Thermal Design of a Europa Lander Mission Concept

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Europa Lander is a mission concept that would explore the surface of Europa, which possesses an icy surface and likely subterranean ocean. The current objectives of the mission concept are to search for biosignatures, assess conditions for habitability, and support future exploration missions. The mission would leverage data gathered from the planned Europa Clipper mission, which would study Europa prior to the lander's arrival. The radiation environment at Europa would limit the planned surface phase of the mission to 20 days. NASA's Jet Propulsion Laboratory is designing the lander and descent stage, which assists with the de-orbit, descent, and landing (DDL) phase of the mission. Both the lander and descent stage are passive thermal designs. This paper will describe the overall thermal designs of the lander and descent stage, various trade studies, and key features of the designs to meet requirements. The information presented about the Europa Lander mission concept is pre-decisional and is provided for planning and discussion purposes only.

Nomenclature

AFT	=	allowable flight temperature
CBE	=	current best estimate
CS	=	carrier stage
DDL	=	deorbit descent landing
DOS	=	deorbit stage
DOV	=	deorbit vehicle
DS	=	descent stage
DSM	=	deep space maneuver
EGA	=	Earth gravity assist
EOI	=	Europa orbit insertion
GSFC	=	Goddard Spaceflight Center
ICEE	=	instrument concepts for Europa exploration
IR	=	infrared
JPL	=	Jet Propulsion Laboratory
MCR	=	mission concept review
MEV	=	maximum expected value
MGA	=	Mars gravity assist
MLI	=	multi-layer insulation
MSFC	=	Marshall Spaceflight Center
PDM	=	periapsis drop maneuver
PDV	=	powered descent vehicle
SLS	=	space launch system
SMAP	=	soil moisture active passive

I. Introduction

EUROPA was first identified in 1610 by Galileo Galilei. Since then, this Jovian moon has been imaged by the Pioneer 10, Pioneer 11, Voyager 1, Voyager 2, Galileo, and New Horizons missions. Images from Voyager 1

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and 2 were the first to offer possible evidence of an icy surface, which spawned a conversation about the possibility of life on Europa. Since then, the moon has been suspected of having an icy surface, a subterranean ocean, and a solid interior.[1] The Hubble Space Telescope has also detected multiple plume ejections from the surface of Europa.[2] Several future planned missions, such as Europa Clipper[3] or Jupiter Icy moons Explorer (JUICE),[4] are in development to study Europa for at least a portion of their respective project science objectives. However, the Europa Lander concept would be the first to study Europa from its surface. This paper will discuss the thermal design as of the mission concept review (MCR), which was successfully passed in October 2018.

II. Science Objectives and Mission Design

The primary mission science objectives for the conceptual Europa Lander mission are (in prioritized order) to search for biosignatures, assess habitability, and characterize the surface environment for future missions. The notional payload suite would chiefly include an organic compositional analyzer, a microscope, and a spectrometer.[5]

Due to the large amount of propellant needed to reach Jupiter and land on Europa, the total spacecraft mass would be roughly 15,500 kg. At this time, only the Space Launch System (SLS) rocket has the confirmed capability to enable the trajectory from Earth to Jupiter. 2026 is the planned launch year, and there is a backup launch opportunity in 2028. The baseline 2026 trajectory uses Earth and Mars gravity assists as part of a five year cruise to Jupiter (Figure 1A). Cruise would be followed by a two year tour around Europa (Figure 1B). Once the vehicle is in position to land on Europa, the lander vehicle would separate from the cruise stage to initiate the landing sequence. For the cruise stage, the driving hot case thermal environment is at 1 AU with solar, Earth IR, and Earth albedo loads. The driving cold case thermal environment is an eclipse during the Europa tour at 5.6 AU.

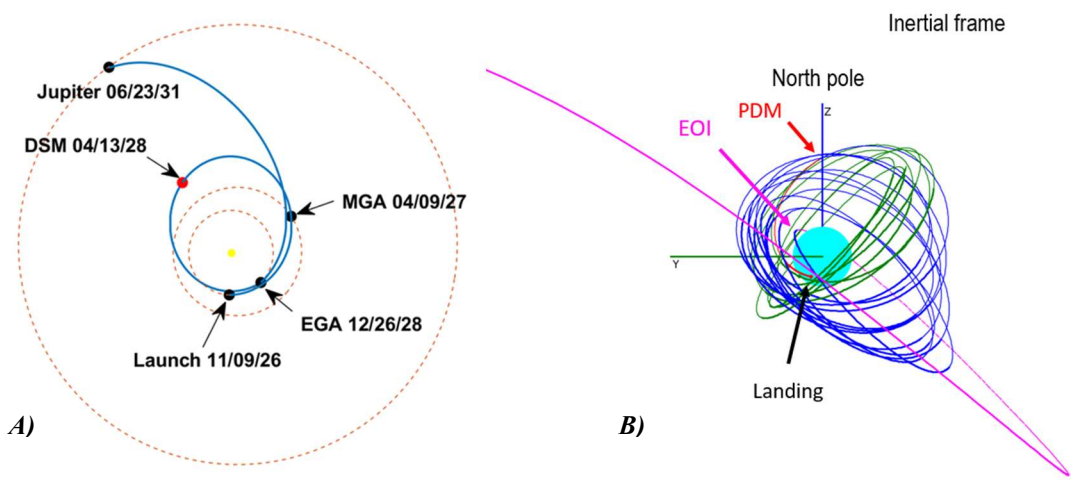


Figure 1. A) The mission trajectory for a 2026 launch. B) The two year Europa tour.

The deorbit, descent, and landing (DDL) sequence (Figure 2) borrows JPL design heritage from the Mars Science Laboratory and planned Mars 2020 missions. Unlike Mars, Europa has a thin atmposphere around 10^{-6} Pa,[6] which provides negligble aerobraking benefit. Therefore, the supersonic parachute associated with Mars landings would be replaced with a solid rocket motor. After the solid rocket stage burn, a descent stage equipped with thruster engines and navigational equipment would slow the lander to a low speed for touchdown. As the lander touches down on the surface, the four legs would independently detect contact and lock into place. Also included would be terrain relative navigation technology, which would debut during the landing sequence of the planned Mars 2020 mission.

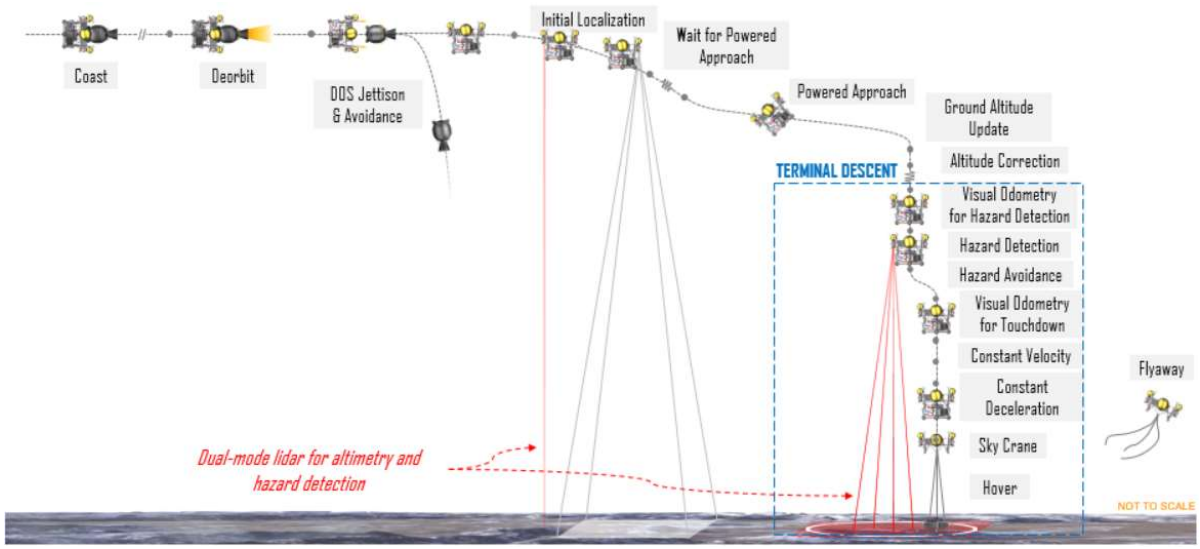


Figure 2. The deorbit, descent, and landing (DDL) sequence that transitions the lander from its Europa tour to the surface.

On the surface, the lander would excavate and collect at least three ice samples. Each sample must originate at least 10 cm below the surface and be at least 7 cm^3 in volume. The one-way light-time from Earth to Europa is approximately 40 minutes, which motivates the need for surface autonomy and efficient telecom transmission. Human ground-in-the-loop operations would be reserved for science evaluation and critical anomaly resolution. Telecom capability would be provided by a 0.7 m^2 high gain antenna, which enables direct-to-Earth communication. The lander would be battery powered, which, although it imposes a finite limit on mission time, is more attractive than solar arrays for this mission. To meet a roughly 50 W time-averaged surface power need at Europa with solar arrays would require almost 1 m^2 of cells (assuming 44 W/m^2 solar flux at 5.6 AU), but this assumes 100% efficiency, no radiation or micrometeriod damage, and sun always normal to the array. A more realistic size might exceed 5 m^2, which is more difficult to accommodate. A key disadvantage with solar arrays on this mission is that the capability would be degraded prior to surface phase by radiation during the two year Europa tour. Batteries, however, can be shielded from radiation by mechanical structure.

The limited time on the surface necessitates urgency for mission concept of operations, which is a contributing factor to the surface phase flight system design. By extension, the thermal design must support a rapid operating cadence (and avoid temperature requirement violations) while conserving survival heater power needed during quiescent sequences (to preserve the finite energy available for the mission).

III. Design Architecture

A. Overall Configuration

The Europa Lander concept is divided into several key assemblies (Figure 3). The spacecraft carrier stage (CS) used in cruise is primarily developed by the Johns Hopkins University Applied Physics Lab. This cruise stage vehicle would handle telecom duties during cruise and transport the the lander from Earth to Europa orbit. The carrier includes an ejectable "bio barrier" shield, which addresses planetary protection requirements. Ensconsed within the carrier stage is the deorbit vehicle (DOV), which itself is comprised of a solid rocket motor deorbit stage (DOS) and the powered descent vehicle (PDV). The PDV is made up of the descent stage (DS) and the lander, which together are responsible for executing the DDL manuever. NASA Goddard Spaceflight Center (GSFC) is cognizant of the carrier stage thermal design, which was the subject of a 2017 ICES paper.[7] The thermal design of the DOV is split between two entities: JPL for the powered descent vehicle (PDV) and NASA Marshall Spaceflight Center (MSFC) for the deorbit stage (DOS). Only the JPL portions of the Europa Lander thermal design will be discussed within this paper.

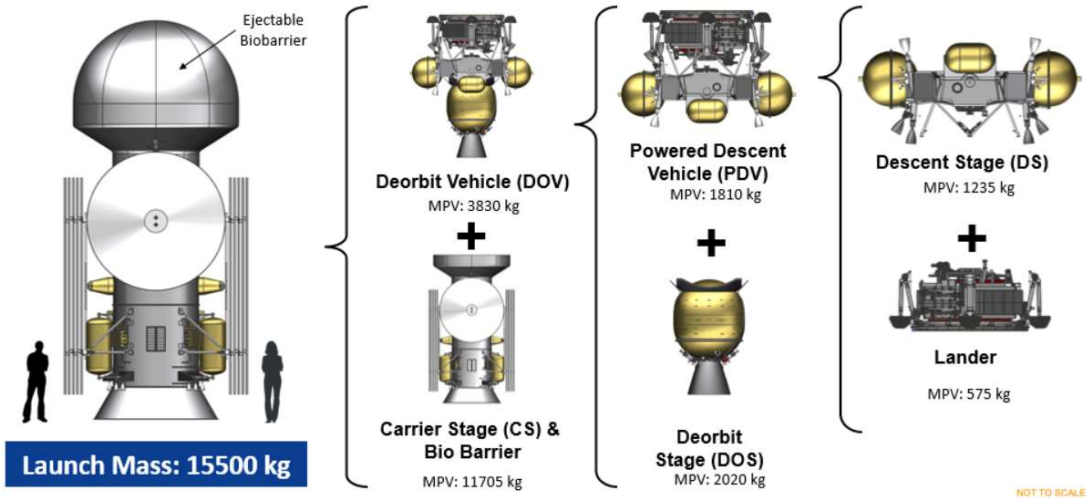


Figure 3. The mechanical configuration of the spacecraft concept with breakdown of stages.

The cruise stage of the mission would be powered by a solar array. The notional array size is 128 m^2, which is based on the planned Europa Clipper design but with the maximum possible area of solar cells. The nominal current best estimate (CBE) power draw by the spacecraft during cruise is 762 W, and the peak instantaneous CBE power draw is 1300 W. The carrier stage would primarily function as a telecom link during cruise and be largely devoid of "intelligent" electronics. The descent stage would contain its own propellant and electronics to execute DDL. Thermal batteries on the DS would provide short-term energy needed for the high power modes and thruster engine firing just before touchdown. On the surface, the lander would have its own high gain antenna and energy sources, which enable independence from the carrier. Radioisotope thermoelectric generators or radioisotope heater units were considered for the lander and cruise stage, but trade studies determined that batteries would be a better option.

B. Overview of Thermal Design Architecture and Methodology

All JPL portions of the present thermal design are passively controlled. The baseline design uses multi-layer insulation (MLI) blankets to conserve heater power, heaters to maintain survival temperatures, and thermal isolation via thin-walled struts or non-conductive interfaces. Notional allowable flight temperature (AFT) requirements for components were derived by analogy from previous JPL missions (Table 1). The majority of thermal modeling was done in ThermXL (a spreadsheet-based modeling tool) and detailed analysis, such as that needed for component level or temperature gradient analysis, was performed in Thermal Desktop®. Due to the transient nature of the mechanical configuration, particularly the lander, the use of spreadsheet tools was valuable for quickly assessing the feasibility of concepts.

Considerations for fault protection and off-sun maneuvers have not matured and are not considered in the thermal

Europa Lander Hardware	Temperature Requirements, °C											
	Allowable Flight				Protoflight/Qual				Flight Acceptance			
	Operational		Nonoperational		Operational		Nonoperational		Operational		Nonoperational	
	min	max	min	max	min	max	min	max	min	max	min	max
Lander												
Electronics	-40	50	-40	50	-55	70	-55	70	-45	55	-45	55
Batteries	0	70	-40	70	-15	90	-55	90	-5	75	-45	75
Cameras	-55	50	-100	50	-70	70	-115	70	-60	55	-105	55
Actuators	-55	50	-100	50	-70	70	-115	70	-60	55	-105	55
Descent Stage												
Electronics	-40	50	-40	50	-55	70	-55	70	-45	55	-45	55
Batteries	0	30	-20	40	-15	50	-35	60	-5	35	-25	45
DDL Brake	-120	80	-135	100	-135	100	-150	120	-125	85	-140	105
Propellant Tanks	15	50	15	50	0	70	0	70	10	55	10	55
Pressurant Tanks	-40	50	-40	50	-55	70	-55	70	-45	55	-45	55
Thruster Catbed	15	50	-40	50	0	70	-55	70	10	55	-45	55
Thruster Valve	15	50	0	50	0	70	-15	70	10	55	-5	55

Table 1. The notional temperature requirements table for the mission.

design to date. The current nominal orientation of the spacecraft places the DOV in a sun-shaded position during cruise. Increases in environmental heating may require a greater heat rejection capability along with a penalty to the heater power needed for cold case environments. Since the spacecraft orientation and fault protection requirements have not matured, the DOV thermal analysis done so far has focused on cold cases.

C. Deorbit Vehicle Thermal Design

For most of cruise, the DOV would be in a quiescent mode with steady state operating power. Therefore, the primary objective of the DOV thermal design is to minimize heater power needed to maintain components at survival temperature limits. The stressing case is to provide enough heat rejection for temperature compliance during high power modes in the leadup to and during DDL.

The PDV would be attached to the DOS with a conical titanium adapter ring. This interface provides a low thermal conductance of approximately 0.3 W/K, which limits heat transfer between the PDV and DOS and is useful for two reasons. First, the PDV is largely comprised of electronic components, which have temperature limits between -40 °C and +50 °C, whereas the DOS must be maintained between -2 °C and +10 °C. A non-conductive interface there reduces the parasitic heat leak that would increase heater power needed for the DOS. Second, the temperature of the DOS outer casing increases to roughly +500 °C during firing. Therefore, a weak thermal coupling between these two assemblies is desirable for both sides of the interface.

The lander and DS are connected by two sets of interfaces. There are eight 38 mm outer diameter, 1 mm thick titanium struts with thermal conductance on the order of 0.001 W/K each. There is also a coupling from the umbilical cable of the DDL brake to the lander. The thermal impact of this coupling is unknown, but it is assumed to be a negligible heat path. Due to the strong thermal isolation between them, the lander and descent stage can be treated as independent thermal designs, which is advantageous for producing a robust design.

The propellant and pressurant tanks achieve a similar thermal isolation from the descent stage with 38 mm outer diameter, 2 mm thick titanium struts. Miniature avionics vaults reside above the pressurant tanks and are isolated with 0.08 W/K interfaces each. Since the mini vaults are aluminum enclosures, heat generated by components is effectively spread thoughout the panels to reduce heater power.

The total CBE heater power needed by the PDV for a cold case cruise environment is predicted at 143 W. This analysis excludes solar heating (comparable to an eclipse scenario) and assumes poor performing MLI for conservatism. The propellant and pressurant tanks alone, which together are 5.4 m^2, account for 81 W of the total 143 W. A modified MLI scheme was modeled on the tanks to reduce heater power to

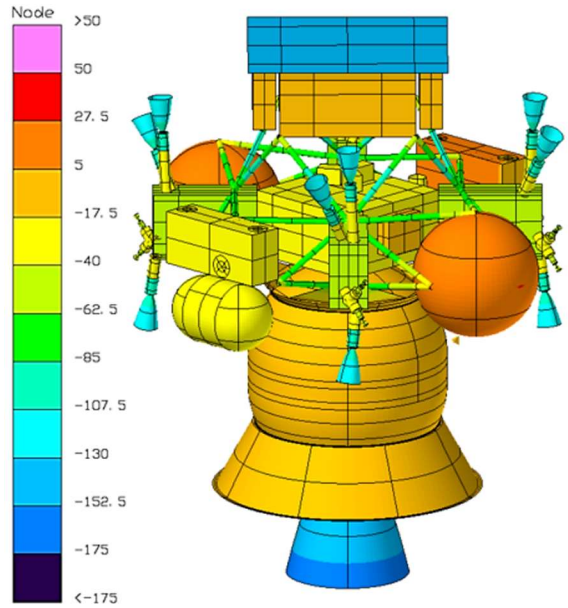


Figure 4. The DOV thermal model and prediction results for a worst case cold environment and low power dissipation.

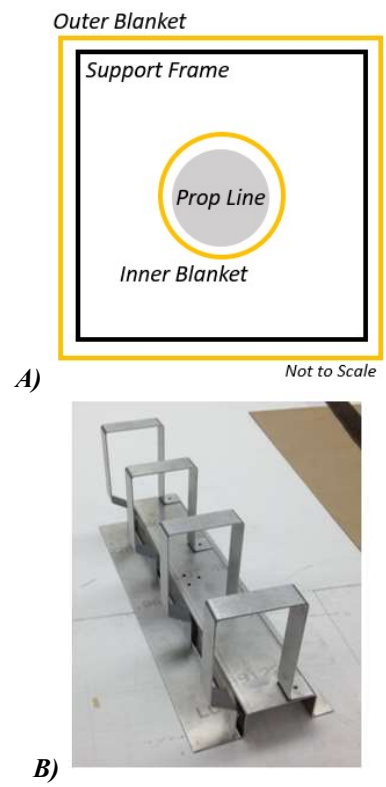


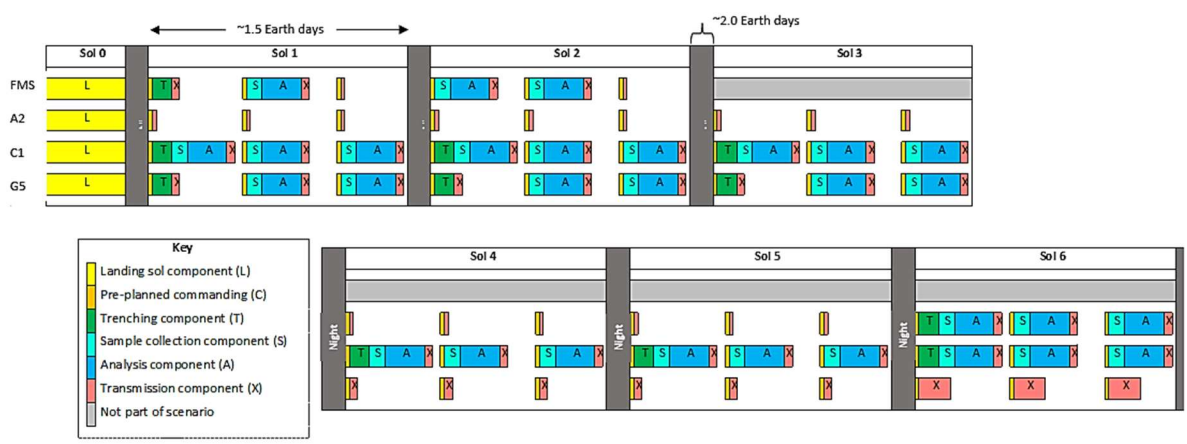
Figure 5. A) The proposed prop line MLI control scheme and B) a development unit tested for the SMAP mission.

these values. Instead of a standard 15 to 20 layer blanket, the tanks are modeled with two separated blankets in a dual MLI scheme: 5 layers in series with 15 layers and a two inch (5 cm) gap between the blankets. Previous development testing for the planned Europa Clipper mission demonstrated a ~50% reduction in heat loss with this blanket implementation.[8] In the DOV thermal model, the normal 20 layer blankets are modeled with $\varepsilon^* = 0.05$ and the separated dual blankets are modeled with overall $\varepsilon^* = 0.035$, which are both conservative assumptions for the cold cases. A non-traditional MLI design was also invoked for the propellant lines in which one MLI blanket is conformally wrapped around the lines and a second blanket is used around a frame enclosure structure surrounding the lines. A single MLI blanket conformally wrapped around prop lines could have large variability in performance, with an effective blanket emissivity ranging from 0.02 to 0.22.[9] For that range of blanket performance, a one inch (2.5 cm) diameter prop line would require 1.6 to 17.9 W/m of heater power. To improve performance, a blanketing scheme previously flown on the Soil Moisture Active Passive (SMAP) mission was invoked (Figure 5). If both the inner blanket conformal to the prop line and the outer blanket around the frame could optimistically achieve ε^* of 0.04, the SMAP scheme would require 1.9 W/m. For 25 meters of prop lines, of which roughly one-third would be heated in cruise, 16 W of heater power is required. This power savings does come at the expense of added mass, which will be evaluated once a notional prop line layout is created. A low ε tape is an alternative solution with less mass, although the Jovian radiation environment tends to preclude the use of adhesives.

So far, the DOV thermal design has accommodated hot case transients for electronics components by relying upon the lateral heat spreading capability of the aluminum vaults in which those components are contained. For instance, high power electronics dissipation during DDL has typically not produced AFT violations due to conduction through the aluminum panels of the vault. In the cases with AFT non-compliance, the thermal team has imposed constraints on operating duration. The flight system has responded by breaking up checkout periods (e.g. 4 hours of landing camera checkout with 60 W power) into several shorter checkout periods (e.g. two separate 2 hour checkouts). Due to the currently passive design architecture, there is less flexibility to accommodate large changes to the PDV design for hot case transients without incurring penalties for cold case heater power. Exploration of active thermal design features, such as louvers, radiators, or heat pipes, could be performed as part of future work.

D. Lander Thermal Design

The lander thermal design is largely driven by the flight system concept of operations and mechanical configuration. Typically, the flight system team architects a set of mission timelines (Figure 6) that each achieves full mission success (FMS) in different ways. These profiles are analyzed by the thermal subsystem to check compliance with AFT requirements and energy available. The majority of thermal modeling was done in ThermXL while detailed analysis, often for evaluating temperature gradients, was done in Thermal Desktop®. Due to the highly transient nature of the lander mechanical configuration at this early stage of the design, the use of spreadsheet tools and back of the envelope calculations was highly valuable for assessing the feasibility of concepts.



FMS = full mission success, A2 = cold case, C1 = hot case, G5 = sizing case

Figure 6. Sizing case surface phase mission concept of operations timelines. C1 is the hot case with maximum expected value (MEV) power profiles and the most ambitious mission activity. A2 is the cold case with minimal mission activity and current best estimate (CBE) power profiles. G5 is the driving case for battery sizing that exceeds full mission success (FMS) with a realistic operating cadence.

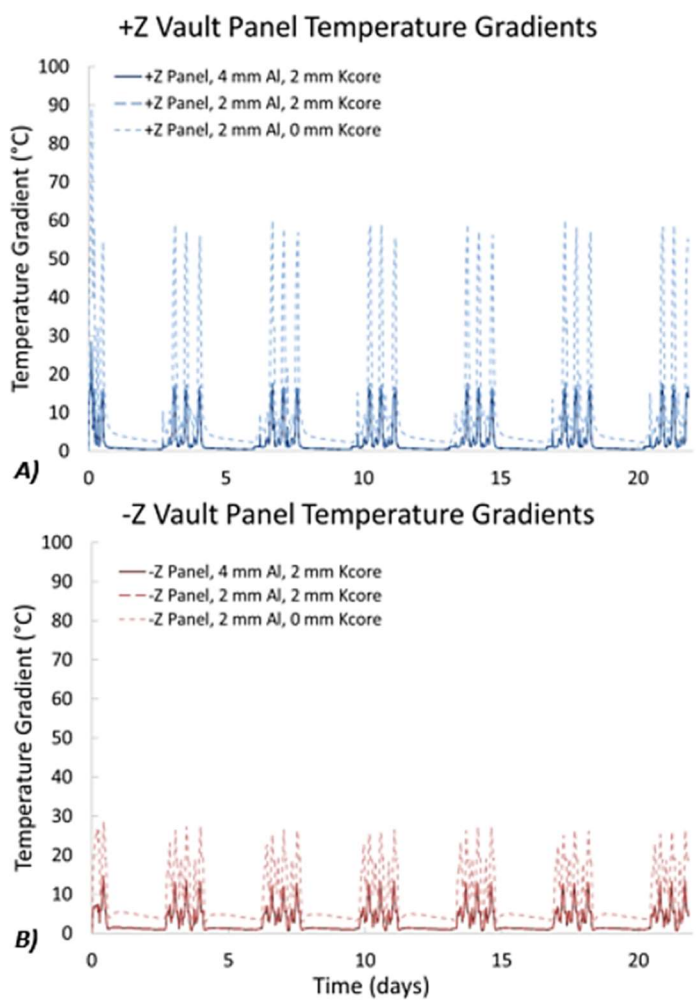


Figure 7. Temperature gradients (max minus min) as a function of kcore thickness for a driving hot case profile in the lander vault's A) +Z panel and B) –Z Panel.

The planned lander batteries have a high energy density chemistry that provides 630 Wh/kg per cell. Furthermore, the battery is electrically inefficient, with almost 1 W of heat generated per 1 W of electrical power output. This heat can be utilized as sensible heat for the lander. To take advantage of this, the battery chassis, which would be mounted on the outside of the lander's main vault, would be coupled to the vault with a 0.4 W/K interface. This value was chosen such that the maximum heat can be transferred from the battery to the vault, which results in no temperature margin to the battery's upper AFT limit. The physical interface between the batteries and vault is planned to be a mechanically stiff mounting panel. The conductance value would remain adjustable by changing the thickness and area of the material as the lander design matures.

Aside from the batteries, other key aspects of the lander thermal design are temperature isothermality of the vault and reducing heat leaks. The former would be achieved with a six-sided vault, which is comprised of aluminum and k-core annealed pyrolytic graphite sandwiched between aluminum. To protect the lander electronics and instruments from radiation, a minimum thickness of material must be maintained. This has a thermal advantage of reducing in-plane thermal gradients, which simplifes box-level temperature requirement compliance if all locations in the vault are nearly the same temperature. High power components are concentrated on the –Z (payload, avionics)

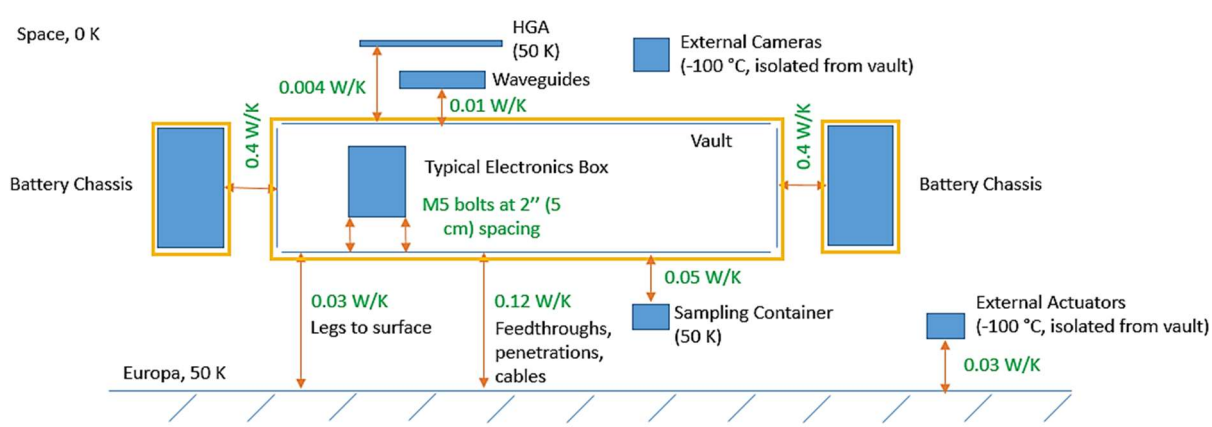


Figure 8. The lander phase block diagram with key thermal interfaces identified (for categories with multiple components, the total conductance is shown).

and +Z (telecom) panels. A sensitivity study was conducted to determine the thickness of k-core needed to maintain low thermal gradients (Figure 7). For both the +Z and –Z panels, using 2 mm of kcore resulted in quiescent thermal gradients less than 1 °C and peak gradients less than 15 °C. This is an attractive result that would be monitored in the future. Second, the lander vault would be isolated from other components outside the vault, including the telecom antenna, sampling arm, and legs. The notional CBE steady state heat leak predicted from the lander vault to external components is 14 W (8 W = actuators, 1 W = cameras, 5 W = telecom) and to feedthroughs is 17 W (9 W = cable harness, 8 W = structure). See Figure 8 for a block diagram of the lander surface phase.

Each of the driving power profiles (Figure 6) were analyzed in the lander surface phase model. The driving hot case mission profile (profile C1 in Figure 6) uses the fastest possible sampling cadence while remaining within the total energy available. This produces three samples per sol, which quickly increases the temperature of the lander vault and batteries. Predictions from the lander model show that the batteries exceed their 70 °C upper AFT limit by around 10 °C (Figure 9). Although this AFT violation is undesirable, the sampling cadence is at a maximum and the total number of samples is unrealistic (FMS is three samples total – not three per day). The thermal design could be improved in future work if this robustness is required by the flight system. Alternatively, the flight system could design profiles with longer time of low power dissipation between sampling events. As such, the battery temperature violation is not viewed as a noncompliance due to the unrealistic nature of the C1 profile. The driving cold case profile (profile A2 in Figure 6) achieves FMS but with the most heater power needed. The cold case represents a mission timeline with sampling activity at the end of the mission so as to not take credit for latent heat due to sampling at the beginning of the mission. Since components are heater controlled to lower AFT limits for much of the timeline, this is a benign profile for assessing temperature compliance. The total CBE energy usage (electronics + heaters) in the cold case is 32 kWh compared to an available 50 kWh, which satisfies energy requirements. The battery sizing case (profile G5 in Figure 6), which is the most important to the flight system, meets the AFT requirements, by design, with no margin on the batteries. The total CBE energy usage is 34 kWh, which is within the 50 kWh available. At this phase of the design, the project prefers to maintain more margin on energy than temperature.

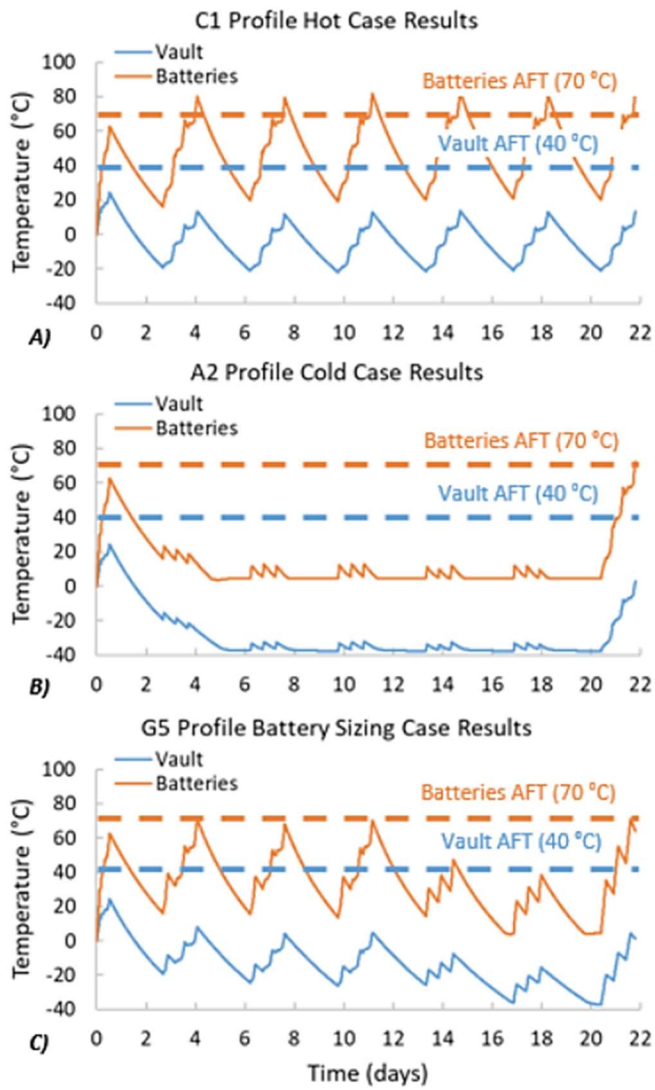


Figure 9. Transient temperature predictions for A) worst case hot (C1), B) worst case cold (A2), and C) battery sizing (G5) power profiles.

IV. Future Work

A. Mission Concept Status

The future of the Europa Lander mission concept is uncertain. Pre-project work was completed and successfully reviewed in a mission concept review (MCR) in preparation for a formal transition to NASA lifecycle phase A. Since the thermal design is coupled to the mechanical configuration, any changes and additional design maturity will initiate

new thermal design challenges. At this time, the thermal design meets applicable requirements. Impacts to the schedule may require changes to time sensitive aspects, such as mission trajectory, of the mission design.

B. Technology Maturation

Due to the ambitious nature of the mission and Europa representing a new environment for a landed mission, there are several technology areas to mature. The thermal design itself does not require any specific technologies to mature at this time, however there is thermal work associated with the technology maturation efforts of other subsystems. Related to thermal engineering are the development of a battery chassis thermal design and cold temperature actuators. A task is in progress for development testing of the current baseline battery chemistry. Different cell packaging arrangements could be explored as well. Cell level and module level tests could be performed to characterize the thermal gradients within the cell and mature the packaging concepts for flight use. Cold temperature actuators with dry lubricant would be especially valuable at Europa where the surface temperature could range from 70 to 120 K. This would reduce the heater power needed to excavate and sample for surface missions in which temperatures approach -100 °C or lower.

Instrument concepts for the the mission concept have been solicited under the second round of the Instrument Concepts for Europa Exploration (ICEE-2) opportunity. Even if the Europa Lander mission concept does not proceed, instrument concepts could still be matured for a future icy worlds mission. Thermal considerations may focus on cryogenic management of the sample ice as it interacts with the payload suite and flight system.

C. Advanced Development

If work on the flight system architecture continues, there are several open design trades related to the thermal design. This includes the use of louvers on the lander (design robustness vs added power and mass), prop subsystem power savings for heaters (steady state power savings vs transient heater power increase for thruster use). A higher fidelity set of temparture requirements would also be drafted over time as component level maturity is increased.

Both the thirty day preparation sequence prior to DDL and the DDL sequence itself require further thermal analysis. The DOV thermal design has met AFT requirements for notional timelines and individual checkout phases in the past, but an end-to-end sequence has not been analyzed. Prevous work was focused on determination of heater power required during each spacecraft power mode. This was motivated by a desire to converge on a solar array area for MCR. To date, a stable order of operations and duration for each mode has not been determined, which prevents a realistic analysis of hot case temperature violations. Once an operations sequence is identified and thermal analysis is completed, the thermal design would accommodate all modes in the thirty day sequence prior to landing and identify any limiting cases or operational constraints on the flight system.

There are also thruster plume impingements on the descent stage tanks from the DOS control thrusters. Unfortunately, there is no viable orientation for those thruster nozzles that avoids plume impingement. A high temperature blanket layup could be a possible solution. Such a blanket could include 15 embossed aluminized kapton layers followed by 5 alternating layers of ceramic fibers, aluminum foil, and ceramic glass separators. This blanket design would weigh more and likely be worse in thermal performance than a standard 20 layer kapton blanket, so this represents a mass and power threat to the flight system. This blanket must also meet radiation and electrostatic discharge requirements relevant to a Jupiter environment.

A higher level trade study would examine the impact of incorporating active thermal management. The lander and descent stage thermal designs both depend heavily on MLI blanket performance and thermal isolation via struts, which results in little design flexibility. Therefore, active thermal technologies, such heat switches and freezable radiators would be examined to improve the thermal design's robustness to future changes in the mechanical, power, and environment designs.

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