsion	416
Avionics	18
Power	266
GN&C	16
Thermal	122
RF Communications	18
Harness	44
Total Dry (Estimated)	1040
Total Dry (30% Reserve Margin)	1486
Consumables (Xenon, Hydrazine, Helium)	1737
Total Wet (30% Margin)	3223

Stack Mass	Est. Mass (kg)
Total Stack (30% Margin)	5540
Maximum Launch Mass ATLAS V 541	5770

Power Summary

The tables below break out the power summary for the chemical cruise stage and SEP cruise stage. Note that summary tables for the lander and cruise stages are separate since they do not share power across the interface for cruise. The total power needed for cruise is the summation of the lander and cruise portions.

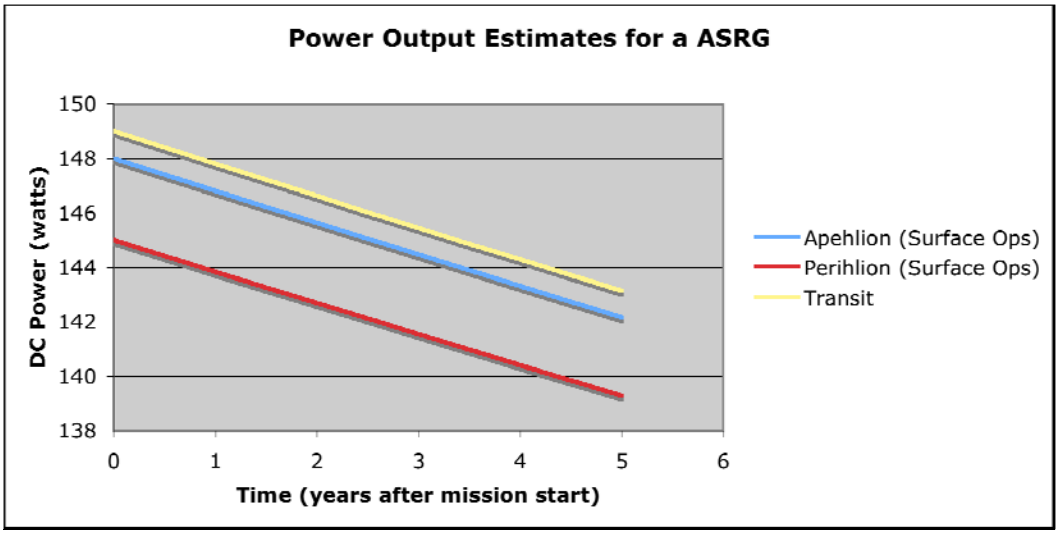
Chemical Cruise Stage Power Summary

Cruise Stage - Chemical Option	Cruise	TCMs		Eclipse	
		SA	Battery		
Propulsion	0	0	148.4	0	
Power System	6.5	6.5		6.5	
Guidance, Navigation, and Control	32.7	32.7	0	32.7	
Thermal Control	430	430		0	
RF Communications	60	60		0	
Harness Loss (3%)	15.9	15.9	4.5	1.2	
Total Power Required	545.1	545.1	152.9	40.4	
Power Available	779	779	0.033	0.5800	Duration (hrs)
Margin	233.9	233.9	5.0	23.4	Total Load W-Hrs
%Margin	30%	30%	224	224	Battery Capactity W-Hrs
			2.25%	10%	Depth of Discharge

SEP Cruise Stage Power Summary

SEP	Cruise/ Flyby	EP Thrust	Eclipse	
EP Thruster Power				
Ion Engine	186	28854	0	
Harness Loss (1%)	1.9	288.5	0.0	
Total Power Required	187.9	29142.5	0.0	
Power Available	30677	30677	30677	
Margin (EP Thruster Power)	30489.1	1534.5	30677.0	
%Margin (EP Thruster Power)	99.39%	5.00%	100.00%	
Bus Power				
Power System	6.5	6.5	6.5	
Guidance, Navigation, and Control	32.7	32.7	32.7	
Thermal Control	354	354	80	
RF Communications	60	60	0	
Harness Loss (3%)	13.6	13.6	3.6	
Total Load	466.8	466.8	122.8	
Total After Power System Loss	547.6	547.6	127.9	
Power Available	781.9	781.9	28.63	Duration (hrs)
Margin (Bus Power)	234.4	234.4	3661.0	Total Load W-Hrs
%Margin (Bus Power)	30%	30%	4574.2	Battery Capactity W-Hrs
Total Power Available from S/A	31459	31459	80%	Depth of Discharge

The next figure shows the power curve used for the study developed by GRC specifically for the Mercury mission (850°C hot end temperature performance). This curve was used to generate all lander power estimates in the study. The surface operations lines represent power available when first landing with the Sun low in the horizon with a 21°C surface temperature and with direct sunlight on the ASRG. If the temperature were increased to the maximum of 70°C, it would affect performance by about 3 W.



ASRG Power Curve Specific for Mercury Environment

The table below breaks out the power summary for the lander powered by the ASRG. This is representative for all options except small reductions in the instrument power for the reduced payload case (mostly removal of robotic arm power). The system was sized to 30% margin for all power modes. Note that landing is performed with battery power.

Lander Power Summary

Lander Subsystem	Cruise		Stereo Imaging	In Situ		Surface Comm.	Landing
	Receive only	Transmit/ Receive		13-hour science cycle	3-hour recharge		
Instruments	0	0	19.3	18.3	0.5	0	2.15
Propulsion	0	0	0	0	0	0	210
Avionics	24	24	21	21	21	19	35
Power System	22.8	22.8	22.8	22.8	22.8	22.8	22.8
Guidance, Navigation, and Control	7.6	7.6	0	0	0	0	56.6
Thermal Control	25	25	25	20	25	25	25
RF Communications	13	20	3	3	3	41.75	41.75
Harness Loss (3%)	2.8	3.0	2.7	2.6	2.2	3.3	11.8
Battery Recharge Load (W-Hrs)	14.1	0	0	51.1	72.7	0	0
Battery Recharge Power each hour (W)	4.2	0	0	11.1	24.2	0	0
Total Lander Load	99.4	102.4	93.8	98.7	98.7	111.8	405.1
Power Available from ASRG	142.0	142.0	141.0	141.0	141.0	141.0	141.0
Power used from battery	0	3.0	0	0	0	13.1	264.1
Duration (hrs)	0	4	0	0	0	8	0.0403
Total Load (W-Hrs)	0	11.9	0	0	0	104.9	10.643
Battery Capactity (W-Hrs)	224	224	224	224	224	224	224
Depth of Discharge	0.0%	5.3%	0.0%	0.0%	0.0%	46.8%	4.8%
Margin	42.6	42.6	47.2	42.3	42.3	42.3	0
% Margin	30%	30%	33%	30%	30%	30%	0%

Technology Description

Radio Frequency

At the aperture size chosen, the minimum RF output power at Ka-band to support the required data rate is 8 W. This output power must be generated while consuming no larger than about 20 W DC input. Such an amplifier does not now exist. We recommend making investments in technology development to improve the power efficiencies of existing technologies. Recent traveling wave tube amplifier (TWTA)-based options are listed below. The mass and power include that of a single TWTA, associated power converter, and cable connecting the two.

Item	Mass (kg)	RF Output (W)	DC Input (W)	Efficiency (%)
TWTA (Cassini)	3.50	10	29	34.5
TWTA (LRO)	2.92	40	100	40

With some technology investment it is possible to reduce the mass of the TWTA + power converter + cable harness to about 2.5 kg. Further development of a 20-W Ka-band TWTA is under way at the NASA GRC. Alternative high-efficiency solid-state technologies should also be explored.

A design based on a Ka-band downlink and X-band uplink is not as robust as one that includes an X-band downlink. The lack of robustness arises from reliance on the two-axis gimbal to point the Ka-band beam at Earth. A less directional X-band transmitter adds robustness if the gimbal fails. This comes at the expense of added mass on the lander. For this study Ka-band was assumed to minimize mass since that was the primary driver.

Avionics

The landing co-processor works in conjunction with the main Integrated Electronics Module (IEM) general-purpose processor and the various terrain sensors to select a safe landing location within the capabilities of the Mercury lander vehicle. Sensors acquire information about the topology of the surface and algorithms process that data and guide the vehicle to suitable terrain. Based on experience with the lunar ALHAT project, we anticipate that the required Mercury landing algorithm will require a processing throughput of several 10s of MFLOPs (millions of floating point operations/second), well beyond the capabilities of current space-qualified general-purpose processors as well as those likely to become available in the next decade. However, a purpose-designed co-processor optimized for the key high-rate kernel calculations of landing algorithm can satisfy that computational requirement. Such a co-processor is well within the capabilities of space-qualified hardware expected to be available within the next five years.

A high-density, high-performance, space-qualified field programmable gate array (FPGA) will form the heart of the co-processor. Current single-chip FPGA qualified technology is already capable of implementing designs with several million gates, and at least one government-funded program (SIRF) is developing an even higher density radiation-tolerant part. Hence, the main technology development required to support a Mercury lander is development of the hazard avoidance algorithm and support co-processor architecture and implementation. Qualified FPGAs with suitable capability may exist now and certainly will in the near future. Implementing the co-processor logic in an FPGA for the landing algorithm should be straightforward; however, that task will be a new engineering development and should have an appropriate budget and schedule to reduce the risk of this new technology.

SEP Solar Array

The arrays for the SEP stage assume high-temperature cells based on GRC technology configured in a very lightweight array configuration similar to the parasol design for the Orion spacecraft. The cells themselves have been demonstrated to 230°C. However, such a large lightweight array has never been developed for high-temperature applications. Therefore, significant technology development should be

performed to evaluate and demonstrate the high-temperature cell technology integrated at the array level and tested specifically to the high-temperature environment expected for the Mercury mission.

Propulsion

100-lbf KEW-5 (DACS) thruster

A PWR 100-lbf KEW-5 thruster was developed for the Divert Attitude Control System (DACS) for the Missile Defense Agency (MDA). As an option for the Mercury mission, the thruster could be the main thruster for the Mercury surface landing. Although the DACS thruster has been in operation, the requirements for the MDA application are somewhat different for the Mercury mission. The DACS thruster could not only provide long-duration burns and but also be in deep space for 6 years, while the thrusters on the current application only operate for short-duration burns right after launch. To enhance the engine performance and reduce the system mass for the Mercury mission, the DACS thruster will be used with a cold MON-25/MMH propellant system with low engine inlet pressure. The current DACS thruster uses MON-15/MMH at a relative high engine inlet pressure. Subsequently, TRL for the thruster is about 5 and 6. The thruster valve and thrust chamber and nozzle will be modified to enhance the performance in the Mercury mission. Furthermore, the thruster performance and operation will be assessed through hot-fire tests at relevant environments.

XRL-132 pump-fed engine:

A PWR XRL-132 pump-fed engine is used as an engine baseline to generate thrust for TCM deterministic burns on the cruise stage. The engine went through a hot-fire demonstration under vacuum conditions in the 1980s under an Air Force program. For the Mercury mission, the TRL level for this engine can be considered as between 5 and 6. The technology advancement investment for the engine should emphasize the completion of the engine technology development phase relevant to the Mercury mission requirements. Engine performance and operations for a complete mission duty cycle (MDC) will be fully assessed and characterized within the relevant environment.

Risk List

The primary risks identified in this study are discussed below. Some risks apply to all concepts and some are specific to the ballistic/chemical and SEP options. For a CML-3 study, there was not enough definition to develop a full risk table with likelihood and consequence. This will need to be addressed in a more detailed study. The purpose of this list is to identify the risk and roughly examine what the potential consequence could be if the risk became a problem. It also provides a starting point for future studies.

- All Concept Options
  - Soft landing with hazard avoidance
    - Potential Consequence—Mission cost and schedule could be impacted to ensure safe landing.
  - Long-term storage of solid rocket motor
    - Potential Consequence—If qualification is not successful, options would be limited to lower performance liquid option. Mission feasibility could be questioned.
  - Very limited uplink rate of 32 bps for surface operations
    - Potential Consequence—May limit science operations. Operate at a lower cadence.
  - Raman spectrometer readiness
    - Potential Consequence—Cost and schedule impact or loss of science measurement.
  - Complexity of three-stage system
    - Potential Consequence—Development cost and schedule impact
  - Thermal environment at landing
    - Potential Consequence—Mass penalty for additional protection, arrival restrictions (e.g., near aphelion).
- Ballistic/Chemical Option
  - Level of performance improvement over reference trajectory case
    - Potential Consequence—Launch mass may be higher than $V_\infty = 0$ option evaluated in this study. Payload mass and mass margin could be affected.

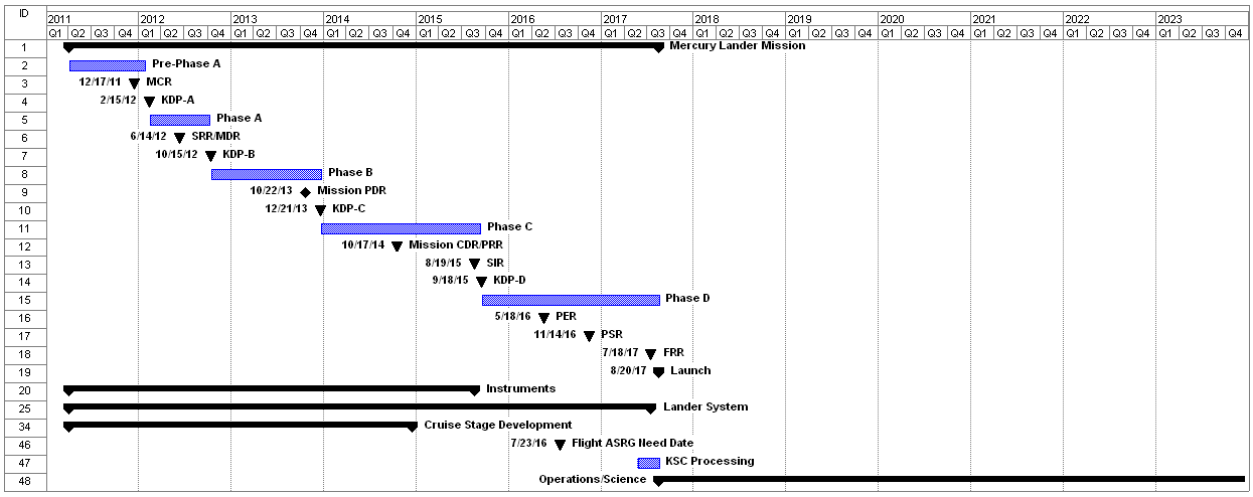
- Mass margin very tight since already carrying a reduced payload set to fit in ATLAS V 551
  - Potential Consequence—Grow into Delta IV Heavy.

- SEP
  - Ability to develop high-density, high-temperature solar arrays with performance significantly better than what has been demonstrated with MESSENGER using high-temperature cell technologies that have been already developed
    - Potential Consequence—If performance improvements in array density and mass are not achieved, the SEP approach is not feasible.
  - Thermal management of a high-power SEP stage
    - Potential Consequence—Mass and volume increase in stage.

4. Development Schedule

High-Level Mission Schedule

The high-level mission schedule is based on previous mission experience and recent concept development efforts, which was deemed a good approximation to the anticipated schedule for the Mercury Lander Mission. The schedule was, however, updated to take into account the required additional cruise stage development. Note that the mission schedule below is for the chemical propulsion case. The SEP schedule is very similar (the chemical propulsion and SEP cruise times are very similar, i.e., 5 years vs. 4.8 years).



Key Phase Duration Table

Project Phase	Duration (Months)
Phase A – Conceptual Design	8
Phase B – Preliminary Design	14.5
Phase C – Detailed Design	21
Phase D – Integration & Test	23.4
Phase E/F –Mission Operations	76.9
Start of Phase B to PDR	12.4
Start of Phase B to CDR	24.5
Start of Phase B to Delivery of Instrument #1 (Suite)	34
Start of Phase B to Delivery of Flight Element #1 (Lander)	57
Start of Phase B to Delivery of Flight Element #2 (Cruise Stage)	26
System Level Integration & Test	24.1
Project Total Funded Schedule Reserve	
Total Development Time Phase B - D	58.9

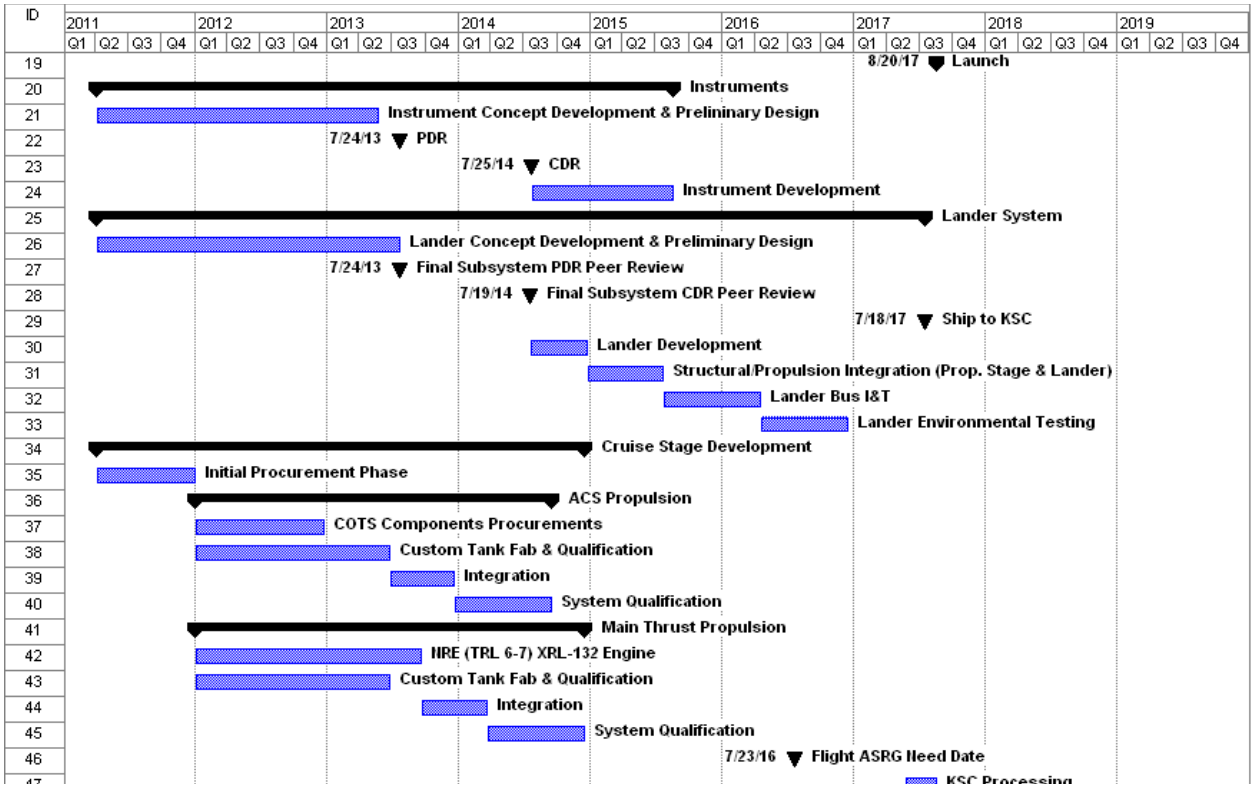
Technology Development Plan

For a CML-3 study, enabling or enhancing technologies have been identified in Sections 2 and 3. A technology development plan would be developed as part of a more detailed study.

Development Schedule and Constraints

The schedule below shows the schedules for instrument, lander, and cruise stage development from concept development through launch. Note that instrument and lander concept developments, as well as initial procurement activities for the cruise stage, need to commence in early 2011 to meet the launch readiness date (LRD) in 2017.

The development schedule further assumes that flight ASRGs will be readily available as government off-the-shelf equipment in mid-2016 and that, therefore, no additional development and/or qualification will be needed.



5. Mission Life-Cycle Cost

Costing Methodology and Basis of Estimate

In support of the ACE Laboratory, CML-3 cost estimates were generated for six mission options. Results are shown for the following options described above:

- Option 2: Robust payloads, including robotic arm, HIPAT chemical propulsion cruise stage, Delta IV-Heavy-equivalent launch vehicle.
- Option 4: Robust payloads, including robotic arm, pump-fed chemical propulsion cruise stage, Delta IV-Heavy-equivalent launch vehicle.
- Option 5: Identical to Option 2 but reduced payloads, Atlas V 551-equivalent launch vehicle.
- Option 6: Robust payloads, including robotic arm, SEP cruise stage, and Atlas V 541-equivalent launch vehicle.

In terms of estimated cost including reserves, all four options shown are mini-Flagship missions. Options 2, 4, and 6—which provide a robotic arm and robust instrument suite—are estimated to cost within 6% of each other. Option 5 is less expensive than the other three, but it provides the least science because of its reduced instrument suite and no robotic arm.

Ground Rules and Assumptions. Ground rules and assumptions for the Mercury Lander estimates are based on the revision 2 draft of "Groundrules for Mission Concept Studies in Support of Planetary Decadal Survey (dGRPDS)."

Cost estimates are presented in fiscal year 2015 (FY15) dollars. Initial estimates were generated in FY10 dollars and adjusted to FY15 dollars. Availability of FY10 estimates enable ready comparison of the initial estimates with cost data from current and recently completed programs and with recently prepared estimates for other programs and trade studies. The inflation adjustment from FY10 dollars to FY15 dollars is based on a 2.7% annual inflation rate presented in dGRPDS.

The cost estimates assume that NASA will fund all Mercury lander mission costs and that all significant work will be performed in the United States. The estimates as presented include all costs, including fees.

The mission cost estimates cover activities through the end of Phase E, including the following:

- Phase A
- Project management, systems engineering, and safety and mission assurance
- Science, including science team members and Science Operations Center (SOC) preparation
- Instruments and, where specified, design and development of the robotic arm
- Spacecraft hardware and flight software development
- Mission operations, including development of ground data systems, DSN charges, and Phase E activities
- Launch vehicle and services
- Systems integration and test
- Education/public outreach (E/PO)
- Cost reserves

All options specify lander electrical power will be provided by an ASRG. Per dGRPDS, we assume that an ASRG will be ready for flight by March 2014 at a unit cost of approximately $20 million (FY10). The dGRPDS also specifies inclusion of a $15 million charge for nuclear launch compliance.

Based on past experience, we assumed that Phase A expenditures would be $2.5 million. We also assumed that E/PO would be approximately 1% of the baseline mission cost, that is, mission cost excluding cost reserves.

Technology development cost estimates cover investments for components needed to achieve a TRL of 6. Except in two cases, costs of those investments are included in the technology development element. In two cases, lightweight solar arrays for the SEP mission and the Raman spectrometer, technology development costs could not be separated from non-recurring design and development effort, and those costs are included in the spacecraft and instruments cost elements, respectively. Also, no dollars were included for Ka-band SSPA or TWTA.

Estimates reflect, where possible, schedules developed for SEP and non-SEP options. For example, the estimated Phase-E costs and DSN charges reflect different transit durations and number of in-transit flybys.

The integration and test element covers effort and expenditures to assemble and check out spacecraft stage subsystems as well as effort and expenditures to integrate and test the three stages, instruments, robotic arm, and ASRG.

Launch vehicle and services costs are based on the provided dGRPDS table.

Our cost reserves posture is based on dGRPDS, released September 21, 2009.

- 50% reserves on the baseline technology development cost estimates.
- 50% reserves on other Phase A–D costs except for launch vehicle and ASRG
- 50% reserves on DSN charges
- 30% reserves on Phase E costs
- No cost reserves on the launch vehicle and services or ASRG
- No reserves on E/PO.

Cost Methodologies. The methods used to estimate mission costs are summarized in the following two tables. The first table covers non-spacecraft-hardware costs; the second, spacecraft subsystem hardware.

Cost Estimating Methodologies for Non-Spacecraft Elements

Element	Method	Comments
Phase A	Engineering estimate	Based on APL experience
Technology Development	Engineering estimates, vendor ROMs	Quality of estimates varies widely
Management, Systems Engineering, S&MA	Cost factors using spacecraft hardware as basis, labor estimate (MD&A)	Factors based on MESSENGER, New Horizons actuals, RBSP trends
Science Team	Level of Effort, by phase (A–E)	Includes instrument planning effort, preparation of SOC
Payloads, Instruments	NICM II (NASA Instrument Cost Model) estimates, engineering estimates	Instrument costs were estimated using the NICM parametric model and crosschecked with historical cost data. An engineering estimate was the basis for the robotic arm cost and includes significant system design and development
Mission Operations	Cost factor (pre-launch spt.), engineering estimates (Phase E)	Phase E estimates adjusted for duration, #flybys
Launch Vehicle & Services	dGRPDS Ground Rules (LV, NEPA compliance)	Engineering estimate used for LV I/F engineering support
Ground Data Systems	Analogies to previous APL missions	
Flight Software	Engineering build-up	Includes development & test

Many of the lander and other stage hardware subsystems and components are similar to those examined during trade studies for the International Lunar Network (ILN) and related Lunar Polar Volatiles (LPV) missions. ILN cost estimates were presented in the summer of 2009 to a NASA Program Analysis & Evaluation (PA&E) review team. Those estimates in turn were derived from a combination of component cost histories and PRICE-H parametric cost analyses.

Costing Methodologies for Spacecraft Elements

Element	Method	Comments
SEP Cruise Stage	NAFCOM parametric model	Hardware components only; GRC estimate
Mechanical & Structural	PRICE-H parametric model	Model originally developed & calibrated for International Lunar Network (ILN) and Lunar Polar Volatiles (LPV) trade studies
Propulsion	Vendor ROMs, engineering estimates (oversight labor)	ROMs, labor estimates provided by
GN&C	Analogy to MESSENGER	Engineering estimate for LIDAR
IEM, Avionics, PSE, BME, Battery, PDU, Testbed hardware	PRICE-H, analogies, vendor ROMs (IEM, testbed h/w)	Estimates at board level, results checked against RBSP & launched mission actuals
Thermal Control	Analogies to MESSENGER	Includes cruise stage solar shield
RF Communications	Analogy to MESSENGER	SSPA requires tech. development
Integration & Test	Cost factor applied to spacecraft costs	Based on STEREO actuals and subsequent engineering analyses

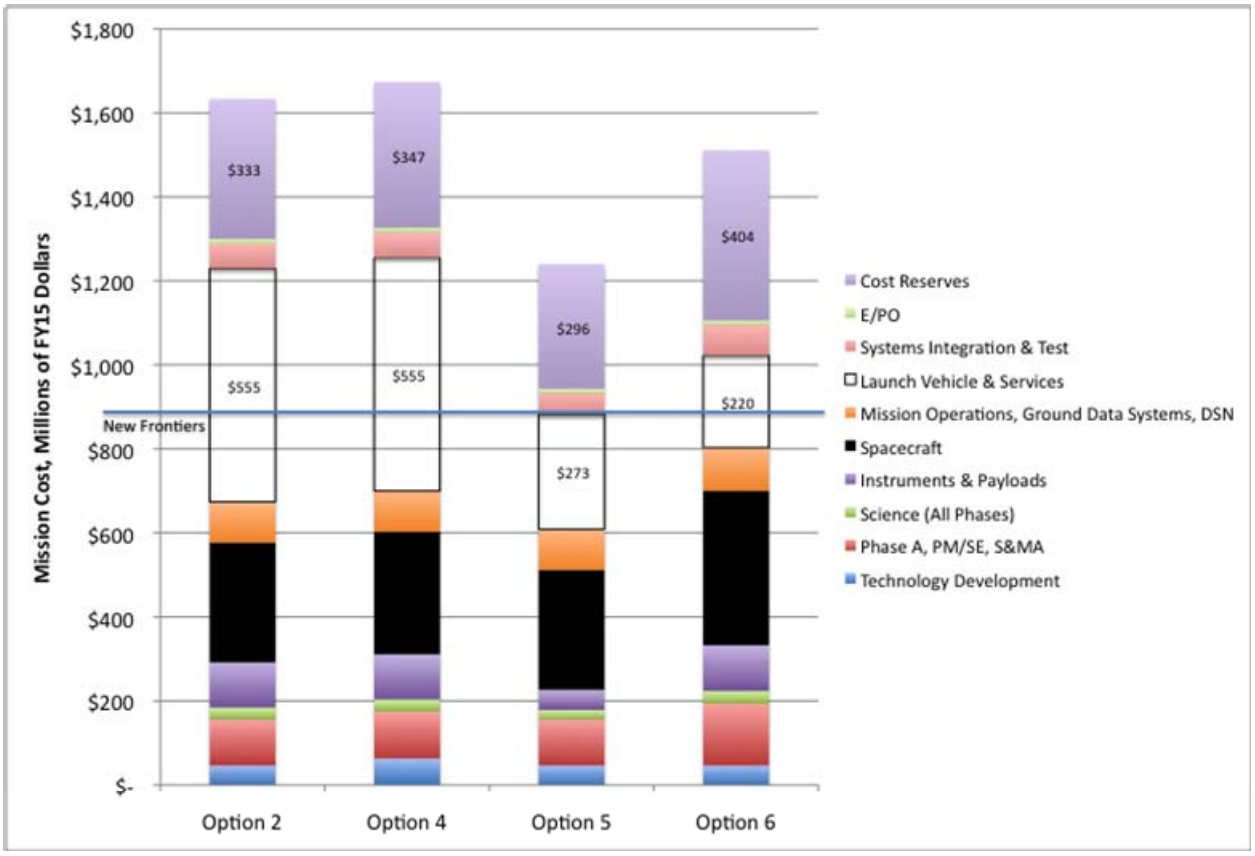
Cost Estimates

Cost estimates for Options 2, 4, 5, and 6 are shown in the figure below. In terms of estimated cost including reserves, all four options are mini-Flagship missions. The three most expensive, Options 2, 4, and 6, which are estimated to cost $1.5–1.7 billion in FY15 dollars, provide a robotic-arm capability and robust instrument suite. Option 6 is the least expensive of the three because its SEP-powered cruise stage enables the use of a less-expensive Atlas V-sized launch vehicle instead of the more expensive Delta IV-Heavy-sized launch vehicle required by Options 2 and 4.

Option 5 is the least expensive option overall at slightly more than $1.2 billion, but it provides the least science because of its reduced instrument suite.

The following figure shows a comparison of the four options, with costs disaggregated in major cost elements.

Cost Comparison Summary in FY15$



The major cost elements are the spacecraft (black bar), launch vehicle and services (white bar), and cost reserves (top purple bar). Options 2 and 4 exceed $1.6 billion, with launch vehicle and cost reserves accounting for more than half of the total cost. Option 5, the reduced payload option, is less expensive both because of fewer payloads and being able to use a smaller launch vehicle. Option 6, the option with an SEP cruise stage, has the highest estimated spacecraft cost. Total mission cost is less than those of the other two robust-payload options because the SEP cruise stage that accounts for the higher spacecraft cost enables the use of a smaller, less expensive launch vehicle.

The table below presents estimated mission costs for the four options mapped to the NASA Level-2 WBS defined in Appendix G of NPR 7120.5D.

Cost Comparison in FY15$ Using NASA WBS

Mercury Lander	Robust Science, Chemical HIPAT Cruise/Delta IV-Heavy LV	Robust Science, Chemical Pump-Fed Cruise/Delta IV-Heavy LV	Reduced Science, Chemical HIPAT Cruise/Atlas 551 LV	Robust Science, SEP Cruise/Atlas 541 LV
Description	**Option 2 FY15$M**	**Option 4 FY15$M**	**Option 5 FY15$M**	**Option 6 FY15$M**
Phase A	**3**	**3**	**3**	**3**
Enabling Technology Development	**50**	**68**	**51**	**51**
Project Management	**34**	**35**	**34**	**44**
Systems Engineering, incl. MD&A & Navigation	**49**	**50**	**49**	**67**
Safety & Mission Assurance	**23**	**23**	**23**	**33**
Science/Technology (Phases A-D)	**17**	**17**	**12**	**17**
Payloads	**108**	**108**	**48**	**109**
Spacecraft (1 lander)	**285**	**292**	**285**	**368**
Cruise Stage (Hardware)	102	108	102	185
SRM Stage (Hardware)	10	10	10	10
Lander (Hardware, excl. ASRG)	129	129	129	129
Flight Software (FSW), Autonony Development & Test	21	21	21	21
ASRG	23	23	23	23
Mission Operations	**69**	**70**	**68**	**79**
Launch Checkout, Early Operations Support (Phase D)	14	15	14	19
Mission Ops (Phase E) [excluding DSN]	55	55	54	60
Phase E Management, Eng. Support	*20*	*20*	*20*	*19*
Phase E Mission Ops, incl. GDS maint.	*19*	*19*	*19*	*18*
Phase E MD&A and Navigation Support	*4*	*4*	*4*	*10*
Phase E Science Team	*12*	*12*	*11*	*13*
Launch Vehicles & Services, incl. LVA, I/F, NEPA	**555**	**555**	**273**	**220**
Ground Data Systems	**16**	**16**	**16**	**16**
Systems Integration & Test	**59**	**60**	**50**	**72**
Space Communications Services (DSN)	**23**	**23**	**23**	**20**
E/PO	**11**	**11**	**11**	**11**
Subtotal	**1304**	**1331**	**948**	**1111**
Excluding ASRG, nuclear launch compliance	*1266*	*1293*	*910*	*1073*
Excluding LV, ASRG, nuclear launch compliance	*727*	*754*	*653*	*869*
Cost Reserves	**335**	**349**	**299**	**407**
Phases A-D (excl. LV, ASRG) : 50%	319	332	282	389
Phase E: 30%	17	17	16	18
Total, including Reserves	**1639**	**1680**	**1247**	**1517**

It was not possible given the degree of schedule definition to distribute costs by fiscal year to present costs in real-year dollars. Estimated costs of Technology Development required to bring required technologies to TRL 6 are shown in the table below.

Estimated Technology Development Costs Without Reserves FY15$

Mercury Lander	Robust Science, Chemical HIPAT Cruise/Delta IV-Heavy LV	Robust Science, Chemical Pump-Fed Cruise/Delta IV-Heavy LV	Reduced Science, Chemical HIPAT Cruise/Atlas 551 LV	Robust Science, SEP Cruise/Atlas 541 LV
Description	**Option 2 FY15$M**	**Option 4 FY15$M**	**Option 5 FY15$M**	**Option 6 FY15$M**
Phase A	**3**	**3**	**3**	**3**
Enabling Technology Development	**50**	**68**	**51**	**51**
Efficient Ka-band SSPA or TWTA	4	5	5	5
Xenon Flash Lamps	2	2	2	2
High-speed Graphics Processing (FPGA)	7	7	7	7
Raman Spectrometer	9	9	9	9
Lightweight/Low-Power LIDAR	2	2	2	2
Solid Rocket Qual. For Long In-Space Storage	4	4	4	4
Ultra-light High-temperature Solar Arrays	11	11	11	11
Bi-Prop Pump-Fed Engine Development	0	17	0	0
DoD DACS for Lander Use	12	12	12	12

Appendix A – Study Team

Role	Name	Organization
Science Champion	Steven Hauck	Case Western
Deputy Science Champion	Allan Treiman	LPI
NASA HQ POC	George Tahu	NASA HQ
Decadal Program Manager	Kurt Lindstrom	JHU/APL
APL Science POC	David Blewett	JHU/APL
Project Manager	Helmut Seifert	JHU/APL
Study Lead/Systems Engineer	Doug Eng	JHU/APL
	Kate Stambaugh	JHU/APL
	Greg Chavers	NASA MSFC
	Steve Oleson	NASA GRC
	Melissa Mcguire	NASA GRC
Mission Design	Yanping Guo	JHU/APL
	John Dankanich	NASA GRC
	Chris Dong	JHU/APL
	Laura Burke	NASA GRC
Cost Estimation	Larry Wolfarth	JHU/APL
	Meagan Hahn	JHU/APL
	John Drexler	NASA GRC
Operations	Mark Holdridge	JHU/APL
	James Cockrell	NASA GRC

Role	Name	Organization
Instruments	Tim Miller	JHU/APL
Propulsion	Huu Trinh	NASA MSFC
	James Fittje	NASA GRC
	Tim Verhey	NASA GRC
Mechanical	Role	JHU/APL
	John Gyekenyesi	NASA GRC
Thermal	Jack Ercol	JHU/APL
	Elizabeth Abel	JHU/APL
	Tony Colozza	NASA GRC
RF	Brian Sequeira	JHU/APL
	Joe Warner	NASA GRC
Avionics/Power	Marty Fraeman	JHU/APL
Power	Glenn Williams	NASA GRC
	Paul Schmitz	NASA GRC
	Eric Lowery	NASA MSFC
	Geoff Landis	NASA GRC
	Jeff Hojniki	NASA GRC
GN&C	Dewey Adams	JHU/APL
	Mike Martini	NASA GRC
Software	Steve Williams	JHU/APL
SEP Configuration	Jon Drexler	NASA GRC

Appendix B – Concept Maturity Level Definitions

Concept Maturity Level	Definition	Attributes
CML 6	Final Implementation Concept	Requirements trace and schedule to subsystem level, grassroots cost, verification and validation approach for key areas
CML 5	Initial Implementation Concept	Detailed science traceability, defined relationships and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy
CML 4	Preferred Design Point	Point design to subsystem-level mass, power, performance, cost, risk
CML 3	Trade Space	Architectures and objectives trade space evaluated for cost, risk, performance
CML 2	Initial Feasibility	Physics works, ballpark mass and cost
CML 1	Cocktail Napkin	Defined objectives and approaches, basic architecture concept

Appendix C – Presentation Material

Mercury Lander Mission Concept Study

Results Briefing

Steven Hauck, Science Champion

Doug Eng, Study Lead

Revision 2/16/2010

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Agenda

- **Study Team**
- **Scientific Objectives**
  - Study Rationale
  - Science Questions
  - Objectives & Priorities
  - Instrument Payload & Options
- **High Level Mission Concept**
  - Concept Maturity Level
  - Mission Overview
  - Mission Trades
  - Technology Maturity Overview
- **Technical Overview**
  - Mission Design & Conops
  - Flight System
  - Risk List
- **Development Schedule**
  - High-level Mission Schedule
- **Mission Life Cycle Cost**
  - Cost Methodology & Basis of Estimate
  - Cost Estimates for Options Considered

Mercury Decadal Study Team

Role	Name	Organization
Science Champion	Steven Hauck	Case Western
Deputy Science Champion	Allan Treiman	LPI
NASA HQ POC	George Tahu	NASA HQ
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GN&C	Dewey Adams	JHU/APL
	Mike Martini	NASA GRC
Software	Steve Williams	JHU/APL
SEP Configuration	Jon Drexler	NASA GRC

SCIENCE OBJECTIVES

Rationale

- **Decadal Survey is looking at Science and Mission priorities for the 2013-2022 timeframe.**
- ***MESSENGER* will complete its mission.**
- ***BepiColombo* hopefully will be launched by ESA.**
- **Following *MESSENGER*'s (and *BepiColombo*'s) reconnaissance of Mercury, operations on the surface are likely to yield answers to some of the highest-priority questions about Mercury.**
- **Bottom-line: <u>Is there a credible, science floor, landed mission to Mercury that is feasible and preferably fits in a PI-led cost-box (New Frontiers)?</u>**

Science Questions

1. What is the bulk composition of Mercury?
2. What is the nature of Mercury's magnetic field?
3. What is the history of Mercury's surface?
4. What is the internal structure of Mercury?
5. What is the character of surface – solar wind interactions on Mercury?

Science Objectives & Priorities

	Science Objective	Science Questions	Priority
A	What is the chemical composition of Mercury's surface (major and minor elements)?	1, 3	1
B	What is the mineralogy and structural state of the materials at Mercury's surface?	1, 3, 5	1
C	What is the magnitude and time-dependence of Mercury's magnetic field, at least for a point on the surface?	2, 4, 5	1
D	What is the character of geological activity (e.g., volcanism, tectonism, impact cratering) at scales ranging from regional to local (i.e., lander environment).	3	1
E	What is the rotational state of Mercury?	4	1
F	What is the magnitude and time-dependence of magnetic fields induced in the interior of Mercury?	2, 4	2

- **Priority 1 is the Science Floor, Priority 2 is additional science as architecture may allow.**
  - Priority 2 Science is no longer under consideration.
- **Objectives from Questionnaire document, but reordered and reworded to reflect refinements in prioritization.**

Payload – Robust Lander

- **Panoramic Stereo Camera**
- **Fluxgate Magnetometer**
- **Mini-Thermal Imaging Spectrometer (Mini-TES)**
- **Communications system for Radio Science**
- **Descent Imager**
- **Robotic Arm that hosts:**
  - Alpha Proton X-ray Spectrometer (APXS)
  - Raman Spectrometer
  - Microscopic Imager

Payload – Minimal Lander

- Panoramic Stereo Camera
- Fluxgate Magnetometer
- Alpha Proton X-ray Spectrometer (APXS)
- Raman Spectrometer
- Communications system for Radio Science
- Descent Imager

Payload Functional Requirements - 1

Science Objective	Measurement	Instruments	Functional Requirements
What is the chemical composition of Mercury's surface (major and minor elements)?	Elemental measurement of surface materials	APXS	Surface contact Calibration target
What is the mineralogy and structural state of the materials at Mercury's surface?	Mineralogical measurement of surface materials	Raman Spectrometer Imaging/Spot Spectrometer Microscopic Imager	Raman: Clear view to, or near contact with, surface Spec: Near and far-field view of surface 360 degree MI: Near surface contact All: Imagery for targeting and context
What is the character of geological activity at scales ranging from regional to local?	Imagery of surface	Descent imager Stereo imager Imaging/spot spectrometer	Image collection during descent Stereo & Spec: Near and far-field view of surface 360 degree

Payload Functional Requirements - 2

Science Objective	Measurement	Instruments	Functional Requirements
What is the magnitude and time-dependence of Mercury's magnetic field, at least for a point on the surface?	Magnetic field strength in 3-directions with time	Fluxgate magnetometer	Magnetically clean spacecraft and deployment away from lander (boom)
What is the rotational state of Mercury?	Tracking of lander position with time	Communications subsystem	Communication with Earth
What is the character of surface-solar wind interactions on Mercury	Structural measurements of surface materials Magnetic field strength in 3-directions with time.	Raman Spectrometer Imaging/Spot Spectrometer Fluxgate Magnetometer	Raman: Clear view to, or near contact with, surface Spec: Near and far-field view of surface 360 degree Mag: Magnetically clean spacecraft and deployment away from lander (boom)

Payload Prioritization

- **Tier 1a – Highest priority, cannot be split**
  - Panoramic Stereo Camera
  - Alpha Proton X-ray Spectrometer
  - Communications Subsystem
- **Tier 1b – Highest priority, can be split**
  - Fluxgate Magnetometer
  - Raman Spectrometer
  - Descent Imager
- **Tier 2 – High priority**
  - Mini-TES
  - Microscopic imager

Robotic arm vs. no arm

- **Robotic arm (Robust Lander Case)**
  - Robotic arm does not require all possible degrees of freedom to meet science goals
  - Raman spectrometer (Wash U/JPL implementation)
    - Arm hosts fiber optics
  - Microscopic imager
  - APXS
- **No robotic arm (Minimal Lander Case)**
  - Raman spectrometer (Hawaii, standoff type)
  - APXS
    - must be extended to surface in at least one spot and have access to calibration target

Potential alternative instruments

- **Thermal camera (rather than visible-NIR) for descent imager – would partially alleviate the problem of very low Sun/long shadows.**
- **Imaging spectrometer – Vis-NIR**
  - Mercury so far shows little in the way of spectral features in the vis-near-IR (other than albedo and slope variations).
  - Long shadows and low solar incidence angles at likely landing sites greatly complicate calibration of VNIR observations, though active illumination could be considered.
  - Because of these two issues, a VNIR imaging spectrometer is not warranted.
- **A thermal-IR imaging spectrometer offers more promise**
  - Sun-facing and even flat surfaces would likely be hot enough to provide good signal.
  - *LRO* Diviner instrument is mapping mineralogical differences on the Moon.
  - Paul Lucey (Univ. of Hawaii) has a Sagnac thermal-IR imaging spectrometer that is very mature. See Lucey et al. (2008) *Appl. Opt.* paper.

Potential alternative instruments

- **New in-situ XRD/XRF instrument (contact only, no sample prep) could potentially replace APXS and Raman.**
  - Determines elemental composition and mineralogy.
  - Like Raman, would require multiple placements for "point counting" in order to get representative composition.
  - See Sarrazin et al. (2009) *LPSC* abstract.
  - In-situ XRD/XRF probably needs further proving that it can obtain the quantitative elemental composition and mineral abundances we desire.

- **Contact vs. Remote Raman Spectrometer**
  - Remote Raman instrument (Sharma/Hawaii) likey cannot measure opaque oxide minerals (e.g., ilmenite). *MESSENGER* and other data suggest that opaque phase(s) are a key constituent of the surface. The Washington Univ/JPL contact Raman has demonstrated ability to measure opaques.

HIGH-LEVEL MISSION CONCEPT

Concept Maturity Level

- **This study was conducted as a Concept Maturity Level of 3**

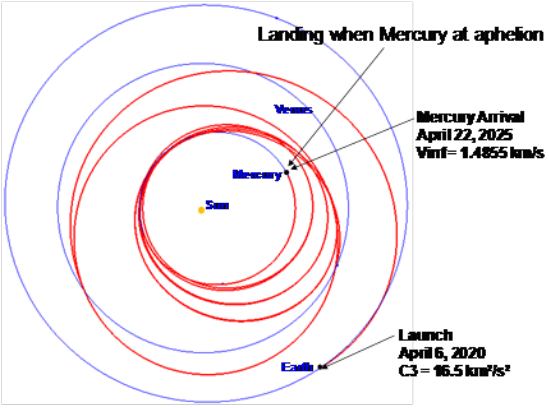
Concept Maturity Level	Definition	Attributes
CML 6	Final Implementation Concept	Requirements trace and schedule to subsystem level, grassroots cost, V&V approach for key areas
CML 5	Initial Implementation Concept	Detailed science traceability, defined relationships and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy
CML 4	Preferred Design Point	Point design to subsystem level mass, power, performance, cost, risk
CML 3	Trade Space	Architectures and objectives trade space evaluated for cost, risk, performance
CML 2	Initial Feasibility	Physics works, ballpark mass and cost
CML 1	Cocktail Napkin	Defined objectives and approaches, basic architecture concept

Basic Constraints and Assumptions

1. **Launch in the 2018-2023 timeframe**
2. **Landing site constrained by lander thermal design, high latitude should be the first-order target**
3. **Ultimate precision landing is not required**
4. **Direct-to-Earth communications is the first-order desire, however it is an open trade as to whether a relay spacecraft would be required**
   - If a relay spacecraft is required, the science it might carry is limited, e.g. an additional magnetometer
5. **Mission duration is modest. It is estimated that a two-week minimum duration could achieve the science goals, though mission length is an open trade**
6. **Landing site should be sunlit at landing and initial operations, though a duration that eventually passes into night is scientifically acceptable and would be benefit some experiments**
7. **Assume that RTGs are available if they are necessary to enable the mission**

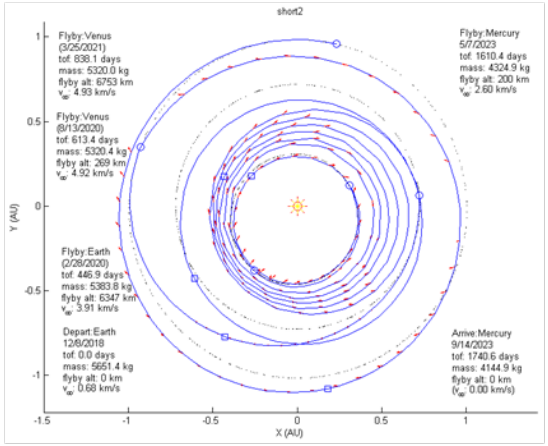
Mercury Landed Mission Overview

Ballistic Trajectory
- Launch C_3 = 17.5
- 2 Venus + 4 Mercury Flybys
- 5 years

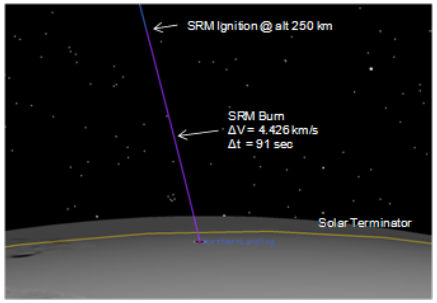


or

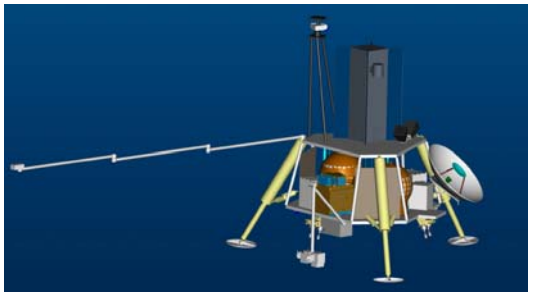
SEP Trajectory
- Launch C_3 = 0.46
- 1 Earth + 2 Venus + 1 Mercury Flybys
- 4.8 years



Direct Landing
- Solid Braking Stage
- Soft landing with Hazard Avoidance
- ΔV ~ 4,400 m/s



Landed Operations
- >22 days landed science
- Two days of limited daylight
- 100 Mbits daily science return



Primary Mission Challenges

- **Extremely high ΔV required to land on Mercury**
  - Depends on on V∞ at arrival
  - 4.4 – 4.7 km/s including finite burn penalty
  - 1.4 km/s (Ballistic) – 12.4 km/s (SEP) of additional cruise ΔV required
  - Basically launching a launch vehicle within a launch vehicle!
- **Propulsion**
  - High ΔV environment drives flight configuration to multiple stages
  - Strong need for high Isp and propellant mass fraction
  - Space storage issues
- **GN&C for landing**
  - Airless body with more than twice the gravity as the moon
  - Surface for safe landing not well characterized
- **Thermal environment**
  - Need to manage Mercury perihelion at 0.3 AU (11 Suns)
  - Trying to land and operate prior to sunset (Surface temp ~70deg C, 4 Suns incident)
  - Operate during Mercury night
- **Operate at low power on the surface**
  - High ΔV limits mass therefore limiting landed power
  - Only adequate mass for single ASRG

Mission Trade Space Examined-1

Mission Area	Options	Results
Landed Power Source	• Solar • ASRG	• Mission only feasible in low Sun or no Sun environment • Eternal points of light near poles theoretically possible but not reliable for practical mission concept
Landing Approach (Final Descent)	• Precision Navigation • Hazard Avoidance • Basic landing	• Precision navigation not required. Not adequate resolution of Mercury surface to pinpoint safe landing area a priori • Some level of basic hazard avoidance deemed necessary
Landing Approach (Touchdown)	• Soft landing with propulsion • Air bags	• Airbags are mass prohibitive • Propulsion needed anyways to take out large ΔV
Landing Lighting	• Sunlit • Dark	• Initial thermal analysis indicates the possibility of landing up to three days prior to sunset. Surface temps and incident Sun should be manageable.
Landed Communications	• Direct to Earth • Relay spacecraft	• Direct to Earth communications possible for required mission duration • Relay spacecraft is mass prohibitive and not considered feasible

Mission Trade Space Examined-2

Mission Area	Options	Results
Landing Location	• Many possibilities	• Mid-latitude and high latitude options may be feasible
Staging	• 2 Stage • 3 Stage	• Three stages necessary to meet mass constraints
Cruise Stage Propulsion (Chemical Trajectory)	• Pump fed + pressure fed bi-propellant system • Pressure fed bi-propellant system	• Pressure fed system provides a more compact design • Pump fed minimizes propulsion mass and has higher Isp providing about 68 kg of system mass savings
Braking Propulsion	• Solid rocket motor • Pressure fed liquid • Pump fed liquid	• Pressure fed liquid system does not provide the Isp or mass performance to meet mission needs • A pump fed liquid system provides significant improvement but is still not competitive with a solid in performance. It also is physically large and this pump fed engine has only been tested at sea-level • Solid meets mission needs but would need to be qualified for long space mission storage
Trajectory Approach/Propulsion	• Ballistic with 2 Venus and 4 Mercury flybys • Low thrust (SEP) with 1 Earth, 2 Venus, and 1 Mercury flybys	• Both options may be possible with differing constraints and risks • Both were defined and costed

ASRG Justification (1)

- Severe thermal constraints are imposed on a lander operating on the surface of Mercury. These constraints drive the location (latitude) and solar elevation angle at the landing site. In this study, the trade analysis converged on landing at relatively high latitude (~70° N or S), two Earth days before sunset.

- Panoramic imaging and thermal spectroscopy of the landing site will be accomplished before sunset, but most of the in-situ measurements (by the APXS, Raman spectrometer, microscopic imager, magnetometer) will be performed in the dark during the ~22 Earth-day primary mission phase.

ASRG Justification (2)

- The thermal design demands for operation at low Sun elevation and during night preclude the use of solar panels for power. "Peaks of eternal light" (PEL) may exist near the mercurian poles, similar to locations mapped on the Moon. It is conceivable that a lander targeted to a PEL could itself land in darkness, and deploy a solar panel on a mast to a height sufficient to catch the Sun on the horizon. However, current and future knowledge of Mercury's topography (at the scale of a lander) is insufficient for identification of such small locations.

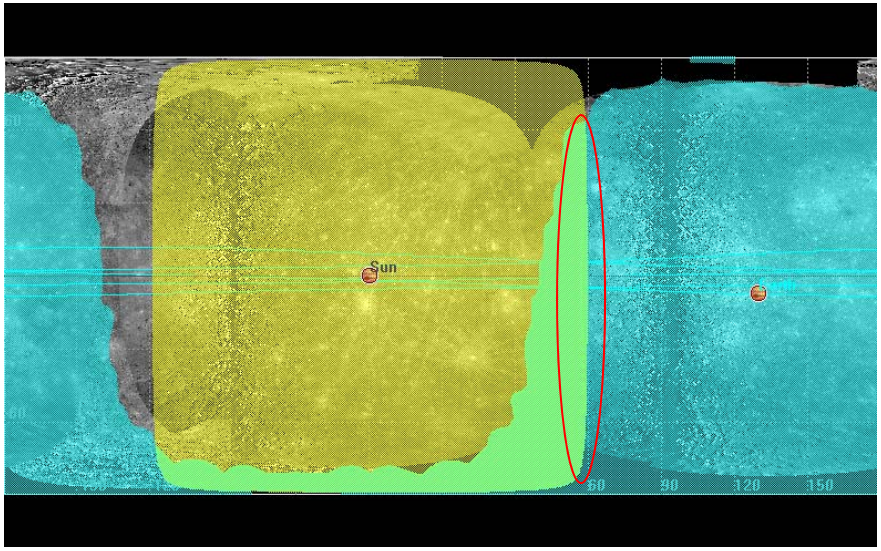
- Further, peaks of eternal light are likely to be found in areas of rugged topography that are unfavorable in terms of risk for safe landing. Guidance and navigation uncertainties on performing a highly pinpoint landing are yet another major challenge for a PEL landing. Therefore, a nuclear power source is the only option for the Mercury lander.

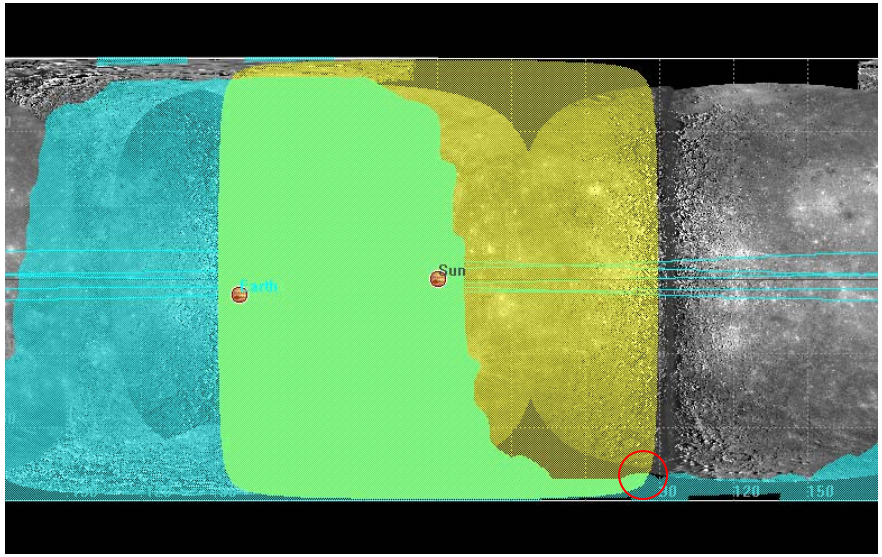
Airbag vs Soft Landing Trade

Considerations	Soft Landing	Airbag Landing
Landing location	Can precisely control landing location	Less precision in landing location
Landing orientation	Lands in controlled orientation	Lands in any orientation & **requires reorientation of lander after impact**
Terrain sensitivity	Sensitive to the nature of the terrain	Sensitive to the nature of the terrain
Mass	Requires DACS engines	Requires DACS engines plus **airbag system mass**
Science	For geochemical payload, may contaminate landing site	No chemical contamination, but may end up in a terrain depression

- Airbags not mass-efficient
  - Mass savings on fuel do not offset the additional mass required for an airbag system.
- Airbag landing prevents the ability to land in specific orientation

Landing Site Trades



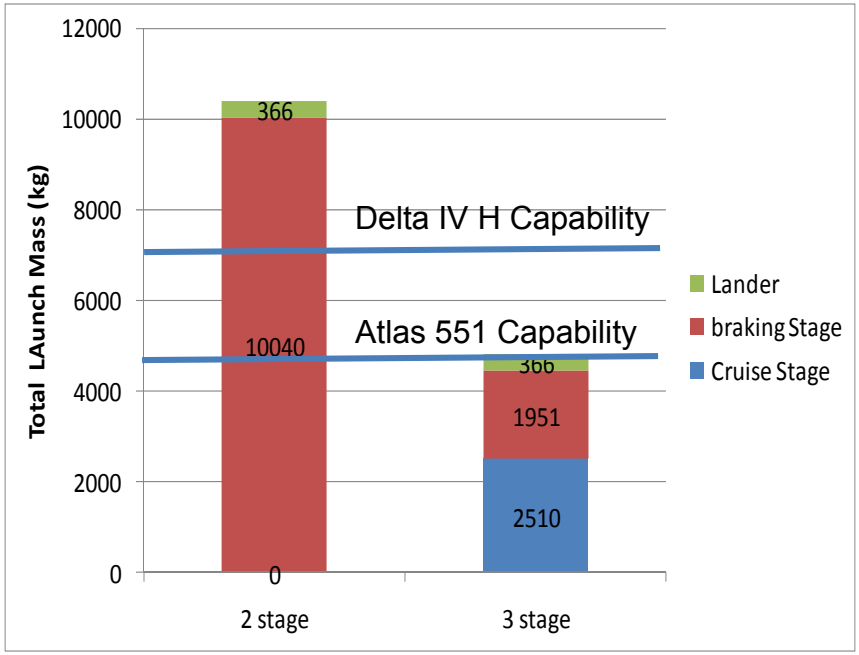


- **Seasonal Opportunities Based on Sun-Earth-Mercury Geometry**
  - Allows for large range of latitude landing sites
  - Available every 3 to 4 months
  - Window of opportunity ~ 50 days
- **Daily Opportunities**
  - Limited to either north or south pole
  - Earth communication access switches poles every ~180 days depending on sub-Earth latitude

Blue: Earth Access
Yellow: Sun Access

Number of Stages Trade

- To achieve ~5800 m/s ΔV within reasonable mass and volume, a multiple stage spacecraft is required
- A trade was performed between a 2 and 3 stage vehicle
- A two stage vehicle would reduce propulsion cost and staging complexity if it could meet mass requirements
- For this high of ΔV required, a two stage vehicle is prohibitively heavy even if the cruise heat shield and solar panels are staged for the braking burn
- A three stage chemical system is near the edge of the Atlas 551 performance as developed in this study with a robust instrument payload
- Further stage optimization may improve mass performance of both stages, although it is very unlikely based on this trade that a 2 stage vehicle will be able to fit in an Atlas 551



Mass Growth Sensitivity

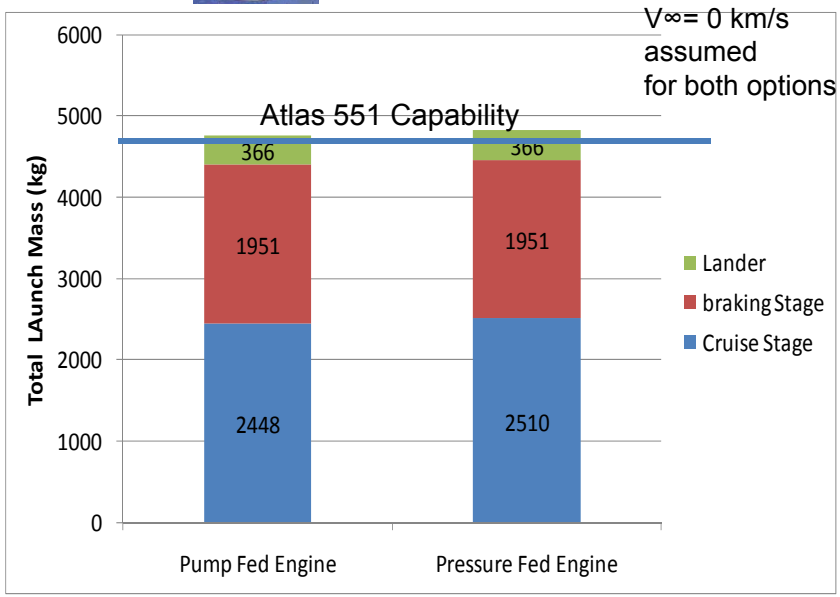
- Reduction of 1 kg of lander dry mass reduces total launch mass by ~8-11 kg
- Reduction of 1 kg of cruise stage dry mass reduces total launch mass ~2.5 kg
- Minimizing mass in the lander is the highest priority even if it means adding more mass to the cruise stage

Cruise Stage Propulsion Trade (Chemical/Ballistic Options)

- **A trade was performed to compare the benefit of using a higher thrust pump fed engine against using multiple lower thrust pressure fed engines**
  - Deep space maneuvers have traditionally been performed using pressure fed engines
- **The benefit of exploring the use of a pump fed engine is to reduce the tank mass with a lower operating pressure as well as a higher Isp of 340 sec vs 323 sec for the pressure fed system.**
- **A stage with a pump fed engine also requires a 2nd propulsion system (bi-propellant pressure fed) for ACS and small maneuvers adding significant complexity to this option.**
- **The overall propulsion system mass was considerably less for the pump-fed system, but because of the engine size, the spacecraft structure and heat shield had to grow significantly resulting in a launch mass benefit of about 62 kg**

Pump-Fed
1 XLR-132+ACS
• Isp = 340 s

Pressure-Fed
6 HiPAT+ACS
• Isp = 323 s



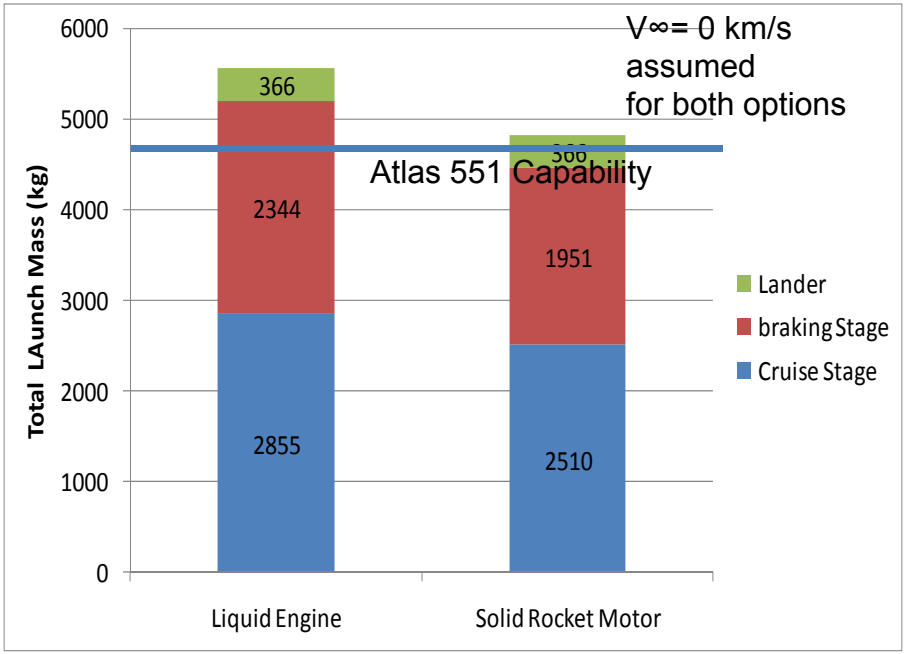
The pump fed option did not provide enough mass improvement to justify the additional complexity and cost. Both options require reduced lander payload to fit in A551

Braking Stage Trade

- **A trade was performed between using solid and liquid propulsion for the braking stage**
- **The braking stage is responsible for removing the majority of ΔV for landing (>4 km/s)**
- **The requirement for thrust to minimize finite burn penalty is in the 50-70 kN range**
- **The liquid propulsion is based off the Aestus RS 72 pump fed hypergolic engine.**
  - A pump fed system significantly reduces tank mass over a pressure fed system
  - A pump fed system has significant Isp advantages over a pressure fed system
  - This engine is currently designed for launch vehicle applications and would need to be qualified for deep space missions
- **The solid is based on the STAR 48 V**
- **The results favor the solid braking stage**
  - The high propellant mass fraction is of greater benefit than the higher Isp
  - The solid is also much more volumetrically compact reducing mass of other spacecraft components
  - The solid still needs to be qualified for long space mission durations

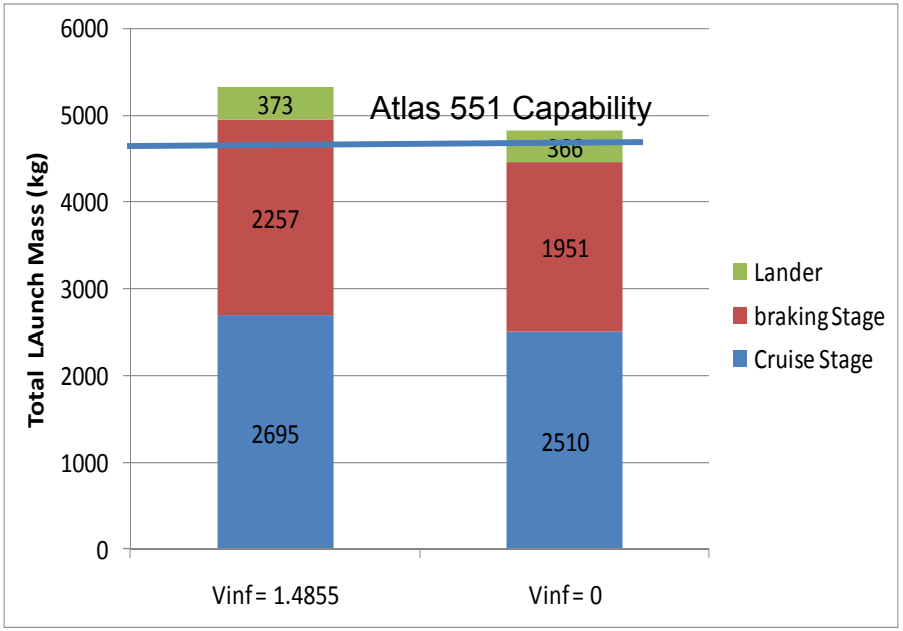
Aestus RS 72
- Isp = 340 s

STAR 48 V
- Isp = 292 s



Arrival V∞ Sensitivity (Ballistic Trajectory)

- **The arrival V∞ is an important parameter in identifying the required landing ΔV**
  - ΔV= 4.71 km/s for V∞=1.4855 km/s (Reference Trajectory)
  - ΔV= 4.42 km/s if V∞=0 km/s can be achieved
- **This ΔV difference applied to the concepts showed a reduction of 472 kg for the V∞ of 0 km/s each carrying the robust science payload**
- **Both concepts can be reduced to fit within the Atlas V 551 capability with the following assumptions**
  - V∞= 1.486: Minimum science payload, 25% margin
  - V∞= 0: Remove robotic arm and microscopic imager, 30% margin

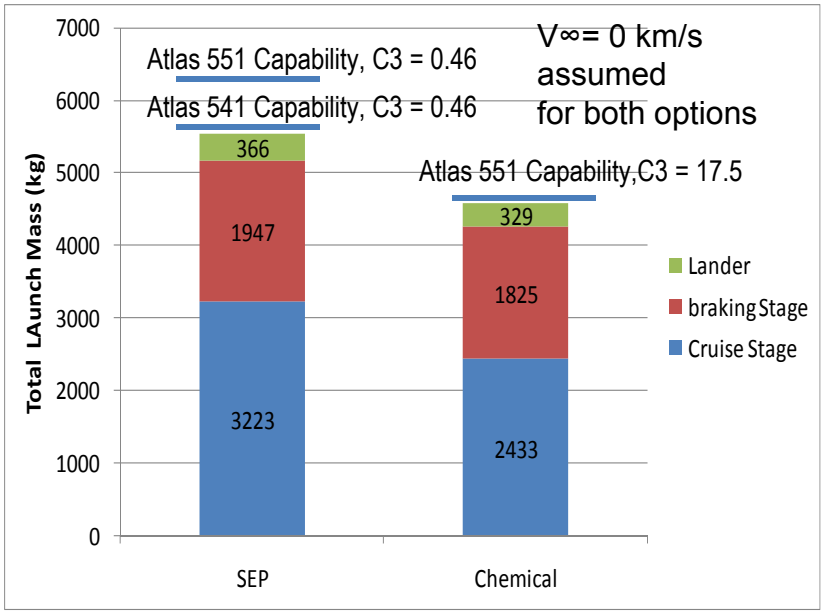


The trajectory should be optimized to reduce V∞ as low as possible to maximize science payload and reduce mass risk given constraints in cruise ΔV

SEP/Ballistic Chemical Cruise Trades (Technical Considerations)

- SEP case is based on the reference trajectory.
- Chemical case shown assumes example of V∞ of 0 (not reference case)

Metric	SEP	Chemical
Instrument Payload (CBE)	37 kg + margin Robust Payload	21 kg Remove robotic arm and micro imager Requires stand-off Raman
Launch C_3	0.46 km2/s2	17.5 km2/s2
Launch Mass (with Margin)	5460 kg	4587 kg
Launch Vehicle	Atlas V 541	Atlas V 551
Cruise Duration	4.8 yrs	5.0 yrs
Primary Risks	• Mass risk on Solar array • Stage heat dissipation	• How well trajectory can be optimized to reduce mass



SEP offers the best payload capability if risks areas can be successfully mitigated. Both options should be looked at in more detail in further study

Concept Options Summary

- All masses include 30% margin

Parameters	Option 1	Option 2	Option3	Option 4	Option 5	Option 6
Instrument Payload	Robust	Robust	Robust	Robust	Reduced	Robust
Trajectory/Prop	Ballistic	Ballistic	Ballistic	Ballistic	Ballistic	Low Thrust
V∞ at Mercury Arrival	1.486	0	0	0	0	0
Cruise Stage Propulsion	Press. Bi-prop	Press. Bi-prop	Press. Bi-prop	Pump Bi-prop	Press. Bi-prop	SEP
Braking Stage Propulsion	Solid	Solid	Pump Bi-prop	Solid	Solid	Solid
Instruments Mass (kg)	53	53	53	53	30	53+
Lander Dry Mass (kg)	331	323	323	323	289	323
Lander Wet Mass (kg)	373	366	366	366	329	366
Braking Stage Dry Mass (kg)	181	181	351	181	181	181
Braking Stage Wet Mass (kg)	2257	1951	2344	1951	1813	1951
Cruise Stage Dry Mass (kg)	726	725	793	722	704	1486
Cruise Stage Wet Mass (kg)	2695	2510	2858	2448	2373	3223
Launch Mass (kg)	5325	4827	5568	4765	4515	5540
Launch Vehicle	Delta IV H	Delta IV H	Delta IV H	Delta IV H	Atlas V 551	Atlas V 541
Launch C3 km^2/s^2	17.5	17.5	17.5	17.5	17.5	0.46
Launch Vehicle Capability (kg)	6915	6915	6915	6915	4630	5770

Recommended Future Trades/Analysis

- **Continue to explore trajectory optimization for the ballistic case**
  - Improve arrival V∞
  - Improve cruise ΔV
  - Improve C3
- **Explore the viability of liquid propulsion for braking with SEP stage if long duration space qualification of solid stage becomes an issue**
  - It is less efficient than the solid but may still work with an ATLAS V 551
- **Further explore optimization of ΔV split braking stage and lander**
- **Analyze off nominal landing performance and hazard avoidance to develop a more refined estimate for lander ΔV and resulting propellant load**
  - A 10% margin was used for this study to cover these areas
- **Refine thermal estimate for landed configuration with incident sunlight**
  - Can this be managed by high temp MLI alone?
  - Does this assumption put restrictions on aphelion vs perihelion landing?
- **Explore possibility of using an ATLAS vehicle with 4 m fairing**
  - E.g. ATLAS 451 (Why isn't this possible?)

Possible Enabling Technologies

Technology	Need	TRL	Development Needed
Efficient Ka-band SSPA or TWTA	Mission is severely mass and power constrained. NASA encouraging K-band use.	3-7	Reduce mass and improve efficiency of existing technologies. Invest in low power TWTA and solid state amplifier technologies.
High speed graphics processing	Hazard avoidance algorithms using cameras and LIDARs need significant processing beyond normal space qualified processors.	4	Develop Co-processor based on high density FPGA technology. Demonstrate with algorithms.
Raman Spectrometer	Provide composition information considered necessary for mission success. Close contact version developed by Wash U and JPL would provide the best return, but requires a robotic arm.	3-4	Two options are being developed. Wash U/JPL has a space design prototype. University of Hawaii has done some field testing with non-space hardware. Both need to be advanced to TRL 6
Lightweight/Low Power LIDAR	LIDAR is needed for hazard avoidance, altimetry, and local slope calculations necessary to assure safe landing. Mission is very mass and power constrained	3-7	Different LIDARs are in various stages of development. Improvements need to be made to minimize power and mass for this type of application.
Solid rocket qualification for long space storage	The rocket motor itself is not new technology but it is not currently qualified for long duration missions	5	No technology development needed. Needs to be qualified for this type of mission.
High temperature, high power density solar arrays	An SEP concept depends on significant power and cannot afford in mass to use demonstrated MESSENGER SA technology	4	High temperature cells have been demonstrated. Array level design and demonstration needs to be performed.
Bi-Propellant pump fed engine for long term space use	May provide some mass savings if used in cruise stage. May be easier to approve than solid for long space storage.	5	Qualify engine for mission environment
DoD DACS thrusters for Lander use	Provides high thrust to weight needed for mission	5	Qualify thrusters for mission environment

Technical Findings Summary (1)

- A landed mission on Mercury is extremely challenging from a ΔV perspective
- To keep this mission even close to a PI led cost cap mission, the largest launch vehicle has to be limited to an ATLAS V 551
- A three stage vehicle is likely required to perform this mission
- A solid rocket motor provides the best braking stage solution and may be enabling for the mission

Technical Findings Summary (2)

- **A ballistic/chemical approach may be feasible but current analysis puts it on the edge of being able to fit within the constraints of an ATLAS V 551**
  - Currently requires a reduction in instrument payload closer to the minimum and margins depend on actual trajectory performance achieved
  - A more detailed study with additional trajectory and vehicle optimization may improve this some
- **An SEP cruise stage approach has the potential to offer more payload to the surface within the ATLAS V 551 constraints**
  - Current concept fits in ATLAS V 541 with the robust payload
  - This concept depends on the development of high density, high temperature solar arrays. Cell technology exists, but array level performance needs to be demonstrated
- **Current cost estimates put all options above a PI led mission cost cap**
  - A ballistic/chemical option is $1.2B if for cases that fit with an ATLAS V 551 including reserves in FY15$M
  - The SEP option is over $1.5B including reserves in FY15$M
  - A ballistic/chemical option on a Delta V H is approximately the same as the SEP option
- **More detailed study efforts are needed to further characterize the feasibility and determine the preferred implementation approach**
  - This study was severely limited in scope and time to develop the concepts

TECHNICAL OVERVIEW

Trajectory Development Assumptions

- **All trajectory options assume direct landing on Mercury**
  - Allows for solid braking stage
  - Not adequate time to explore orbital options

- **Minimizing V∞ at Mercury arrival provides the lowest ΔV requirement for landing**
  - V∞ = 0 km/s achieved for low thrust trajectory
  - ΔV= 1.4855 km/s achieved for reference ballistic case. V∞ approaching 0 km/s may be achievable for ballistic case with further optimized trajectory.

- **Ballistic Trajectory (Reference Case)**
  - 2 Venus + 4 Mercury flybys
  - 5.0 years

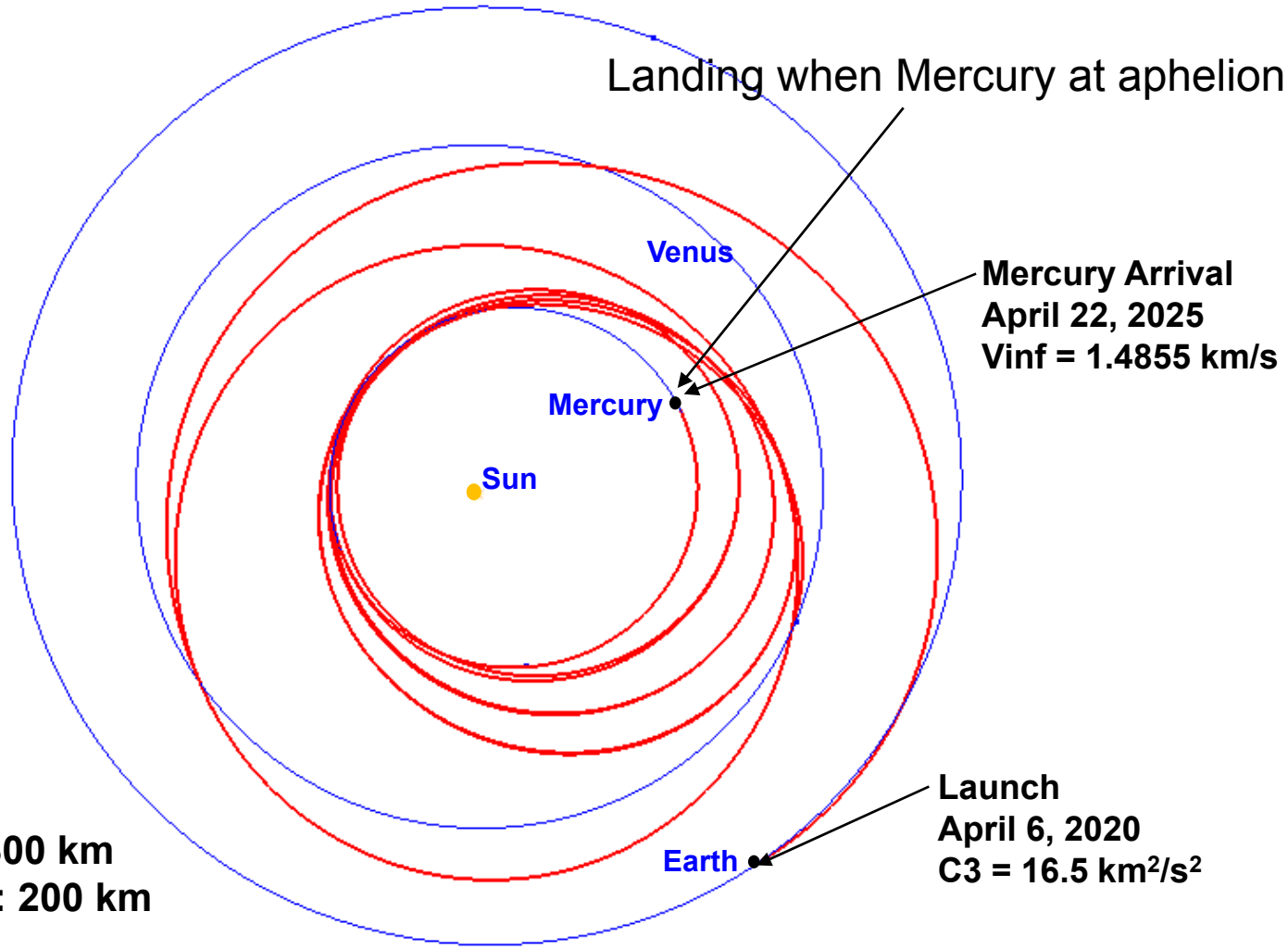
- **Low Thrust Trajectory (Reference Case)**
  - 1 Earth + 2 Venus + 1 Mercury flybys
  - 4.8 years

Ballistic Trajectory Approach (Reference Mission Design)

- **Launch**
  - Date: April 2020
  - C3: 16.5 km^2/s^2 , estimated Max C3 of 17.5 km^2/s^2 for a 20-day period
- **Trajectory to Mercury**
  - A ballistic trajectory including two Venus and four Mercury flybys
  - 3 deep space maneuvers, with a total Delta-V of 1238.2 m/s
- **Mercury Arrival**
  - Vinf: 1.4855 km/s
  - Solar distance: Mercury at aphelion
- **Landing**
  - Direct descent to surface
  - Landing site near latitude of 70 deg north
  - Direct-to-Earth communications for ~ 22 days
  - Landing at sunlit for 2 days, then passes into night
- **Mission Timeline**
  - Cruise (launch through Mercury arrival): 5.0 years
  - Landed operations: > 2 weeks

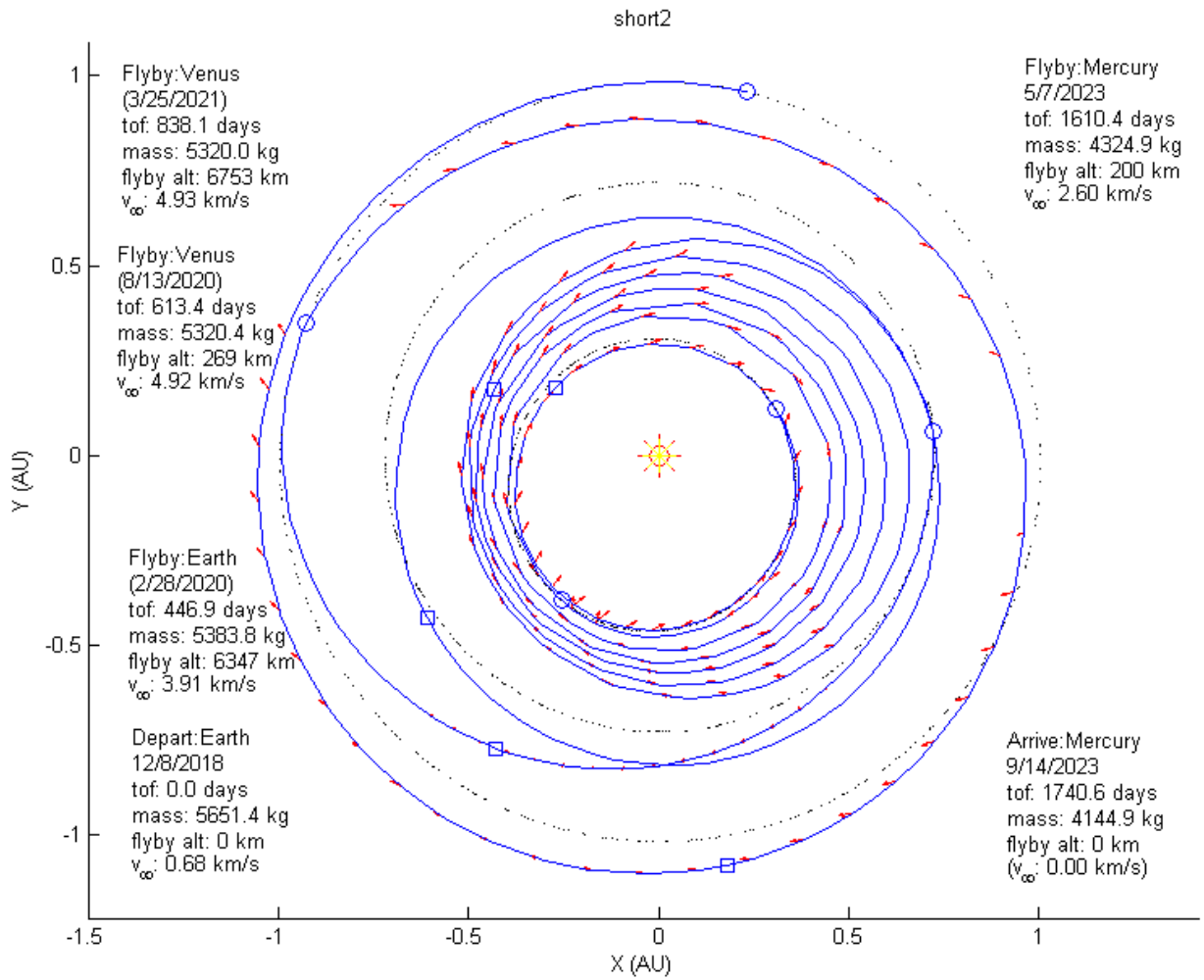
Ballistic Trajectory Approach (Reference Trajectory)

- Launch 4/6/2020
- Venus Flyby #1 10/14/2020
- Venus Flyby #2 8/10/2021
- Mercury Flyby #1 10/1/2021
- DSM-1 12/18/2021
- Mercury Flyby #2 6/24/2022
- DSM-2 8/27/2022
- Mercury Flyby #3 6/14/2023
- DSM-3 8/13/2023
- Mercury Flyby #4 3/6/2025
- Mercury Arrival 4/22/2025



Venus flyby altitude: 300 km
Mercury flyby altitude: 200 km

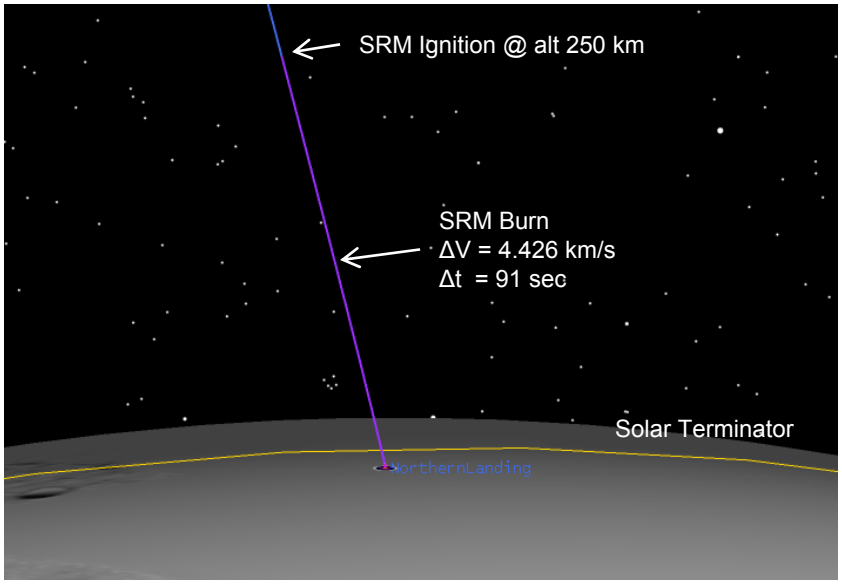
Low Thrust (SEP) (Reference Trajectory)

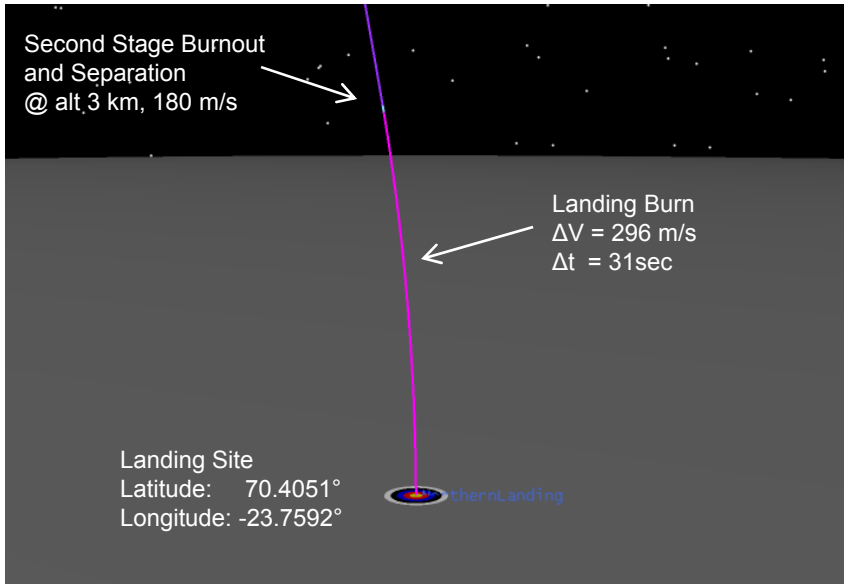


Four flybys (1 Earth, 2 Venus, 1 Mercury)
Total cruise duration: 4.8 years

Landing Approach

- **Solid rocket motor used for braking following a gravity turn profile**
  - TVC assumed to control attitude
- **Solid rocket motor is ejected and lander continues to soft landing using bipropellant propulsion system**
  - Optical sensors and LIDAR used for hazard avoidance
  - Lander touches down at ~ 1m/s

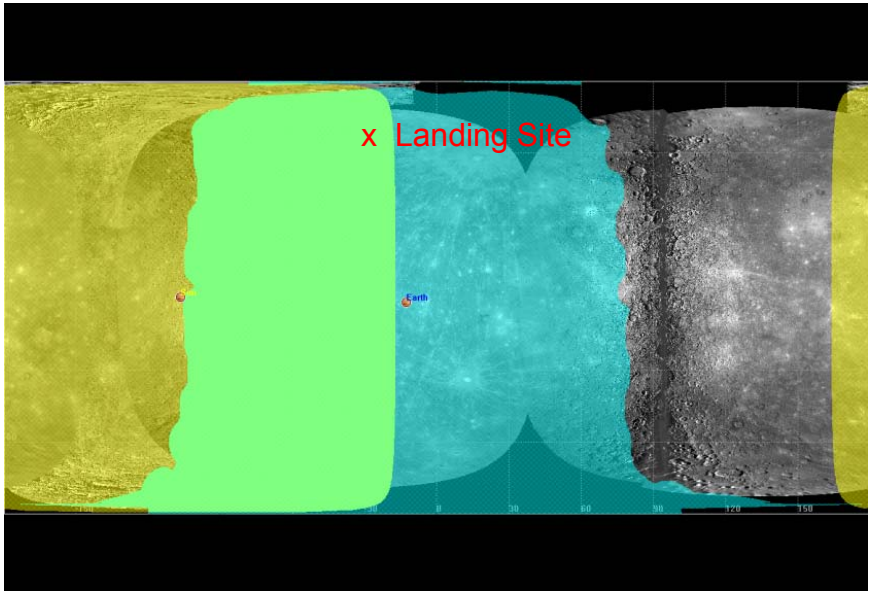




Graphics depict ballistic reference trajectory case. Other cases would differ in flight path, initial velocity, and altitude

Landing Conditions (Reference Trajectory Case)

- Latitude 70.4 deg, Longitude -23.8 deg
- Allows for 2 days prior to sunset
- Sun returns 88 days later
- Earth contact for first 22 days
- Earth contact resumes 29 days before sunrise



Ballistic Trajectory ΔV Requirement

Segment	V∞ = 0 km/s	V∞ = 1.4855 km/s
Cruise		
Deep Space Maneuvers	1238	1238
Navigation (Statistical)	124	124
Cruise Margin	38	38
Cruise Total	**1400**	**1400**
Landing	4420	4709
Total	**5820**	**6109**

Concept of Operations

- **Cruise assumed to be managed similar to MESSENGER**
  - Two 4 hour contacts per week for normal cruise operations
  - SEP will require a little more caretaking for propulsion and navigation operations
  - SEP engines need to be shut down prior to flybys
- **Landing**
  - Real-time communications managed by MGA on Lander
  - Once initialized, landing is autonomous
- **Landed Operations**
  - Contacts scheduled for 8 hours per day using DSN 34 m dish
  - Magnetometer will operate continuously throughout the mission
  - Other instruments will be operated serially starting with imagers while still in daylight
  - Primary mission ends after 22 days
  - Extended mission is possible until dawn but will require many days out of contact

Instrument Mass and Power

Instrument	Mass (kg)	Ave. Power (W)
Descent Imager (2)	0.3	2.2
Stereo Camera (2)	0.3	2.2
Microscopic Imager	0.4	12.9
Mini-TES	2.1	5.6
Camera Actuators	3.0	15 movement
Raman Spectrometer	3.8	15
APXS	1.5	4
Robotic Arm	15	45 movement
Magnetometer	0.2	0.5
Magnetometer Boom	3.0	-
Flash Lamps	3.0	Negligible due to low duty cycle
Instrument Component Electronics (5)	4.0	Included above
Total	37	

Instrument Data Volume Estimates

Data Product	Data per "frame" (Mbits)	Frames	Total Data Raw (Mbits)	Compression factor	Total compressed	Total for 8 bit images	Total for 8 bit/3-color Bayer images
3 color stereo pan (1024x1024 pix, 12 bits)	1.26E+01	436	5486.1	4	1371.5	914.4	457.2
2 Hz Mag in 14 days	5.65E-05	2419200	136.7	2	68.3	68.3	68.3
Additional images	1.26E+01	200	2516.6	4	629.1	419.4	419.4
APXS spectra	3.20E-02	10	0.3	1	0.3	0.3	0.3
Raman spectra	1.23E-02	200	2.5	4	0.6	0.6	0.6
100 MI images	1.26E+01	100	1258.3	4	314.6	209.7	209.7
Mini TES spectra	1.00E-03	500	0.5	4	0.1	0.1	0.1
Descent images (1024x1024 pix, 8 bits)	8.39E+00	50	419.4	4	104.9	104.9	104.9
	Total (Mbits)		9820.4		2489.5	1717.8	1260.6
	Days to Downlink				20	20	20
	Minimum Downlink Rate (Mbits per 8-hr Pass)				**124.5**	**85.9**	**63.0**

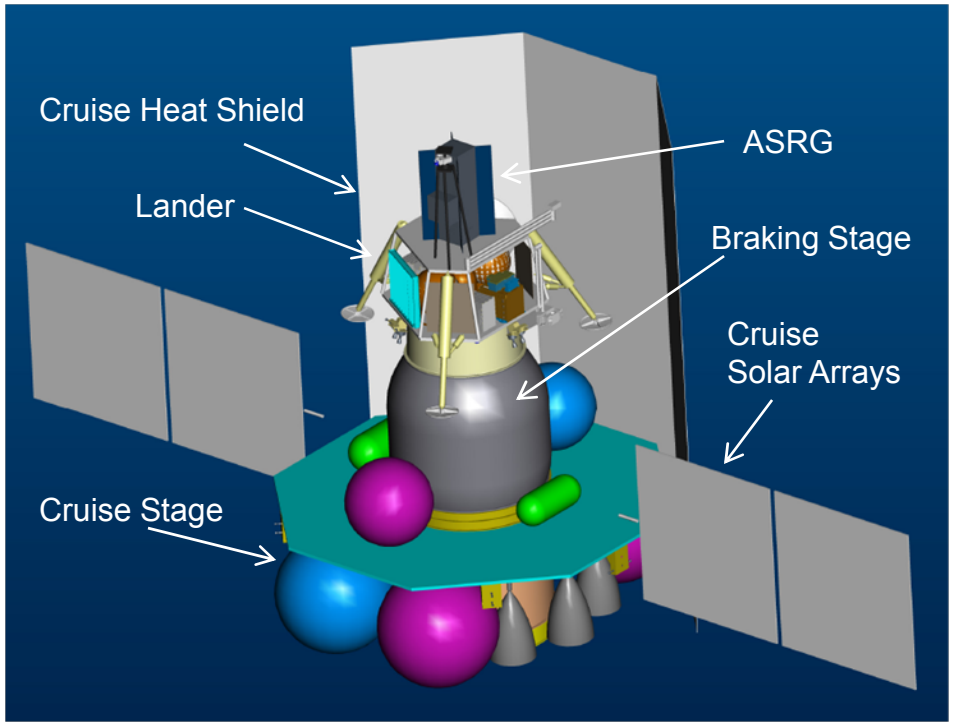
- **100 Mbits per day average downlink is sufficient for robust science**

Concept Architecture (Functional Breakdown)

- **Cruise Stage**
  - Propulsion and ACS reaction wheels for cruise phase including any TCMs
  - Thermally manage cruise and braking stages
  - Provide power for cruise propulsion, and thermal management of cruise and 2nd stages
  - Provide thermal protection for entire stack during inside 0.6 AU
  - Accommodate RF antennas as needed for cruise phase
- **2nd Stage**
  - Provide braking propulsion for direct Mercury landing
- **Lander**
  - Final descent propulsion for soft landing
  - C&DH and GN&C processing for entire mission
  - GN&C hardware for entire mission
  - Thermal management of lander for entire mission
  - Power to lander for entire mission
  - Accommodate instruments and science measurements

Lander Integrated with Cruise Stage (Chemical Propulsion)

- **Pressure fed bipropellant system**
  - Six 445 N engines +12 ACS engines
- **Solar Arrays based on MESSENGER technology**
- **Heat shield based on MESSENGER technology**
- **4 reaction wheels**
- **Sun Sensors**
- **X-Band Medium and low gain antennas for cruise communications**
- **Small Li-Ion battery for eclipses**
- **MLI insulation on braking stages that separates with cruise stage**

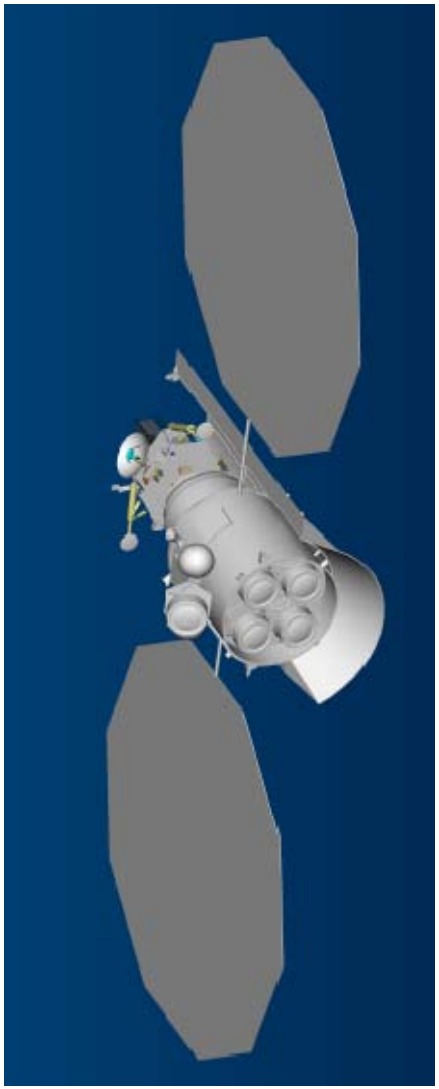


Chemical Cruise Stage Characteristics

Parameter	Summary/Value
Primary Structure	Aluminum, Aluminum-Li (SEP)
Cruise to Braking Stage Separation	4 point pyro separation
RF Hardware	X-band SSPA, 2LGA, 2 MGA
Cruise telemetry w/LGA	X-band, 720 bps
Cruise command w/LGA	X-band, 2000 bps
GN&C Hardware	Reaction Wheels (4), Sun Sensors
Attitude Determination During Cruise	Star Trackers – Inertial attitude, IMU – Rates, Sun sensors – safe-hold
Attitude Control During Cruise	3-Axis using reaction wheels +12 thrusters
TCM Engines	6 445 N thrusters, MMH-NTO, 323 s Isp
ACS Engines	12 thrusters 22 N each, MMH-NTO
Solar Array Power	780 W
Solar Array Type	High temp arrays based on MESSENGER technology
Solar Array Size	8 sq. m
Battery	Li-Ion, 8 A-Hr
Thermal Management	MLI, Heaters, software controlled
Propulsion Stage	+20°C to +40°C
Antennas	-50°C to +250°C
Solar Arrays	-150°C to +200°C

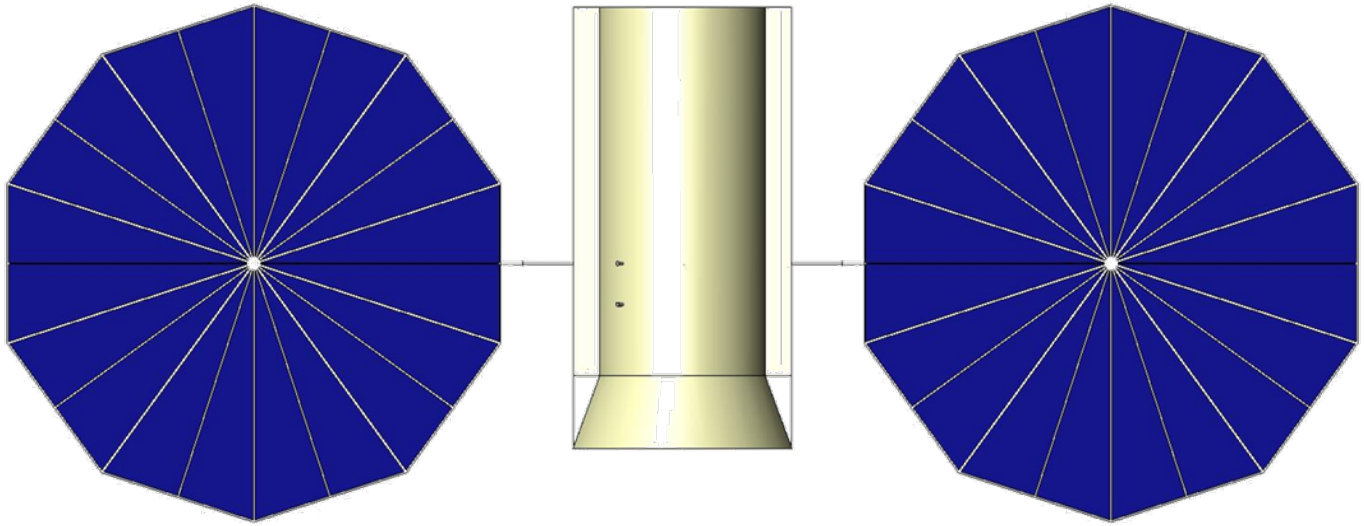
Lander Integrated with Cruise Stage (SEP)

- **4+1 gimbaled 7kW NEXT ion propulsion system**
  - ~1600 kg xenon propellant
  - Provides ACS during thrusting
- **Small Hydrazine RCS**
- **High temperature solar arrays**
  - 5.5 m diameter each
  - 30kW of power at Mercury (10.2 kW at 1 AU)
  - Based on GRC high temperature cell technology
  - Feathered up to 67° near Mercury
- **Heat shield based on MESSENGER technology**
- **Variable conductance heat pipes to radiator panels**
- **MLI insulation on braking stage that separates with cruise stage**
- **4 reaction wheels**
- **Sun Sensors**
- **X-Band Medium and low gain antennas for cruise communications**
- **Li-Ion battery for eclipses**

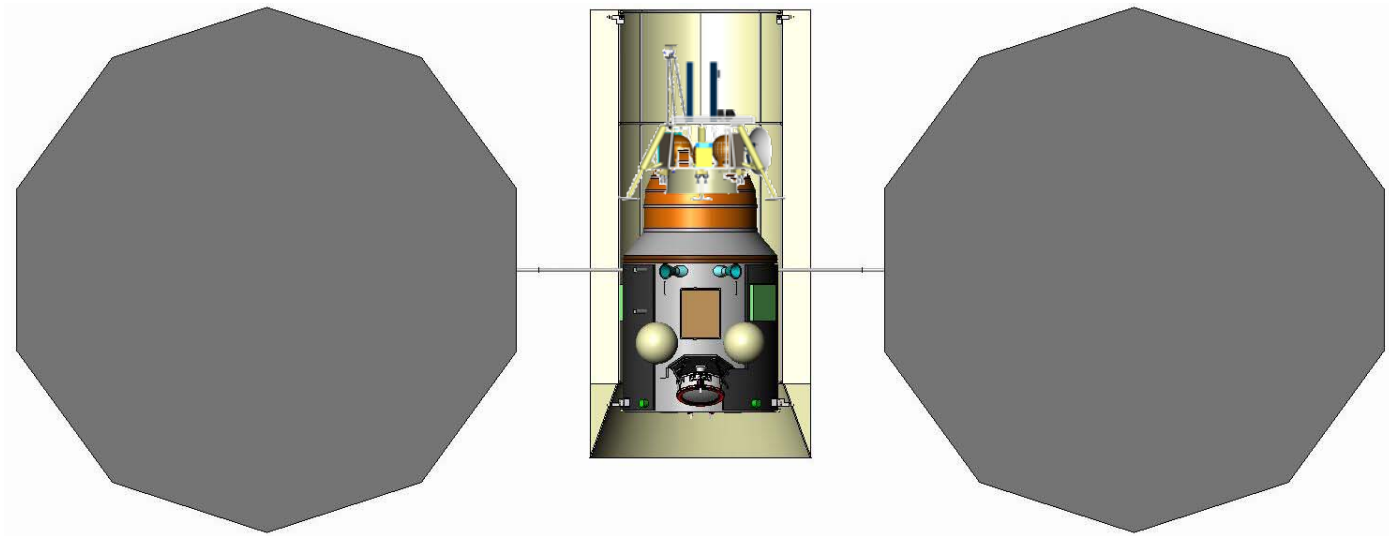


SEP Cruise Stage Concept

Sun Side



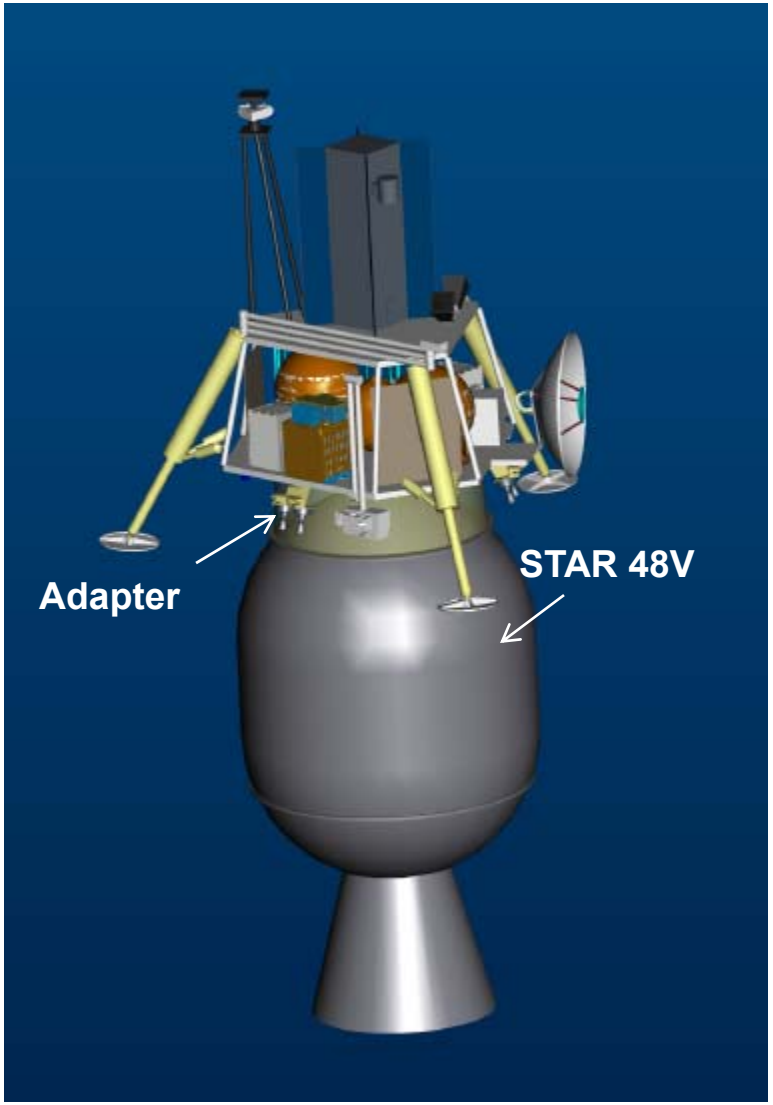
Shaded Side



SEP Cruise Stage Characteristics

Parameter	Summary/Value
Primary Structure	Aluminum, Aluminum-Li (SEP)
Cruise to Braking Stage Separation	4 point pyro separation
RF Hardware	X-band SSPA, 2LGA, 2 MGA
Cruise telemetry w/LGA	X-band, 720 bps
Cruise command w/LGA	X-band, 2000 bps
GN&C Hardware	Reaction Wheels (4), Sun Sensors
Attitude Determination During Cruise	Star Trackers – Inertial attitude, IMU – Rates, Sun sensors – safe-hold
Attitude Control During Cruise	3-Axis using NEXT engines, reaction wheels +12 thrusters
TCM Engines	NEXT 4+1 ion propulsion, Xenon propellant, 4100 s
ACS Engines	16 thrusters 4 N each, Hydrazine
Solar Array Power	10,400 - 31,350 W
Solar Array Type	High Temp Cells based on GRC tech, Array type based on Orion
Solar Array Size	2 arrays, circular 5.5 m diameter each
Battery	Li-Ion, 130 A-Hr
Thermal Management	MLI, variable conductance heat pipes
Propulsion Stage	+20°C to +40°C
Antennas	-50°C to +250°C
Solar Arrays	-150°C to +230°C

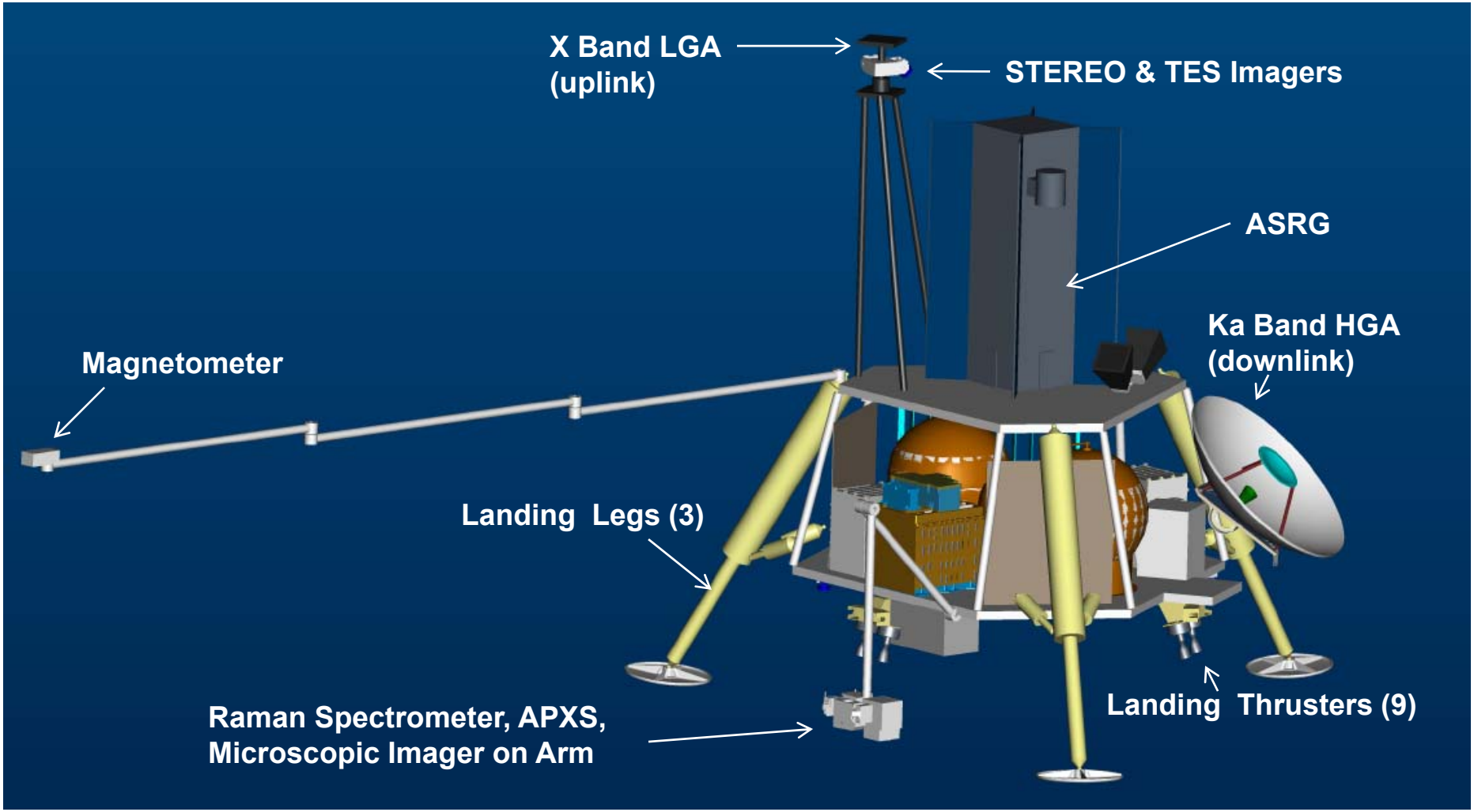
Lander With Braking Stage



Braking Stage Characteristics

Parameter	Summary/Value
Adapter	Aluminum, 4 point pyro separation
Landing telemetry w/MGA	16 bps, X-band
Landing command rate w/LGA	15.6 bps, X-band
Rocket Motor	STAR 48V (Custom loaded)
Propulsion Stage Thermal Range	+20°C to +40°C

Lander Concept (Deployed)



Lander Stage Characteristics

Primary Structure	Composite panels
Primary Structure	Composite Honeycomb Panels
Landing Legs	Based on Apollo, Aluminum with honeycomb to absorb energy
Robotic Arm	3 DOF, Instruments mounted side-by-side
Magnetometer Arm	3 m, multi-segmented, composite boom
RF Hardware	X/Ka-band coherent transceiver, SSPA or TWTA, HGA, LGA, MGA
Oscillator	Ultra-stable Oscillator (USO)
RF Power	8 W
HGA Diameter	60 cm
Landed telemetry w/HGA	3.5 kbps, Ka-band
Landed command w/LGA	31 bps, X-band
Processor	RAD 750 (133 MIPS)
Digital Signal Processor	High density FPGA based (>20MFLOPS)
Data Storage Capacity	32 Gb SDRAM
Landing Sensors	Star Trackers- Inertial attitude, IMU-attitude rates, Descent Imagers – surface relative rates, hazard avoidance, LIDAR- relative slope, hazard avoidance, surface relative altitude &altitude rate
ASRG Power	142 W – Cruise, 141 W- Landed
Battery	Li-Ion, 8 A-Hr
Landing Engines	9 Engines based on MDA DACs Technology, 445N each, MMH-MON-3
ACS Engines	9 Engines based on MDA DACs Technology, 22N each, MMH-MON-3
Thermal Management	“Thermos bottle” approach, Louvers, heaters for external instruments, high temp. MLI
Lander Bulk Temperature	+20°C to +40°C
ASRG (interface Temperatures)	0°C to +60°C

Mass and Power Margin Calculation

- **Margins are calculated using the Decadal Mission Study Ground Rules**
  - 30% using the following definition
  - Margin = Max Possible Resource Value – Proposed Resource Value
  - Margin (%) = $\frac{\text{Margin}}{\text{Max Possible Resource Value}}$ X 100
- **30% applied to all hardware with the following exceptions**
  - STAR 48V inert mass since it a known mass with a finite tolerance
  - Power use on NEXT PPTs. Assumed 5% uncertainty on efficiency similar to margins on DAWN
- **Note that this margin calculation method is significantly more conservative than JHU/APL practices, which uses 30% growth margin (divide by proposed resource value)**
- **For such high energy/low power missions it may be impractical to set margins at this high of level and introduce more rigor into the development process**
  - Using APL standards, the ballistic option would fit in the A551 with the robust payload

Mass Summary

Lander Stage	Est. Mass (kg)
Instruments	21
Mechanical	50
Propulsion	23
Avionics	11
Power	41
GN&C	11
Thermal	14
RF Communications	22
Harness	10
Total Dry (Estimated)	203
Total Dry (30% Reserve Margin)	289
Consumables (Propellant, Helium)	39
Total Wet (30% Margin)	329

Braking Stage	Est. Mass (kg)
Motor Case and Nozzle	154
Adapter, S&A, and break-up	19
Total Dry (Estimated)	173
Total Dry (30% margin –not motor)	181
Propellant	1632
Total Wet (30% margin)	1813

- **Chemical Propulsion Cruise Stage**
- **Reduced Payload – No robotic arm or microscopic imager**

Cruise Stage	Est. Mass (kg)
Mechanical	60
Propulsion	215
Avionics	0
Power	81
GN&C	28
Thermal	69
RF Communications	16
Harness	24
Total Dry (Estimated)	493
Total Dry (30% Reserve Margin)	704
Consumables (Propellant, Helium)	1669
Total Wet (30% Margin)	2373

Stack Mass	Est. Mass (kg)
Total Stack (30% Margin)	4515
Maximum Launch Mass ATLAS V 551	4630

Mass Summary

Lander Stage	Est. Mass (kg)
Instruments	37
Mechanical	55
Propulsion	23
Avionics	11
Power	41
GN&C	11
Thermal	14
RF Communications	22
Harness	12
Total Dry (Estimated)	226
Total Dry (30% Reserve Margin)	323
Consumables (Propellant, Helium)	43
Total Wet (30% Margin)	366

Braking Stage	Est. Mass (kg)
Motor Case and Nozzle	154
Adapter, S&A, and break-up	19
Total Dry (Estimated)	173
Total Dry (30% margin –not motor)	181
Propellant	1770
Total Wet (30% margin)	1951

- **SEP Cruise Stage**
- **Robust Payload**

Cruise Stage	Est. Mass (kg)
Mechanical	140
Propulsion	416
Avionics	18
Power	266
GN&C	16
Thermal	122
RF Communications	18
Harness	44
Total Dry (Estimated)	1040
Total Dry (30% Reserve Margin)	1486
Consumables (Xenon, Hydrazine, Helium)	1737
Total Wet (30% Margin)	3223

Stack Mass	Est. Mass (kg)
Total Stack (30% Margin)	5540
Maximum Launch Mass ATLAS V 541	5770

Power Summary

- ## Chemical – Cruise Stage

Cruise Stage - Chemical Option	Cruise	TCMs		Eclipse	
		SA	Battery		
Propulsion	0	0	148.4	0	
Power System	6.5	6.5		6.5	
Guidance, Navigation, and Control	32.7	32.7	0	32.7	
Thermal Control	430	430		0	
RF Communications	60	60		0	
Harness Loss (3%)	15.9	15.9	4.5	1.2	
Total Power Required	545.1	545.1	152.9	40.4	
Power Available	779	779	0.033	0.5800	Duration (hrs)
Margin	233.9	233.9	5.0	23.4	Total Load W-Hrs
%Margin	30%	30%	224	224	Battery Capactity W-Hrs
			2.25%	10%	Depth of Discharge

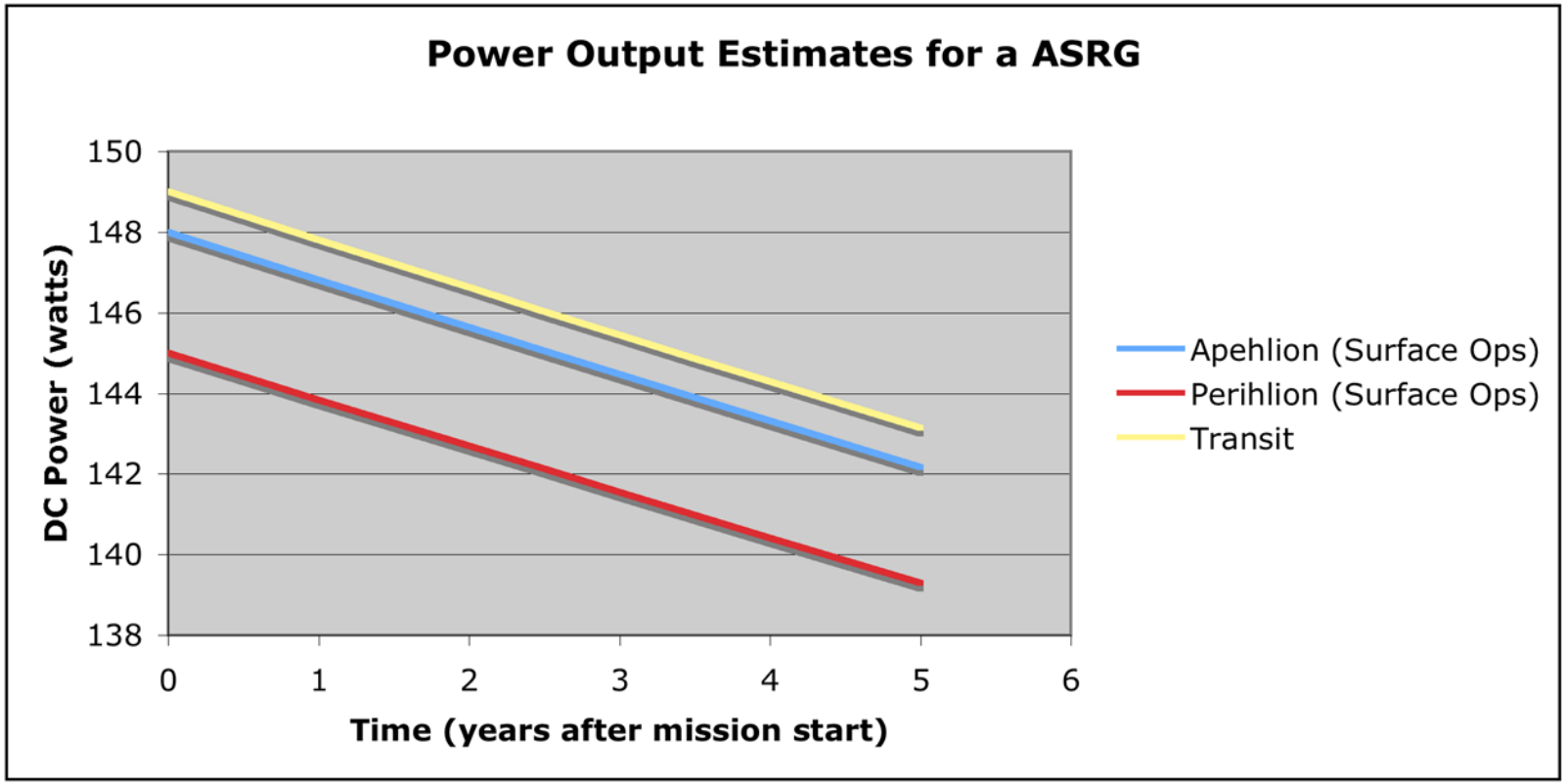
Power Summary

- ## SEP Cruise Stage

SEP	Cruise/ Flyby	EP Thrust	Eclipse	
EP Thruster Power				
Ion Engine	186	28854	0	
Harness Loss (1%)	1.9	288.5	0.0	
Total Power Required	187.9	29142.5	0.0	
Power Available	30677	30677	30677	
Margin (EP Thruster Power)	30489.1	1534.5	30677.0	
%Margin (EP Thruster Power)	99.39%	5.00%	100.00%	
Bus Power				
Power System	6.5	6.5	6.5	
Guidance, Navigation, and Control	32.7	32.7	32.7	
Thermal Control	354	354	80	
RF Communications	60	60	0	
Harness Loss (3%)	13.6	13.6	3.6	
Total Load	466.8	466.8	122.8	
Total After Power System Loss	547.6	547.6	127.9	
Power Available	781.9	781.9	28.63	Duration (hrs)
Margin (Bus Power)	234.4	234.4	3661.0	Total Load W-Hrs
%Margin (Bus Power)	30%	30%	4574.2	Battery Capacity W-Hrs
Total Power Available from S/A	31459	31459	80%	Depth of Discharge

ASRG Power Performance

- **Provided by GRC specific for Mercury surface environment at landing (Sun low in horizon)**
  - 850 deg C hot end temperature
  - 0.1 absorptivity radiator coating
  - Direct sunlight
  - Estimated surface temperature (21 deg C for perihelion case)



Power Summary

- **Lander**

Lander Subsystem	Cruise		Stereo Imaging	In Situ		Surface Comm.	Landing
	Receive only	Transmit/ Receive		13-hour science cycle	3-hour recharge		
Instruments	0	0	19.3	18.3	0.5	0	2.15
Propulsion	0	0	0	0	0	0	210
Avionics	24	24	21	21	21	19	35
Power System	22.8	22.8	22.8	22.8	22.8	22.8	22.8
Guidance, Navigation, and Control	7.6	7.6	0	0	0	0	56.6
Thermal Control	25	25	25	20	25	25	25
RF Communications	13	20	3	3	3	41.75	41.75
Harness Loss (3%)	2.8	3.0	2.7	2.6	2.2	3.3	11.8
Battery Recharge Load (W-Hrs)	14.1	0	0	51.1	72.7	0	0
Battery Recharge Power each hour (W)	4.2	0	0	11.1	24.2	0	0
Total Lander Load	99.4	102.4	93.8	98.7	98.7	111.8	405.1
Power Available from ASRG	142.0	142.0	141.0	141.0	141.0	141.0	141.0
Power used from battery	0	3.0	0	0	0	13.1	264.1
Duration (hrs)	0	4	0	0	0	8	0.0403
Total Load (W-Hrs)	0	11.9	0	0	0	104.9	10.643
Battery Capactity (W-Hrs)	224	224	224	224	224	224	224
Depth of Discharge	0.0%	5.3%	0.0%	0.0%	0.0%	46.8%	4.8%
Margin	42.6	42.6	47.2	42.3	42.3	42.3	0
% Margin	30%	30%	33%	30%	30%	30%	0%

Telecommunications Technology

- **Ka-band communications was chosen based on a NASA commitment to supporting K-band in the future as well as the mass constraints of the lander**
- **Since this mission is severely limited on power during landed operations, a high efficient Ka-band amplifier**
  - Output power is limited to about 20 W for an RF power of 8W
  - This level of efficiency currently does not exist in Ka-band systems
- **It is recommended to improve power efficiencies of existing technologies.**
  - Mass reduction of existing TWTA technologies to ~2.5 kg
  - Development of low power Ka-band TWTA technologies
  - Alternative high efficiency solid state technologies

Avionics Technology

- **Landing on a high gravity body such as Mercury requires significant real-time data processing if hazard avoidance is required**
  - Significant work has been done under the Constellation ALHAT effort for lunar applications
  - ~20 MFLOPS required based on that work
- **This is beyond current space-qualified general purpose processors**
- **A co-processor designed for this application is well within the capabilities of space-qualified hardware expected to be available in the next 5 years**
- **A co-processor based on a high density FPGA**
  - Single chip with several million gates exits
  - Funding exists for even higher density FPGAs
- **Algorithms need to be developed and tested together with the co-processor to demonstrate the capability**

Power Technology

- **SEP Arrays**
  - Assume high temperature cells based on GRC technology
  - Cells tested to 230° C
  - High temperature arrays using these cells have not been developed and tested
  - Technology development should focus on an a high temperature, high density array design that uses this or similar technology and maintain the lightweight characteristics of the array currently being developed for the Orion Spacecraft

Propulsion Technology

- **SEP**
  - NEXT engine technology is currently considered at TRL-6 and ready for flight program development
- **Pump-fed Bi-propellant Engines**
  - Not baselined in the trades but may improve efficiency and mass in cruise stage
  - XRL-132 engine developed by Air Force in 1980s, TRL 5 or 6
  - Further work may show applicability for Mercury mission
  - The mission may benefit if a smaller engine with the same technology was developed since the cruise stage did not greatly benefit from the 16 kN of thrust provided
- **Lander Propulsion**
  - High thrust to weight bi-propellant engines developed by the Missile Defense Agency (MDA)
  - TRL 5-6
  - Ongoing technology effort associated with ILN for Lunar landing
  - MON-25/MMH being evaluated with low inlet pressure
  - Thruster valve, thrust chamber, and nozzle could be modified for Mercury Mission
  - Thruster performance needs to be characterized with hot-fire tests at relevant environments

Propulsion Technology

- **Qualifying a STAR-48V for 6 year Cruise**
  - Previously, no solid rocket motor has been stored in space for this duration
  - Magellan used a STAR-48 that was fired 15 months after launch to capture into Venus orbit
  - LDEF provided long duration exposure (5 years, 9 months) of STAR motor materials with results showing favorably to long term space storage.
    - Ground based vacuum aging studies produced similar results
  - Risk could be managed by
    - Tight temperature controls on solid during cruise (assumed in study)
    - Plugging the nozzle to limit exposure to Vacuum environment
    - Heat the motor to near uniform temperature prior to firing
  - Not a new technology but the motor would need to be qualified for long duration space storage

Primary Risk Areas (1)

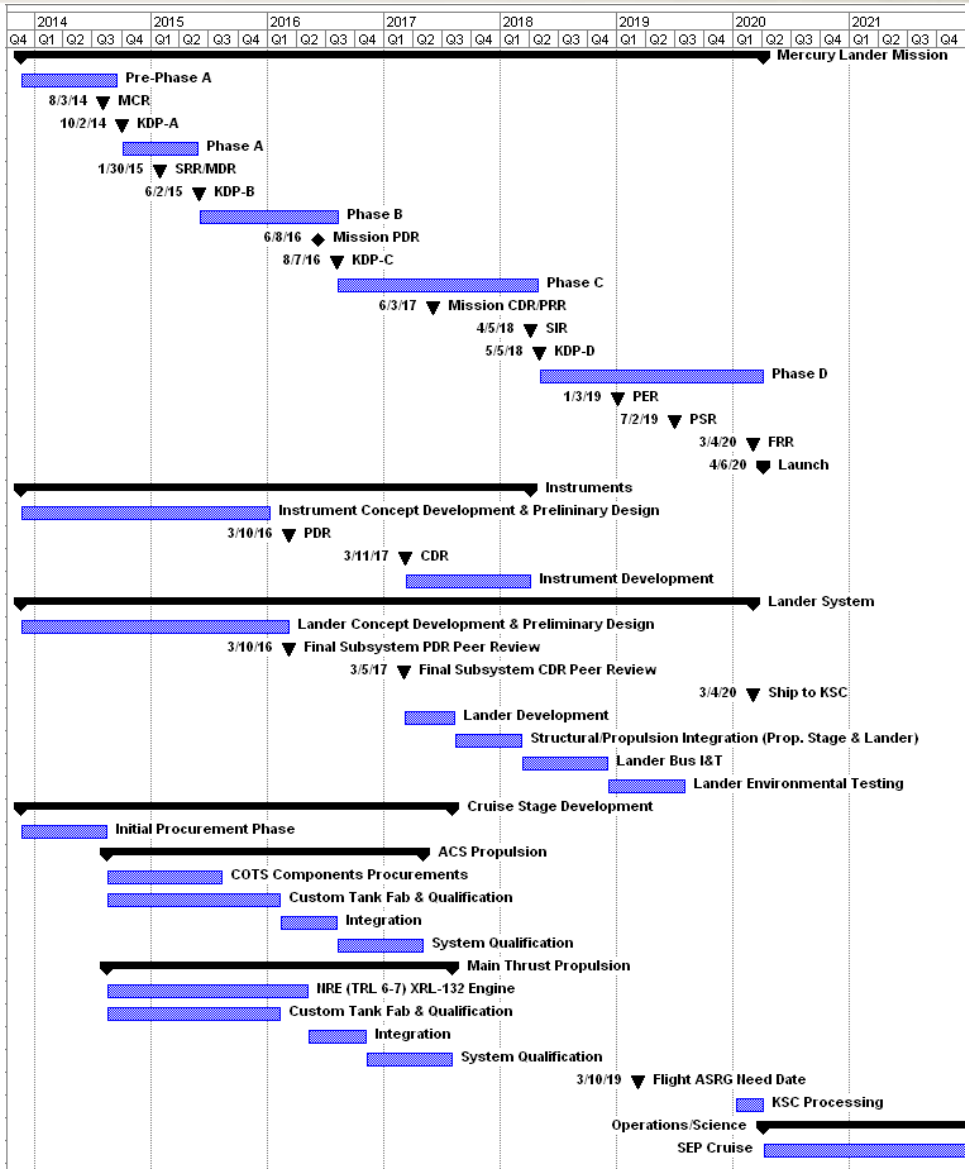
- **All Concept Options**
  - Soft landing with hazard avoidance
    - Potential Consequence- Mission cost and schedule could be impacted to ensure safe landing
  - Long term storage of solid rocket motor
    - Potential Consequence- If qualification not successful, options would limit to lower performance liquid option. Mission feasibility could be questioned.
  - Very limited uplink rate of 32bps for surface operations
    - Potential Consequence- May limit science operations. Operate at a lower cadence.
  - Raman spectrometer readiness
    - Potential Consequence- Cost and schedule impact or loss of science measurement
  - Complexity of 3-Stage System
    - Potential Consequence- Development cost and schedule impact
  - Thermal environment at landing
    - Potential Consequence- Mass penalty for additional protection, arrival restrictions (e.g. near aphelion)

Primary Risk Areas (2)

- **Ballistic/Chemical Option**
  - Level of performance improvement over reference trajectory case
    - Potential Consequence- Launch mass may be higher than $V_\infty = 0$ option evaluated in this study. Payload mass and mass margin could be affected.
  - Mass margin very tight since already carrying a reduced payload set to fit in ATLAS V 551
    - Potential Consequence – Grow into Delta IV Heavy
- **SEP**
  - Ability to develop high density, high-temperature solar arrays with performance significantly better than what has been demonstrated with MESSENGER using high temperature cell technologies that have been already developed
    - Potential Consequence- If performance improvements in array density and mass not achieved, the SEP approach is not feasible
  - Thermal management of a high power SEP stage
    - Potential Consequence – Mass and volume increase in stage

HIGH LEVEL MISSION SCHEDULE

High Level Mission Schedule



COST ESTIMATES

Ground Rules and Assumptions

- **Ground Rules and Assumptions derived from revision 2 draft of "Groundrules for Mission Concept Studies In Support of Planetary Decadal Survey (dGRPDS)**
- **Cost estimates presented in Fiscal Year 2015 (FY15) dollars**
  - Estimates generated in FY10 dollars
  - Prior-year costs adjusted to FY10 dollars using historical inflation rates
  - FY15 adjustment based on 2.7% annual inflation rate provided in dGRPDS
- **Cost Estimates cover Phases A-E, including**
  - Technology Development
  - Launch Vehicle and Services
  - ASRG procurement & integration
  - Cost Reserves
- **[Per dGRPDS] ASRG will be ready for flight no earlier than March 2014 at a unit cost of ~$20M [FY10], with $15M for nuclear launch compliance**

Ground Rules & Assumptions (con't)

- **Phase A: $2.5M, based on APL experience**
- **Education/Public Outreach based on 1% of mission cost**
- **Technology Development cost estimates cover effort to achieve TRL 6**
  - 50% cost reserves on Technology Development estimates
  - Instrument, spacecraft estimates exclude Tech. Dev. Estimates
- **Phase-E costs and DSN charges reflect Phase-E duration, number of flybys**
- ***Cost reserves posture based on Version 2 of dGRPDS, released September 21, 2009***
  - 50% reserves on Phase A-D costs except for launch vehicle and ASRG
  - No cost reserve on launch vehicle and services, ASRG
  - 30% reserves on Phase E costs, except 50% reserves on DSN charges
  - No reserves on E/PO

Cost Estimating Methodologies: Non-Spacecraft Hardware

Element	Method	Comments
Phase A	Engineering estimate	Based on APL experience
Technology Development	Engineering estimates, vendor ROMs	Quality of estimates varies widely
Management, Systems Engineering, S&MA	Cost factors using spacecraft hardware as basis, labor estimate (MD&A)	Factors based on MESSENGER, New Horizons actuals, RBSP trends
Science Team	Level of Effort, by phase (A-E)	Includes instrument planning
Payloads, Instruments	NICM estimates, engineering estimates	Robotic arm
Mission Operations	Cost factor (pre-launch spt.), engineering estimates (Phase E)	Phase E estimates adjusted for duration, #flybys
Launch Vehicle & Services	dGRPDS Ground Rules (LV, NEPA compliance)	Engineering estimate used for LV I/F engineering support
Ground Data Systems	Analogies to previous missions	
Flight Software	Engineering build-up	Includes development & test

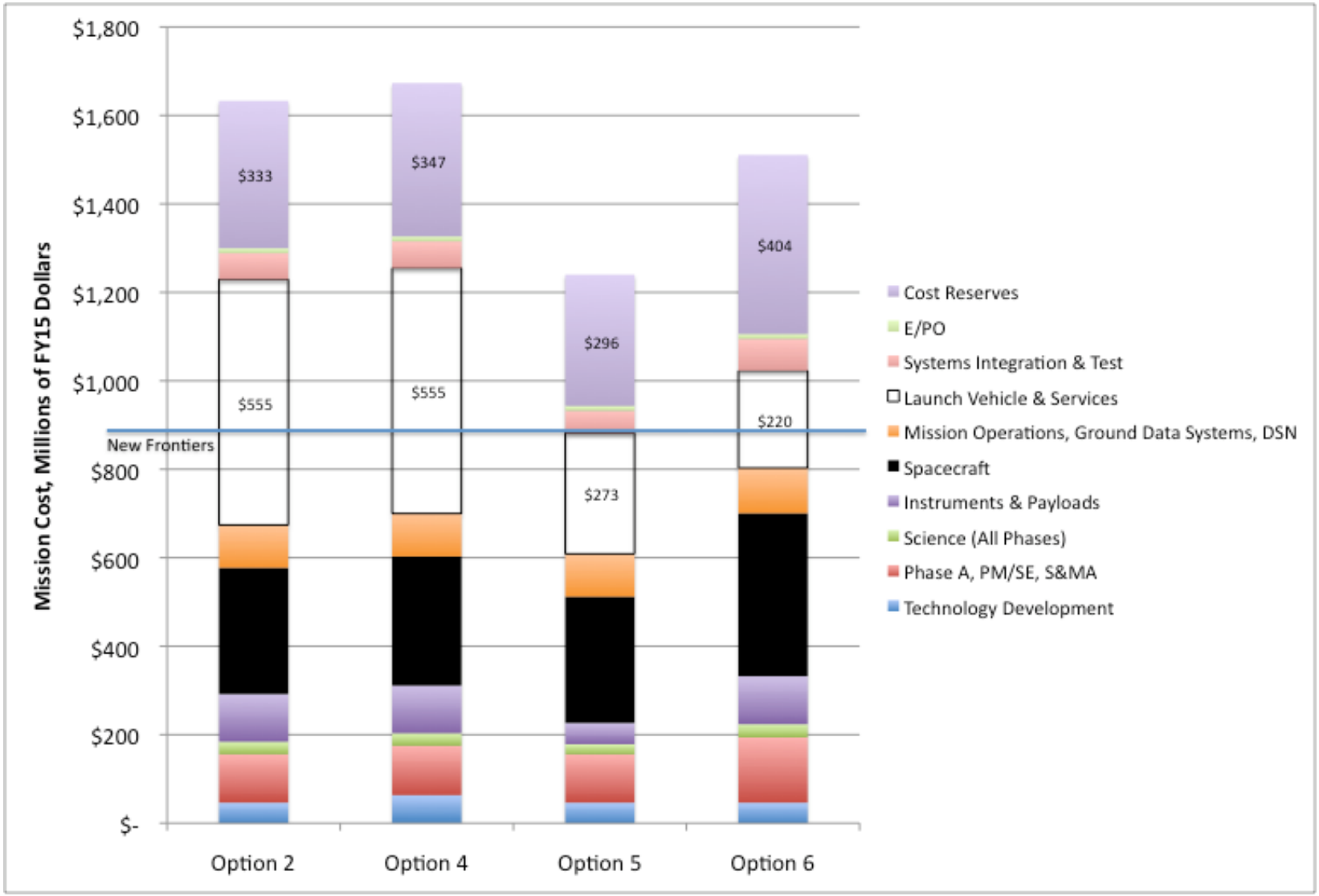
Cost Estimating Methodologies: Spacecraft Hardware

Element	Method	Comments
SEP Cruise Stage	NAFCOM parametric model	Hardware components only; GRC estimate
Mechanical & Structural	PRICE-H parametric model	Model originally developed & calibrated for ILN, LPV trade studies
Propulsion	Vendor ROMs, engineering estimates (oversight labor)	ROMs, labor estimates provided by
GN&C	Analogy to MESSENGER	Engineering estimate for LIDAR
IEM, Avionics, PSE, BME, Battery, PDU, Testbed hardware	PRICE-H, analogies, vendor ROMs (IEM, testbed h/w)	Estimates at board level, results checked against RBSP & launched mission actuals
Thermal Control	Analogies to MESSENGER	Includes cruise stage solar shield
RF Communications	Analogy to MESSENGER	SSPA requires tech. devel.
Integration & Test	Cost factor applied to spacecraft costs	Based on STEREO actuals and engineering analysis

Summary of Options Used in Cost Comparison

	Cruise Stage	Payload	Launcher
Option 2	Chem./Press. fed	Robust	Delta IV H
Option 4	Chem./Pump fed	Robust	Delta IV H
Option 5	Chem./Press. fed	Reduced	Atlas V 551
Option 6	SEP	Robust	Atlas V 541

Cost Comparison of Mission Options (In Millions of FY15 Dollars)



Cost Comparison of Mission Options NASA Level-2 WBS (In FY15$M)

Mercury Lander	Robust Science, Chemical HIPAT Cruise/Delta IV-Heavy LV	Robust Science, Chemical Pump-Fed Cruise/Delta IV-Heavy LV	Reduced Science, Chemical HIPAT Cruise/Atlas 551 LV	Robust Science, SEP Cruise/Atlas 541 LV
Description	**Option 2 FY15$M**	**Option 4 FY15$M**	**Option 5 FY15$M**	**Option 6 FY15$M**
Phase A	**3**	**3**	**3**	**3**
Enabling Technology Development	**50**	**68**	**51**	**51**
Project Management	**34**	**35**	**34**	**44**
Systems Engineering, incl. MD&A & Navigation	**49**	**50**	**49**	**67**
Safety & Mission Assurance	**23**	**23**	**23**	**33**
Science/Technology (Phases A-D)	**17**	**17**	**12**	**17**
Payloads	**108**	**108**	**48**	**109**
Spacecraft (1 lander)	**285**	**292**	**285**	**368**
Cruise Stage (Hardware)	102	108	102	185
SRM Stage (Hardware)	10	10	10	10
Lander (Hardware, excl. ASRG)	129	129	129	129
Flight Software (FSW), Autonony Development & Test	21	21	21	21
ASRG	23	23	23	23
Mission Operations	**69**	**70**	**68**	**79**
Launch Checkout, Early Operations Support (Phase D)	14	15	14	19
Mission Ops (Phase E) [excluding DSN]	55	55	54	60
Phase E Management, Eng. Support	*20*	*20*	*20*	*19*
Phase E Mission Ops, incl. GDS maint.	*19*	*19*	*19*	*18*
Phase E MD&A and Navigation Support	*4*	*4*	*4*	*10*
Phase E Science Team	*12*	*12*	*11*	*13*
Launch Vehicles & Services, incl. LVA, I/F, NEPA	**555**	**555**	**273**	**220**
Ground Data Systems	**16**	**16**	**16**	**16**
Systems Integration & Test	**59**	**60**	**50**	**72**
Space Communications Services (DSN)	**23**	**23**	**23**	**20**
E/PO	**11**	**11**	**11**	**11**
Subtotal	**1304**	**1331**	**948**	**1111**
Excluding ASRG, nuclear launch compliance	*1266*	*1293*	*910*	*1073*
Excluding LV, ASRG, nuclear launch compliance	*727*	*754*	*653*	*869*
Cost Reserves	**335**	**349**	**299**	**407**
Phases A-D (excl. LV, ASRG) : 50%	319	332	282	389
Phase E: 30%	17	17	16	18
Total, including Reserves	**1639**	**1680**	**1247**	**1517**

Estimated Technology Development Costs, without Reserves (FY15$M)

Mercury Lander	Robust Science, Chemical HIPAT Cruise/Delta IV-Heavy LV	Robust Science, Chemical Pump-Fed Cruise/Delta IV-Heavy LV	Reduced Science, Chemical HIPAT Cruise/Atlas 551 LV	Robust Science, SEP Cruise/Atlas 541 LV
Description	Option 2 FY15$M	Option 4 FY15$M	Option 5 FY15$M	Option 6 FY15$M
Phase A	**3**	**3**	**3**	**3**
Enabling Technology Development	**50**	**68**	**51**	**51**
Efficient Ka-band SSPA or TWTA	4	5	5	5
Xenon Flash Lamps	2	2	2	2
High-speed Graphics Processing (FPGA)	7	7	7	7
Raman Spectrometer	9	9	9	9
Lightweight/Low-Power LIDAR	2	2	2	2
Solid Rocket Qual. For Long In-Space Storage	4	4	4	4
Ultra-light High-temperature Solar Arrays	11	11	11	11
Bi-Prop Pump-Fed Engine Development	0	17	0	0
DoD DACS for Lander Use	12	12	12	12