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Triton Hopper: Exploring Neptune's Captured Kuiper Belt Object

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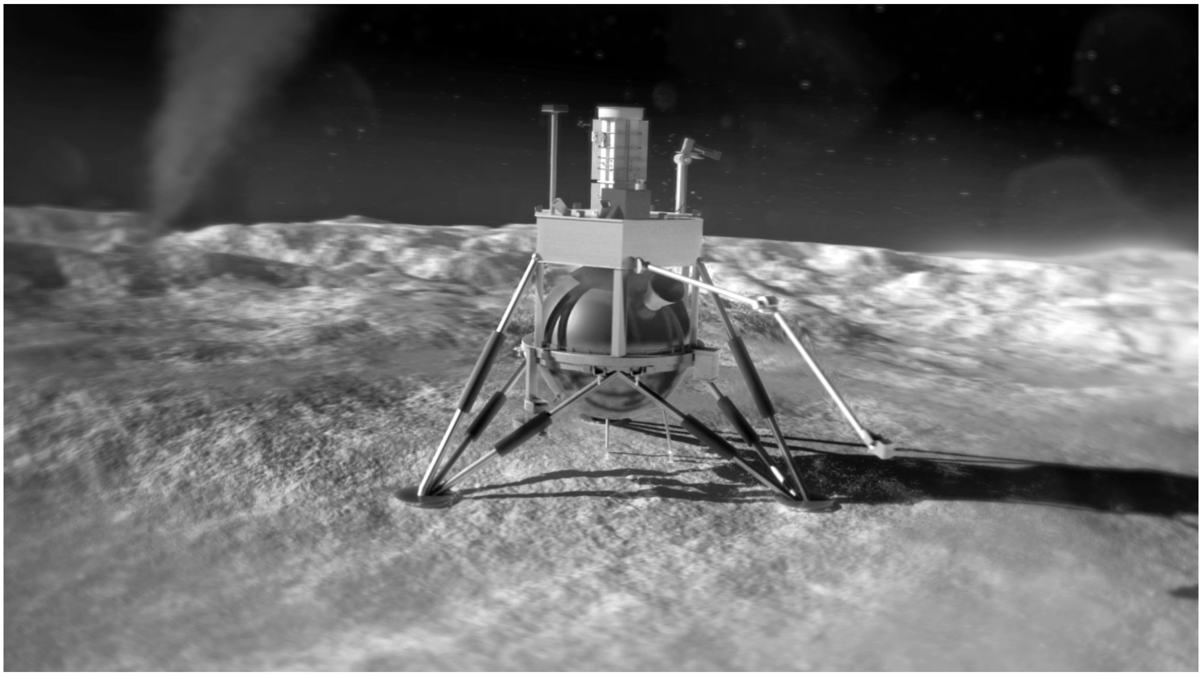

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Triton Hopper: Exploring Neptune's Captured Kuiper Belt Object

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1.0 Executive Summary

The Triton Hopper is a mission concept study (Figure 1.1) funded as a Phase 1 effort by the NASA NIAC program. This work serves as the final report for this effort.

Neptune's moon Triton is a fascinating object, a dynamic moon with an atmosphere, and geysers (Ref. 1). Triton is unique in the outer solar system in that it is most likely a captured Kuiper belt object (KBO)—a leftover building block of the solar system (Ref. 1). When Voyager flew by, it was the coldest body yet found in our solar system (33 K), yet had volcanic activity, geysers, and a thin atmosphere (see Figure 1.2 and Figure 1.3). It is covered in ices made from nitrogen, water, and carbon dioxide, and shows surface deposits of tholins, organic compounds that may be precursor chemicals to the origin of life. At a distance of over 30 AU, it would be by far the most distant object ever landed on by a spacecraft (S/C).

The Triton Hopper effort set out to design a mission to not merely land, but repeatedly fly across the surface of Triton, utilizing the volatile surface ices (primarily nitrogen) as propellant to launch across the surface and explore all the moon's varied terrain. We have determined that such a Hopper can be developed using simple methods of collection and propulsion. Gathering 100 kg of surface nitrogen ice with either a robotic scoop, or atmospheric gas using a cryopump provides sufficient propellant to allow this ~300 kg, highly instrumented lander to hop 5 km across the surface once a month, using a low-temperature radioisotope-thermal engine. More sophisticated propulsion could increase the flight range. Two years of hopping would allow this lander to hop 150 km and visit 30 sites! The final Hopper design is shown in Figure 1.4.

Figure 1.1.—Triton Hopper Mission animation.

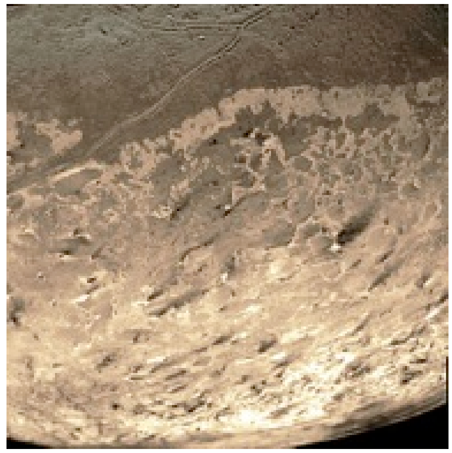


Figure 1.2.—Voyager image of South Pole of Triton showing geysers.

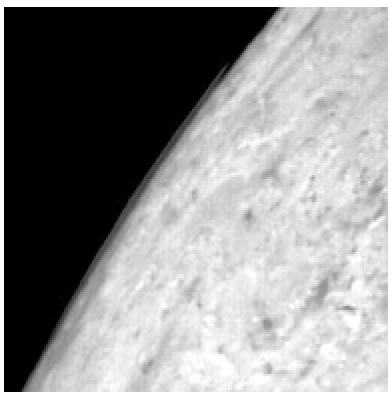


Figure 1.3.—Voyager image showing thin atmosphere.

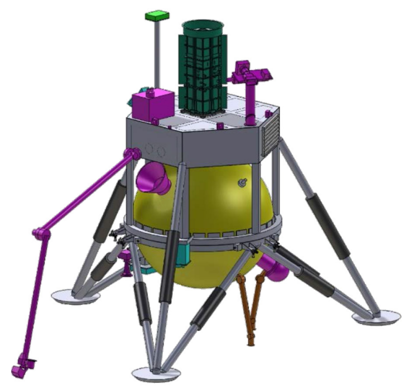


Figure 1.4.—Triton Hopper Concept.

The concept of operations for the Hopper is shown in Figure 3.2 and detailed in Section 3.1.

While the phase I effort focused primarily on design of the Hopper vehicle, not the delivery to Triton, a top-level analysis was performed to show that it is possible to deliver the vehicle from Earth to Triton and land it on the surface. Figure 2.3 summarizes the delivery trades. While use of nuclear electric propulsion would allow getting into low Triton orbit (and thereby minimizing the Triton descent propulsion system size), the baseline design choice was use of a solar electric propulsion (SEP) stage and aerocapture system, leveraging the investment in SEP by other NASA technology efforts, including the piloted Mars mission's use of SEP and aerocapture.

Using the SEP and aerocapture system produced a launch and delivery concept similar to that explored in "Mission Trades for Aerocapture at Neptune" (Ref. 2). As such, the concept developed in the reference was assumed to be the delivery concept to Neptune. New trajectories to Neptune using SEP and aerocapture for a 2029 launch date were developed as was a notional mission and combined solid/biprop landing stage to get the Hopper nearly to the surface. The baseline conops for the delivery to Triton is shown in Figure 3.2.

The main focus of this NIAC effort was to determine feasible methods of gathering, processing and using in situ propellant. According to Voyager the surface is predominantly nitrogen, in the form of both ice and snow form on the surface (Figure 1.5). Triton has a very thin atmosphere (~1 Pa), again mostly of nitrogen. Both surface and atmosphere sources were analyzed for propellant collection.

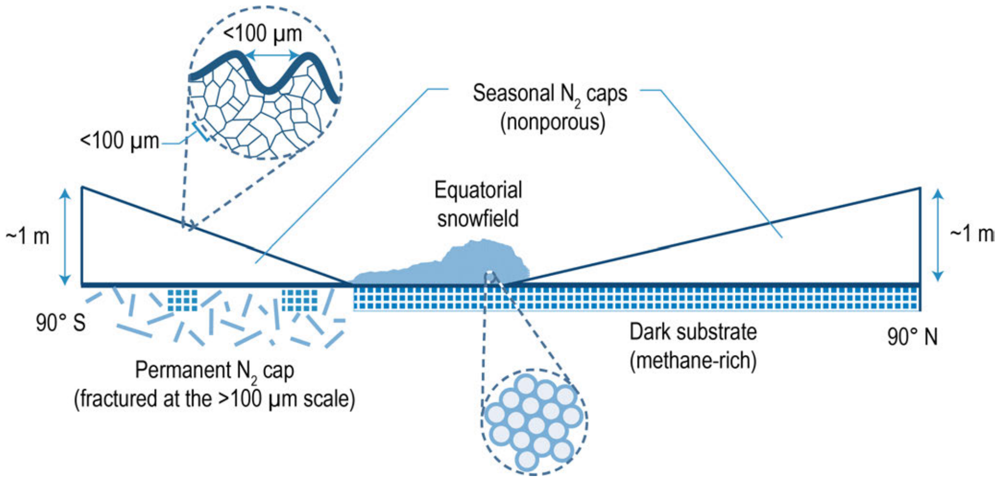


Figure 1.5.—Triton surface conditions. (Adapted from Eluszciewicz, 1991, in Neptune and Triton edited by D. P. Cruikshank. © 1995 The Arizona Board of Regents. Reprinted by permission of the University of Arizona Press (Ref. 1)).

Since nitrogen was accessible both frozen on the surface and in the atmosphere, it was selected as the easiest propellant to utilize and process. Four main approaches to gathering propellant were considered, each having various options (e.g., shovels or drills or scrapers). These methods are shown in Figure 2.4.

The obvious choice is to gather the propellant in its frozen form on the surface. Since the science system requires a robotic arm with a scoop to gather samples for analyzing, a simple approach is to reuse this scoop to gather frozen nitrogen. Even at low temperatures, nitrogen ice is no harder than water ice (at 0 °C), and could be even easier to gather in the form of snow. However, the low gravity of Triton (~1/2 lunar gravity) and icy terrain might make it difficult to gather nitrogen at all locations. Additionally, the issue of other contaminants in surface ices needs to be addressed. An alternate source, the atmosphere, is available regardless of surface conditions, and could be compressed using a cryopump. In order to reduce power requirements for the cryopump it was determined that the cryopump should operate at very low temperatures and reject its waste heat (~90% of the 50 W of power) through a deployable arm to the surface thermal sink. Good conduction through the arm and a large opening (~10 cm diameter) are key to making the cryopumping option work. In the baseline design, both propellant gathering techniques were incorporated. The cryopump atmospheric inlet doubled as the funnel for frozen nitrogen from the shovel. Either system can gather the amount of nitrogen in roughly a week with 50 W or less of power. Once filled, a unique interior door system will seal the tank and the pressure will keep it sealed (Figure 1.6).

Once the tank is filled, the propellant is heated to gas. Nitrogen gas has been used as a very simple, although low performance, propellant for reaction control systems. Besides the low I_{sp} (~60 s I_{sp}) the high pressure required makes relatively heavy tanks necessary, but such high pressure simplifies the propellant feed system. Temperature and pressure of the cold gas was optimized, trading off the tank mass against the engine performance. Approximately 100 kg of nitrogen is vaporized and heated to 300 °K and 2000 psi in about 11 days using 60 W of heaters. Use of waste heat from the radioisotope generator was complex and was detrimental to power production, while use of RTGs would provide four times the waste heat but would also be more challenging to integrate to the low temperature platform during propellant collection. Heating the tank will also heat the propulsion system so it will not need to operate at extremely low temperatures.

Use of additional propellant heating by a resistojet was also analyzed. This would improve the specific impulse (>100 s I_{sp}), but requires large amounts of electrical power, and the trade analysis indicated little or no net performance gain from this approach. Increasing the gas temperature from heat stored in a thermal mass could also provide performance. The thermal mass would be heated slowly over the week spent gathering propellant, and the nitrogen propellant is then passed over it during flight. While analysis of this approach showed that net improvement in performance could be achieved, with over 100 s of specific impulse possible, detailed results were not ready until near the end of the project, and the simpler pressurized gas system was used for the Phase I design effort. The more advanced engine with improved I_{sp} performance could quadruple the distance achieved per hop, so this approach will be further investigated in a Phase II.

Given the cold gas nitrogen thruster approach the Hopper can still hop surprising distances due to Triton's low gravity. Figure 1.7 shows a nominal hop.

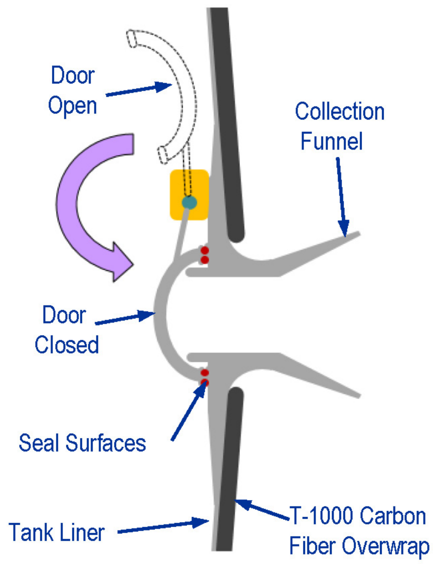


Figure 1.6.—Propellant tank door concept.

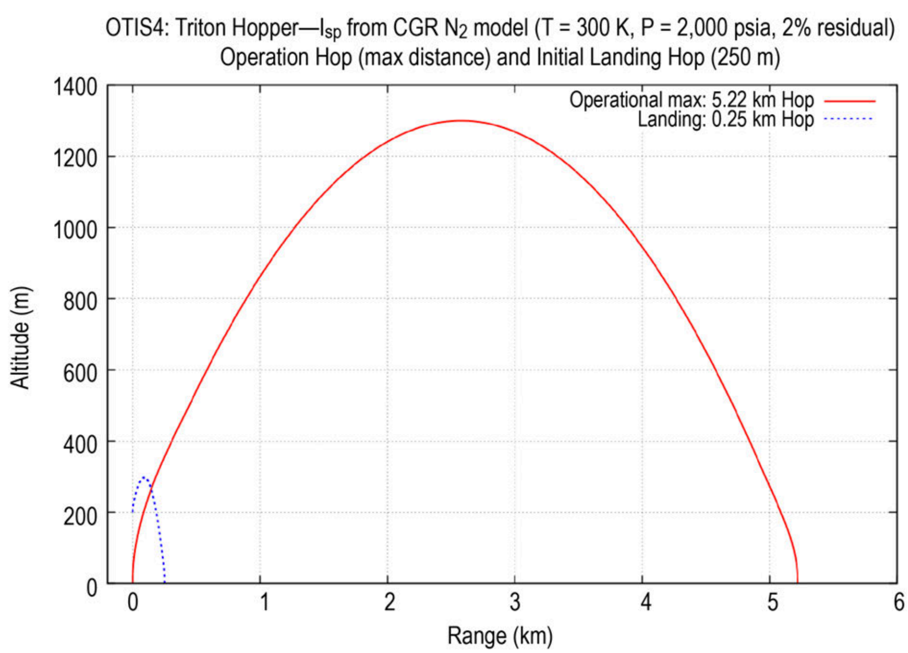


Figure 1.7.—Hop trajectory.

A thrust to weight of 4:1 was required by the trajectory. Six 220 N cold gas thrusters were used, along with six 4.5 N thrusters for reaction control. The landing sites will be mapped by the orbiter during flybys to allow mission controllers to program the Hopper (with use of an IMU and star trackers) to precisely hop to a 100-m diameter landing zone. A combination of lidar and radar systems will provide soft landing and allow the Hopper to autonomously avoid landing on debris > 10 cm in size or extreme slopes. The orbiter will also relay the estimated 1 Gb of data that the science suite gathered on the Hopper and sent using a 10 W X-Band communications system.

1.1 Science

- Narrow-field and wide-field stereo cameras.
- Visible/infrared spectroscopy
- Surface sample collection, with associated sample analysis by:
  - Microscopic imager
  - Thermal Evolved Gas Analysis
  - Gas chromatography
  - Mass spectrometry
  - X-ray crystallography
  - APXS
  - Organic sample analysis
  - Meteorology
- In-flight atmospheric sampling and analysis
- Ground-penetrating radar
  - Look under surface for geyser sources
- Seismometer

The huge of the impact of such a Hopper is clear when considering that it replaces 30 single-site landers on Triton's surface! Even a Triton rover would be hard pressed to cover 150 km in 2 years (for comparison, the MER rover Opportunity covered 42 km in 11 years.)

This report will go into detail on the trades and point design created for the Triton Hopper. A summary of the requirements, assumptions, design, and trades to be discussed is shown in Table 1.1.

The conclusion of the design study was that it is possible to design an innovative Hopper vehicle propelled by in situ resources that could accomplish a roving mission on Triton, a captured Kuiper belt object of high scientific interest.

The approach developed for the Triton Hopper concept can be adapted utilized on other bodies, whether icy or not. By the use of cryopump grounded to the cold surface even thin atmospheres can be pumped to create propellants. Ices are even easier to collect and process given the right tools.

Phase II efforts plan to explore the higher performing high temperature thrusters, lightening the lander, and/or speeding up propellant collection/processing in order to extend hop distance and frequency. A complete launch and delivery analysis to Triton is also planned. Finally, use of a Triton-type Hopper on Pluto should be analyzed.

TABLE 1.1.—SUMMARY OF TRITON HOPPER REQUIREMENTS, ASSUMPTIONS, DESIGN, AND TRADES

Item	Requirements	Assumptions	Design	Trades
Top-Level	Scientific lander which can hop multiple times over Triton's surface using in situ propellants.	Focus on Hopper design – scale delivery system and assume an orbiter in a 12-day resonant orbiter with 4 m antenna for relay. FOMs: Probe mass, complexity, science	Hopper capable of 5 km hops every 24 days, covers 150 km distance and surveys 30 sites. Extended mission possible (no propellant limit)	Less science, lighter rover, further hops
Science	Investigate wide variety of Triton surface features, sample subsurface and analyze with oven, Image surface, sample atmosphere during hops, monitor weather, observe geysers.	Triton environment based on Voyager flyby. 0.779 m/s^2 gravity, 1 Pa atmosphere of nitrogen, global surface temperature of 35 K		Drill
System	Hopper part of flagship class mission, off-the-shelf equipment where possible, TRL 6 cutoff 2026	Mass growth per updated AIAA S-120-2006 (add growth to make system level 30%, 10% launch margin), single fault tolerant		
Mission	Launch 2029, deliver ~ 400 kg Hopper to near Triton surface. Hop 5 km multiple times	Past conceptual mission design (Ref. 2) for launch and delivery to Neptune	Hot nitrogen used as cold gas for thrusters to hop 5 km horizontal (1 km vertical). Precision landing with collision avoidance.	
Operations	Two-year operations. Perform Science. Travel to as many diverse sites as possible. Collect 100 kg of nitrogen for hop propellant at each location	Triton Hopper lands with minimal propellant.	Gathers nitrogen either by cryopumping atmosphere or digging up nitrogen snow/ice (8 Earth days) and heating (11 Earth days) to ~300 °K, 2000 psia.	Hop distance, landing accuracy
Guidance Navigation & Control	Provide soft landing (1 m/s vertical and horizontal), hazard avoidance (>10 cm debris, <13° slope), Landing site size ~100 m diameter	Orbital imagery 10 cm resolution, Earth uploaded landing location, Hopper performs final landing based on hazard avoidance inputs		
Launch Vehicle	Flagship class	Delta IV Heavy to C_3 4.3 km^2/s^2 Launch loads: 4 to 5gs axial		Falcon Heavy
Propulsion	In situ nitrogen propellant, T/W ~ 4, ΔV for take-off and landing = 150 m/s, RCS ΔV ~ 5 m/s	Landing stages (solid and chemical – notional designs only)	Hopper: Single ~ 1 m high pressure COPV nitrogen tank with 10 cm opening for nitrogen collection (dig or cryopump), nitrogen cold gas thrusters, six 220 N thrusters for takeoff-landing, six 5 N thrusters for RCS	Heated liquid lithium (or Be or other) block to raise I_{sp} of collected nitrogen propellant to reduce size/mass of propellant tank.
Power	~100 W continuous, ~450 W during hops (~3 min)		1 ASRG with capacitor for peaks	Two MMRTGs
Comms/ Command & Data Handling (C&DH)	1 Gb science data storage, uplinked in 20 hr every 12 days	4 m antenna on Orbiter which is in view for 20 hr at ~210,000 km	X-Band UHF radio with omni antenna, Rad750 with valve and motor drivers	Data rate/day vs. satellite contact time vs. antenna
Thermal/ Planetary Protection	Keep Bus at 300 °K, isolated from surface and propulsion system. Propulsion system kept cold (~35°) during propellant collection, heated to (~300 °K) during propellant processing. Planetary Protection Class II*	Hopper: Avionics heated by running electronics Assume liquid water far below the surface	Utilize low conductivity Ti-alloy (7 W/m-K) Hopper structures, high conductivity (6000 W/m-K) thermal shorts (deployable) to the surface to cool tank/cryocooler during propellant collection. Retract thermal shorts and use 60 W heaters during propellant processing in the tank.	ASRGs self-radiating or coolant loop (needed for cruise deck portion of mission), use ASRG waste heat to vaporize nitrogen.
Mechanical	Low conductivity structure to reduce heat leak from bus and propulsion system (tank), 1 m/s landing speed, 3g limit, 2 cm deflection		Primary: Low conductivity (7 W/m-K) Ti alloy, ~500 kg lander, Secondary: 4% of components Mechanisms: digging arm for science/propellant collection, landing gear, deployable camera mast	Use Phoenix heritage as much as possible

2.0 Study Background, Assumptions and Approach

2.1 Introduction

Neptune's moon Triton is a fascinating object, a dynamic moon with an atmosphere, and geysers (Ref. 3). Triton is unique in the outer solar system in that it is most likely a captured Kuiper belt object (KBO)—a leftover building block of the solar system (Ref. 4). When Voyager flew by it was the coldest body yet found in our Solar System (33 K) and had volcanic activity, geysers, and a thin atmosphere. It is covered in ices made from nitrogen, water, and carbon dioxide, and shows surface deposits of tholins, organic compounds that may be precursor chemicals to the origin of life.

Exploring Triton will be a challenge well beyond anything done in previous missions; but the unique environment of Triton also allows some new possibilities for mobility. The Hopper designed here (Figure 1.4) will land in the southern hemisphere in 2040 where geysers have been detected. The Hopper will hop every 24 days while in view of the orbiter, exploring the surface and thin atmosphere on its way. The Hopper will hop north toward the equator, exploring as many diverse regions as possible. This craft will autonomously carry out detailed scientific investigations on the surface, below the surface (drilling) and in the upper atmosphere to provide unprecedented knowledge of a KBO turned moon and expanding NASA's existing capabilities in deep space planetary exploration to include Hoppers using different ices for propellant.

Triton is roughly 2700 km in diameter with a surface of mostly frozen nitrogen, mostly water ice crust and core of metal and rock (Ref. 3). Its gravity is half that of Earth's Moon and its atmosphere is $1/70{,}000^{th}$ of Earths or 0.3% of Mars.

The mission will investigate the full surface and atmospheric phenomenon: chemical composition of surface and near subsurface materials, the thin atmosphere, volcanic and geyser activity. Measurements of all these aspects of Triton's unique environment can only be made through focused in situ exploration with a well-instrumented craft. And this craft is provided revolutionary mobility, nearly global, using in situ ices as propellants.

2.2 Previous Studies

Before the Hopper concept was selected for detailed analysis, a large variety of alternate concepts for Triton exploration were evaluated, including remote sensing from orbiters; stationary landers; impactors; wheeled, walking, hopping, and hovercraft rovers; and several alternative approaches to rocket-powered Hoppers (Refs. 3 to 8).

At least one past Neptune mission evaluated a Triton lander design (Ref. 9), but remote sensing and stationary landers cannot achieve the baseline science, which will require in situ sample analysis and the ability to move from an initial landing site to a site featuring accessible surface organic compounds, which may not be available at all surface locations.

Because of the variety and global distribution of scientific targets, long-distance mobility is critical to achieving the scientific goals. The probe needs to be able to investigate multiple features: traversing to geyser regions and investigating plumes which will give access to samples from the interior; moving to tholin deposit regions to investigate primordial organic chemistry; investigating unique geological features such as the cantaloupe terrain. While Mars rovers have demonstrated record traverse distances of 26 km on another world, Triton exploration demands orders of magnitude better traverse capability: we need pole to pole mobility to fully explore all the features of this fascinating moon. The rocket-propelled Hopper approach was the only one with the combination of long-distance mobility with the ability to take shorter hops to achieve the full range of desired science. As discussed later in Section 3.2.4, the ability to achieve multiple hops requires refueling. These considerations drive the selection of the in situ refueled rocket Hopper for exploration.

2.3 Triton Overview

Triton is an object of great scientific interest. Unique among the large moons of the solar system, it is in an inclined retrograde orbit, rather than the direct orbit of all but the smallest of solar system moons. This, and other clues, indicates that Triton is very likely a KBO. Thus, a mission to explore Triton will be the first-ever exploration of a KBO, giving us in situ insight in to one of the least understood and most interesting kinds of body in the solar system. The only detailed information we have about the surface of Triton is from a fly-by of a single S/C, the Voyager 2, in 1989, which only imaged one hemisphere of the moon. Nevertheless, what we know shows that the object is complicated and interesting.

As a KBO, Triton is of interest because it contains material from the outer solar system, and thus allows study of composition of areas of the primordial solar system that are relatively unchanged by thermal alteration processes that occur nearer to the Sun. Of particular interest are surface deposits of tholins, complex organic materials that are dark red in color, and which are not present on the surfaces of inner solar system bodies, but ubiquitous on the larger KBOs. These organic deposits may hold keys to the biochemical origin of life.

In addition to its interest as a KBO, Triton is a dynamic object. It is one of the few bodies in the solar system that is seen to have active eruptions. Near the South Pole, cryo-volcanism is in evidence, with geysers shooting up to 8 km high that may be a link to a hypothesized subsurface ocean. The plumes of these geysers give access to the interior of the moon. Other regions of the surface show geological evidence of low-temperature volcanism. It is one of the few moons with a noticeable (although thin) atmosphere, with clouds believed to be composed of frozen nitrogen crystals at altitudes of 1 to 3 km from the surface, and a tropospheric haze, hypothesized to be composed of hydrocarbons and nitriles created by photochemical alteration of methane, ammonia, and nitrogen. Triton also includes surface features of unknown origin such as “Cantaloupe terrain.”

2.4 Triton Operational Environment

Triton, as shown in Figure 2.1, is the largest moon of Neptune. It has some unique characteristics. Its surface is a mix of smooth planes near heavily cratered areas. The reasoning for this significant difference in terrain is not fully understood. Also Triton is the only moon in the solar system that orbits its planet in a direction opposite to its rotation. It was also recently discovered, from Earth ground observations, that Triton experiences seasons causing its thin atmosphere to vary in thickness (Ref. 10). Each season lasts approximately 40 Earth years due to the long solar orbital period of Neptune. The surface of Triton has a number of different materials, normally gaseous in Earth’s environment, that are frozen and cover the surface. These include mainly nitrogen with some water, carbon monoxide and carbon dioxide. Triton’s physical properties are summarized in Table 2.1.

Triton’s rarified atmosphere is mostly nitrogen. However, methane and carbon monoxide have also been detected. It is believed that a layer of frozen nitrogen on the surface with higher concentrations of carbon monoxide, up to ten times greater than seen below the surface, vaporizes from the surface during the summer time causing the increase in seasonal atmospheric density that was detected.

Data from the Voyager 2 S/C indicated that cryo-volcanic activity is likely occurring on the surface of Triton in the form of gaseous nitrogen geysers. It is also believed that an ammonia-rich water ocean exists beneath the icy surface.

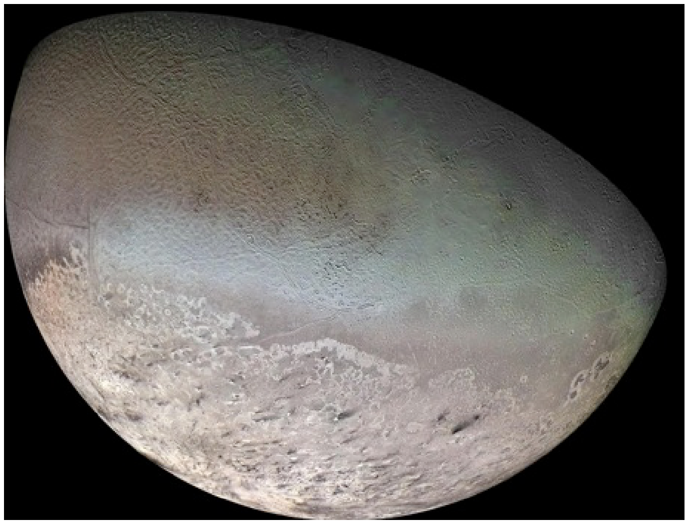


Figure 2.1—Image of Triton (Ref. 11).

TABLE 2.1—TRITON PHYSICAL PROPERTIES

Property	Value
Diameter	2707 km (about ¾ the diameter of Earth's Moon)
Solar flux at Triton	1.5 W/m^2
Average surface temperature	38 K
Solar orbital period	165 Earth years
Orbital period around Neptune	141.6 hr (retrograde motion opposite planets rotation, same as rotation period (day), tidally locked with Neptune)
Atmosphere composition	Nitrogen, carbon monoxide, methane, carbon dioxide
Atmosphere pressure	14 to 40 microbar (depending on the season), (7.5 to 30 mTorr)
Gravitational acceleration	0.779 m/s^2
Surface composition	55% frozen nitrogen; 15 to 35% water ice; 10 to 20 % frozen carbon dioxide
Surface material thermal conductivity (Refs. 12, 13, and 14)	Nitrogen: 0.25 W/mK at 38 K; Water ice: 23.9 W/mK at 38 K; Carbon dioxide: 5 W/mK at 38 K
Orbital inclination	To Neptune's equator: 156.9°; To the ecliptic: 129.8°
Orbital eccentricity	0.0
Atmosphere gas constant (R_a)	296.8 J/kg-K
Atmosphere ratio of specific heats (γ_a)	1.4
Atmosphere specific heat (c_{pa})	1039 J/kg-K

The operating environment of the Hopper will be on the surface of Triton where it will collect propellant and then use a ballistic hop to another location. The components on the Hopper have to withstand the deep space transit to Triton and operation within the cold surface environment. While on the surface the Hopper will collect material either from the atmosphere or by directly scooping it from the surface into a tank where it will be stored and used as propellant. The condition of the surface material is not completely known, it can range from a fluffy snow like consistency to solid ice. The thermal conductivity of the main components of the surface in solid form is given in Table 2.1.

The cold environment of Triton provides a number of challenges in the operation of equipment and materials. Operating within this environment requires all of the electronics and science equipment to be insulated and isolated as much as possible from the environment.

The gravitational acceleration on Triton (0.779 m/s^2) is less than half that of Earth's Moon. This low gravity presents challenges in using force to break into the icy surface and extract material for propellant. Due to the low gravity, the weight of the S/C is limited and in turn the amount of downward force that can be applied. To overcome this, approaches such as scraping or using opposing scoops to dig into the surface could enable enough force to be generated to break into the surface ice.

2.5 Hopper Science

The science instrument package was chosen to address the major scientific enigmas of Triton. It made use of the ability of the Hopper to sample widely varied terrain, including both horizontal sampling as well as vertical sampling through the plumes of geysers and through layers of the thin nitrogen atmosphere.

Experiences from the Mars Exploration Rovers and other missions have shown how critical visual photography is, not merely for operations planning, but as a capable scientific tool for understanding geomorphology and surface processes. Rover operations have also shown the value of stereo imagery. Thus, the first instrument is a highly capable camera. In addition to scientific use, the camera will also produce high-definition images for use in the public outreach, to bring the public into "virtual presence" on the surface of the icy moon. The camera will not only characterize the geology of the surface locations, it will also be used for in-flight imagery during the hops, allowing detailed aerial surveying of the geological context of each location and three-dimensional reconstruction of terrain from stereo imagery.

Likewise, the value of visible and near-infrared spectrometry has been shown to be of great value for remote mineralogy. These tools will be the main remote-sensing instruments.

At a landed location, detailed measurements will be done using the in situ instruments. The vehicle is uniquely suited for surface analysis, since as a part of the refueling operation it will take surface samples into the vehicle. This will allow complete characterization of surface material. A significant characterization will be the use of mass spectrometry. In addition, classic in situ instruments will be included, including a microscopic imager, x-ray crystallography, and alpha-particle x-ray spectrometer (APXS), all of which have been proven to be of great value during rover missions.

Measurements when landed will be supplemented by measurements during flight. These instruments will characterize the vertical profile of the atmosphere and cloud layers. The vehicle will also be able to fly through the plumes of geysers, taking samples of the geyser activity as a method of probing the deep interior of the moon.

Finally, three additional tools will give information on what is below the surface. A small ground-penetrating radar will allow a vertical profiling of the local surface. This will determine the amount of layering, which gives information about climate. Depending on the landing site, the frequency chosen, and the signal to noise ratio, it may be also be possible to characterize the subsurface liquid reservoir which feeds the geyser activity.

Seismometry can also give valuable information. Geyser eruptions are typically accompanied with localized crustal seismic activity, and measuring this activity will yield information about the liquid reservoir feeding the geyser and about the mechanisms of geyser action. If Triton is subject to global seismic activity, measuring seismic motions may give information about the deep interior as well. The lander will be equipped with microelectromechanical (MEMS) accelerometers on the footpads, measuring three components of acceleration; accelerometers on each foot give redundancy to the system, as well as some ability to filter out S/C induced disturbances from ground motion.

Finally, precision navigational measurement will give a measure of the wobble of Triton in its orbit, which is a powerful tool for determining size of the liquid core.

The major science instruments of the baseline science package are described in Section 4.1.

2.6 Study Purpose and Approach

The purpose of this NIAC phase I study was to develop a conceptual Hopper vehicle design to explore Triton by using volatiles gathered in situ in the 33 °K environment. Timeframe guidelines include a launch in the latter half of the 2020s (to utilize a Jupiter flyby), 2 years of surface science and hops of multi-km in distance. A radioisotope power system was assumed for all operations including gathering and melting propellants, power and thermal support. Due to the many hour communications delay the Hopper will need to take off, fly and land autonomously, although ground controllers will plan each hop. An orbiter that will also provide high-resolution imagery of potential landing sites will relay data and instructions. The Hopper would carry out detailed scientific investigations to as many different Triton regions as can be reached. Science activities would be to a level similar to a rover but with the added capability to sense the atmosphere during hops and investigate Triton's geysers. A main science objective would be to determine if Triton is captured Kuiper belt object. Figures of merit that guided the team were: feasibility, distance hopped, science carried, and cost (be part of a Flagship mission.)

The approach included launch and delivery using a past 2004 Neptune orbiter design (SEP/Aerocapture), a <500 kg wet Hopper mass, and evaluating volatile gathering techniques as well as how these volatiles might best be used as propellant to provide hopping.

2.7 Growth, Contingency, and Margin Policy

The COMPASS Team follows a standard set of definitions for mass, growth and contingency for each study executed by the team. Those definitions appear below, followed by a graphical representation in Figure 2.2.

Mass*:* The measure of the quantity of matter in a body.

Basic Mass (aka CBE Mass): Mass data based on the most recent baseline design. This is the bottoms-up estimate of component mass, as determined by the subsystem leads. *Note 1*: This design assessment includes the estimated, calculated, or measured (actual) mass, and includes an estimate for undefined design details like cables, multi-layer insulation (MLI), and adhesives. *Note 2*: The mass growth allowances (MGA) and uncertainties are not included in the basic mass. *Note 3*: COMPASS has referred to this as current best estimate (CBE) in past mission designs. *Note 4*: During the course of the design study, the COMPASS Team carries the propellant as line items in the propulsion system in the Master Equipment List (MEL). Therefore, propellant is carried in the basic mass listing, but MGA is not applied to the propellant. Margins on propellant are handled differently than they are on dry masses.

CBE Mass*:* See Basic Mass.

Dry Mass*:* The dry mass is the total mass of the system or S/C when no propellant is added.

Wet Mass*:* The wet mass is the total mass of the system, including the dry mass and all of the propellant (used, predicted boil-off, residuals, reserves, etc.). It should be noted that in human S/C designs the wet masses would include more than propellant. In these cases, instead of propellant, the design uses Consumables and will include the liquids necessary for human life support.

Inert Mass*:* In simplest terms, the inert mass is what the trajectory analyst plugs into the rocket equation in order to size the amount of propellant necessary to perform the mission Delta-velocities (ΔVs). Inert mass is the sum of the dry mass, along with any non-used, and therefore trapped, wet materials, such as residuals. When the propellant being modeled has a time variation along the trajectory, such as is the case with a boil-off rate, the inert mass can be a variable function with respect to time.

Basic Dry Mass: This is basic mass (aka CBE mass) minus the propellant or wet portion of the mass. Mass data is based on the most recent baseline design. This is the bottoms-up estimate of component mass, as determined by the subsystem leads. This does not include the wet mass (e.g., propellant, pressurant, cryo-fluids boil-off, etc.).

CBE Dry Mass: See Basic Dry Mass.

MGA: MGA is defined as the predicted change to the basic mass of an item based on an assessment of its design maturity, fabrication status, and any in-scope design changes that may still occur.

Predicted Mass: This is the basic mass plus the mass growth allowance for to each line item, as defined by the subsystem engineers.

Note: When creating the MEL, the COMPASS Team uses Predicted Mass as a column header, and includes the propellant mass as a line item of this section. Again, propellant is carried in the basic mass listing, but MGA is not applied to the propellant. Margins on propellant are handled differently than they are handled on dry masses. Therefore, the predicted mass as listed in the MEL is a wet mass, with no growth applied on the propellant line items.

Predicted Dry Mass: This is the predicted mass minus the propellant or wet portion of the mass. The predicted mass is the basic dry mass plus the mass growth allowance as the subsystem engineers apply it to each line item. This does not include the wet mass (e.g., propellant, pressurant, cryo-fluids boil-off, etc.).

Mass Margin (aka Margin): This is the difference between the allowable mass for the space system and its total mass. COMPASS does not set a Mass Margin; it is arrived at by subtracting the Total mass of the design from the design requirement established at the start of the design study such as Allowable Mass. The goal is to have Margin greater than or equal to zero in order to arrive at a feasible design case. A negative mass margin would indicate that the design has not yet been closed and cannot be considered feasible. More work would need to be completed.

System-Level Growth: The extra allowance carried at the system level needed to reach the 30% aggregate MGA applied growth requirement.

For the COMPASS design process, an additional growth is carried and applied at the system level in order to maintain a total growth on the dry mass of 30%. This is an internally agreed upon requirement.

Note 1: For the COMPASS process, the total growth percentage on the basic dry mass (i.e., not wet) is:

$$\text{Total Growth} = \text{System Level Growth} + \text{MGA*Basic Dry Mass}$$

$$\text{Total Growth} = 30\%\text{* Basic Dry Mass}$$

$$\text{Total Mass} = 30\%\text{*Basic Dry Mass} + \text{basic dry mass} + \text{propellants.}$$

Note 2: For the COMPASS process, the system level growth is the difference between the goal of 30% and the aggregate of the MGA applied to the Basic Dry Mass.

$$\text{MGA Aggregate \%} = (\text{Total MGA mass/Total Basic Dry Mass})\text{*100}$$

$$\text{Where Total MGA Mass} = \text{Sum of (MGA\%*Basic Mass) of the individual components}$$

$$\text{System Level Growth} = 30\%\text{* Basic Dry Mass} - \text{MGA*Basic Dry}$$

$$\text{Mass} = (30\% - \text{MGA aggregate \%})\text{*Basic Dry Mass}$$

Note 3: Since CBE is the same as Basic mass for the COMPASS process, the total percentage on the CBE dry mass is:

$$\text{Dry Mass total growth} + \text{dry basic mass} = 30\%\text{*CBE dry mass} + \text{CBE dry mass.}$$

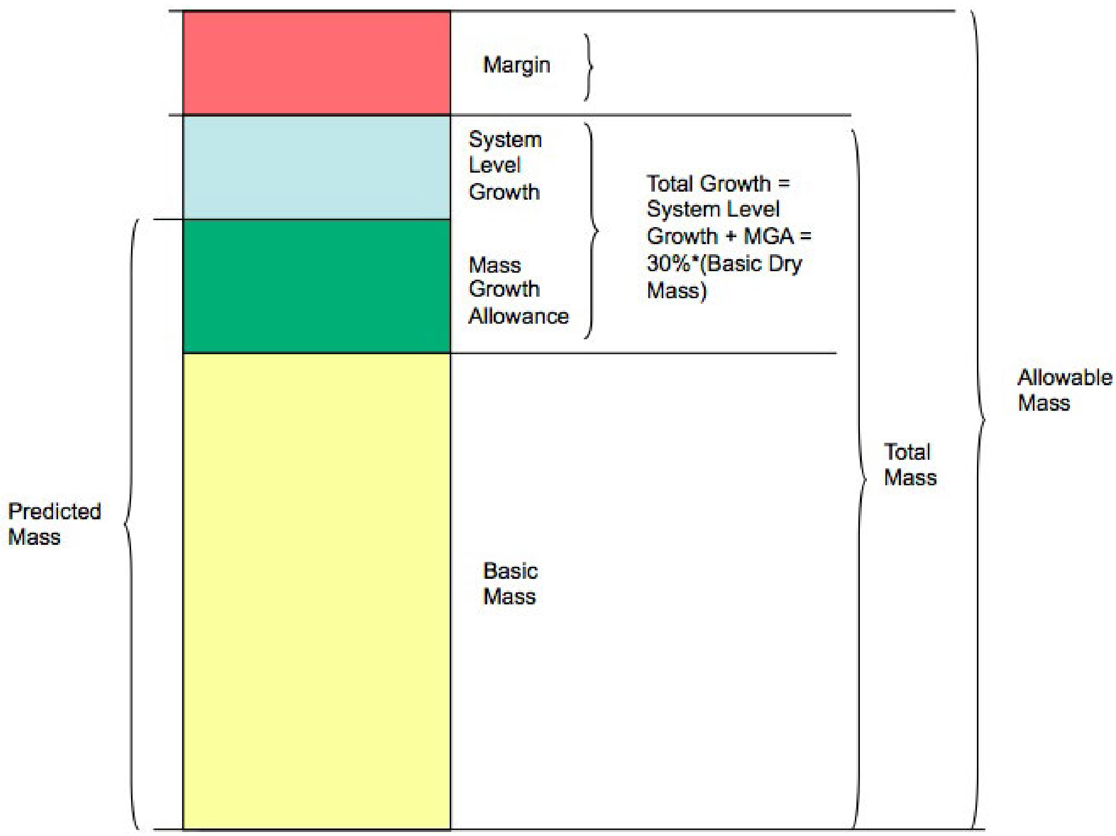
Therefore, dry mass growth is carried as a percentage of dry mass rather than as a requirement for launch vehicle performance, etc. These studies are Pre-Phase A and considered conceptual, so 30% is standard

COMPASS operating procedure, unless the customer has other requirements for this total growth on the system.

Total Mass: The summation of basic mass, applied MGA, and the system-level growth.

Allowable Mass: The limits against which margins are calculated. *Note*: Derived from or given as a requirement early in its duration.

Table 2.2 expands definitions for the MEL column titles to provide information on the way masses are tracked through the MEL used in the COMPASS design sessions. These definitions are consistent with those above in Figure 2.2 and in the terms and definitions. Table 2.2 is an alternate way to present the same information to provide more clarity.



(Basic = bottoms-up estimate of dry mass)
(MGA = applied per subsystem line item)

Figure 2.2—Graphical Illustration of the Definition of Basic, Predicted, Total and Allowable Mass

TABLE 2.2—DEFINITION OF MASSES TRACKED IN THE MEL

CBE mass	MGA growth	Predicted mass	Predicted dry mass
Mass data based on the most recent baseline design (includes propellant)	Predicted change to the basic mass of an item phrased as a percentage of CBE dry mass	The CBE mass plus the MGA	The CBE mass plus the MGA — propellant
CBE dry + propellant	MGA% * CBE dry = growth	CBE dry + propellant + growth	CBE dry + growth

2.7.1 Mass Growth

The COMPASS Team normally uses the AIAA S–120–2006, “Standard Mass Properties Control for Space Systems,” as the guideline for its mass growth calculations. Table 2.3 shows the percent mass growth of a piece of equipment according to a matrix that is specified down the left-hand column by level of design maturity and across the top by subsystem being assessed.

TABLE 2.3—MGA AND DEPLETION SCHEDULE (AIAA S-120-2006)

Major category	Maturity code	Design maturity (basis for mass determination)	MGA (%)												
			Electrical/ electronic components			Structure	Brackets, clips, hardware	Battery	Solar array (SA)	Thermal control	Mechanisms	Propulsion	Wire harness	Instrumentation	ECLSS, crew systems
			0 to 5 kg	5 to 15 kg	>15 kg										
	1	Estimated (1) An approximation based on rough sketches, parametric analysis, or undefined requirements; (2) A guess based on experience; (3) A value with unknown basis or pedigree	30	25	20	25	30	25	30	25	25	25	55	55	23
	2	Layout (1) A calculation or approximation based on conceptual designs (equivalent to layout drawings); (2) Major modifications to existing hardware	25	20	15	15	20	15	20	20	15	15	30	30	15
	3	Prerelease designs (1) Calculations based on a new design after initial sizing but prior to final structural or thermal analysis; (2) Minor modification of existing hardware	20	15	10	10	15	10	10	15	10	10	25	25	10
	4	Released designs (1) Calculations based on a design after final signoff and release for procurement or production; (2) Very minor modification of existing hardware; (3) Catalog value	10	5	5	5	6	5	5	5	5	5	10	10	6
	5	Existing hardware (1) Actual mass from another program, assuming that hardware will satisfy the requirements of the current program with no changes; (2) Values based on measured masses of qualification hardware	3	3	3	3	3	3	3	2	3	3	5	5	4
	6	Actual mass Measured hardware	No mass growth allowance—Use appropriate measurement uncertainty values												
	7	Customer furnished equipment or specification value	Typically a “not-to-exceed” value is provided; however, contractor has the option to include MGA if justified												

The COMPASS Team's standard approach is to accommodate for a total growth of 30% or less on the dry mass of the entire system. The percent growth factors shown above are applied to each subsystem before an additional growth is carried at the system level, in order to ensure an overall growth of 30%. Note that for designs requiring propellant, growth in the propellant mass is either carried in the propellant calculation itself or in the ΔV used to calculate the propellant required to fly a mission.

In AIAA S–120–2006, a timeline shows how the various mass margins are reduced and consolidated over the mission's life span. The system-integration engineer carries a system-level MGA, called "margin", in order to reach a total system MGA of 30%. This is shown as the mass growth for the allowable mass on the authority to precede line in mission time. After setting the margin of 30% in the preliminary design, the rest of the steps shown below are outside the scope of the COMPASS Team.

2.7.2 Power Growth

The COMPASS Team typically uses a 30% growth on the bottoms-up power requirements of the bus subsystems when modeling the amount of required power. No additional margin is carried on top of this power growth. The power system assumptions for this study will be show in Section 3.3 on the Power Equipment List (PEL).

2.7.3 Redundancy Assumptions

The S/C is designed to be single fault tolerant in all vehicle subsystems.

2.8 Design Trades

While getting the Hopper to Triton was considered outside the scope of the Phase I effort some work was performed to find how difficult it is to get a lander onto Triton. All past concepts have had to rely on advanced propulsion techniques to reach Neptune in a timely fashion with existing launchers. Figure 2.3 summarizes the delivery trades. While use of nuclear electric propulsion would allow getting into low Triton orbit (and thereby minimizing the triton descent propulsion system size) it was felt that use of a SEP stage and aerocapture system is far more synergetic with other NASA technology efforts (especially the piloted Mars mission's use of SEP and aerocapture).

Using the SEP and aerocapture system produced a launch and delivery concept similar to that explored in "Mission Trades for Aerocapture at Neptune" (Ref. 2). As such, the concept developed in the reference was assumed to be the delivery concept to Neptune. New trajectories to Neptune using SEP and aerocapture for a 2029 launch date were developed as was a notional mission and combined solid/biprop landing stage to get the Hopper nearly to the surface. The baseline CONOPS is shown in Figure 3.1.

The main focus of this NIAC effort was to determine feasible methods of gathering, processing and using in situ propellant. First the propellant must be gathered. According to Voyager the surface is predominantly nitrogen, whether in ice or snow form on the surface (see Figure 1.5). As mentioned earlier Triton has a very thin atmosphere (~1 Pa), again mostly of nitrogen.

Since nitrogen was so plentiful and easily accessible (both frozen on the surface and in the atmosphere) it was considered the easiest propellant to utilize and process. But how to acquire it? Four main methods were considered, each having various options (e.g., shovels or drills or scrapers). These methods are shown in Figure 2.4.

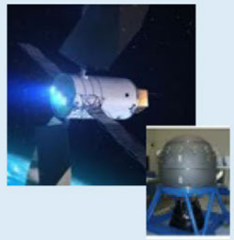
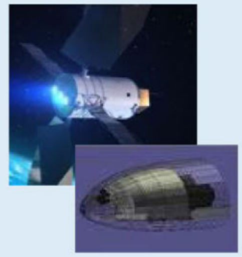
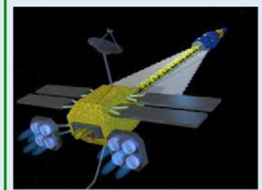
Triton Hopper Delivery Trades	Chemical Capture	SEP/Chemical Capture	SEP/ Aerocapture	Nuclear Electric Propulsion
				
Trip Time to Triton	~18 yr	~ 15 yr	~ 12 yr	~ 17 yr
Triton ΔV	~ 4 km/s	~ 3 km/s	~ 3 km/s	~ 1.2 km/s
Selection Pros	Customer for SLS?	Only requires new SEP stage	Trip Time excellent, Technologies being developed	Smallest Hopper landing stage ~
Selection Cons	Trip Time	Landing stage ~ 3X mass of Hopper	Landing stage ~ 3X mass of Hopper	NEP not currently in development

Figure 2.3—Triton Delivery Trades.

Approach using similar mass/power to gather ~100 kg N_2	Pumping from ~1 Pa atmosphere	Cryopumping from ~ 1 Pa atmosphere	Subliming from surface using waste heat and pumping	Mechanically Gathering from Surface
Simplicity/Heritage/ Global Accessibility	High	High	Low	Medium
Current knowledge of N_2 conditions at surface	Homogeneous atmosphere	Homogeneous atmosphere	Variable: Ice, Snow N_2 Ice conducts as well as stainless steel And is as hard as water ice (0°C)	
Purity of N_2	High	High	Medium	Low (solids, other gases)
Speed	~ 3000 l/s	~ 3000 l/s		7 days
Mass/Power	400 kg/2 kW	~10 kg / 50 W (assuming cryopump waste heat conductively transferred to surface)	TBD	10 kg / 30 W average
Production Rate	~ 15 kg N_2/day	~ 15 kg N_2/day	TBD	~17 kg N_2/day

Figure 2.4—Propellant Collection Trades.

The obvious choice is to gather the propellant in its frozen form on the surface. Since the science system already had a scoop to gather samples for analyzing in an oven it is easy enough to reuse it to gather frozen nitrogen. Nitrogen should only be as hard as water ice (0 °C) and even easier to gather in the form of snow. However, the low gravity of Triton (~1/2 Earth's Moon) and icy terrain might make it difficult to gather nitrogen at all locations. Adding to this the issue of other contaminants caused the team to search for a better primary system. Use of a turbopump to gather the atmosphere and then allow it to frost on the inside of the tank (at slightly higher pressure) will work but to gather roughly 100 kg of nitrogen in a week's time would require too much power and mass. (Assuming a 50 W, 10 kg limit a smaller turbopump could gather the 100 kg but would require over 6 months to do so!) Cryopumping from the thin atmosphere will work regardless of where the Hopper is. In order to reduce power requirements for the cryopump it was determined that the cryopump should operate at very low temperatures and reject its waste heat directly to the surface (~90% of the 50 W of power) through a deployable arm. Good conduction through the arm and a large opening (~10 cm diameter) are key to making the cryopumping option work so it was decided that mechanical options would be used as a backup to the cryopumping if good surface thermal contact between the cryopump and the surface cannot be made. The large opening doubled as the funnel for frozen nitrogen from the shovel. Either system can gather the amount of nitrogen in roughly a week with 50 W or less of power. Once filled, a unique interior door system will seal the tank and the pressure will keep it sealed (see Figure 1.6).

Once the tank is filled the options of how to use it as propellant were explored. Nitrogen gas has been used as a very effective if low performing (~60 s I_{sp}) propellant for both launch vehicles and S/C and was even used for RCS on the Skylab space station. Besides the low I_{sp} the high pressure required for the system requires heavy tanks but such high pressure simplifies the propellant feed system. The ~100 kg of nitrogen can be vaporized and heated to 300 °K and 2000 psi in about 11 days using 60 W of heaters (Use of waste heat piped from the ASRG was complex and was detrimental to power production, use of RTGs would provide four times the waste heat but would also be more challenging to integrate to the low temperature platform during propellant collection.) Heating the tank will also heat the propulsion system so it will not need to operate at extremely low temperatures.

As an alternative the nitrogen can be heated through a resistojet to improve its specific impulse (>100 s I_{sp}) but requires large amounts of electrical power. Heating a thermal sink slowly over a week and then passing the nitrogen over it to improve I_{sp} could also provide such performance. While selection of the thermal sink material was conducted (and lithium or beryllium were chosen as candidates) and initial performance estimated (over 100 s I_{sp}) the simpler high-pressure system was chosen for the Phase I effort. Such improved I_{sp} performance has shown to quadruple hop distance for only doubling I_{sp} so this approach will be further investigated in a Phase II.

3.0 Baseline Design

3.1 Concept of Operations (CONOPS)

As mentioned previously the main focus of the Triton Hopper design was in the operations on the surface and the hops. However, a notional launch and delivery scheme was adapted from a previous reference and the mission recalculated for the appropriate Triton Hopper Launch window around 2030 (when Jupiter will be available for gravity assist flybys.) Figure 3.1 illustrates the launch and delivery conops. Using the past reference a heavy lift launcher (Delta IV Heavy or equivalent) launches the vehicle to escape where a SEP stage performs inner-solar system burns to enable an Earth flyby. Not long after, the SEP stage is jettisoned (the distance to Neptune makes solar arrays ineffective.) A Jupiter gravity assist is next which greatly accelerates the Triton orbiter/Hopper to a 2041 Neptune encounter. Another key technology is the use of an aerocapture system in lieu of a large chemical system to capture the orbiter/Hopper into an elliptical orbit around Neptune. This orbit is reduced/changed to have an apoapsis just below Triton by Neptune by using aerobraking. At this point the Hopper is separated and uses a combination of solid and bipropellant stages to capture into a 200 km Triton orbit. From there the bipropellant 'sky platform' descends to 250 m where the Hopper can perform its first hop to the surface. The sky platform then crashes on the surface (given more mass for legs and landing equipment, the sky platform could make a single landing from which the Hopper could take off from). Once on the surface the Hopper begins its science, propellant collection/processing, and hopping cycle supported by the Neptune orbiter.

The cycle of Triton science, propellant gathering/processing and hopping is illustrated in Figure 3.2. With limited power (~50 W once housekeeping is allotted for) it is estimated that propellant gathering (either by the cryopump or arm/scoop) will require 8 days, as will the propellant heating and vaporization to pressure. Since the orbiter will make a close approach every 12 days it made sense to allot 4 days every twelve for science with the other eight for propellant. So after landing the Hopper will gather propellant and perform science with an orbiter flyby to relay science and assess how much propellant is gathered. (Key components to propellant collection are two thermal conducting legs, one from the tank and one from the cryocooler to keep the tank/cryocooler near surface temperatures (~35 °K).) The orbiter close pass (<200,000 km) should last 20 hr so that controllers on Earth can receive the signals, evaluate them and send back next step instructions in that time period, including next hop instructions.

During the next pass the Hopper will inform the ground controllers of its propellant state (pressure and temperature). Controllers will then signal a go ahead for hop and the Hopper will hop its ~2 min, 5 km horizontal, 1 km high trajectory gathering atmospheric data and relaying it and housekeeping data during the entire hop to the orbiter. Final approach will be programmed from the Earth, based on high-resolution orbiter images, but final touchdown and collision avoidance (>10° slopes and >10 cm debris) will be calculated real-time by the Hopper lidar and main computer. Once down the Hopper will deploy its cooling legs for the tank and cryocooler and begin the process again.

At this rate during a 2 year mission, the Hopper should be able to hop ~30 times for a distance of 150 km. Thirty locations could thus be sampled and evaluated by a single lander. Further hops could be made by improving the I_{sp} of the propulsion system or reducing the Hopper mass. Depending upon Hopper health it should be able to continue hopping given and extending mission duration.

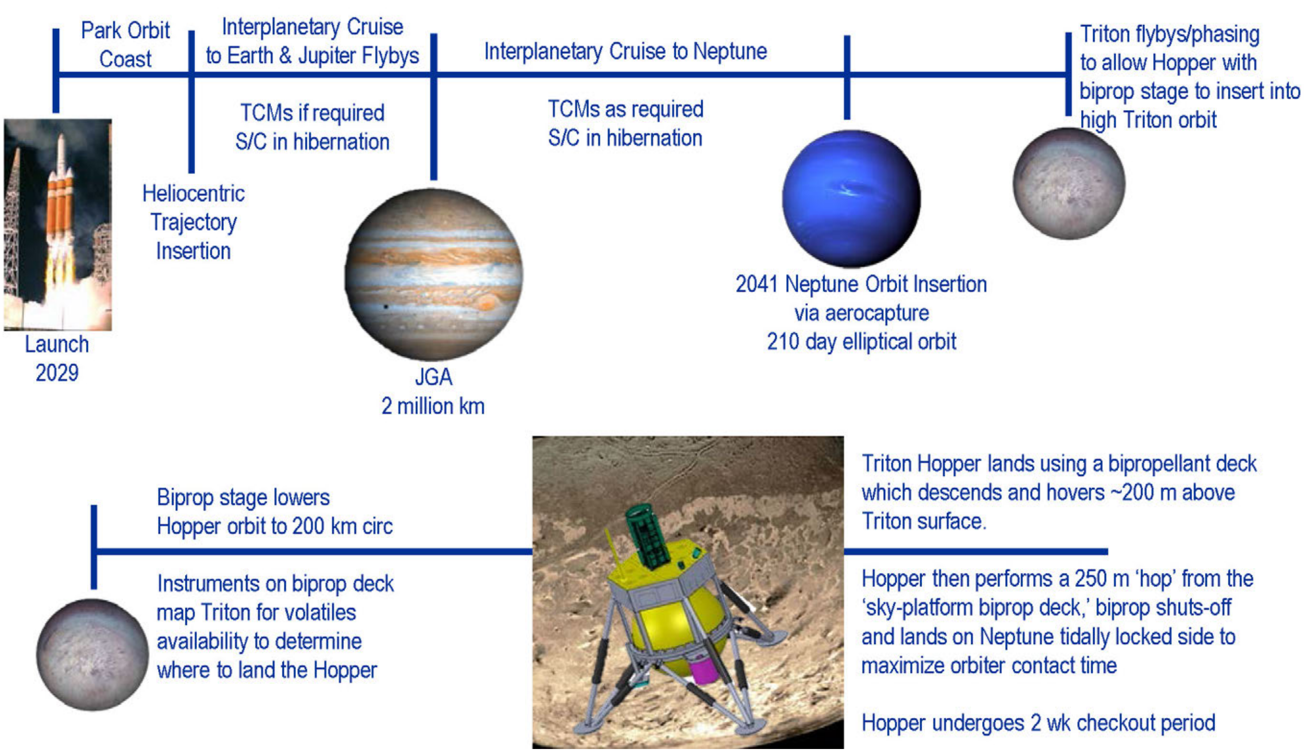


Figure 3.1—Hopper launch and delivery CONOPs.

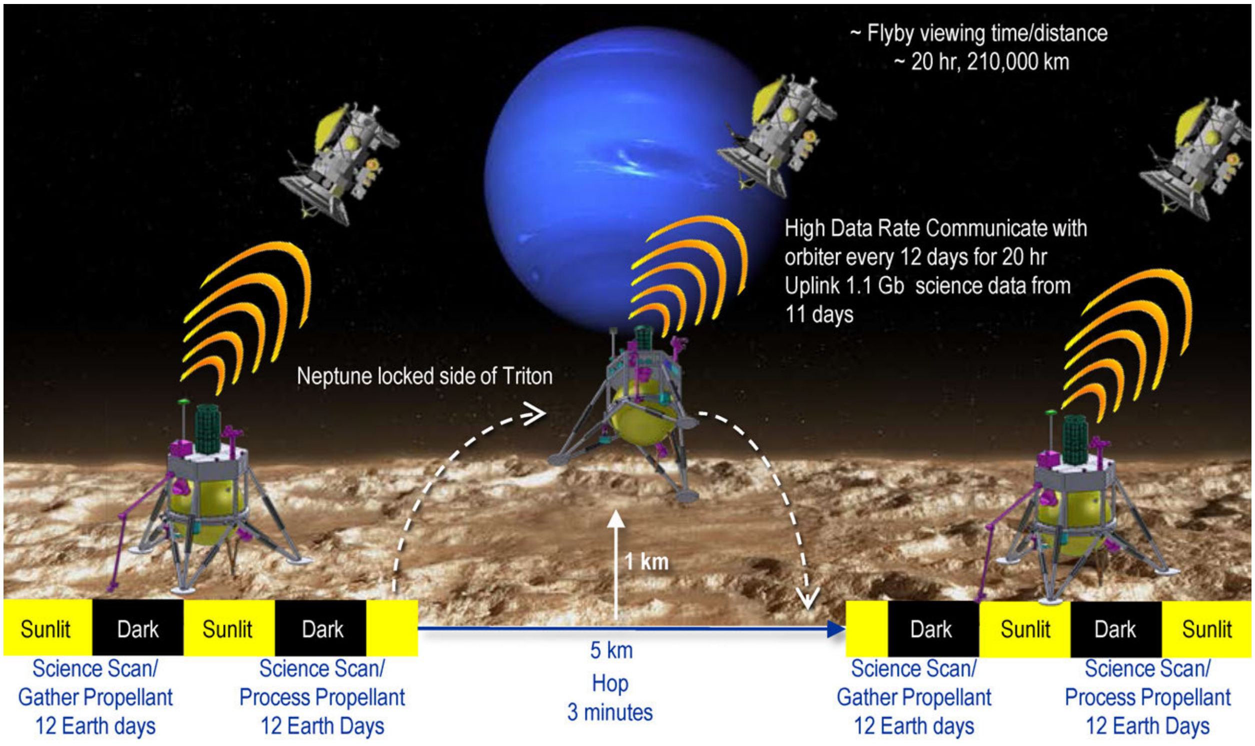


Figure 3.2—Triton Hopper surface CONOPS.

3.2 Mission

3.2.1 LV Ascent, Park Orbit and TTI

The trans-Triton injection (TTI) burn will be performed with the launch vehicle, a Delta IV Heavy. On October 26, 2029, the Delta-V Heavy will deliver 8685.5 kg to a C_3 of 4.3 km^2/s^2 to begin the interplanetary cruise phase to Triton.

3.2.2 Earth to Triton Cruise

The S/C will be placed in hibernation for the majority of the cruise phase except during periods of thrusting, planetary encounters for gravity assists, planned health assessments, or when any trajectory correction maneuvers must be performed. The Earth to Triton trajectory is shown in Figure 3.3.

Following TTI, the S/C will use the SEP thrusters to maneuver the S/C in heliocentric space. Prior to capturing in orbit around Neptune, the S/C performs Earth and Jupiter gravity assist maneuvers to further increase the S/C velocity. The S/C arrives at Neptune on October 23, 2041. The total interplanetary cruise phase of the mission requires 12 years and 2610 kg of Xe propellant to complete.

After arriving at Neptune, the S/C will aerocapture into a 210-day orbit around Neptune. The net delivered mass to Neptune prior to performing the Triton capture burns is 4475 kg (in addition to the now dry 1600 kg SEP vehicle, see Table 3.1). Following the aerocapture, a plane-change burn will be performed at apogee to adjust the inclination of the S/C. The S/C will then perform aerobraking to reduce the apogee altitude from the initial 210-day orbit to an elliptical orbit with apogee at Triton altitude. A powered flyby of Triton followed by a short coast reduce the hyperbolic arrival velocity and adjust the arrival velocity vector to minimize the ΔV required to capture into a circular 200 km Triton orbit.

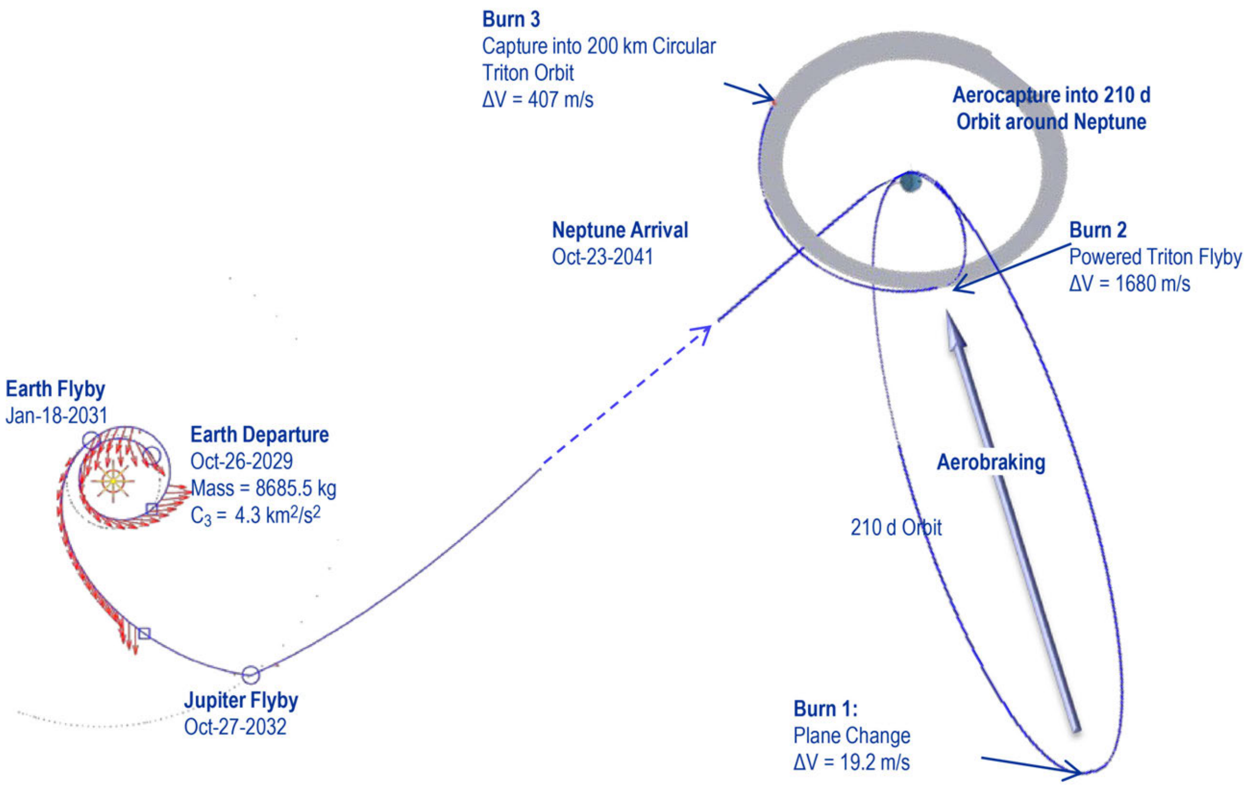


Figure 3.3.—Interplanetary and delivery trajectory.

TABLE 3.1—INTERPLANETARY DELIVERY SUMMARY

EJGA Delivered Mass	
EOL Power to EP System	30 kW
I_{sp}	~2000 s
Arrival Mass at Neptune	6075 kg
SEP Vehicle Dry Mass	1600 kg
Pre-NOI Net Delivered Mass	4475 kg

3.2.3 Triton Entry, Descend, Landing and Operation

All mission analyses for the Triton Hopper's surface operational scenarios were performed using the Optimal Trajectories by Implicit Simulation (OTIS) program, version 4. The exception is for the ΔV estimates of the Earth departure using an ELV; those were taken from analyses with the Copernicus trajectory program.

There were two delivery scenarios that were considered for the initial delivery of the Hopper vehicle. One scenario considers a delivery, via ELV, to a Neptune based orbit or a direct delivery to a Triton orbit. The Hopper then performs appropriate de-orbit and initial landing. The other scenario assumes the Hopper vehicle was delivered to a 200 km circular orbit around Triton in some fashion and proceeds to de-orbit and land. The latter became the baseline due to the large savings in propellant provided by the Neptune aerocapture system. The unpursued direct option is described in Appendix A.

3.2.3.1 Hopper Delivery—From 200 km Circular Triton Orbit

This initial delivery scenario assumes the Hopper has been delivered as described in Section 3.2.2 to a 200 km circular orbit at Triton. The delivered vehicle stack consists of the Hopper vehicle and a bipropellant stage. This Hopper vehicle represents a more mature configuration and is consistent with the

TABLE 3.2—SUMMARY OF DESCENT AND LANDING FROM 200 km TRITON ORBIT

Triton Hopper in the 2040 timeframe—Descend and land from 200 km circular orbit; gets Hopper to the surface														
Final landing via Hopper's cold gas rocket: I_{sp} from CGR nitrogen model (T= 300 K, P= 2,000 psia, 2% residual)														
Stage	Initial (max) thrust, N	Initial (max) thrust, lbf	Initial T/W, Earth	Initial T/W, Triton	Down range, km	Altitude, km	Duration,[a] s	Initial mass, kg	Final mass, kg	Prop used, kg	ΔV,[a] m/s	Final V, m/s	Final T/W, Triton g	Notes
Biprop	14,095.2880	3,168.7396	2.7173	34.2078	1.1192	199.9845	1.3652	528.9462	522.8141	6.1321	36.5916	922.8286	34.2078	De-orbit
Coast	0.0000	0.0000	0.0000	0.0000	3,443.9151	1.0000	3,819.5582	522.8141	522.8141	0.0000	36.5916	1,059.1679	0.0000	Fall
Biprop	14,095.2880	3,168.7396	2.7492	34.6090	3,462.7060	0.2000	3,853.1244	522.8141	372.0473	150.7668	1,104.1995	0.1001	48.6338	Slow down
Delivered Hopper									356.3574					Minus biprop stage[b]
	Initial (max) thrust, N	Initial (max) thrust, lbf	Initial T/W, Earth	Initial T/W, Triton	Down range, km	Peak altitude, km	Duration, s	Initial mass, kg	Final mass,[c] kg	Prop used, kg	ΔV, m/s	Final V, m/s	Final T/W, Triton g Earth g	
Hopper	1,203.8000	270.6344	0.3445	4.3364	0.2500	0.2970	49.7145	356.3575	325.6149	30.7426	42.0626	0.0100	1.4237506 0.1130969	Land[d]
									325.0000					Dry Hopper

[a] Cumulative amount
[b] Equals 10% of prop used
[c] Includes 0.6 kg of unusable (liquid) nitrogen - ~ 2% of 30.7 kg used prop (assume a total load of ~32 kg)
[d] Landing velocity limit of 1 m/s; no landing g-limit

baseline operational vehicle described in the next section. The bipropellant stage has a thrust level of 14,095 N and an I_{sp} of 320 s. While in the 200 km orbit, the bipropellant stage performs a de-orbit burn followed by a coast along a suborbital arc (i.e., fall towards surface). Next, the bipropellant stage performs a slowdown burn to reduce velocity. Finally, the Hopper performs a controlled landing hop on Triton's surface. The delivered Hopper wet mass is approximately 356 kg which accounts for a dry Hopper of 325 kg and a partial propellant load, enough to perform the initial landing hop to the surface 250 m downrange. The Hopper's propulsion system is described (top level) in the next section. Table 3.2 shows a summary of the delivery scenario from a 200 km Triton orbit.

3.2.4 Operation of Triton Hopper

As described above, once on Triton's surface, the Hopper collects resources and produces propellant that is used to perform its hops. The Hopper vehicle has a wet (fully loaded) mass of 450 kg. This includes a dry Hopper weighting 325 kg and a full propellant load of 125 kg with a 2% unusable propellant margin since nitrogen liquefies below a propellant remaining fraction of 0.02. The propulsion system consists of cold gas rockets; these rockets have been used in the mission analysis with an initial tank pressure of 2,000 psia, temperature = 300 K. The propulsion system is described in detail in Section 4.0. The thrust level of the Hopper, derived from the baseline case (and was optimized for maximum range of that case), is 1203.8 N (270.6 lbf).

Initial analyses have been performed to ascertain the required thrust level for the Hopper propulsion system, various hopping distances and propellant loads. The baseline case is a result of using a cold gas nitrogen rocket and maximizing the hopping distance using a 125 kg propellant limit (with 2% unusable). The sequence of events for each operational hop is as follows:

- Vertical takeoff followed by a pitch-over during which flight path angle rate is not to exceed –0.175 rad/s (approximately –10°/s). Constant maximum thrust. Pitch-over profile can be up to second order in shape
- Ballistic coast with a non-impulsive flow at 5% of the nominal maximum burn flow rate to account for attitude control maneuver's propellant usage

- Landing profile: Propulsive landing during which the thrust is allowed to throttle down to 10 percent of maximum thrust, if needed. The throttle profile is an optimized spline shape, subject to a (–0.1 to 0) (1/s) rate limit down to 10 m altitude and vertical position, followed by a (–1.0 to 0) (1/s) rate limit during the final 10 m vertical descent. Final "touchdown" throttle setting (may range from 0 to 0.3 (1/s)) is derived from requiring the vehicle to remain in a vertical orientation with a final altitude ≤ 0.1 m and the final velocity ≤ 0.01 m/s
- Landing pitch-up profile can be up to third order in shape
- Vehicle ends in a vertical position

Table 3.3 shows a summary of this baseline operational hopping scenario on Triton's surface.

Figure 3.4 and Figure 3.5 show the altitude versus time and versus range, respectively, for the initial delivery from a 200 km Triton orbit (final landing performed by Hopper) in blue and the baseline hop operational hop while on the Triton surface in red. The resulting thrust-to-weight ratio is over 3.

TABLE 3.3—SUMMARY OF BASELINE OPERATIONAL HOPPING SCENARIO

Triton Hopper baseline operational scenario using nitrogen CGR propulsion model (T= 300 K, P= 2,000 psia, 2% residual)														
Cold gas rocket: M_0= 450 kg														
Initial (max) thrust, N	Initial (max) thrust, lbf	Initial T/W, Earth	Initial T/W, Triton	Hop distance range, km	Peak altitude, km	Total time, s	Hang time (ballistic), s	Prop used, kg	Final mass, kg	ΔV, m/s	Final V, m/s	Final T/W, Triton g	Final T/W, Earth g	Notes
1,203.7833	270.6207	0.272782	3.433985	5.2190	1.3000	141.6320	117.9537	122.5000	[a]327.5000	154.0605	0.0100	1.4155	0.1124	Max range
									325.0000					Dry Hopper

[a] Includes 2.5 kg of unusable (liquid) nitrogen = 2% of 125.0 kg loaded prop

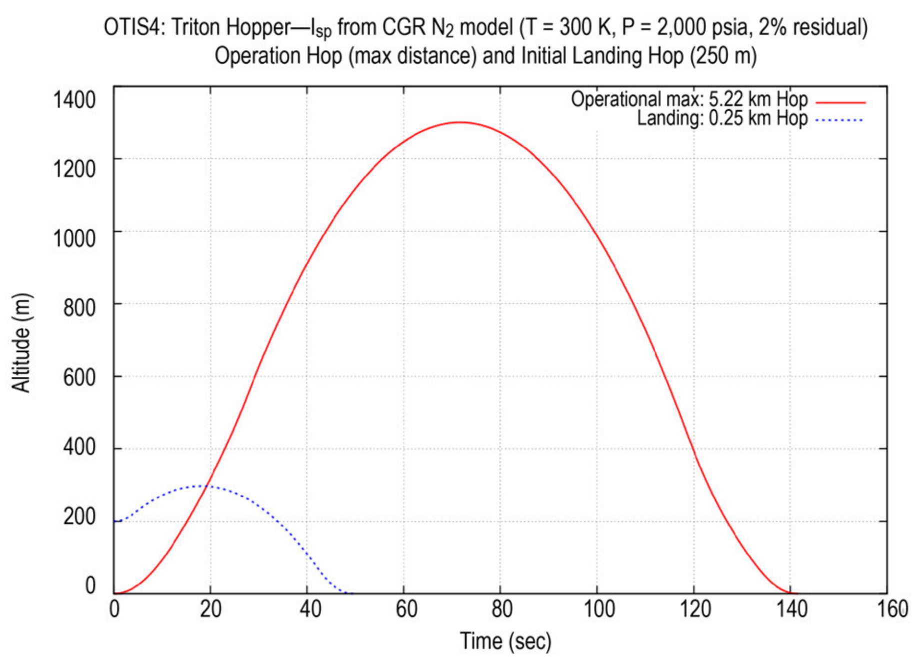


Figure 3.4—Altitude versus time for delivery landing from 200 km Triton orbit (blue) and baseline operational hop.

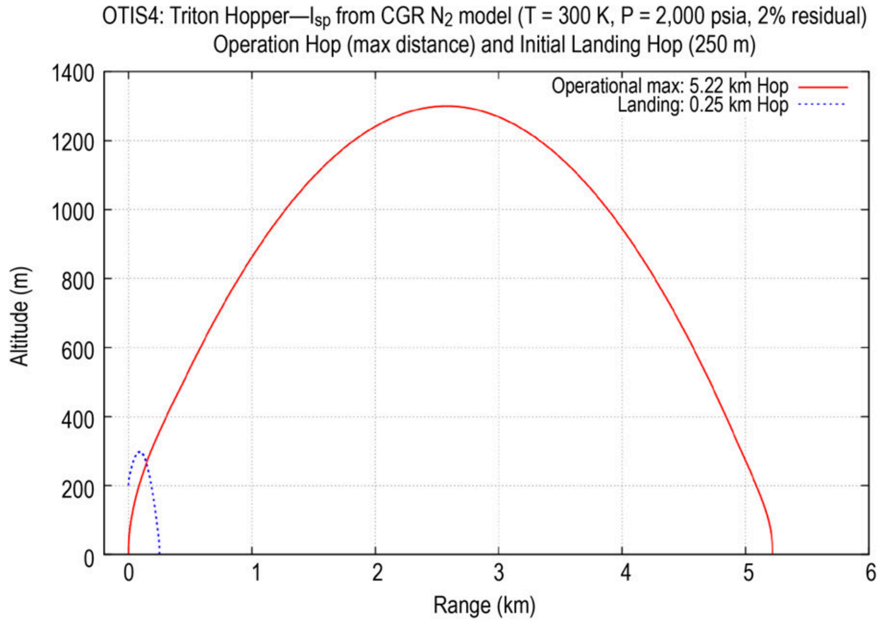


Figure 3.5—Altitude versus range for delivery landing from 200 km Triton orbit (blue) and baseline operational hop.

TABLE 3.4—TRITON HOPPER—CASE 1—MISSION ΔV SUMMARY

Phase no.	Phase name	Pre-burn mass, kg	Main ΔV, m/s	ACS ΔV, m/s	Propulsive element	I_{sp}, s	Main prop, kg	ASC prop, kg	Post burn mass, kg	Change in mass, kg
1	Powered Triton flyby	2442.5	1680	0.0	Solid stage	294	1080	0.0	1362.5	–1080.0
2	Drop solid stage	1362.5	------	----	----------------	-----	------	---	1268.6	–93.9
3	Capture into 200 km circular Triton orbit	1268.6	407	0.0	Descent stage	323	153	0.0	1115.7	–153.0
4	De-orbit from 200 km circular	1115.7	37	0.0	Descent stage	323	13	0.0	1102.8	–12.8
5	Slow down burn	1102.8	1066	0.0	Descent stage	323	315	0.0	787.8	–315.0
6	Drop descent stage	787.8	------	----	----------------	-----	------	---	389.8	–398.0
7	Landing burn, 250 m hop off of descent stage	389.8	41	0.0	Lander	52	30	0.0	359.9	–29.9
8	ACS for landing burn	359.9	0	0.1	Lander	52	0	0.7	359.2	–0.7
9	Process propellant for 5 km hop	359.2	------	----	----------------	-----	–117	0.0	476.6	117.4
10	~5 km hop	476.6	141	0.0	Lander	52	115	0.0	361.7	–114.9
11	ACS for 5 km hop	361.7	0	3.5	Lander	52	0	2.5	359.2	–2.5
	Total, solid stage		1680	0.0			1080	0.0		
	Total, descent stage		1509	0.0			481	0.0		
	Total, lander		182	4.5			145	3.2		

A summary of the mission ΔV's can be seen in Table 3.4. The propellant numbers in Table 3.4 represent usable propellant. Additional propellant, 5% of the usable, is carried as margin and held as inert mass in the both the descent stage and lander. This margin is assumed available for terminal descent collision avoidance with ground debris. No margin on propellant was carried in the solid stage, rather the inert mass of that stage was assumed to be 8% of the total mass of the stage. The propellant required for the Hopper to hop off of the descent stage and perform the initial landing on Triton, 29.9 kg of main propellant and 0.7 kg of ACS propellant, is pre-loaded on the Hopper, while the propellant required for each hop, 114.9 kg of main propellant and 2.5 kg of ACS propellant, is processed and loaded onto the Hopper while on the surface of Triton. The totals for the Lander in the last row of Table 3.4 represent the amount of propellant the Hopper requires to perform the initial landing and one 5 km hop.

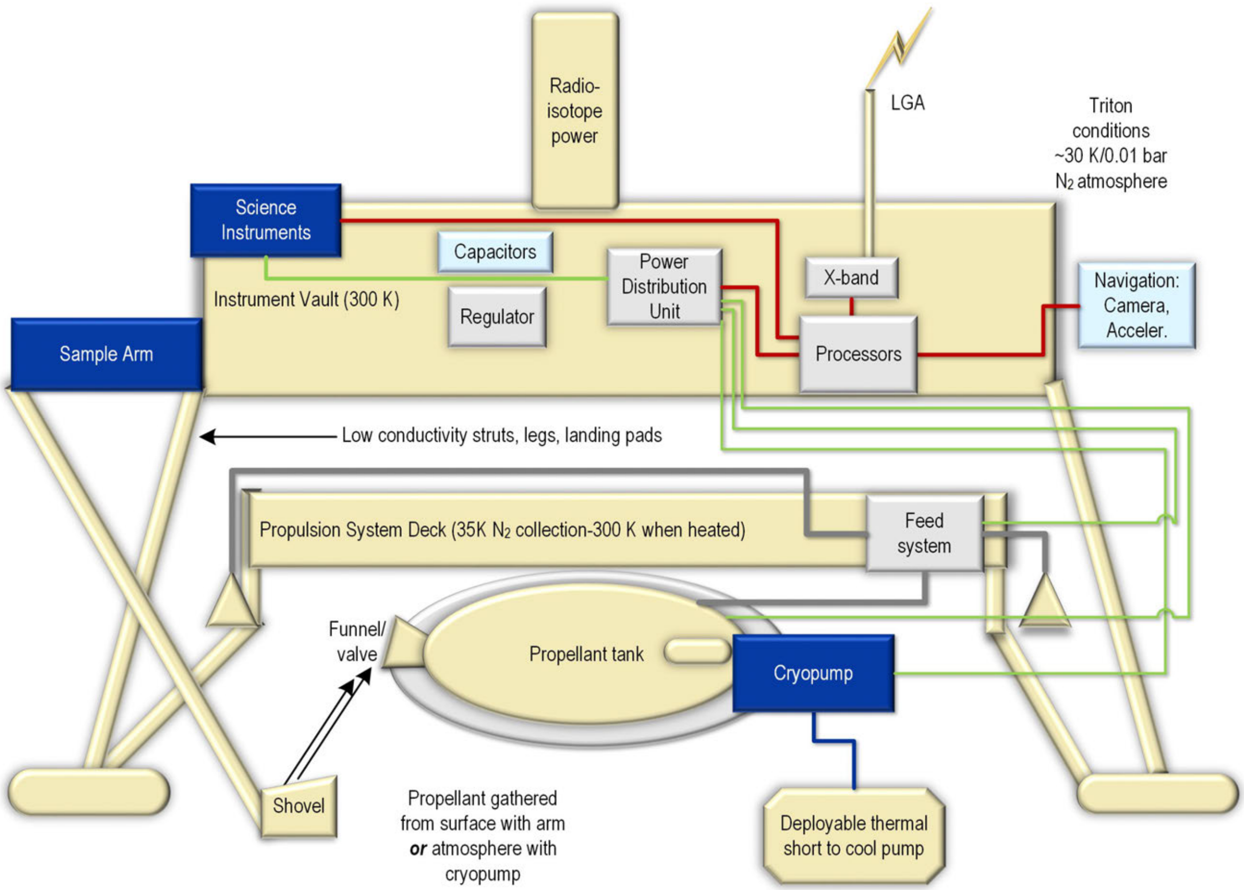


Figure 3.6—Triton Hopper Block Diagram

3.3 System Level Summary

The Triton Hopper system uses a radioisotope power system that powers two ISRU systems to generate the nitrogen propellant necessary to propel the Hopper across Triton. Figure 3.6 shows the Triton Hopper block diagram and illustrates how all of the major vehicle subsystems are integrated.

The Triton Hopper MEL contains the mass roll up; including a subsystem aggregate MGA, for each vehicle subsystem. The Triton Hopper MEL can be found in Table 3.5.

The MEL shown in Table 3.6 captures the bottoms-up estimation of CBE and growth percentage of the Triton Hopper that the subsystem designers calculated for each line subsystem. In order to meet the total required mass growth of 30% an allocation is necessary for growth on basic dry mass at the system level, in addition to the growth calculated on each individual subsystem. This additional system-level mass is counted as part of the inert mass to be flown along the required trajectory. Therefore, the additional system-level growth mass impacts the total propellant required for the mission design.

3.4 Triton Entry System/Cruise Stage

Four primary architecture elements are necessary for the Triton Hopper to reach the surface of Triton: SEP Cruise Stage, Neptune Aerocapture System and Orbiter, Solid Propellant Triton Orbit Insertion Stage and a Bipropellant Triton Descent Stage. Details of the mission and associated ΔV for each element can be found in Section 3.2, Mission.

TABLE 3.5—TRITON HOPPER MEL

WBS number	Description Case 1 CD-2015-127	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06	Triton Hopper	1206.81	4.1	49.32	1256.13
06.1	Hopper	402.84	10.5	42.49	445.33
06.1.1	Science Payload	22.99	3.8	0.87	23.86
06.1.2	Attitude Determination and Control (AD&C)	14.61	8.1	1.19	15.80
06.1.3	C&DH	14.70	29.0	4.26	18.96
06.1.4	Communications and Tracking	4.20	10.0	0.42	4.62
06.1.5	Electrical Power Subsystem	39.50	11.5	4.53	44.03
06.1.6	Thermal Control (Non-Propellant)	33.00	15.0	4.95	37.95
06.1.7	Propulsion (Chemical Hardware)	87.21	17.8	15.56	102.77
06.1.8	Propellant (Chemical)	127.12	0.0	0.00	127.12
06.1.9	Propulsion (EP Hardware)	0.00	0.0	0.00	0.00
06.1.10	Propellant (EP)	0.00	0.0	0.00	0.00
06.1.11	Structures and Mechanisms	59.51	18.0	10.71	70.22

TABLE 3.6—TRITON HOPPER SYSTEM SUMMARY

WBS	Main subsystems	Basic mass, kg	Growth kg	Predicted mass, kg	Aggregate growth, %
06.1	Hopper	402.84	42.49	445.33	11
06.1.1	Science Payload	22.99	0.87	23.86	4
06.1.2	AD&C	14.61	1.19	15.80	8
06.1.3	C&DH	14.70	4.26	18.96	29
06.1.4	Communications and Tracking	4.20	0.42	4.62	10
06.1.5	Electrical Power Subsystem	39.50	4.53	44.03	11
06.1.6	Thermal Control (Non-Propellant)	33.00	4.95	37.95	15
06.1.7	Propulsion (Chemical Hardware)	87.21	15.56	102.77	18
06.1.8	Propellant (Chemical)	127.12		127.12	0
06.1.9	Propulsion (EP Hardware)	0.00	0.00	0.00	TBD
06.1.10	Propellant (EP)	0.00	------	0.00	TBD
06.1.11	Structures and Mechanisms	59.51	10.71	70.22	18
System Level Growth Calculations Hopper					Total growth
	Dry Mass Desired System Level Growth	275.72	82.72	358.44	30
	Additional Growth (carried at system level)	--------	40.23	--------	15
	Total Wet Mass with Growth	402.84	82.72	485.56	---

During mission analysis, the SEP Cruise stage and Neptune Aerocapture System and orbiter were resized based on the assumptions presented in the paper “Mission Trades for Aerocapture at Neptune” (Ref. 2). Additional details on the system design assumptions, with any changes noted in Section 3.2.

The solid propellant Triton Orbit Insertion stage was sized based on a propellant mass fraction (PMF) assumption of 0.93 with and I_{sp} assumption of 293 s. The commercially available ATK Star 37GV is close to what is needed (but would need to be modified for deep space use.) Further work is necessary to fully address all design issues associated with using a solid rocket motor for this application, but those issues are outside the scope of the current study.

The Bipropellant Triton Descent Stage was sized to have a PMF of approximately 0.55 using an I_{sp} of 303 s. A PMF of 0.55 is within the expected range for this type of stage based on historical stage designs. To meet this desired stage PMF, a detailed design was completed for the propulsion system to understand the design complexities associated with this type of system. The remaining inert mass of the system was then adjusted to achieve the target PMF for the descent stage. It is assumed that this stage would use the Triton Hopper avionics and landing sensor systems. It is therefore assumed the descent stage would contain only the propulsion, structure and minimal control avionics necessary to perform its mission. A dedicated follow-on design study would be necessary to fully work out the design details of this stage, as the detailed descent stage design is outside the scope of this study.

Based on the mission assumptions used to deliver the Hopper to the Triton surface, Table 3.7 shows how the Triton Hopper mass compares to the available delivery mass at Neptune/Triton. Additional details on the various components used to deliver the Triton Hopper to the surface of Triton are listed in Section 3.4.

3.5 Triton Concept Drawings and Descriptions

This section focuses solely on the layout of the Triton Hopper in its landed configuration. Design of the launch vehicle packaging and all other elements of the mission architecture are beyond the scope of this study. The configuration of the Triton Hopper while on the surface can be seen in Figure 3.7.

TABLE 3.7—TRITON HOPPER DELIVERED MASS SUMMARY

	Mass, kg	Assumptions (if applicable)
Net mass delivered to Neptune orbit	4475.0	COMPASS team trajectory analysis
Aerocapture System	1119.0	From reference Neptune mission paper
Payload left in Neptune orbit	792.0	From reference Neptune mission paper
Available mass for Triton Hopper and descent stages	2564.0	
Triton orbit insertion stage, solid	1189.2	Solid stage I_{sp} = 293 s, PMF = 0.93
Triton descent stage, bipropellant	885.7	Bipropellant I_{sp} = 323 s, PMF = 0.55
Triton Hopper, mass at launch	391.4	Loaded propellant at launch enough for short landing hop, ~33 kg
Available launch margin	97.8	
Available launch margin, %	3.8%	

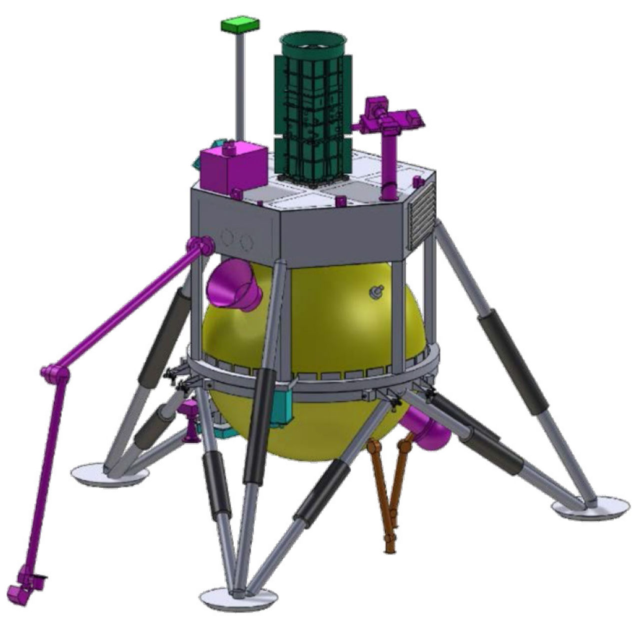


Figure 3.7—Isometric View of the Triton Hopper

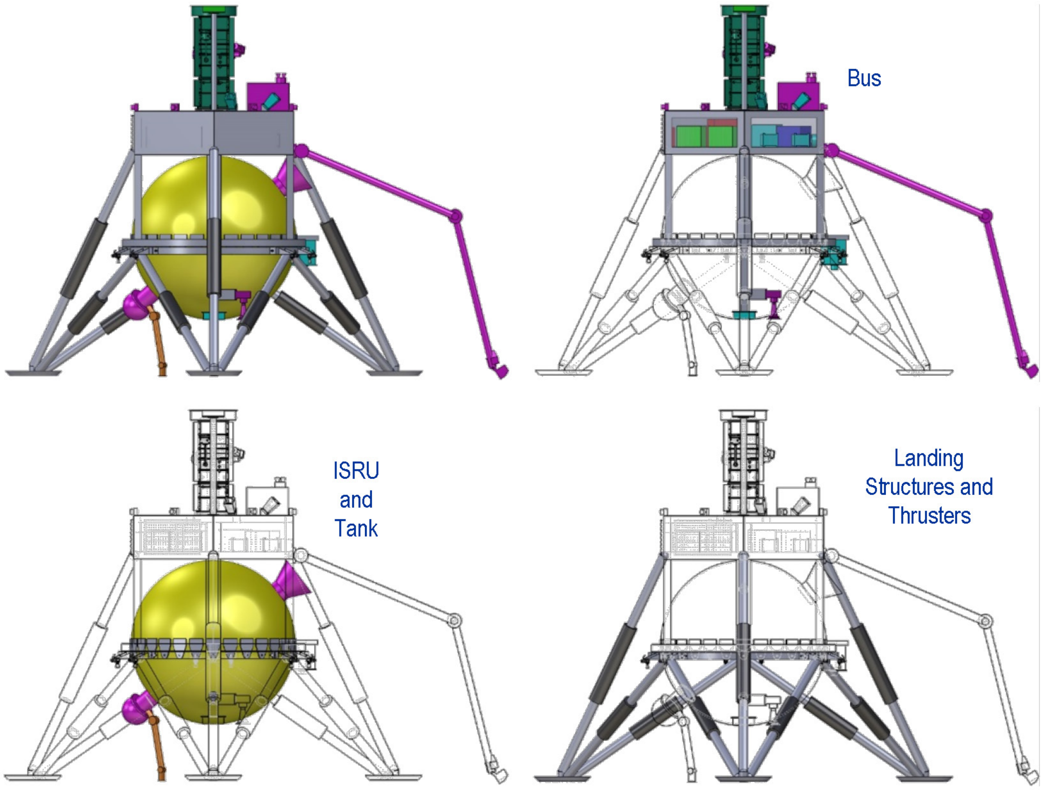


Figure 3.8—Major Elements of the Triton Hopper.

The Triton Hopper can be separated into three major elements: the bus, the ISRU system with propellant tank, and the landing structure with thrusters. Figure 3.8 shows the breakout of these three major elements. All of the Triton Hopper components not dedicated to the ISRU, propulsion, and landing structures are contained on the bus. The cryocooler and tank collection funnel for the ISRU system are mounted directly to the propellant tank itself. Finally, the landing element is comprised of the landing legs, thrusters, and the tank mounting structure.

These three major elements of the Triton Hopper are thermally isolated from one another through the structural layout and material properties of the structures. This isolation is required to prevent heat transfer between the "hot" and "cold" elements that comprise the Triton Hopper.

All of the electronics are contained on the bus element and due to the waste heat generated by these electronics; the bus is considered the "hot" element of the Hopper. In order to prevent the heat from the bus entering the propellant tank while on Triton's surface, the only conductive path from the bus into the tank is down to the landing pads through the three longer landing legs, then up through the six shorter legs and into a square tubular ring that provides the interface to the mounting tabs contained on the tank. While this is a conductive path for heat transfer from the bus to the tank, the long path through low conductive structures keeps the transfer to a minimum. The structure of the bus is comprised of a hexagonal space frame with closeout panels that provide the interfaces for a majority of the components contained on the bus. The vertical members located at the vertices of this hexagonal space frame extend down to a tubular ring that goes around the circumference of the tank. This ring does not come in direct contact with the tank and "floats" above the second ring that interfaces directly to the mounting tabs contained on the tank while the Hopper is on the Triton surface. During the thrusting and landing phases of the mission, both of which do not require thermal isolation between the bus and tank, these two rings come together and allow the loads to be carried through the vertical members of the bus structure. Once landed on the surface, the three longer members of the landing legs will “push” the bus structure upwards slightly to provide the separation between the two rings, and thus provide the thermal isolation between the bus and tank. Additional structure extends outward from this ring to mount the LIDAR, and downward from this ring to mount the landing radar and the MARDI used for surface science. Details on the components contained on the bus are discussed later in this section while a more detailed description on the structural design of the Triton Hopper can be found in Section 4.8, Structures and Mechanisms.

The propellant tank, cryocooler, and tank collection funnel comprise the second major element of the Triton Hopper. The tank is mounted to the tubular ring to which the six shorter landing legs are integrated. Again, this provides a thermal barrier between the propellant tanks and the “hot” bus structure, with the only conductive path between the two being down the three longer landing legs, into the landing pads, and up the six shorter landing legs into the ring to which the tank is mounted. This conductive path minimizes the heat transfer between the bus and tank by utilizing low conductive materials for the structures. Mounted directly to the tank is the “cold” end of the cryocooler. Waste heat from the “hot” end of the cryocooler is transferred through a mechanical arm that extends down to the cold Triton surface. A tank collection funnel is located on the tank nearly opposite from the cryocooler and is used to allow the frozen surface from Triton to be dumped into the tank from a robotic arm containing a scoop in order to turn the frozen surface into propellant required for the hopping to another location. The propellant tank is heated, when needed, by a series of heaters located on the tank (not shown in the CAD design), and is cooled, when needed, by a mechanical arm attached to the tank that transfers heat from the tank down to the cold Triton surface. Those components included on the propellant tank and ISRU element of the Triton Hopper can be seen in Figure 3.9. A more detailed discussion on the thermal control for the tank can be found in Section 4.6, Thermal Control System.

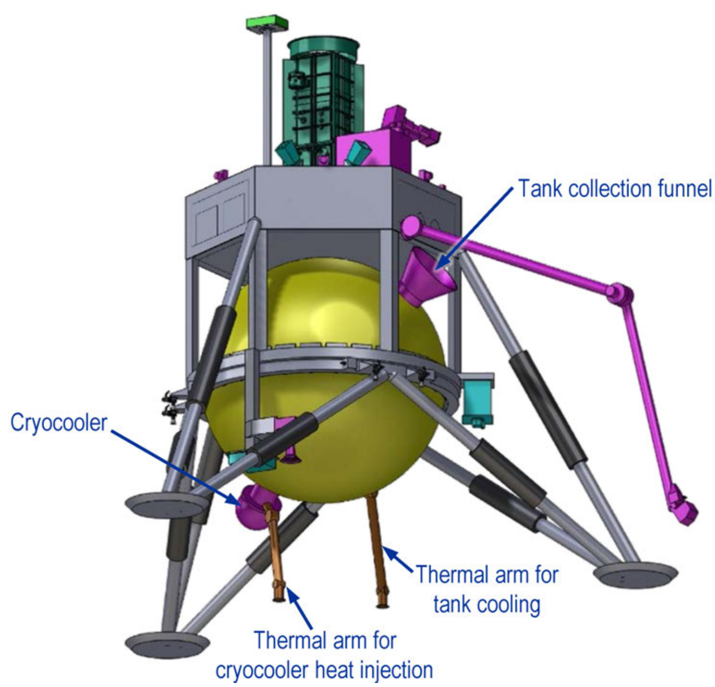


Figure 3.9—Components Contained on the Propellant Tank and ISRU Element of the Triton Hopper.

The landing structure is the third major element that comprises the Triton Hopper. It is comprised of the three landing pads, landing legs, and square tubular ring that contain the thrusters and to which the tank is mounted. The landing legs and pads provide a 3-m diameter footprint. Each of the three landing legs is comprised of one long member extending from the pad to the hexagonal bus structure, two shorter members that extend between the pads and the ring used to mount the tank, and a single 16-in. diameter landing pad. The six main thrusters and six RCS thrusters are mounted to structure contained on the ring to which the tank is mounted. The main thrusters are grouped into three pairs, each (pair) 120° from one another around the ring, and provide an upward thrust vector. The RCS thrusters are also grouped into three pairs, each (pairs) 120° from one another around the ring, and provide a tangential thrust vector in opposing directions for each pair. Thruster locations and orientation can be seen in Figure 3.10. A more detailed discussion on the thrusters can be found in Section 4.7, Propulsion System, while more details on the landing legs can be found in Section 4.8, Structures and Mechanisms.

Figure 3.11 shows two images of the lander with a color-coded breakdown of all the components by subsystem. Note that the components related to the ISRU system (cryocooler and tank collection funnel) are included in Section 4.1.

Those components contained on the bus element but located external to the bus structure include the ASRG, communications antenna, two radiator panels, two star trackers, six cameras that comprise the 360-camera, meteorology package, Mastcam, IR spectrometer, chemical analysis package, robotic arm (containing the APXS sensor head, scoop, and a camera), the MARDI camera, landing radar antenna, and the LIDAR. All of these components are called out in Figure 3.12.

The ASRG is mounted to the center of the top closeout panel of the bus structure. This orientation, while adding height to the lander, allows the existing mounting interface to be utilized and mounted directly to the closeout panel, and allows all of the radiator fins to be used for heat rejection, something that could not be done if the ASRG was "laid down" on the panel to minimize the lander height. A fixed boom is extended from the top closeout panel of the bus structure to which the X-band omni antenna is mounted to the top. The boom is required to keep the antenna above the ASRG with an unobstructed view of the sky.

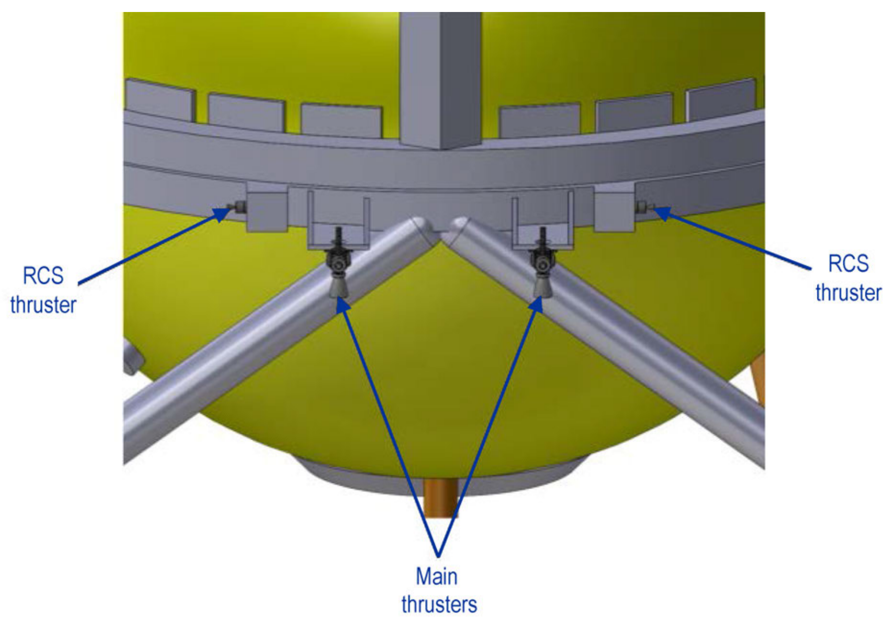


Figure 3.10—Thruster Location and Orientation

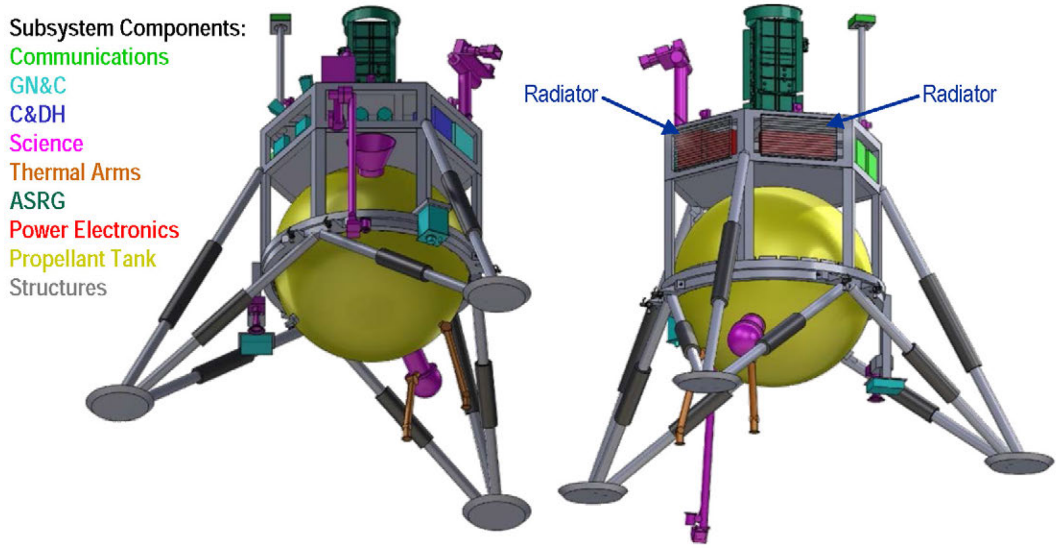


Figure 3.11—Subsystem Breakdown of the Triton Hopper

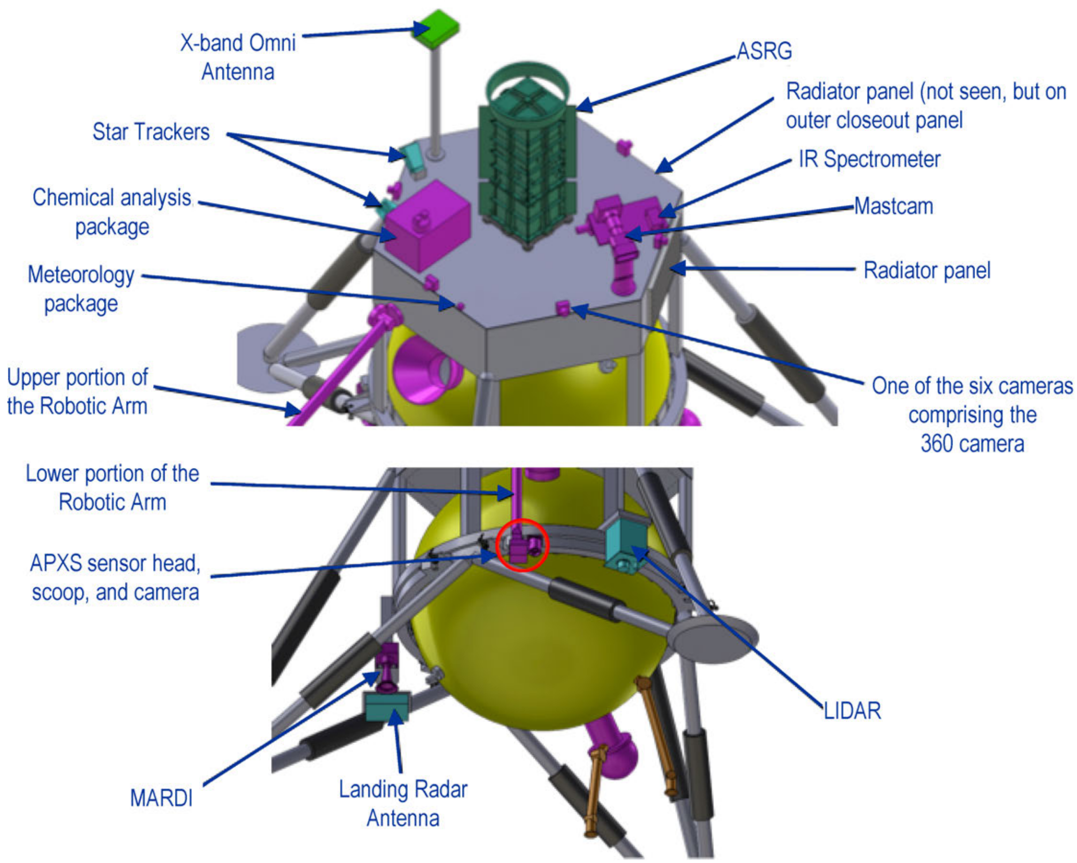


Figure 3.12—Components Located External to the Triton Hopper Bus

The two radiator panels that provide the heat rejection for the internal electronics are mounted to adjacent closeout panels on the sides of the bus structure. While this location provides a view of the Triton surface, the cold temperature of the surface does not reduce the heat rejection capability of the radiators. Both star trackers are mounted out at the edge of the top closeout panel of the bus structure opposite from one of the radiator panels in order to avoid any potential thermal distortion of their view. Each star tracker is angled upward at a 45° angle and both are angled away from one another by 90°. This location and orientation ensures that each have an unobstructed view of deep space, and that in the event one have the sun in its field-of-view, the other will not.

Each of the six individual cameras that comprise the 360°-camera is mounted out at the edge and in the center of each side of the hexagonal shaped closeout panel. Each camera is pointed parallel to the Triton surface. These locations and orientation of each camera provide a full 360° view while on the surface of Triton. The meteorology package is also mounted to the top closeout panel out near the edge, in order to keep it away from the ASRG. The Mastcam and IR Spectrometer are both mounted to a platform located on the top of a mast extending up from the top closeout panel of the bus. The mast has the capability to rotate 360° providing a panning capability, while the platform on top of the mast provides a tilt capability. While the view of the Mastcam will be blocked when panned back towards the ASRG, the mast was located directly opposite the antenna boom so that this boom will not provide an additional blockage other than that from the ASRG. The chemical analysis package is also mounted directly to the top closeout panel and is located out near the edge of that panel. The side closeout panel relative to the chemical analysis package provides the interface for the base of the robotic arm. By locating these two components on the same side of the bus, the scoop contained on the end of the arm can be used to place surface samples into the chemical analysis package, in addition to the tank collection funnel, also located on this side of the bus, though mounted to the tank. A camera and APXS sensor head are also located on the end of the robotic arm.

The MARDI camera from the science subsystem, along with the landing radar antenna, is mounted to a tubular structure that extends down from the ring contained on the bus structure. This allows those components to have an unobstructed view of the surface while also maintaining a thermal barrier between them and the propellant tank. Finally, the LIDAR used by the landing subsystem is mounted off of the ring portion of the bus structure. Again, this allows for an unobstructed view of the surface and provides a thermal barrier between itself and the propellant tank.

All of the remaining subsystem components are all contained inside the hexagonal bus structure and are mounted to the inside face of the side closeout panels. These components are all called out in Figure 3.13. For the Power Subsystem, the ultracapacitor and DC/DC converter, ACU Stirling controller, and the PMAD-shunt are located on the faces directly behind the two radiator panels, as these components provide a majority of the waste heat. The cPCI enclosure of the C&DH subsystem, and the landing radar and velocimeter for the landing subsystem both share a face. Both of the IMUs are mounted to a single closeout panel, while the star tracker electronics are mounted to the closeout panel directly underneath the location of the star tracker optical heads. Finally, both X-band transceivers are mounted to a single closeout panel near the location of the boom containing the antenna. More detailed discussions on all of the subsystems can be found in their corresponding sections of this document.

The overall dimensions of the landed Triton Hopper are shown in Figure 3.14, while some additional views of the Hopper configuration can be seen in Figure 3.15 and Figure 3.16.

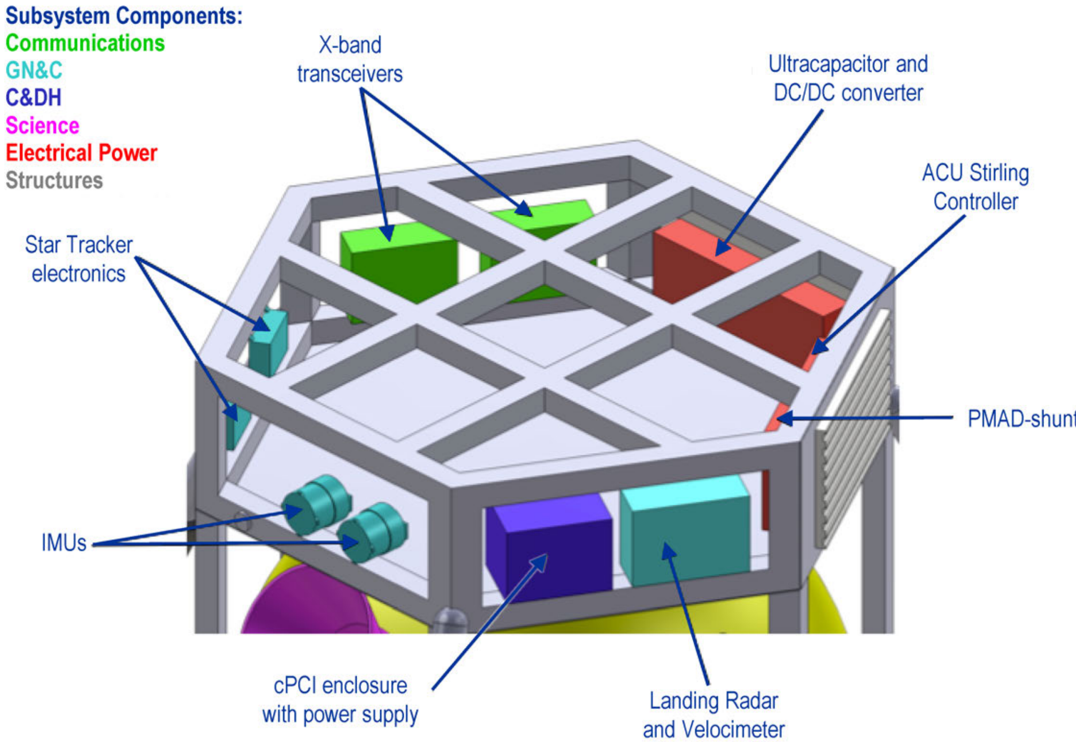


Figure 3.13—Components Located Inside the Triton Hopper Bus

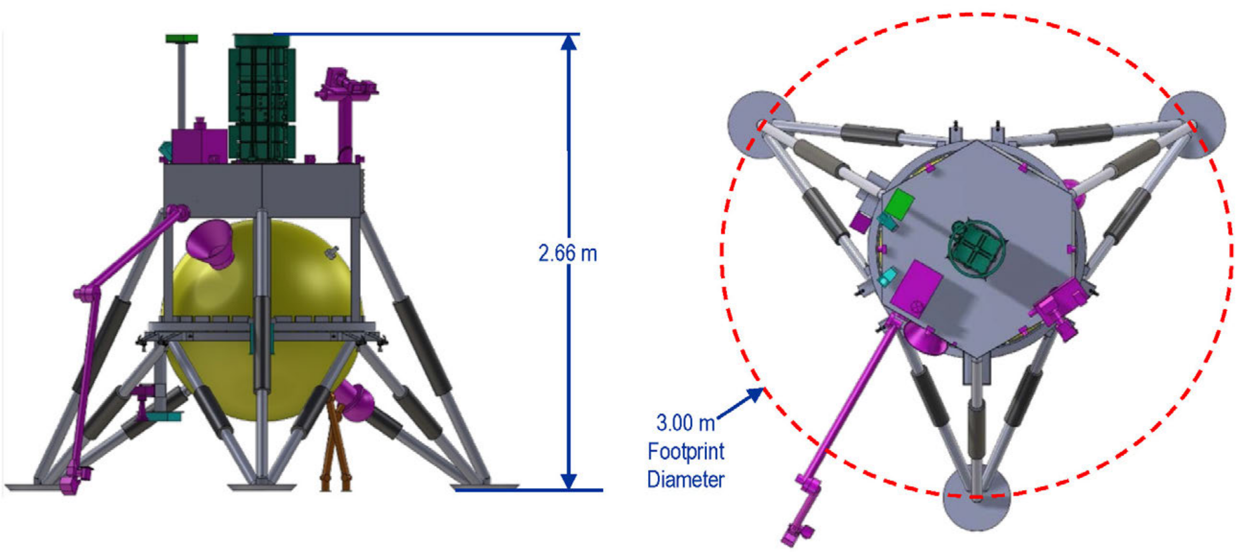


Figure 3.14—Overall Dimensions of the Landed Triton Hopper

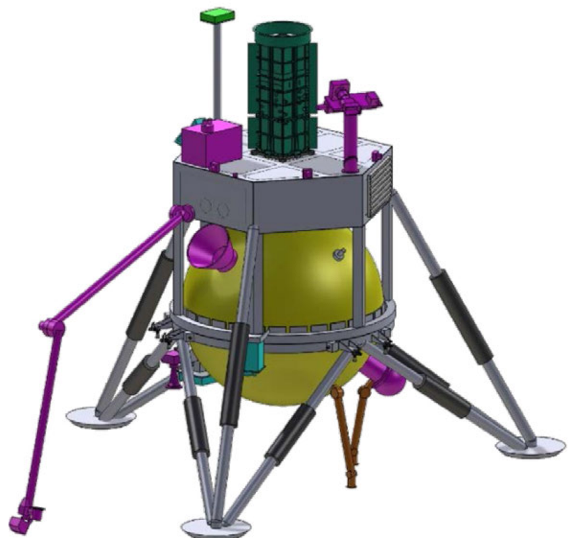


Figure 3.15—Isometric view of the Triton Hopper

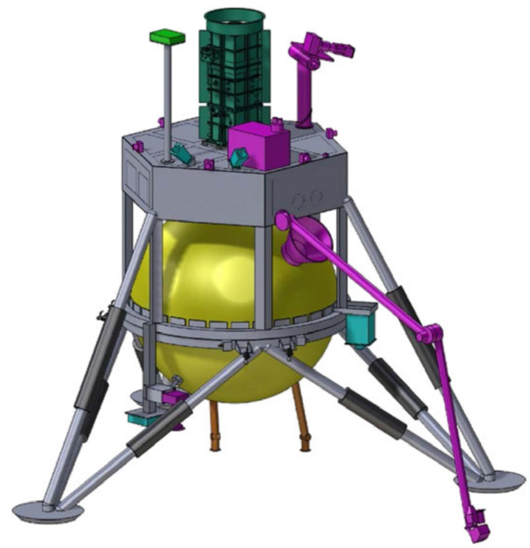


Figure 3.16—Additional isometric view of the Triton Hopper

4.0 Subsystem Breakdown

4.1 Science Payload and ISRU System

The science instrument package was chosen to address the major scientific enigmas of Triton. It will make use of the ability of the Hopper to sample widely varied terrain, including both horizontal sampling as well as vertical sampling through the plumes of geysers and through layers of the thin nitrogen atmosphere.

The major science investigations will include atmospheric science, remote measurements of the surface (including geomorphology and spectroscopy), and detailed in situ analysis of surface materials.

The electronics packages for the instruments will be in a warmed electronics enclosure heated during operation. Motors for pointing of instruments will incorporate electric heaters to allow the pointing motors to be preheated before use.

4.1.1 Science Requirements

The goal of the science payload on the Triton Hopper was to hop 30 times in 2 years a distance of 150 km total. Investigations on the surface and subsurface will be pursued as well as sensing and imaging during the hops. See Section 2.5 for a detailed description of the science mission goals.

4.1.2 Science Instruments

Instead of a high-resolution color stereo camera ***and*** the high-resolution high-speed camera, we will use a copy of the Mastcam from the Mars Science Laboratory along with a set of six small cameras to allow full visibility around vehicle. The camera weighs about 1 kg and draws 13 W when imaging[15]

The landed science suite consists of

Figure 4.1—MARDI Descent imager

- **Descent Imager** based on the Mars Descent Imager (MARDI) shown in Figure 4.1. This will be used to provide Aerial photography during landing and hops.

In addition, this downward-looking camera will be incorporated to take detailed images during landing, and also during each hop. Estimated mass of the descent imager is 1 kg with a 13 W power draw (but it runs only during the last few minutes of landing, and during the hops). The MARDI camera was designed to take wide angle, color images during descent of the landing site for the Mars Surveyor 2001 lander mission.

  - http://www.nasa.gov/mission_pages/phoenix/spacecraft/mardi.html

- The **landing-site chemical analysis package** will include a Quadrupole Mass Spectrometer (QMS), and Gas Chromatograph (GC), based on an evolved version of the instruments flown on the Surface Analysis at Mars (SAM) instrument suite on Mars Curiosity rover. The QMS and the GC can work together for definitive identification of organic compounds. The instrument’s estimated mass is 9.7 kg, with an average power requirement during operations of 7 W.
- **Robotic arm (RA)** to acquire samples. The robotic arm will be based on Phoenix robotic arm, but redesigned slightly for the lower gravity of Triton. Mass estimation is 8.0 kg. The RA is 2.35 m in length with an elbow joint in the middle allowing the arm to trench about 0.5 m (1.6 ft) below the Triton surface, also allowing for collection of ice and snow to be processed for propellant. This instrument was designed to be able to penetrate through the ice/soil mixtures at Mars polar landing sites, but will need to be modified for Triton conditions similar to those expected at the landing site. It has an average power requirement when moving of 42 W, and peak of 55 W.
  - http://www.nasa.gov/mission_pages/phoenix/spacecraft/robotic-arm.html
- **Mastcam (Mast Camera)**, shown in Figure 4.2 from the Mars Science Laboratory (MSL). The Mastcam will take color images and color video of the surface. The Mastcam will be mounted on a pointable mast, so that images can be stitched together to form a panoramic view of the camera’s surroundings.
  - http://mars.jpl.nasa.gov/msl/mission/instruments/cameras/mastcam/

Figure 4.2—MastCam

- **360° cameras**: In addition to the high-resolution camera, six small cameras will provide a 360° panoramic view in all directions, to show the lander’s surroundings. Each camera weighs 30 grams.
- **Spectroscopy**: V/UV/NIR point spectrometer based on an instrument designed at NASA Glenn, with optics added. Estimated mass: 100 grams. Spectroscopy is the study of optical spectra, and will be used to examine the mineralogy of the surface materials, as well as the optical spectra of the Triton geyser ejecta. The V/UV/NIR spectrometer will be mounted on the same mast as, and co-pointed with, the Mastcam.
- **Meteorology Package** to measure pressure, temperature and wind speeds. Pressure measurements devices: weight estimate one gram (plus read-out electronics). Wind speed measurement device is estimated at 55 grams.
  - Measurements of the composition of atmosphere will use APXS instrument (see next bullet)
- **Alpha-particle X-ray spectrometer (APXS) instrument**.
- APXS has proved in Mars missions to be a remarkably effective, reliable, and lightweight instrument to measure elemental composition of both surfaces and atmosphere. The APXS will be mounted on the robotic arm, and determines the elemental chemistry of rocks and soils using alpha particles and x-rays. Its physical details were as follows: Mass 0.1 kg, power 2.5 W, size 8.5- long by 5-cm-diam. (Based on MER instrument, Rieder et al., 2003 (Ref. 16).
  - http://en.wikipedia.org/wiki/APXS

- http://marsrovers.jpl.nasa.gov/mission/spacecraft_instru_apxs.html

4.1.3 ISRU: Propellant Gathering Systems

4.1.3.1 Cryocooler for Triton Atmosphere Collection

One means of collecting the gas required for Hopper propulsion is to freeze out the nitrogen atmosphere inside the propulsion tank. To accomplish this a cold surface, both below the freezing point of the atmosphere in temperature, and capable of removing the heat of sublimation from the solidifying nitrogen is required. Designs similar to this known as “cryopumps” are used to trap residual nitrogen in vacuum chambers. Their main components are a mechanical cryocooler to cool the surface and remove heat, and an extended surface heat exchanger to trap and hold the frozen nitrogen. However, since the Triton atmosphere is much colder, a new design is required.

First Approach: The first design started with the “cryopump” concept, but replaced the conventional single stage cryocooler with a two stage cryocooler capable of reaching 30 K. A temperature of 30 K was picked as sufficiently below the atmospheric freezing point temperature to promote rapid freezing. It was calculated that a heat removal of 5 W was required to freeze sufficient nitrogen for propulsion (roughly 100 kg) within 8 days (the desired fill time). There was assumed an additional 5 W of heat load into the tank, but this was removed via a thermal link to the surface. Please note the thermal link strategy cannot be used directly to freeze atmosphere, as the atmosphere and surface are quite close in temperature. Using the GRC Cryogenic Analysis Tool (CAT) (Refs. 17 and 18) to model a conventional room temperature based cryocooler resulted in the characteristics shown below:

- Cryocooler input power: 457 W
- Cryocooler mass: 18 kg
- Radiator area 3.05 m^2 (281 K)
- Radiator mass: 16.8 kg
- Cryocooler control electronics 5.3 kg

Figure 4.3 shows an image of a similar design (Ref. 19).

Figure 4.3—Two stage conventional 20 K flight cryocooler used in GRC ground testing.

Because the planned propellant tank is a composite overwrap design, and hence fairly low thermal conductivity, it was decided to include an extended surface to act as a tank internal heat exchanger. This was designed as a 6061-T6 aluminum (Al) vane structure. The overall structure was sized to hold the entire volume of solid nitrogen required for the propellant load with no more than 5 cm of nitrogen between vanes. The resultant design was 14 semicircular (11 cm radius) 6061-T6 Al vanes 0.5 mm thick mounted around a support tube. The cold head of the cryocooler was then thermally linked to the support tube. Mass of this element was 0.3 kg (estimated from the structure volume and an Al density of 2.7 g/cm^3). The element design has not been checked for launch and landing loads. Accommodating those loads may require mass growth.

Final Approach: In order to reduce the power requirements of the atmospheric collection system, a second design was undertaken. The second design retained the basic concepts of the first design, but replaced the radiator with a thermal link to the surface. This allowed for design of a single stage cryocooler operating from a cold platform at 60 K (due to limitations of CAT 60 K was the coldest temperature still analyzable) to the 30 K required for freezing nitrogen. This resulted in a cryocooler with the much more modest characteristics:

- Cryocooler input power: 53 W
- Cryocooler mass: 2.4 kg
- Cryocooler control electronics 0.5 kg
- Thermal link 2.1 kg (design of thermal links is discussed in the thermal control section)

A quick estimate of size based on similar cryocooler parts yielded the following physical dimensions:

- 13 cm diameter by 15 cm long cylinder cold end heat exchanger
- 20 cm diameter by 15 cm high compressor dome
- 1.3 cm thick by 18 cm diameter interface ring acts as hot side heat exchanger

The cold end heat exchanger was estimated from the size of the second stage of a room temperature cryocooler with similar capacity. The compressor size was estimated from the warm compressor driving the second stage, although this estimate is less certain since a cold compressor might have different volume requirements (the higher density of cold gas will tend to make them smaller). Although many different materials are used within the cryocooler, the exterior of the cryocooler will be 300 series stainless steel. Any material mismatch within the cryocooler will have been accounted for in the cryocooler design.

Although the mass and size of the first design seemed feasible, providing an additional 457 W from a 110 W power system was a completely unacceptable solution. The second design proposed is the recommended design. Key risks of this design are as follows. Although working conditions are the same as the second stage of existing cryocoolers and cold compressors are part of existing designs, operation without a warm compressor will require a custom design. Integrating the cryocooler and tank internal heat exchanger into composite tank will be challenging, and may require embedding metal bosses into the tank weave.

4.1.3.2 ISRU Ice and Snow Collection System

Two primary methods are being considered to recover surface material for ISRU: scoop and drilling. Using a scoop has the potential to gather a greater quantity of material over time when considering loose material. However, it faces a limitation of material hardness. Harder, more consolidated materials may require an additional tool to break up the deposit so that is accessible to the scoop. The best example of this is the scoop on the Phoenix rover, which landed in the higher latitude region of Mars where permafrost exists near the surface. Retrieving the ice-cemented regolith required the use of rasp tool in addition to the percussive scoop blade. This method was successful, but reduced the amount of material that could be recovered in each scoop. In terms of flight hardware, the phoenix scoop is the best baseline for this type of material recovery (most other examples involved loose regolith material on the Moon and Mars).

Drilling is more rugged and effective in harder materials. It also allows greater access to subsurface resources, which can be especially advantageous when overburden is significant. Several concepts are being developed for drilling resource recovery, but are still unproven in flight. This includes core drilling, where material is captured within the hollow center of the drill bit, and augering where drill cuttings are collected either on the auger flights or otherwise conveyed. Narrower drill bits have a higher rate of penetration, but result in less material recovered. For core drilling, the consistency of the core will reflect that of the deposit itself. For consolidated icy material, the sample will be solid or in large chunks, unlike the more granular material that would result from auguring. This would impact the selection of the resource processing method used.

For the Triton Hopper, the resource is available at or near the surface so scooping is feasible. The phoenix scoop for the initial trade study since it has been demonstrated in permafrost with strength in the range of what may exist at Triton surface. However, resource recovery maybe is slow, requiring 400 scoops to meet the system nitrogen needs.

4.1.4 Science Payload and ISRU System MEL

After going through several design iterations in order to reduce mass and power as far as possible and still maintain a high level of science experiments performed on the Hopper, Table 4.1 captures the final science payloads and ISRU System.

4.2 Communications and Tracking

4.2.1 Communications Requirements

The communications requirement for this scenario is the design of a science data uplink from the Triton Hopper Surface platform to an orbiting platform.

TABLE 4.1—SCIENCE PAYLOAD MASSES

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.1	Science Payload			22.99	3.8	0.87	23.86
06.1.1.a	Instruments			20.09	0.0	0.00	20.09
06.1.1.a.a	Descent Imager	1	1.0	1.00	0.0	0.00	1.00
06.1.1.a.b	Robotic arm	1	8.0	8.00	0.0	0.00	8.00
06.1.1.a.c	Landing site chemical analysis package	1	9.7	9.70	0.0	0.00	9.70
06.1.1.a.d	Mastcam	1	1.0	1.00	0.0	0.00	1.00
06.1.1.a.e	360° camera	6	0.0	0.18	0.0	0.00	0.18
06.1.1.a.f	Spectroscopy	1	0.1	0.10	0.0	0.00	0.10
06.1.1.a.g	Meteorology package	1	0.1	0.06	0.0	0.00	0.06
06.1.1.a.h	APXS instrument	1	0.1	0.05	0.0	0.00	0.05
06.1.1.b	ISRU			2.90	30.0	0.87	3.77
06.1.1.b.a	Cryopump	1	2.4	2.40	30.0	0.72	3.12
06.1.1.b.b	Cryopump Electronics	1	0.5	0.50	30.0	0.15	0.65

4.2.2 Communications Assumptions

The communications subsystem design (Figure 4.4) will consist of a single fault tolerant space qualified small deep space transceiver (SDST) (Figure 4.5) configured for X-band operations (8.0 to 8.4 GHz) mounted internal to the Hopper hexagonal platform structure. An omnidirectional antenna (Figure 4.6) is used and mounted on top of the platform. The communications uplink is closed with a 4 m dish antenna mounted Triton nadir facing on the orbiter approximately 210,000 km distance (Figure 4.7). Further assumptions are a standard 3 dB margin, included in the communications link for the link budget analysis, which is typical for space design applications due to the uncertainty of the components performance and available real effective isotropic radiated power. The Hopper platform will communicate with the orbiter every 12 days for 20 hr to uplink approximately 1.1 Gb of science data accumulated over the past 11 days. This Hopper subsystem link design does not include a space to Earth relay. All communications are assumed direct Hopper platform to orbiter. All Hopper platform hardware is assumed to be minimum TRL 5 or greater.

Figure 4.4—Communications System representation.

Figure 4.5—Small Deep Space Transponder

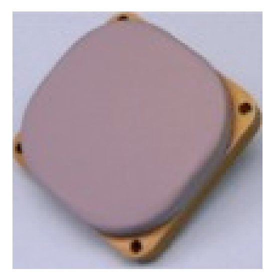


Figure 4.6—Omnidirectional/ Hemispherical Antenna

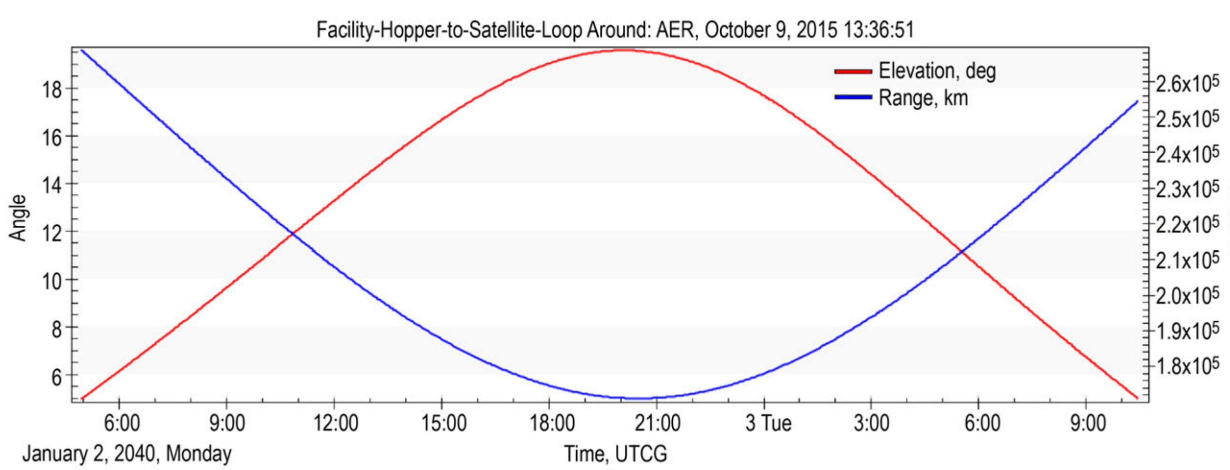


Figure 4.7—Typical Orbiter—Hopper Data Pass (Every 12 Days)

4.2.3 Communications Design and MEL

X-band is always an attractive option for higher capacity data downlinks because the data throughput matches or exceeds other comparable use frequency bands for the same size antenna, with minimal impact from atmospheric influences. Also, X-band has heritage use and less frequency crowding as compared to some other frequency bands for higher capacity downlink operations. In this case, the communications subsystem design is based around the use of a space qualified SDST configured for X-band operations. The SDST was chosen as the transmitter because it is highly configurable and can be modified to accommodate the necessary parameters, including proper data coding and modulation, to ensure mission success. It also has a built in beacon mode for ranging and emergency communications. The SDST is vibration and radiation tolerant, respectively to 25 g_{rms} and 100 krad and can be modular configured up to a maximum of 15 Mbps. An omnidirectional antenna with approximately 3 dBi of gain is mounted zenith facing on top of the Hopper platform. A 4 m antenna with approximately 47 dBi of gain is mounted Triton nadir facing on the orbiter platform. A slightly higher gain in both the Hopper platform and orbiter antenna is anticipated, although the minimum worst case is used. For analysis, we use the amount of data to be uplinked from the Hopper to the orbiter, and the orbiter period of viewing (usually minimum 5° elevation on each horizon) such that the minimum speed of the link is calculated as follows:

$$[1.1 \text{ Gb} / 20 \text{ hr}] \times [1 \text{ hr} / 3600 \text{ s}] = [1.1 \text{ Gb}] / [20 \times 3600] \text{ s} = 15277.78 \text{ b/s or} \sim 16 \text{ kbps}$$

Link budget analysis shows that the X-band subsystem with 10 W (10 dBW) of radio frequency (RF) power can accommodate up to a 9 kbps data transfer rate. Therefore, for higher data rates to close the link and meet the required 16 kbps, the RF power would have to increase to a minimum of 17 W (12 dBW), reduce the link margin by 2 dB or use a 2:1 data compression ratio. Since the Hopper is power constrained and increased power is not an option, it is recommended that data compression be used rather than link margin reduction. The Communication System MEL is shown in Table 4.2 and link budget calculations are provided in Figure 4.8, Figure 4.9, and Figure 4.10.

TABLE 4.2—COMMUNICATION SYSTEM MEL

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.4	Communications and Tracking	--	----	4.20	10.0	0.42	4.62
06.1.4.a	X-Band System	--	----	4.20	10.0	0.42	4.62
06.1.4.a.a	X-band Transceiver	2	2.00	4.00	10.0	0.40	4.40
06.1.4.a.b	X-band Omni Antenna	1	0.20	0.20	10.0	0.02	0.22

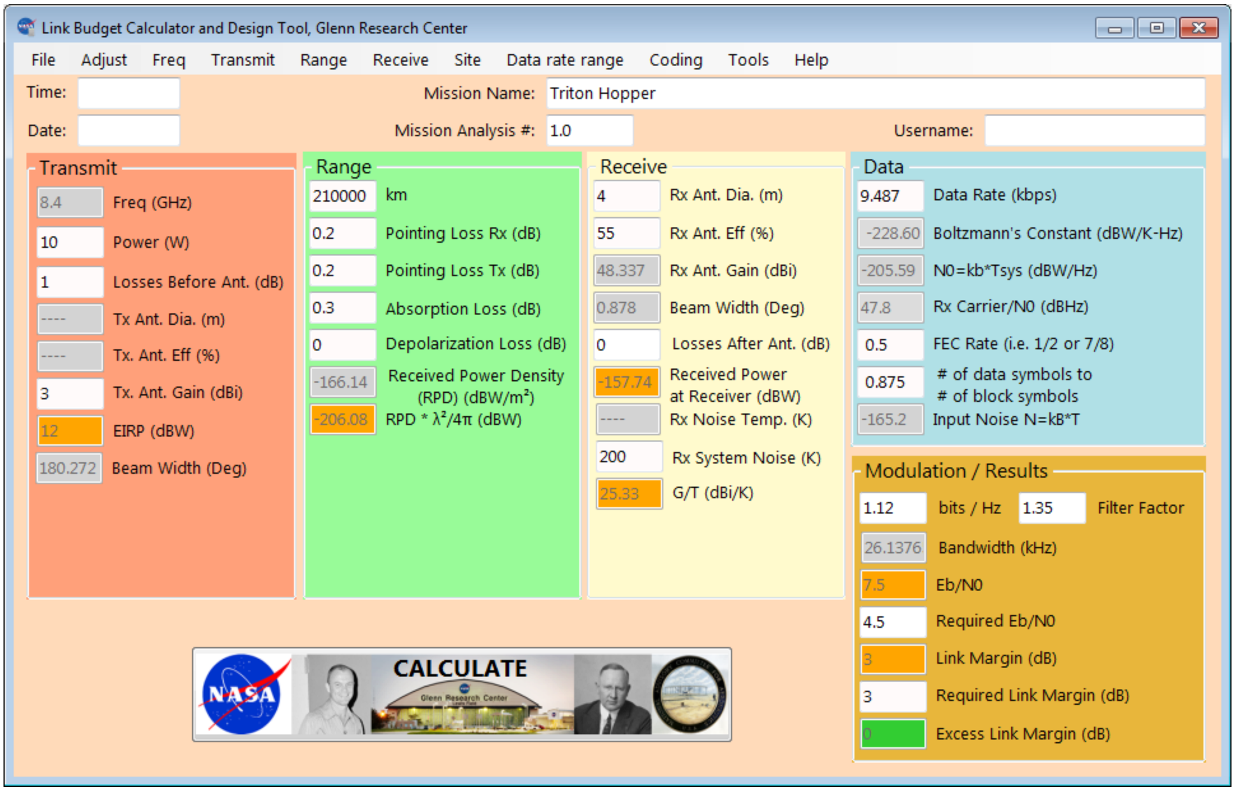


Figure 4.8—Triton Hopper Communications System, 10 W ~9 kbps

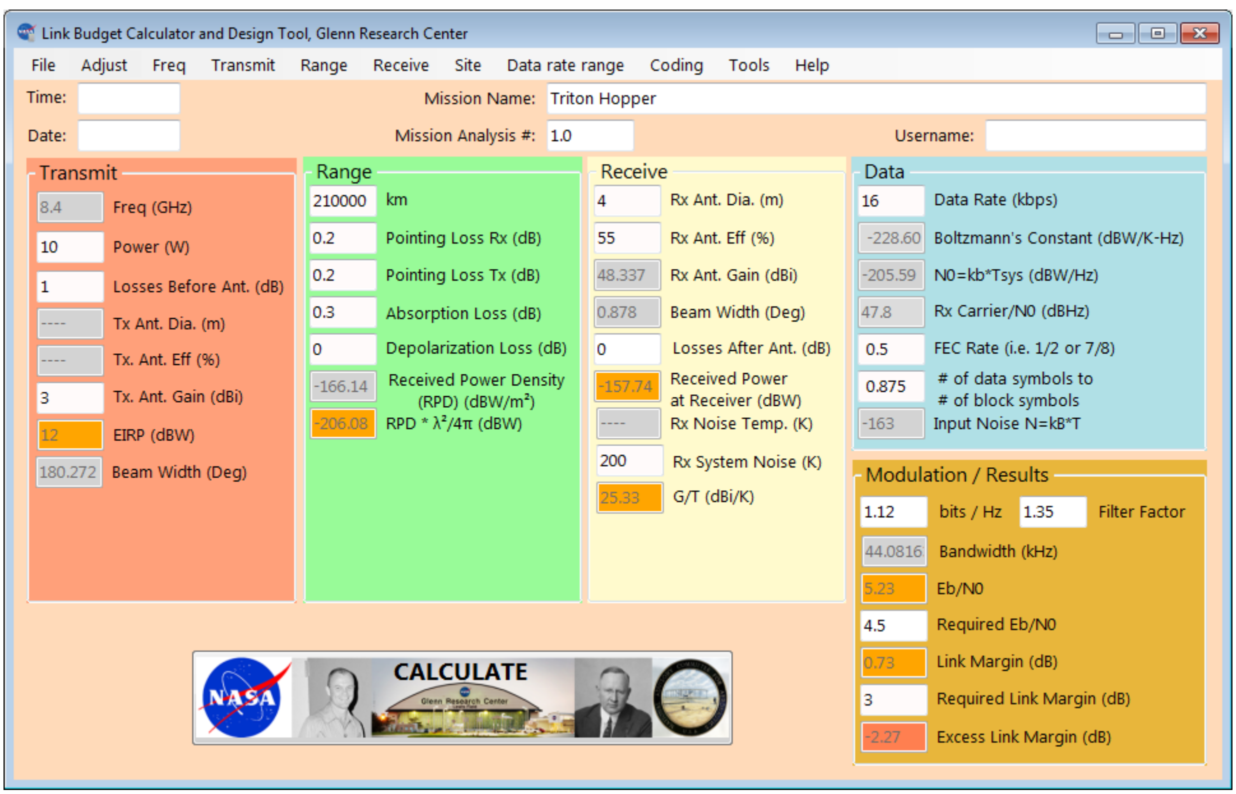


Figure 4.9—Triton Hopper Communications System, 10 W—16 kbps

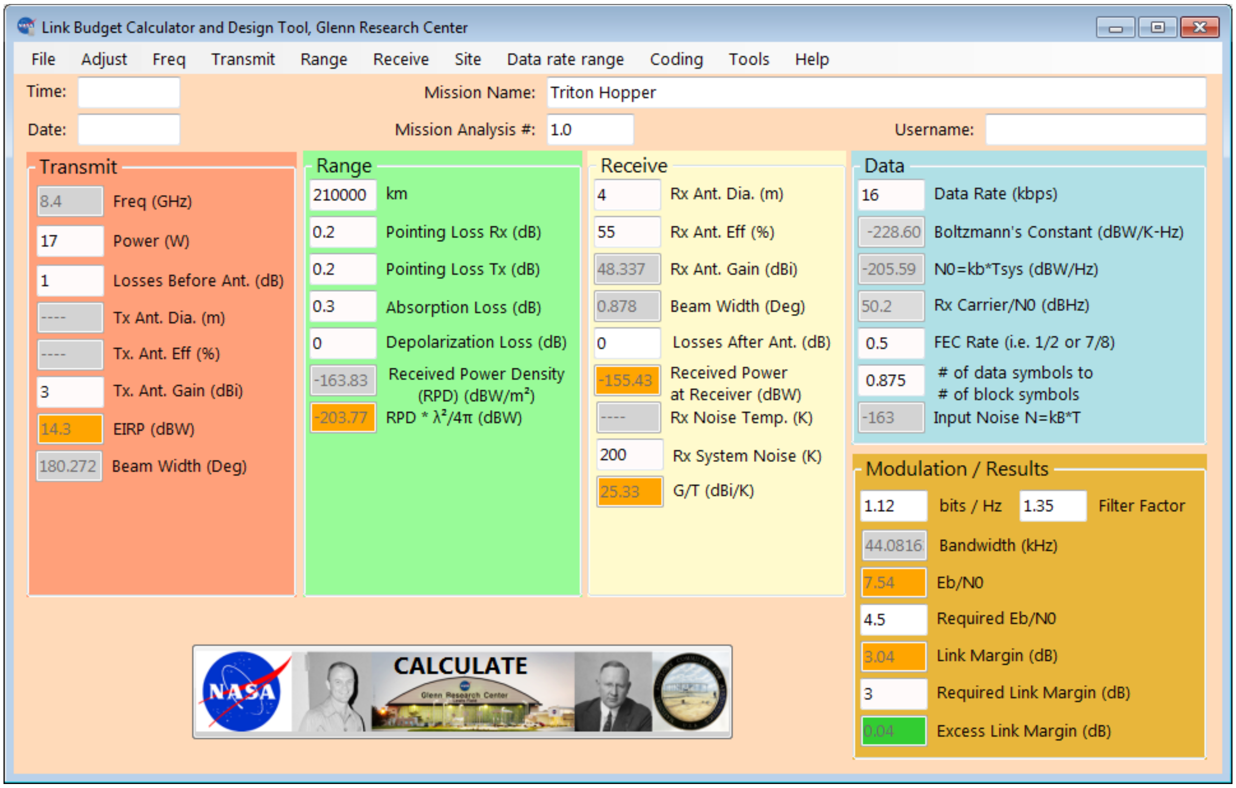


Figure 4.10—Triton Hopper Communications System, 17 W—16 kbps

4.2.4 Communications System and Risk Analysis

Two inherent risks to the communications subsystem are:

(1) Due to the uncertainty of the Triton atmosphere, it could contain gaseous components that might interfere with closing the communication uplink from the surface to the orbiter. Given that it is widely accepted that the main components of the atmosphere are nitrogen, methane and carbon monoxide with trace amounts of other substances whose interactions with RF energy are well characterized, the probability of this risk is low.

(2) There is the possibility that Triton Hopper surface operations during the propellant expulsion phase could disturb particulate matter and distribute it throughout the local area with some portion of it settling on the antenna. Over time, this could degrade the RF signal acquisition to a point where the resulting attenuation cannot be overcome, thus the data transfer rate would be reduced. The probability of this risk is low to medium, depending on the possible particulate matter concentration. Including margin into the communications system link to overcome unforeseen circumstances such as this mitigates both of these risks.

4.3 Command and Data Handling (C&DH) System

The C&DH subsystem is used to provide computer control and data storage for the Triton Hopper.

4.3.1 C&DH Requirements

The C&DH System provides the processing capability and data storage to operate the equipment on the Triton Hopper. The system concept has a single fault tolerant main processor rated for 100-krad total dose radiation environment. The system has a science data storage capability large enough to accommodate 1 Gb every 12 days.

4.3.2 C&DH Assumptions

The C&DH enclosure will be maintained in a temperature environment of 300 K.

4.3.3 C&DH Design and MEL and Risks

The concept design solution includes a flight controller enclosure populated with electronic boards selected based on their capability to meet mission goals and to survive the environmental conditions. The main components are:

- Two processor cards utilizing Power PC 750 radiation hardened cards, or equivalent.
- Watchdog switcher.
- Solid-state memory card.
- cPCI enclosure with power supply.
- Atomic Clock module/Ultra oscillator module.
- Valve/Motor drivers.
- Data interface cards (RS422/485) for communications, science and navigation instruments.
- Wiring harness and connectors.

The system concept is illustrated in Figure 4.11, and the C&DH MEL is shown in Table 4.3.

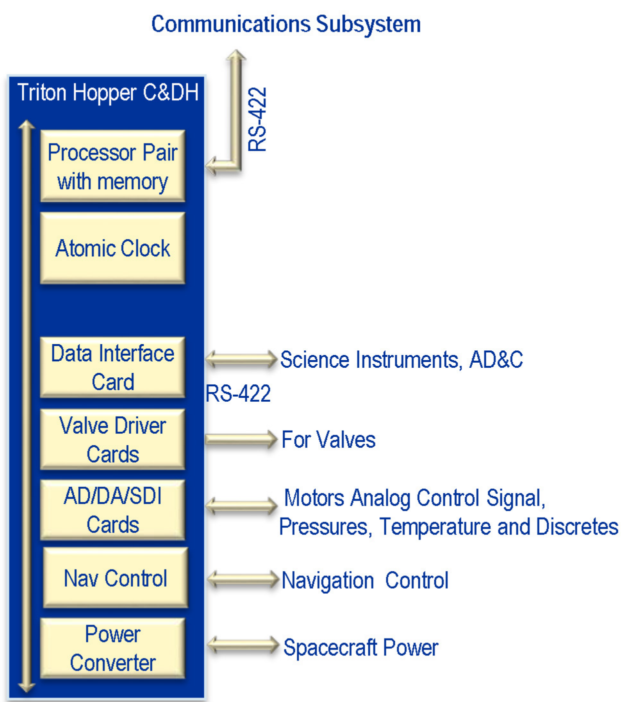


Figure 4.11—Triton Hopper Command and Data Handling System.

TABLE 4.3—COMMAND & DATA HANDLING SYSTEM MEL

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.3	C&DH	--	----	14.70	29.0	4.26	18.96
06.1.3.a	C&DH Hardware	--	----	12.70	33.5	4.26	16.96
06.1.3.a.a	Processor board	2	1.00	2.00	30.0	0.60	2.60
06.1.3.a.b	Data Interface Unit	0	0.30	0.00	30.0	0.00	0.00
06.1.3.a.c	Time Generation Unit	1	0.50	0.50	30.0	0.15	0.65
06.1.3.a.e	Command and Control Harness (data)	1	3.00	3.00	55.0	1.65	4.65
06.1.3.a.f	cPCI enclosure with power supply	1	3.00	3.00	20.0	0.60	3.60
06.1.3.a.g	Valve drivers	2	1.35	2.70	30.0	0.81	3.51
06.1.3.a.i	Separation drivers	1	1.50	1.50	30.0	0.45	1.95
06.1.3.b	Instrumentation and wiring	--	----	2.00	0.0	0.00	2.00
06.1.3.b.a	AD/DA/SDI card	2	0.50	1.00	0.0	0.00	1.00
06.1.3.b.d	Pressure and temperature sensors	20	0.05	1.00	0.0	0.00	1.00

4.3.4 C&DH System Risk

Title: SBC failure due to Radiation

Risk statement: The failure of one or more of the SBC board components can occur due to excessive radiation during the mission, specifically Solar Particle Events (SPE).

Context: SPE are injections of energetic electrons, protons, alpha particles, and heavier particles into interplanetary space. These particles are accelerated to near relativistic speeds by the interplanetary shock waves which precede fast coronal mass ejections and which exist in the vicinity of solar flare sites. This S/C will travel through the radiation belts and operate where radiation is prevalent. SPE cannot be avoided and will cause electronics to fail.

Approach: Research / Accept / Watch / Mitigate

The C&DH subsystem is one fault-tolerant, and the recommended PowerPC 750 is radiation hardened with additional onboard fault detection and recovery circuitry.

Mitigation of total ionizing dose induced failure is accomplished by using the 100 krad rated SBCs. Shielding for SPE is not really possible, but the likelihood of being exposed to one is statistically low. The risk of failure based on SPE is accepted.

4.4 Guidance, Navigation, and Control (GN&C)

4.4.1 GN&C Requirements

The GN&C requirements for the Triton Hopper were as follows:

- Land on a slope no greater than 13°
- Vertical velocity at landing no greater than 1 m/s
- Horizontal velocity at landing no greater than 1 m/s
- Land within 50 m of the targeted area on the initial landing and subsequent hops
- Actively avoid landing on debris >10 cm

While the angle at which the Hopper would actually tip over is estimated to be 49°, the requirement of landing on a slope no greater than 13° was taken from the Apollo Lunar Lander and was related to being able to takeoff safely from the surface. The requirement for the vertical velocity at landing to be no more than 1 m/s is a derived requirement based off of a 3 g limit at landing assumed by the structures subsystem and a 20 cm displacement distance of the landing legs upon landing. The requirement for the horizontal velocity to be no more than 1 m/s is also a derived requirement based off of the assumption of landing on a 13° slope and applying a factor of safety to ensure that the Hopper does not tip over when landing.

4.4.2 GN&C Assumptions

The following assumptions were made in the design of the Triton Hopper:

- Orbital imagery would be performed with a 10 cm resolution to aid personnel on Earth in choosing safe landing locations
- Desired landing and subsequent hop locations would be uploaded from Earth, with the Hopper performing hazard avoidance to find a safe area to land within 50 m of the desired landing location
- Simple geometric shapes in estimating vehicle moments of inertia

4.4.3 GN&C Design Summary

The GN&C design of the Hopper is based off of heritage from the Mars Phoenix Lander and the Origins, Spectral Interpretation, Resource Identification, Security, Regolith Explorer (OSIRIS-REx) sample return S/C. The navigation hardware consists of:

- Two DTU Micro-Advanced Stellar Compass Star trackers with one being redundant
- Two Northrop Grumman LN-200s Inertial Measurement Units (IMU) with one being redundant
- One landing radar and velocimeter with heritage from the Phoenix Lander
- One GoldenEye 3D Flash LIDAR Space Camera with heritage from OSIRIS-REx

The star tracker provides estimates of the vehicle’s attitude in inertial space. The IMU contains three solid-state silicon accelerometers that are used to estimate propulsive ΔV’s and three solid-state fiber optic gyros to estimate the vehicle’s angular velocity and aid in attitude estimates. The landing radar and velocimeter provides estimates of the vehicle’s altitude and ground relative velocity during landing. The Flash LIDAR is used for hazard avoidance, aiding in determining a safe landing site by generating elevation maps during landing.

Control of the Hopper is provided by 6 Moog 58-126, 222 N (50.0 lbf) thrusters for main propulsion which can be pulsed to provide pitch and yaw control, and 6 Moog 58-113, 4.5 N (1.0 lbf) thrusters that provide roll control. The GN&C MEL can be seen in Table 4.4, with the control thrusters being held in the Propulsion Subsystem MEL.

While the navigation hardware is located on the Hopper, the IMU(s) and Star Tracker(s) are also used during the following phases of the mission, providing estimates of the vehicle’s position, velocity, attitude and body rates:

- Powered Triton Flyby—propulsively performed by the solid stage with control also provided by the solid stage
- Capture into 200 km circular Triton orbit—propulsively performed by the descent stage with control also provided by the descent stage
- Deorbit and descent—also performed by the descent stage with control also provided by the descent stage

TABLE 4.4—GUIDANCE, NAVIGATION, AND CONTROL MEL

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.2	AD&C			14.61	8.1	1.19	15.80
06.1.2.a	GN&C			14.61	8.1	1.19	15.80
06.1.2.a.a	IMU	2	0.75	1.50	3.0	0.05	1.55
06.1.2.a.b	Flash LIDAR	1	6.50	6.50	3.0	0.20	6.70
06.1.2.a.c	Landing Radar and Velocimeter	1	5.00	5.00	18.0	0.90	5.90
06.1.2.a.d	Star Tracker Optical Head	2	0.25	0.50	3.0	0.02	0.52
06.1.2.a.e	Star Tracker Electronics Box	2	0.56	1.11	3.0	0.03	1.14

After the descent stage performs the deorbit maneuver from the 200 km circular orbit about Triton, the descent stage / Hopper begin to descend to the surface. The altitude, vertical velocity and horizontal velocity are measured using the landing radar and Doppler velocimeter. At an altitude of approximately 200 m, the descent stage performs a burn to null both the horizontal and vertical velocity of the vehicle at which point the Hopper “hops” off of the descent stage using preloaded propellant. The descent stage falls to the surface as the Hopper begins a controlled descent, nominally up to 250 m away from the point that it detached from the descent stage. During the Hopper’s descent, the flash LIDAR is used for hazard avoidance by generating elevation maps, aiding in determining a safe landing site. The Hopper lands safely on the surface assuming a slope of no greater than 13°. Its vertical velocity should be no greater than 1 m/s with a horizontal velocity no greater than 1 m/s. Once on the surface, all GN&C subsystem components are turned off to save power, however knowledge of the Hopper’s latitude, longitude and heading in relation to a Triton body fixed coordinate frame is stored in the computer. Before a hop takes place, the navigation hardware is powered back on. The star tracker is powered on to estimate the Hopper’s heading in inertial space by imaging the background stars, which is then translated to a Triton body fixed coordinate frame and used to verify the stored heading in the computer. The lander then performs up to a 5 km hop, with the vehicle’s position, attitude and body rates estimated using the IMU, altitude and ground relative velocity estimated using the landing radar and Doppler velocimeter, and the flash LIDAR aiding in determining a safe landing site Figure 4.13 shows the locations of the GN&C hardware on the Hopper.

The estimated maximum horizontal velocity that the Hopper can land with on a 0° slope without tipping over is 2.5 m/s. If a 13° slope is assumed, the maximum horizontal velocity is reduced to 1.9 m/s. By assuming a safety factor of 2, 0.95 m/s was assumed to be the maximum horizontal landing velocity. It is assumed that this is within the capability of the Doppler velocimeter to null the horizontal velocity to less than this limit since the Phoenix lander that landed on Mars had a maximum designed horizontal velocity of 1.4 m/s, with the actual value estimated to be approximately 0.06 m/s.

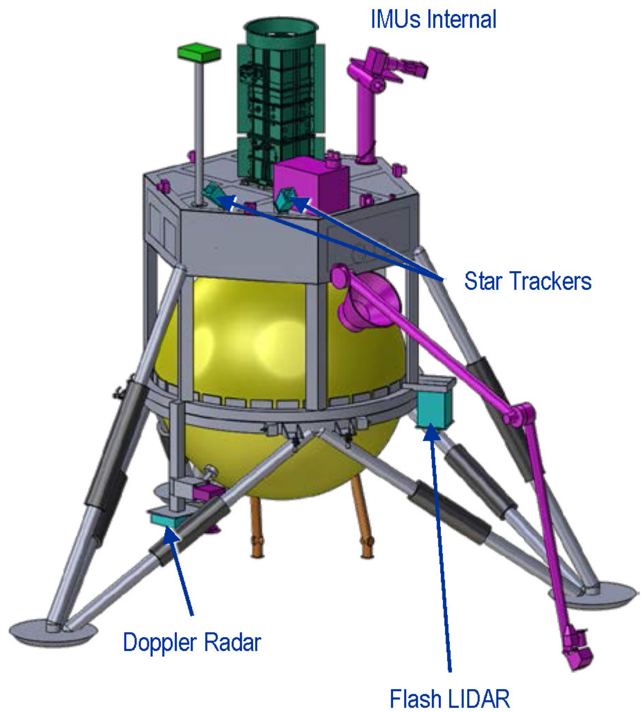


Figure 4.12—Location of the GN&C Hardware on the Hopper

4.4.4 GN&C Risks

One inherent risk for the GN&C subsystem is the uncertainty of Triton's surface features and variation, which could lead to landing in a more hazardous area than expected, possibly leading to the Hopper tipping over upon either the initial landing or landing after a hop. While the Flash LIDAR is used to perform hazard avoidance prior to landing, it is possible that the landing area will be more hazardous than expected, making it difficult for the lander to find a safe spot. Planning the general area to hop to by evaluating images taken from the orbiter will reduce this risk.

Another risk for the GN&C subsystem is that there is a possibility that the landing radar may lock onto the descent stage after the Hopper separates from it, giving erroneous results for both altitude and ground relative velocity. This was also a risk for the Phoenix Lander, where tests showed that the radar could lock onto the heat shield that had been jettisoned, and give erroneous results. Mitigation for Phoenix included both software modifications and delaying the power on of the landing radar. While the Hopper is actually hopping off of the descent stage and moving laterally relative to it rather than the descent stage simply falling underneath the Hopper, modifications similar to those for Phoenix will likely need to be made.

4.4.5 GN&C Recommendations

The requirement of 1 m/s of vertical velocity upon landing leads to requiring the thrusters to null the Hopper's vertical velocity at an altitude of 0.64 m, allowing the vehicle to free fall to impact on the surface. This requirement may be difficult to meet. It is recommended to conduct a trade study to assess the influence of higher impact velocities on the structures subsystem.

4.5 Electrical Power

The Triton Hopper requires a power system that can provide electricity to the Hopper during the 12-year transit from Earth to Triton, through Triton EDL and during the 2 years of surface operations. The Advanced Stirling Radioisotope Generators (ASRG) was selected as the electrical power source for the S/C and lander. Although it is no longer an active program at NASA it represents a reasonable estimate of how a future high performance Radioisotope Power System (RPS) should perform. In addition to the ASRG an ultra-capacitor based energy storage system was added to meet short duration high power output portions of the mission.

4.5.1 Power Requirements

Power varies from 25 W at launch to a peak of 599 W during the Landing Hop and Burn mission phase (3), see Table 4.5(a). Although the power requirement for the Landing Hop-Burn is high relative to nominal power requirements its duration is only 50 s. Nominal power during science is 108 W with communications and propellant collections requiring 67 and 116 W, respectively. The main Hop/Burn phase requires 594 W with duration of 60 s. The Main Hop-Coast and science phase requires 201 W for 120 s. Nominal power output from the ASRG is 117 W. This leads to three relatively short duration phases of the mission (Landing Hop-Burn, Main Hop-Burn and Main Hop-Coast + Science) requiring more power than the ASRG can provide. The relatively short duration of the required power along with the very long duration of dormant activity for the energy storage system required either a new battery technology (other than Lithium-ion) to be used. For this power system Ultracapacitors were used to provide this power peaking capability.

4.5.2 Power Assumptions

It is assumed that the ASRG (Figure 4.14) location on the S/C allows it to have an effective sink temperature < 100 K (Figure 4.14). It was also assumed that the ASRG has degraded over its 14 years of operation at a rate of 1.2% per year (Figure 4.15). Estimated power output for the ASRG at the EOM is 115 W. The ASRG mass will be 32 kg with a volume envelope of 34- by 34- by 77-cm with a Stirling controller that will be located above the main body of the S/C. The energy storage system selected is based upon currently available Maxwell Ultracapacitors and that their "unlimited storage when discharged" duration specification is adequate for the 12 years of storage/transit and their availability and performance are as advertised. It is assumed that the Triton Hopper will operate at a bus voltage is 28 V (± 6) and excess power will be sent to a shunt. This is in addition to the ASRG shunt that is included which is needed when the ASRG is removed from the S/C electrical system. The S/C bus is assumed to lose 2% of its power in line losses and the Ultracapacitors have a 95% charge/discharge efficiency.

TABLE 4.5—POWER EQUIPMENT LIST

(a) Power modes 1 to 5

WBS number	Description Case 1 CD-2015-127	Power mode 1, W	Power mode 2, W	Power mode 3, W	Power mode 4, W	Power mode 5, W
	Power mode name	Launch	Cruise	Landing hop, burn	Science	Communications
	Power mode duration	60 min	10 yr	50 s	4 d	20 hr
06	Triton Hopper	19.38	26.38	460.68	83.38	51.38
06.1	Hopper	19.38	26.38	460.68	83.38	51.38
06.1.1	Science Payload	0	0	17	57	6
06.1.1.a	Instruments	0.0	0.0	17.0	57.0	6.0
06.1.1.b	ISRU	0.0	0.0	0.0	0.0	0.0
06.1.2	AD&C	0	0	94.3	0	0
06.1.3	C&DH	18.5	18.5	22.5	18.5	18.5
06.1.4	Communications and Tracking	0	5	24	5	24
06.1.5	Electrical Power Subsystem	0	0	0	0	0
06.1.6	Thermal Control (Non-Propellant)	0.88	0.88	0.88	0.88	0.88
06.1.7	Propulsion (Chemical Hardware)	0	2	302	2	2
06.1.8	Propellant (Chemical)	0	0	0	0	0
06.1.9	Propulsion (EP Hardware)	0	0	0	0	0
06.1.10	Propellant (EP)	0	0	0	0	0
06.1.11	Structures and Mechanisms	0	0	0	0	0
	Bus Power, System Total	19	26	461	83	51
	30% growth	6	8	138	25	15
	Total Bus Power Requirement	25	34	599	108	67

(b) Power modes 6 to 9

WBS number	Description Case 1 CD-2015-127	Power mode 6, W	Power mode 7, W	Power mode 8, W	Power mode 9, W
	Power mode name	Propellant collection	Propellant processing	Main hop, burn	Main hop, coast/science
	Power mode duration	8 d	11 d	60 s	120 s
06	Triton Hopper	89.38	88.38	456.68	154.68
06.1	Hopper	89.38	88.38	456.68	154.68
06.1.1	Science Payload	56	2	15	15
06.1.1.a	Instruments	2.0	2.0	15.0	15.0
06.1.1.b	ISRU	54.0	0.0	0.0	0.0
06.1.2	AD&C	0	0	94.3	94.3
06.1.3	C&DH	14.5	18.5	20.5	18.5
06.1.4	Communications and Tracking	5	5	24	24
06.1.5	Electrical Power Subsystem	0	0	0	0
06.1.6	Thermal Control (Non-Propellant)	11.88	60.88	0.88	0.88
06.1.7	Propulsion (Chemical Hardware)	2	2	302	2
06.1.8	Propellant (Chemical)	0	0	0	0
06.1.9	Propulsion (EP Hardware)	0	0	0	0
06.1.10	Propellant (EP)	0	0	0	0
06.1.11	Structures and Mechanisms	0	0	0	0
	Bus Power, System Total	89	88	457	155
	30% growth	27	27	137	46
	Total Bus Power Requirement	116	115	594	201

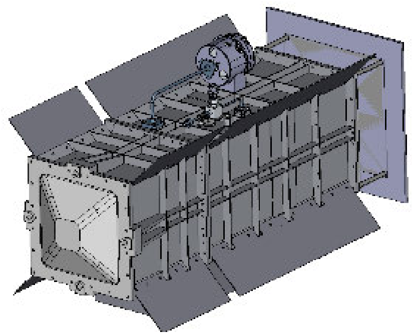


Figure 4.13—ASRG

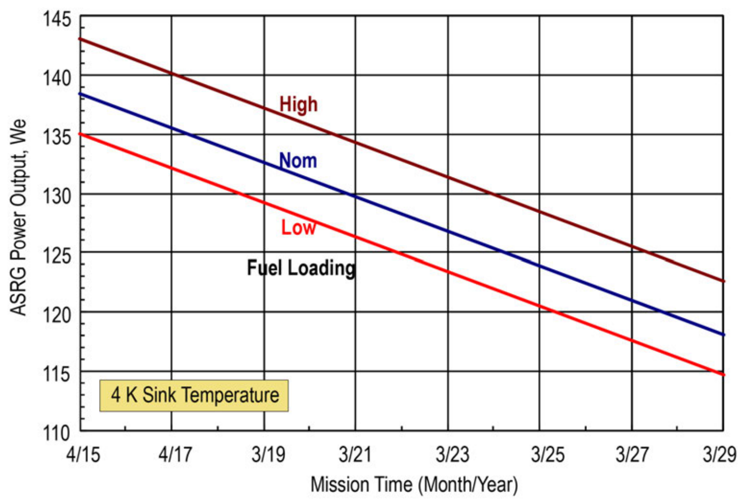


Figure 4.14—ASRG power output versus mission year.

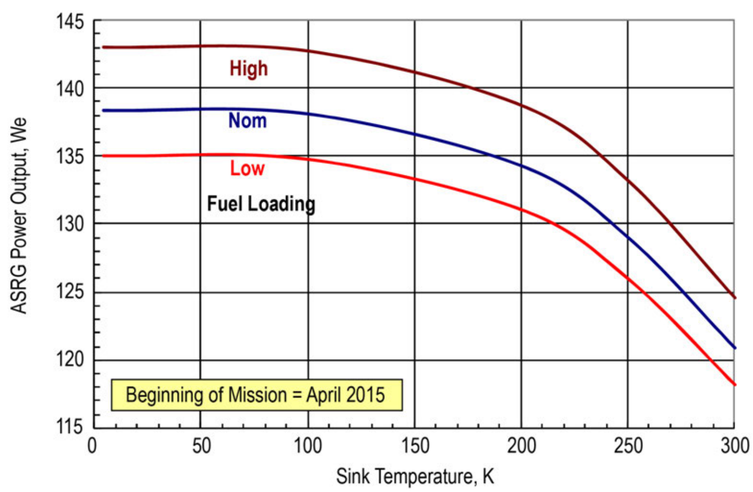


Figure 4.15—ASRG power output versus sink temperature

4.5.3 Power Design and MEL

The Triton Hopper power system uses a single ASRG connected to the S/C bus to provide power (Figure 4.17). Sixteen Maxwell Ultracapacitor Model BCA P0650 are used with eight in series and two parallel strings with each string capable of providing full power peaking. Each capacitor has a peak operating voltage of 2.7 V, have maximum current draw of 25 A and weighs 160 grams. The energy storage envelope is 44- by 7- by 14-cm and has a total mass of 2.56 kg with an operational temperature range from –40 to 65 °C. Voltage range during peak operation during the Main Hop-Burn phase is from 21.6 to 16 V. Total energy stored in each string is 19 W-hr with approximately 10 W-hr used during the 60 s Burn/Hop. Total leakage current from each of the capacitors is 1.5 mA. Because the capacitors do no operate over the same voltage range as the S/C bus, a DC/DC convertor is required. This DC/DC convertor is 90% efficient and converts the nominal 28-V S/C power to the 21-V (max) capacitor voltage. A mass summary for the power system is shown in Table 4.6. This shows but the basic mass along with a column including the margin for each of the components.

4.5.4 Risks

Risks to this mission for the power system are that an ASRG and/or similarly performing radioisotope generator will not be available. This type of system is required due to low solar flux and both limited availability of plutonium for the RPS and a low degradation rate of the RPS due to the very long duration of the mission. An additional risk to the power system is the 14 years of storage life required for the ultracapacitors. The intent is that they will not be charged during the long duration cruise but this needs to be verified.

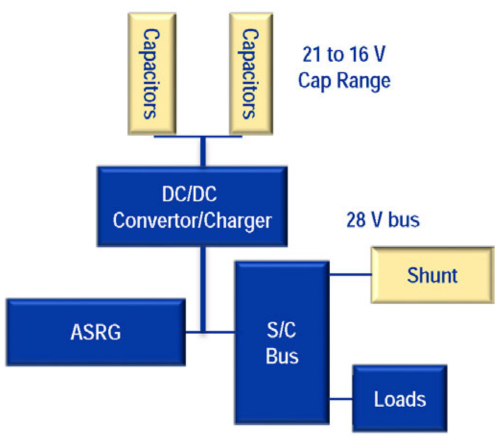


Figure 4.16—Power Schematic

TABLE 4.6—POWER SYSTEM MEL

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.5	Electrical Power Subsystem			39.50	11.5	4.53	44.03
06.1.5.a	Power Generation			32.00	10.0	3.20	35.20
06.1.5.a.a	ASRG	1	25.00	25.00	10.0	2.50	27.50
06.1.5.a.b	ACU-Stirling Controller	1	7.00	7.00	10.0	0.70	7.70
06.1.5.b	Power Management & Distribution			7.50	17.7	1.33	8.83
06.1.5.b.a	Harness	1	1.00	1.00	20.0	0.20	1.20
06.1.5.b.b	PMAD-shunt	1	3.00	3.00	20.0	0.60	3.60
06.1.5.b.c	Ultracapacitor and DC/DC Convertor	1	3.50	3.50	15.0	0.53	4.03

4.5.5 Power Trades

Other power systems were considered for this mission. The currently available Multi-Mission Radioisotope Thermoelectric Generator (MMRTG) was considered but its high thermal output (2000 W of thermal power BOM) and its 4.8% per year degradation rate make it a poor candidate for this mission. This low conversion efficiency generator exacerbates thermal control issues with storage of the cold gas propellant. Additional the 4.8% per year degradation rate would require two or more MMRTG to do the job of a single ASRG. This would increase the plutonium use by a factor of eight or more.

Because ancillary heat is needed during some phases of the mission the Auxiliary Cooling System (ACS) built into the ASRG was considered to supplement the electrical power available to heat the propellant tanks. But for the sake of simplicity was decided not to use the ACS but just an electric heater. Newer concepts using variable conductance heat pipes might be able to move much more heat and will be considered for phase II.

4.6 Thermal Control System

The thermal control system for the Triton Hopper was basically separated into two segments: the thermal control of the electronics enclosure and the thermal control of the propellant tank. Because of the significantly different operating requirements both segments of the vehicle had to be thermally isolated from each other as much as possible. The thermal system design included devising an approach to maintain the electronics and science instruments within their desired temperature operating range and being able to cool the propellant tank to ambient conditions and then bring it up to the desired 300 K propellant temperature prior to a hop within the power budget and operational timeframe available.

The thermal environment on Triton has little variation over the course of the mission (see Sec. 2.4.) Therefore the thermal system was designed for the remotely sensed steady-state environmental conditions on the surface of Triton. The main focus of the thermal design is in controlling the heat flow into and out of the propellant storage tank in order to maintain the correct temperature for collecting, storing and heating the nitrogen propellant. Operating on Triton's surface is challenging due to the low temperature of the surroundings (~38 K). The surface is covered with solid nitrogen ice and snow. To control the temperature of the propellant storage tank during filling the tank will need to be cooled to near surface temperatures. Then during pressurization it will need to be warmed to near 300 K. This wide operating temperature range requires a heat management system that enables control of the tank temperature with a minimal amount of input power. To accomplish this a movable conductive path is used in the form of a highly conductive arm that allows the tank to connect thermally with the surface during cooling and then removed during the heating process.

The main thermal system components are listed below, shown in Table 4.7, and illustrated in Figure 4.17.

- Insulation: electronics and propellant tank
- Radiator for electronics
- Avionics, thermal management, (cooling plates, heat pipes)
- Temperature sensors, controllers, switches, data acquisition
- Conductive arms for tank thermal control
- Electric heaters for electronics thermal control.

TABLE 4.7—THERMAL CONTROL SYSTEM MEL

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.6	Thermal Control (Non-Propellant)			33.00	15.0	4.95	37.95
06.1.6.a	Active Thermal Control			6.90	15.0	1.04	7.94
06.1.6.a.a	Heaters	25	0.20	5.00	15.0	0.75	5.75
06.1.6.a.b	Thermal Control/Heaters Circuit	4	0.20	0.80	15.0	0.12	0.92
06.1.6.a.c	Data Acquisition	1	1.00	1.00	15.0	0.15	1.15
06.1.6.a.d	Thermocouples	10	0.01	0.10	15.0	0.02	0.12
06.1.6.b	Passive Thermal Control			24.60	15.0	3.69	28.29
06.1.6.b.a	Heat Sinks	2	0.14	0.28	15.0	0.04	0.32
06.1.6.b.b	Heat Pipes	2	0.48	0.96	15.0	0.14	1.10
06.1.6.b.c	Radiators	1	1.80	1.80	15.0	0.27	2.07
06.1.6.b.d	MLI Insulation	1	7.39	7.39	15.0	1.11	8.49
06.1.6.b.g	Engine MLI	1	1.81	1.81	15.0	0.27	2.08
06.1.6.b.i	Propellant Tank MLI	1	8.00	8.00	15.0	1.20	9.20
06.1.6.b.j	Tank Conductive Arm	1	2.29	2.29	15.0	0.34	2.63
06.1.6.b.l	Cryocooler Conduction Arm	1	2.08	2.08	15.0	0.31	2.39
06.1.6.c	Semi-Passive Thermal Control			1.50	15.0	0.23	1.73
06.1.6.c.a	Louvers	1	1.10	1.10	15.0	0.17	1.27
06.1.6.c.b	Thermal Switches	2	0.20	0.40	15.0	0.06	0.46

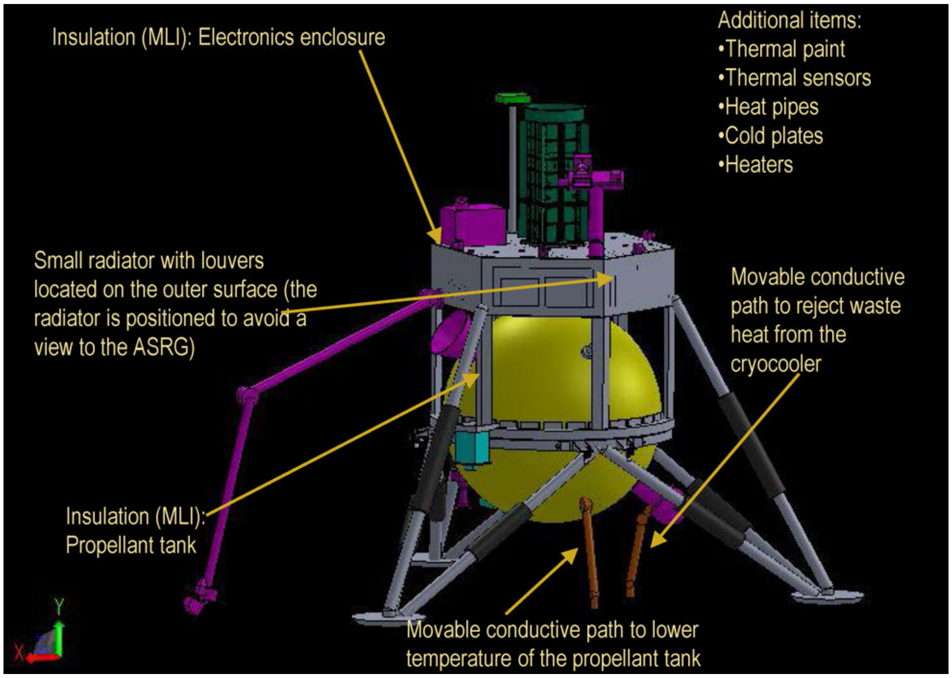


Figure 4.17—Triton Hopper Thermal System Components

4.6.1 Triton Hopper Thermal System Surface Operation

For the Triton Hopper to operate on the cold Triton surface their temperature has to be maintained within their desired operating range. The vehicle is separated into two distinct sections that have significantly different thermal requirements, the electronics and science instrumentation enclosure and the propellant tank. The required temperature range, given in Table 4.8, is much higher than the temperature of the surrounding environment. To maintain the desired operating temperature electric heaters along with the waste heat from the isotope power system is utilized to warm the electronics enclosure interior as well as heat the propellant tank prior to a hop. Any excess heat generated during operation is rejected through a small radiator located on the side of the electronics enclosure. The radiator incorporates louvers so that it can be effectively turned off when not needed. The Hopper must be insulated to provide sufficient thermal control during operation to maintain the desired operating temperatures.

A thermal analysis was performed to determine the internal operating temperature and to size the required insulation layers and radiator needed to maintain the desired internal operating temperature. The heat losses from the interior were broken down into:

- Pass-through wires for operating external instruments
- Heat loss directly through the insulation
- Heat loss through the structure that passes through the insulation.

The radiator sizing was based on an energy balance analysis of the area needed to reject the identified heat load to space. From the area a series of scaling equations were used to determine the mass of the radiator. The radiator is located on the side deck of the Triton Hopper. This provides a good view to deep space with about a 0.5 view to the surface and no view to the radioisotope system radiators. There is insulation between the radiator and electronics enclosure body providing a single surface for radiating. The radiator is connected to the cold plates with heat pipes to move heat from the interior to the radiator. The radiator was sized to remove the waste heat from the electronics enclosure during operation at full power. The radiator will need to reject heat at power levels above 31 W.

Louvers were used on the radiator to help minimize heat loss during times when the internal power consumption and waste heat are low. If electronics thermal output decreases the louvers and heaters will be used to maintain the internal temperature of the electronics. The louvers are a passive device that opens up as the temperature of the radiator increases and then close as it cools. This maintains the desired heat flow from the radiator to keep the internal temperature within its operating limits. The addition of louvers increases the radiator area by approximately 30% over a radiator without louvers. The louver specific mass is 4.5 kg/m^2.

TABLE 4.8—THERMAL SYSTEM SPECIFICATIONS

Specifications/component	Value/approach
General dimensions: Propellant tank Electronics enclosure	 1-m-diameter 1- by 1- by 0.3-m
Electronics system waste heat	100 W
Operating temperature	Electronics enclosure: 230 to 310 K (–43° C to 37° C) Propellant tank: 38 to 300 K (–235 to 27 °C)
Insulation (MLI)	MLI is used to for insulating the electronics enclosure and propellant tank.
Radiator system	A radiator with louvers is used to reject the waste heat from the electronics as needed. Al (0.1- by 0.1- by 0.005-m), Water based heat pipes to move heat from the electronics packages.
Passthroughs and conductive paths: Structure and science	Wires and ports for data collection and control are the assumed pass-through for the electronics enclosure. These passthroughs account for 5% of the insulated surface area. Conductive path for rejecting heat from the cryocooler and propellant tank: Pyrolytic graphite (1600 W/mK)
Heating	Electric heaters and waste heat from the electronics and radioisotope power system are utilized to heat and maintain temperature within the S/C.

The radiator specifications are summarized in Table 4.9.

Cold plates and heat pipes are utilized to move heat from the electronics packages to the radiator. These components consist of conductive plates with integral heat pipes connected to them, as shown in Figure 4.18. The cold plates also incorporated heaters in order to maintain the desired electronics temperature throughout the mission. Due to the low temperatures the Hopper can experience during operation ammonia was selected as the working fluid. The specifications of the heat pipes and cold plates are given in Table 4.10.

TABLE 4.9—RADIATOR SPECIFICATIONS

Variable/characteristic	Value
Radiator solar absorptivity	0.14
Radiator emissivity	0.84
Max radiator Sun angle	0°
View factor to the hopper body and RPS radiator	0
View factor to the surface	0.5
Radiator operating temperature range	223 to 310 K
Power dissipation	68 W (internal electronics)
Radiator area	0.23 m^2 (body mounted)

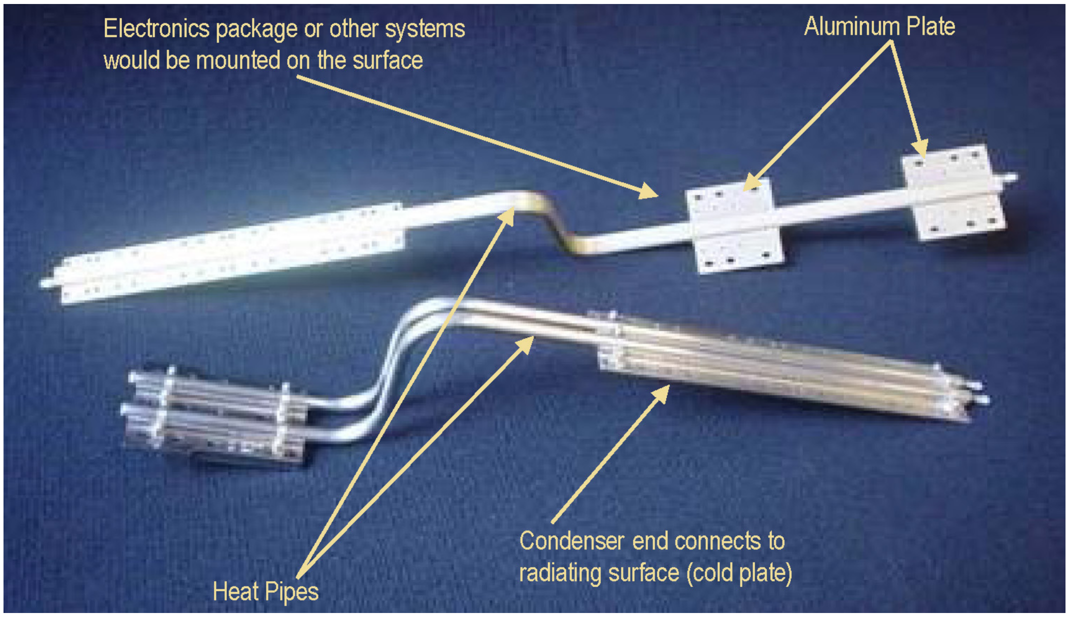


Figure 4.18—Example of Heat Pipes With Integral Cold Plates

TABLE 4.10—HEAT PIPE AND COLD PLATE SPECIFICATIONS

Variable/characteristic	Value
Cooling plate and heat pipe material and density	Al, 2,770 kg/m^3
Number of cooling plates	Four (for electronics)
Cooling plate dimensions	0.1- by 0.1- by 0.005-m
Heat pipe working fluid	Ammonia

Heaters can be configured to different shapes to match the device or component being heated

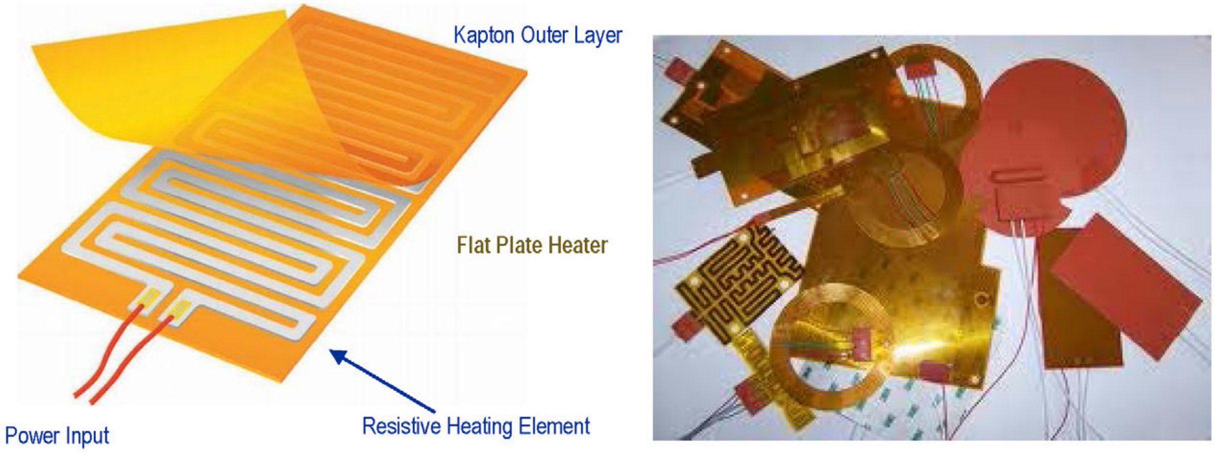


Figure 4.19—Flexible Flat Plate Heaters

Waste heat from the internal components as well as electric heaters is used to provide heat to the electronics when operating. Strip heaters are used to provide heat to the components within the electronics enclosure as well as the propellant tank. Flat plate heaters, as shown in Figure 4.19, are used on the propellant tank and cold plates to provide heat to the electronics if necessary with heaters located on each cold plate. Heaters are also utilized within the propellant tank as a heat source for melting and gasifying the collected nitrogen and bringing the propellant up to 300 K. Thermal control is accomplished through the use of a network of thermocouples whose output is used to control the power to the various heaters. A data acquisition and control computer is used to operate the thermal system. Maximum heater power is approximately100 W.

Due to the low atmospheric pressure radiation is the main mechanism for heat transfer within the Hopper and to the surroundings. Therefore MLI was selected as the means for insulating the Hopper components.

MLI was used to cover the exterior exposed portion of the electronics enclosure and propellant tank, as illustrated in Figure 4.20. The insulation was analyzed to determine the required number of layers and the corresponding mass and heat loss needed to maintain the average 300 K interior temperature during normal operation. A tradeoff was performed between the insulation mass and the required heater power cold time operation. It was determined that 25 layers would provide the best insulating option for this mission. The insulation model was based on radiation heat transfer analysis of the heat transfer from the S/C through the insulation to the surroundings. It should be noted that there is a rarified nitrogen atmosphere on Triton. Although the gas pressure is very low this can affect the operation of the MLI by providing a conductive path between the layers. For atmosphere pressures between 7.5 to 30 mTorr, there will be degradation in the performance of the MLI (Ref. 20). To minimize this affect, it is proposed to design convection barriers between the MLI layers to minimize this convection. A 5% pass-through area was assumed for the insulation heat loss for items passing through the insulation.

The pass-throughs in the insulation around the electronics enclosure allow heat to leak out of the enclosure and therefore need to be minimized. One of the main heat leaks through the insulation are the structural legs that support the electronics enclosure, as illustrated in Figure 4.21. Because of the very cold surrounding environment, conduction down the legs to the surface can be significant. To minimize this, the structural legs that go from the electronics enclosure to the surface are constructed from a titanium (Ti) alloy (Ti-6Al-4V) with a low thermal conductivity of 6.7 W/mK. The heat loss, from the three legs to the surface, is 12.9 W.

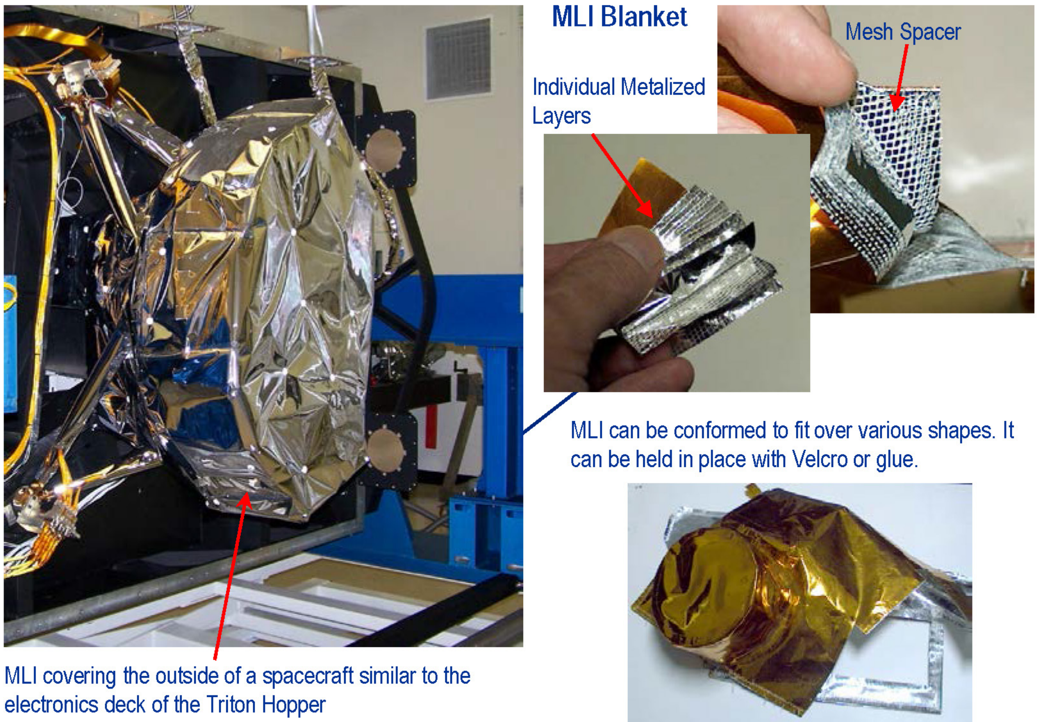


Figure 4.20—MLI Applied to Various S/C Components

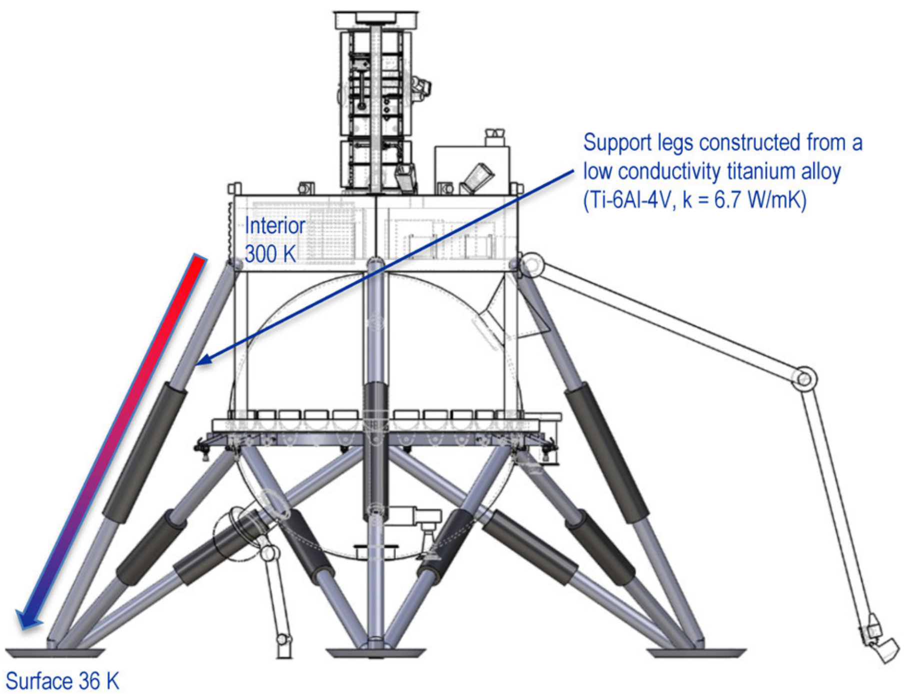


Figure 4.21—Heat Leak Down Electronics Enclosure Support Legs

The structure of the Triton Hopper is also used to isolate different segments of the S/C in order to help maintain thermal control. This is necessary since different sections have greatly different temperature requirements. The long support structure legs connected to the electronics enclosure also provide a long conductive path from the propellant tank to the enclosure. The structure holding the propellant tank is separated from the structure holding the electronics enclosure while on the surface.

While on the surface and the propellant is being collected, the tank is held to near ambient temperature conditions. A conductive rod that is lowered to the surface accomplishes this. This conductive rod is constructed from a high conductivity material, pyrolytic graphite, with a thermal conductivity of 1600 W/mK and 3 cm in diameter. The conductive rod, illustrated in Figure 4.22, conducts any excess heat from the propellant tank during filling to the surface where it is rejected to maintain the tank temperature near ambient conditions. The path is sized to move approximately 3 W of heat from the tank to the surface with an approximate 0.7 K temperature difference between the tank and surroundings. The tank itself is also designed with highly conductive strips along its interior to move heat from over its surface to the conductive rod. The conductive rod details are listed in Table 4.11.

A conductive rod is also used to reject heat from the cryocooler, as shown in Figure 4.22, that is used to collect and solidify nitrogen from the atmosphere. The cryocooler conductive rod works in a similar manner as that for the tank except that it operates at a much larger temperature difference and therefore can conduct significantly more heat. The details on the cryocooler conductive rod are also given in Table 4.11. The cryocooler is described in Section 4.1.3.1.

While on the surface during the propellant collection process, the tank is at near ambient conditions and needs to be isolated from the upper electronics enclosure. To accomplish this the structural ring connected to the support structure of the electronics enclosure is separated from the structural ring connected to the support structure of the tank. This gap between these rings minimizes any direct conduction from the electronics enclosure to the tank. The only conductive path between them is through the support legs, which is a long path through a very low conductivity material.

During takeoff and landing the two rings compress against each other to provide structural support during these high structural loading maneuvers. But then once on the surface and no longer under load they separate. Compression springs between the two rings enable this motion. This is illustrated in Figure 4.23.

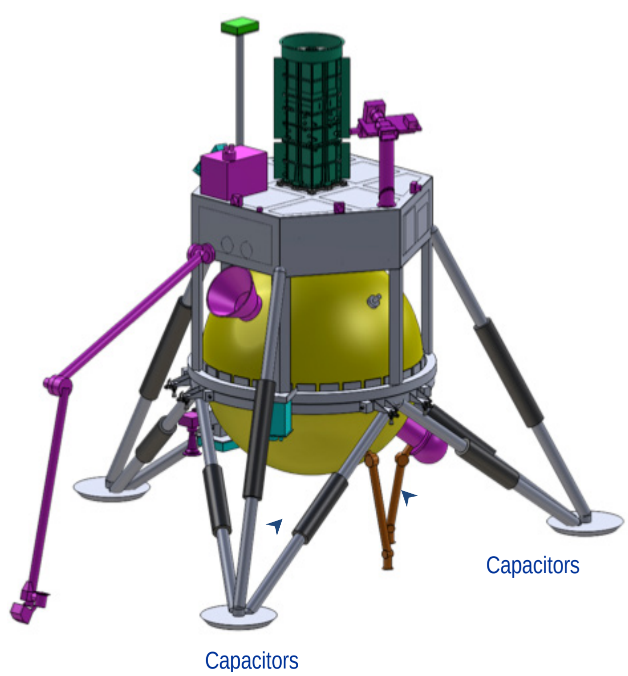


Figure 4.22—Illustration of the Conductive Rods for Rejecting Excess Heat from the Cryocooler and Propellant Tank

TABLE 4.11—TANK AND CRYOCOOLER CONDUCTIVE ROD SPECIFICATIONS

Characteristic	Tank conductive rod	Cryocooler conductive rod
Length	0.5 m	0.5 m
Material	Pyrolytic graphite	Pyrolytic graphite
Thermal conductivity	1600 W/mK	1600 W/mK
Heat rejected	3.1 W	56.6 W
Mass	1.53 kg	0.89 kg
Diameter	4.2 cm	3.2 cm
Temperatures Hot Cold	 38.7 K 38 K	 60 K 38 K

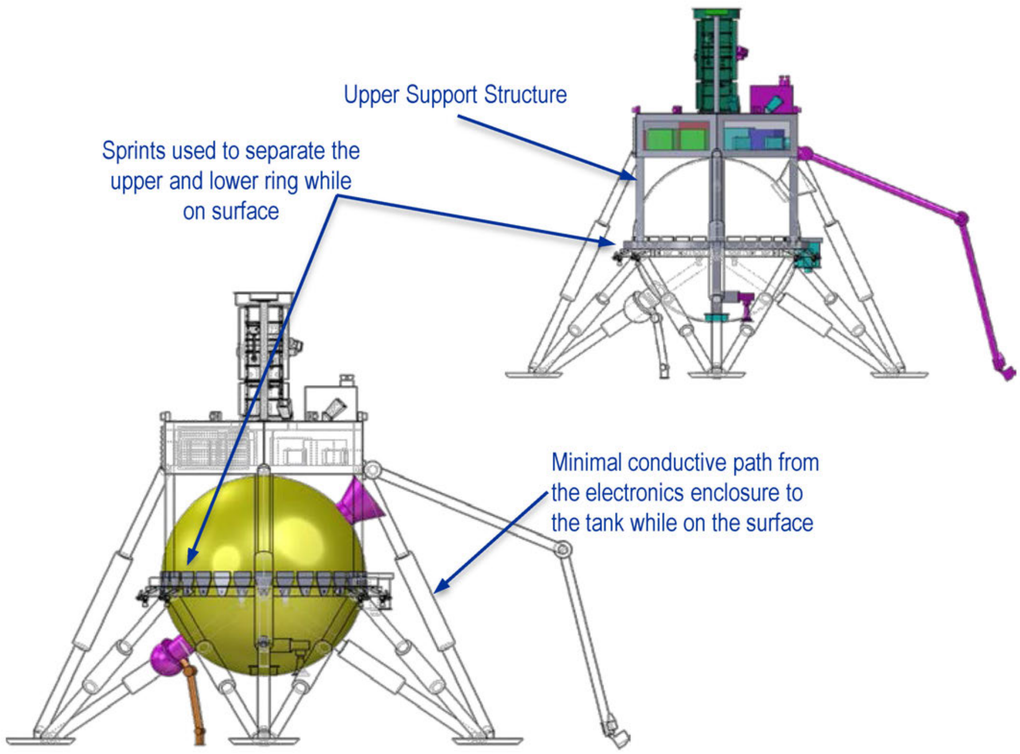


Figure 4.23—Conductive Paths Between the Hopper Upper and Lower Structure

4.7 Propulsion System

The propulsion system for this design consists of three major elements. There is a solid rocket motor based braking stage, a bipropellant descent staged that is used to lower the Hopper close to the Triton surface, and a nitrogen gas based monopropellant system used for terminal descent to Triton, initial landing, and subsequent surface hop maneuvers via replenishing the propellant with in situ resources. Due to funding constraints, complete integrated conceptual designs of the solid rocket motor braking stage and bipropellant descent stage are not conducted, but a conceptual level design of their propulsion systems is conducted in order to ascertain the impact of the required ΔV maneuvers on propellant load, and thus on the entire S/C that must be fitted to the launch vehicle.

4.7.1 Propulsion System Requirements

The solid rocket braking stage and bipropellant descent stage are required to provide both the braking ΔV required and adequate control required to place Hopper close enough to Triton's surface as to allow it to safely land using an initial quantity of nitrogen gas propellant that was brought from Earth. Once on the surface, the Hopper propulsion system is required to provide adequate performance for additional hop maneuvers via cold gas thrusters utilizing warmed nitrogen gas as propellant. The nitrogen used for these maneuvers is required to be obtained as an in situ resource.

4.7.2 Propulsion System Assumptions

In order to reduce integration and thermal flow path complexity, it is assumed that the propulsion system will utilize a single large central propellant tank. Due to the long trip to Triton, it is assumed that pyrotechnic valves are used for propellant isolation during both the launch and cruise phases of the mission. The system is assumed to be single fault tolerant and utilize as many COTS or near COTS components as possible. Multiple thrusters will be located around the Hopper, thus thrust vector control is assumed to be accomplished via thruster pulse modulation. Although heat will be supplied to the Hopper tank to warm the nitrogen gas prior to any hop maneuver, it is assumed that the actual hop itself can be modeled as an adiabatic blow-down. Because nitrogen in the tank will cool and the pressure will decrease as propellant is expelled, it is assumed for analysis purposes that all useable propellant is consumed when either the critical temperature or critical pressure of nitrogen is reached, thus avoiding any potential condensing of the remaining propellant in the tank. The remaining nitrogen in the tank is thus treated as trapped residual propellant.

4.7.3 Propulsion Systems Designs

The propulsion system for this design consists of three major elements. First, there is a Star-37GV based solid rocket motor based braking stage that is used for Triton orbital insertion, a monomethyl hydrazine (MMH) and nitrogen tetroxide (NTO) based bipropellant based descent stage that is used to lower the S/C close to the Triton surface, and a nitrogen gas based monopropellant system used exclusively by the Hopper. This final system is used for both the initial Triton surface landing, and will be refueled via in situ resources such as the nitrogen gas present in the atmosphere or the suspected nitrogen snow on the surface. Once collected, the nitrogen is heated to form pressurized gas that is used for propulsive hop maneuvers with powered landings.

4.7.3.1 Solid Based Breaking Stage

The initial braking stage is based on the existing ATK STAR-37GV solid rocket motor. This motor utilizes an electromechanical flex seal TVC system that can provide ± 4° of thrust vectoring in both planes. This motor's inert mass fraction is 8%, the casing is comprised of graphite-epoxy composite, and the propellant is TP-H-3340, which is 18% Al, 71% ammonium perchlorate, and 11% HTPB. This motor delivers a nominal I_{sp} of 293.5 s at a nominal thrust of 56.9 kN (12.80 klb_f). Due to mission requirements, this motor will require an additional 8.7% (93.7 kg (206.5 lb_m)) propellant than the current 1,076 kg (2,371 lb_m) propellant load listed in the ATK catalog. This quantity of additional propellant may require a minor redesign or even a requalification of the motor, the impacts of which, however, are beyond the scope of this study. One option to circumvent this is issue, is to add additional propellant to the liquid descent stage, thus allowing it to provide the required additional ΔV.

4.7.3.2 Bipropellant Descent Stage

The descent stage used to lower the S/C to Tritons surface is a bipropellant stage utilizing MMH and NTO for propellants. This system has four main thrusters, sixteen RCS thrusters, two fuel tanks, two oxidizer tanks, a single fault tolerant feed system, and is pressurized with helium stored in a single COTS tank. The four main thrusters are Aerojet R-42s that deliver a nominal thrust of 890 N (200 lb_f) at a nominal I_{sp} of 303 s. The 16 RCS thrusters are Aerojet model R-6D, which produces a nominal thrust of 22 N (5.0 lb_f) at a nominal I_{sp} of 294 s, and are assumed to be mounted in four pods containing four thrusters each. The feed and pressurization systems consist of COTS components, a nominal instrumentation suite, and tank and line heaters.

The nominal O/F for both the main and RCS thrusters is ~1.65, which for MMH and NTO, results in near identical fuel and oxidizer volumes. This allows for identical tanks to be used for both propellants, which can be mounted in an opposing pairs configuration to minimize CG shift during operation. For this design, ATK model 80309-1 is selected. This tank is constructed of Ti-6Al-4V, is polar boss mounted, has a MOP of 2.19 MPa (318 psia), has a diameter of 56.2 cm (22.1 in.) and a length of 81.3 cm (32.0 in.), has an integral vane type propellant management device, and operates at 78% rated capacity with the current propellant load. The helium pressurant gas for this system is stored in one COTS ATK model 80459-1 pressurant tank. This tank is a COPV design with a MOP of 31.0 MPa (4,500 psia), is 42.2 cm (16.36 in.) in diameter, and is 66.3 cm (26.1 in.) long. Figure 4.24 shows a preliminary P&ID of the descent stage propulsion system.

4.7.3.3 Hopper Propulsion System

The propulsion system on the Hopper is a nitrogen-based cold gas system that is based around a single 116.7 cm (45.9 in.) diameter spherical custom COPV tank with a Ti alloy liner. The system has six large thrusters for axial propulsion that are located in three groups spaced equally around the

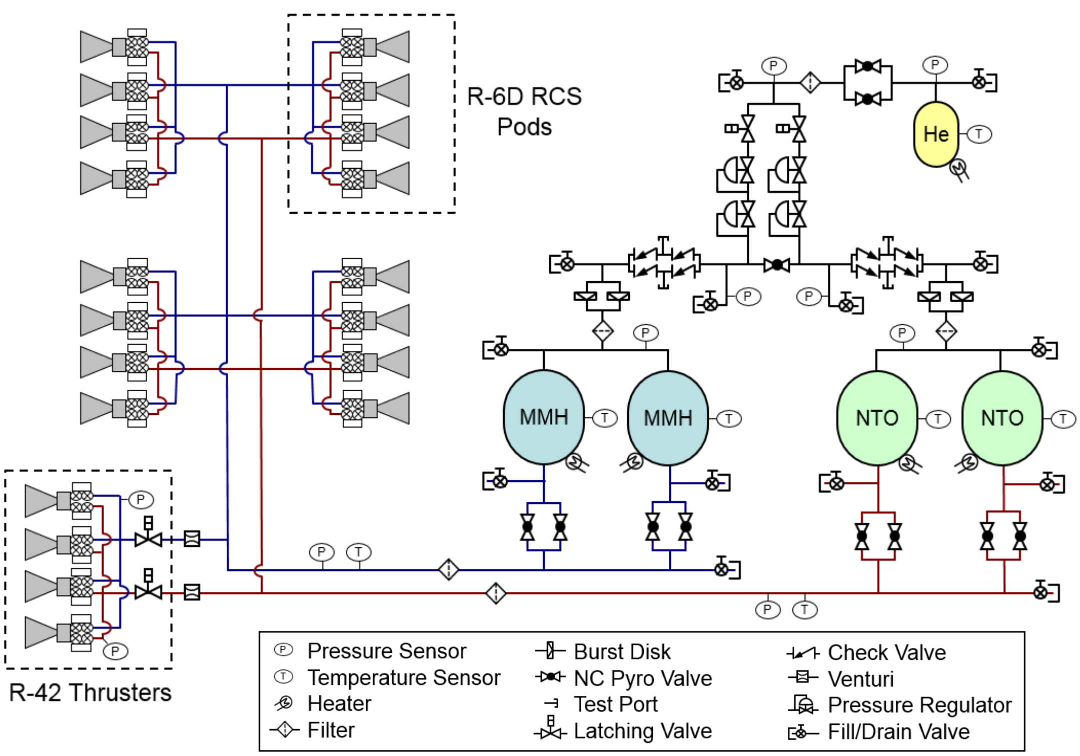


Figure 4.24—Preliminary Descent Stage P&ID.

circumference of the tank. Roll control is accomplished via six smaller thrusters that fire tangentially to the main thrusters and are located in three opposing pairs alongside the main thrusters.

The system is design to be refueled via in situ propellant gathering. The primary method is to use a cryocooler to cool a heat exchanger surface inside the tank. Once cold enough, the ambient atmosphere will collect and freeze inside the tank. A backup system is to use the robotic arm to scoop nitrogen snow from the surface and feed it directly into the tank. To help facilitate propellant collection, a 15.25 cm (6.0 in.) diameter opening is located near the top of the tank. This opening has a collection funnel mounted to the outside of the tank that is designed to help funnel any snow scoped up by the robotic arm fall into the tank. There is a door mounted inside the tank that closes before the propellant is heated and the tank pressurized. The placement of the collection funnel, thrusters, and the other major propulsion elements on the Hopper is shown in Figure 4.25.

Once adequate nitrogen has been collected inside the tank, the door is closed, and heat is applied to the tank in order to vaporize the nitrogen and heat it to 300 K. For a 116.7 cm (45.9 in.) diameter tank, that equates to a pressure of 13.79 MPa (2,000 psia) for a propellant load of 127.0 kg (279.9 lb_m). This warmed nitrogen gas is then fed to the thrusters via a nominal single fault tolerant feed system with line heaters, a nominal instrumentation suite, normally closed pyrotechnic valves for propellant isolation during launch and the coast to Triton, and a set of latching valves that can allow Triton atmosphere to flow into the tank in the event the tank door fails closed. This will greatly increase propellant gathering time, due to the small flow area, but should allow the hoper to continue its mission. The preliminary P&ID for the Hopper propulsion system is shown in Figure 4.26.

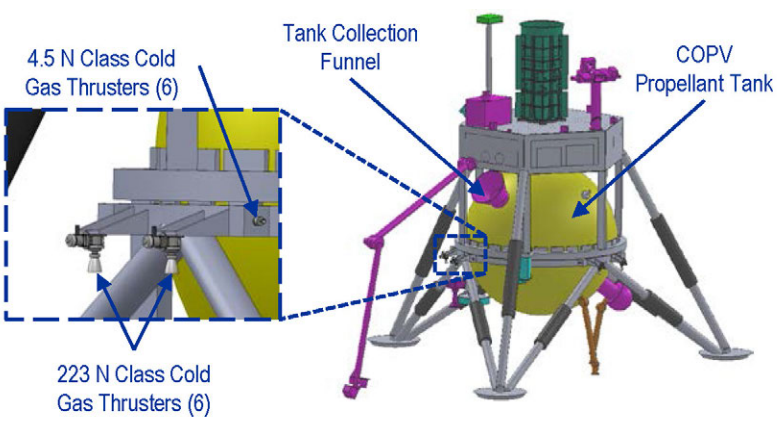


Figure 4.25—Hopper Propulsion Configuration.

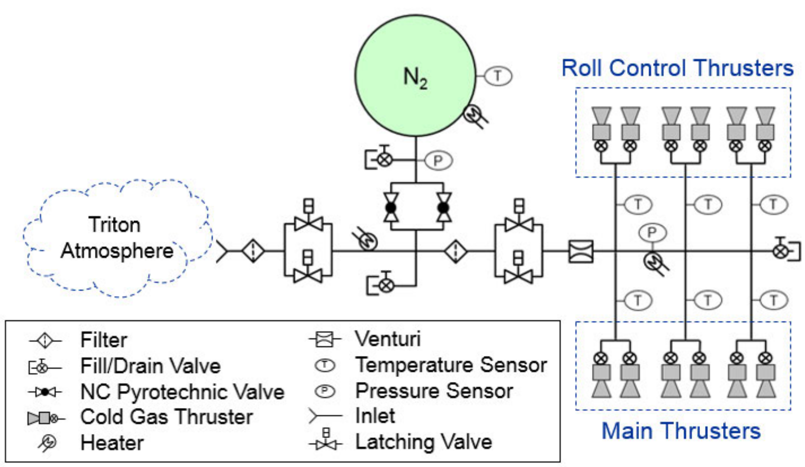


Figure 4.26—Hopper Preliminary P&ID.

There are a total of 12 thrusters on the Hopper, six main thrusters and six smaller roll control thrusters. The six main thrusters are designed to provide a nominal thrust of 223 N (50 lbf) and are based on Moog COTS thruster model 58-126. The six roll control thrusters provide a nominal thrust of 4.5 N (1.0 lbf) and are based on Moog model 58-118. Although all the thrusters are hard mounted to the S/C body, it is planned that the cold gas thrusters can be pulse modulated as to adjust the resultant thrust vector to that required by the S/C during hop maneuvers.

4.7.3.3.1 Propellant Tank Door

The internally mounted propellant tank door is designed to seal the 15.25 cm (6.0 in.) diameter opening into the tank and allow for propellant pressurization. The door actuation system is located inside the tank and has internal redundancy as to meet the single fault tolerant requirement. The door has an ellipsoidal shape as to support the internal tank pressure, is envisioned to retract into the tank when open, and has a seal surface around the rim that seals against a flange on the tank wall when closed. This design utilizes the internal tank pressure to help provide sealing force during propulsion system operation. There is a cylindrical funnel extension that extends into the tank to help guide nitrogen snow into the tank while protecting the seals form accidental contact and potential damage. There are two axisymmetric seals present to provide single fault tolerance. A schematic showing a cross sectional view of the door mechanism is shown in Figure 4.27, and illustrates how the system would operate in both the open and closed positions.

4.7.3.3.2 Tank Door Seals

One key component to the proper operation of the propulsion system is the door seals. If they fail, they vehicle is stranded and the propulsion system rendered inoperative. For this reason, dual concentric seals will be used to provide a level of redundancy. Although the selection and detail design of the seal system is beyond the scope of this study, a survey of potential seal options from various manufactures was conducted to determine what options are commercially available. According to literature available from Parker, several types metal seals can handle temperatures form cryogenics to 1,144 K (1,600 °F) and pressures up to 524.0 MPa (76,000 psi). A typical selection matrix for various seal types is shown in Figure 4.28, and a chart of typical seal leak rates is shown in Figure 4.29.

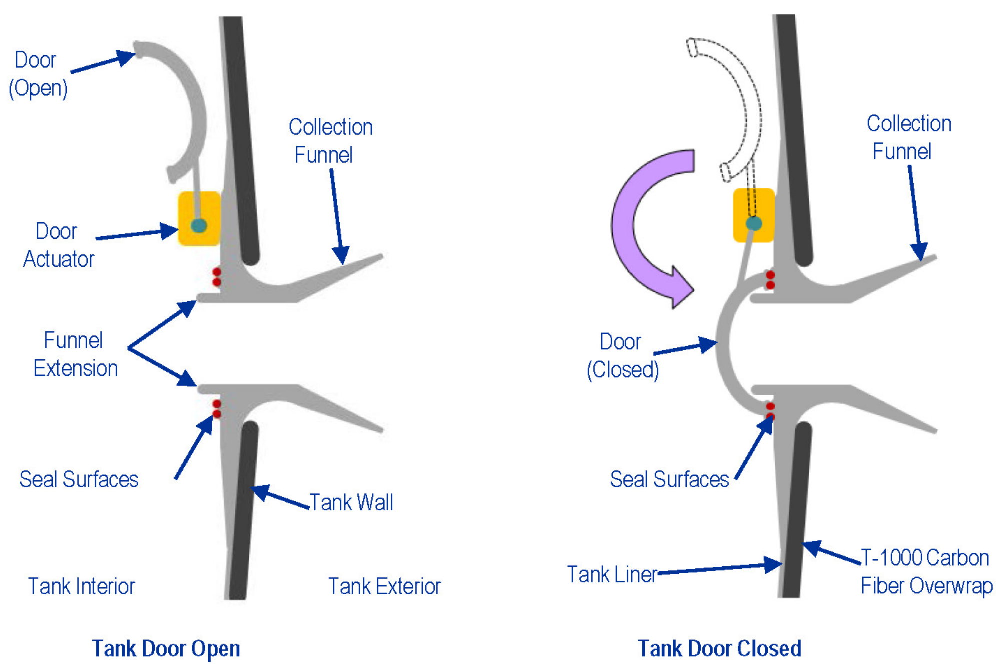


Figure 4.27—Propellant Tank Door Schematic

From the seal options evaluated during this study, the spring energized O-ring and wire ring designs seem to be the most applicable at this stage in the design, as they are typically used in applications requiring mild corrosion resistance and exposure to cryogenic to moderate temperatures. Both are readily available in 304/304L, 316/316L, 321, and 347 stainless steel alloys, with 304 and 321 being the preferred alloys for the metal O-ring and wire ring designs. These types of seals are typically plated depending on the operational environment, and the base material can be tempered various ways including annealing, work hardening, and solution heat-treating.

4.7.4 Hopper Propulsion System Performance

In preparation of a hop maneuver, the propellant tank is heated to 300 K and pressurizes to a design operating pressure of 13.79 MPa (2,000 psia). This is the initial condition of the propellant in the tank for a hop maneuver, and as propellant is expelled through the thrusters, the temperature and pressure in the

Seal Type	Sealing Requirements					
	High Springback	Low Load	High Load	Low Leak Rate	Pressure Capability	Low Cost
Metal C-Ring	⭘	⭘	⭘	◉	●	◉
Metal E-Ring	●	●	⊘	○	⭘	○
Metal O-Ring	○	⊘	◉	◉	●	◉
Metal U-Ring	◉	●	⊘	○	◉	⭘
Metal Wire Ring	⊘	⊘	●	⭘	●	●
Spring Energized C-Ring	⭘	⊘	●	●	●	○

Figure 4.28—Typical seal selection matrix.

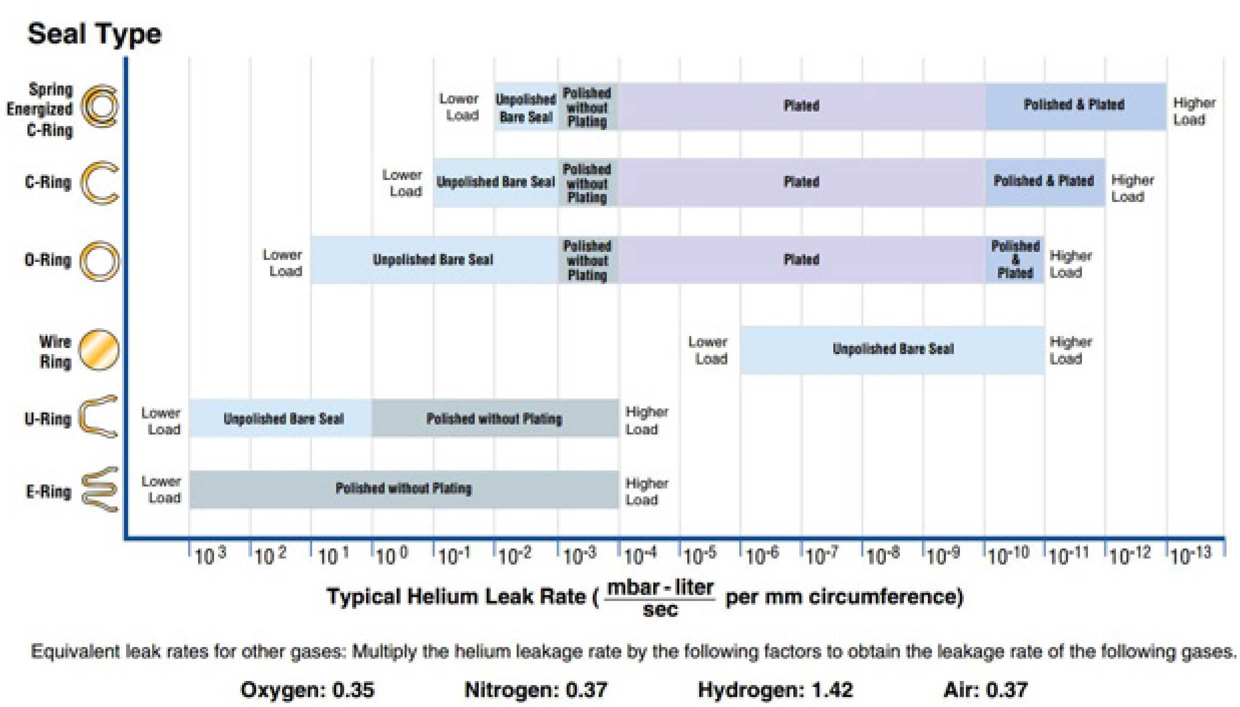


Figure 4.29—Leak rates for various seal types.

tank decrease. This complicates trajectory calculations since thruster I_{sp} is a function of temperature and changes as the propellant is consumed. For the purposes of this design, propellant consumption is modeled as an adiabatic blow-down process, which is limited to a critical point if either critical temperature or pressure met. This critical point limit is in place to help reduce the risk of condensation in the tank or feed system, which could result in a sudden drop in propulsion system performance as the nitrogen propellant enters a region where phase change to a liquid can occur. The resulting nitrogen left in the tank at this point is treated as trapped residual propellant. Calculated tank conditions and estimated thruster I_{sp} are shown in Figure 4.30 and Figure 4.31 as functions of propellant mass fraction for the Hopper system.

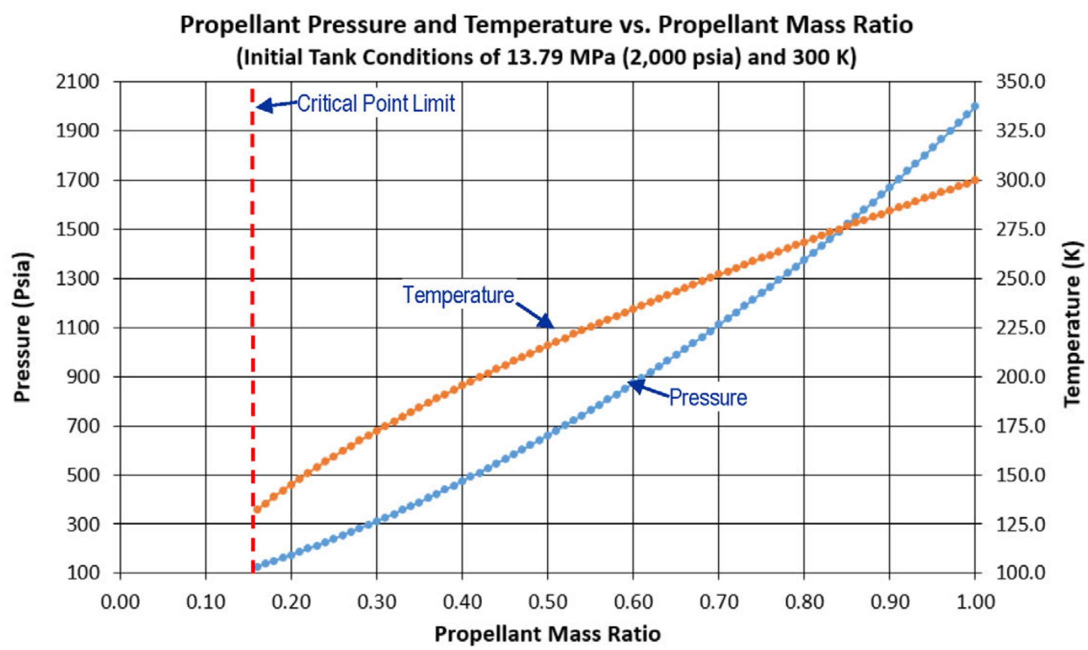


Figure 4.30—Hopper tank conditions during propulsive operations.

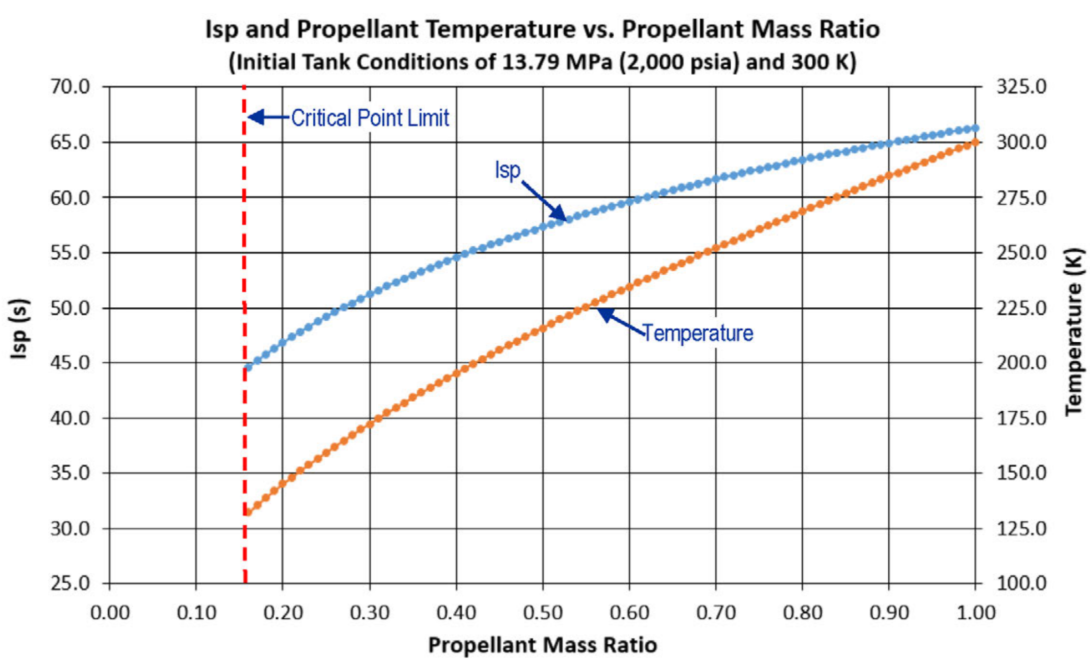


Figure 4.31—Thruster I_{sp} and propellant temperature during propulsive operations.

4.7.5 Hopper Propulsion System MEL

A listing of the various components in the Triton Hopper propulsion system and their corresponding masses is shown in Table 4.12.

4.7.6 Propulsion System Risk

There are several propulsion system related risks pertinent to this design. First, there is the tank itself. The COPV tank used in this design will be exposed to large temperature swings and large pressures during the mission, thus potentially inducing large mechanical and thermal stresses. In addition, this design calls for an internal door assembly and an internal heat exchanger assembly, both of which have fairly low TRLs. The unique features and operating conditions result in a development risk and potential cost and schedule overruns.

Second, there are potential development risks for the cold gas thrusters. This design requires the development of relatively large cold gas thrusters that can operate over a large pressure range and can be reliability pulse modulated. Most cold gas systems are pressure regulated, but this system, however, requires the thrusters to operate at pressures as high as 2,000 psia and still be pulse modulated for thrust vector control. Although these new thrusters are based on existing designs, they still require engineering development and testing of new flight ready hardware.

TABLE 4.12—PROPULSION SYSTEM MASTER EQUIPMENT LIST

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.7	Propulsion (Chemical Hardware)			87.21	17.8	15.56	102.77
06.1.7.a	Primary Chemical System Hardware			87.21	17.8	15.56	102.77
06.1.7.a.a	Main Engine Hardware			86.61	17.9	15.55	102.16
06.1.7.a.a.a	Large Cold Gas Thruster	6	1.50	9.00	25.0	2.25	11.25
06.1.7.a.a.b	Feed System	1	9.14	9.14	25.0	2.29	11.43
06.1.7.a.a.c	COPV Tank	1	56.67	56.67	15.0	8.50	65.17
06.1.7.a.a.d	Mounting Hardware	1	2.00	2.00	3.0	0.06	2.06
06.1.7.a.a.e	Sample Door Assembly	1	6.00	6.00	25.0	1.50	7.50
06.1.7.a.a.f	Heat Transfer Vanes	1	0.30	0.30	25.0	0.08	0.38
06.1.7.a.a.g	Cryocooler Mounting Boss	1	3.50	3.50	25.0	0.88	4.38
06.1.7.a.b	Reaction Control System Hardware			0.60	3.0	0.02	0.62
06.1.7.a.b.a	Cold Gas Thruster	6	0.10	0.60	3.0	0.02	0.62
06.1.8	Propellant (Chemical)			127.12	0.0	0.00	127.12
06.1.8.a	Propellant			124.30	0.0	0.00	124.30
06.1.8.a.a	Fuel			124.30	0.0	0.00	124.30
06.1.8.a.a.a	Fuel Useable	1	114.93	114.93	0.0	0.00	114.93
06.1.8.a.a.b	Fuel Margin	1	5.75	5.75	0.0	0.00	5.75
06.1.8.a.a.c	Fuel Residuals (Unused)	1	3.62	3.62	0.0	0.00	3.62
06.1.8.b	Propellant			2.83	0.0	0.00	2.83
06.1.8.b.a	Fuel			2.83	0.0	0.00	2.83
06.1.8.b.a.a	Fuel Usable	1	2.49	2.49	0.0	0.00	2.49
06.1.8.b.a.b	Fuel Margin	1	0.25	0.25	0.0	0.00	0.25
06.1.8.b.a.c	Fuel Residuals (Unused)	1	0.08	0.08	0.0	0.00	0.08

Third, there is a development risk associated with instrumentation. Monitoring of the Hopper propulsion system will require pressure instrumentation that can accurately measure pressures from multiple mega-Pascal's to single digit Pascal's. This range exceeds that of existing pressure sensors commonly used in propulsion systems, thus requiring the development of flight ready sensors capable of operating in very low pressure and temperature environments.

Finally, there is a development risk associated with the propellant tank door development. The actuated tank door used in this design must close and seal properly for the propulsion system to work. If the door doesn't seal properly, the tank will not pressurize and propellant gas will escape, thus greatly reducing propulsion system effectiveness and effectively crippling the vehicle mobility-wise.

4.7.7 Propulsion System Modeling

The method used to design the propulsion system involves using a mix of published values, empirical data, and analytical tools. Published values for COTS components and empirical data are used wherever possible, with analytical tools being employed as necessary. Empirical data is used to aid in the mass and size estimation of similar systems when published values are not available. Numerous analytical tools are used in this analysis, including NIST tables, fluid and gas property codes such as REFPROP and CEA, as well as custom tools developed form basic physical relationships and conservation equations with empirical based inclusions for real life hardware requirements (mounting bosses, flanges, etc.).

For the purposes of this design, propellant consumption is modeled as an adiabatic blow-down process, with a cut-off if either critical temperature or pressure is met. This cut-off is in place to help reduce the risk of condensation in the tank or feed system, and the resulting nitrogen left in the tank is treated as trapped residual propellant.

4.7.8 Propulsion System Recommendation

It is recommended to consider trading the traditional pyrotechnic valves used for propellant isolation during launch with a new positive isolation valve design currently in development by Vacco. This valve is a normally closed positive isolation valve that is designed to be a drop-in replacement for typical pyrovalve applications. It has low shock, is electrically redundant, eliminates the pyrotechnic charge, uses existing electrical circuits, and has positive retention of seat components. They are designed to operate at MOP of 34.47 MPa (5,000 psia). Unfortunately, this design currently has very limited, if any, flight heritage, and thus a low TRL.

4.8 Structures and Mechanisms

4.8.1 Structures and Mechanisms Requirements

The Triton Hopper structures must contain the necessary hardware for research instrumentation, communications, and power while fitting while minimizing its overall size. The structural components must be able to withstand applied loads from the launch vehicle, operational maneuvers, and landing. In addition, the structures must provide minimum deflections, sufficient stiffness, and vibration damping. The maximum axial load of 3 g is anticipated upon landing on the planet surface. Other parts of the flight may impose a higher axial load with lateral loads. The goal of the design is to minimize mass of the components that comprise the structure of the S/C bus, and must also fit within the physical confines of the launch vehicle. In addition, the structures must provide low thermal conductivity in an effort to thermally isolate components within the lander and the surrounding environment. The Hopper is to operate in a low cryogenic temperature environment.

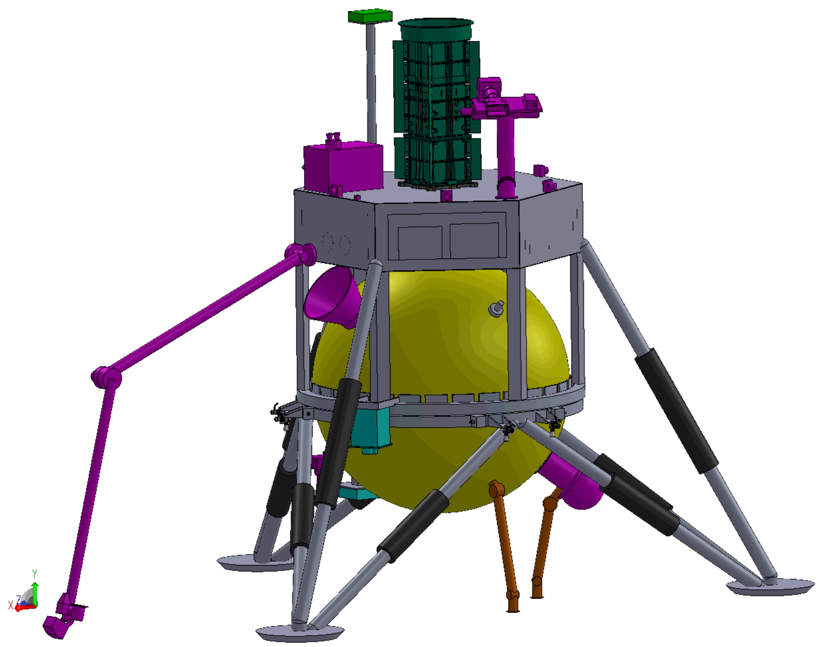


Figure 4.32—A View of the Triton Lander

4.8.2 Structures and Mechanisms Assumptions

The main bus consists of a space frame with tubular and plate members that are assumed to provide the optimum architecture for housing the necessary operational hardware. The bus components are made of Ti-6Al-4V, extra low interstitial (ELI), from the Metallic Materials Properties Development and Standardization (MMPDS), Federal Aviation Administration, 2012 (Ref. 21). Figure 4.32 illustrates views of the Lander and its main bus. Joining of components is by welding, bonding, and threaded fasteners. The analysis performed in this study looks at the landing conditions with an assumed approach velocity of 1 m/s (39 in./s) and a maximum axial acceleration of 3 g.

4.8.3 Structures and Mechanisms Design and MEL

The Lander bus consists of a Ti-6Al-4V ELI space frame. The various components are mounted to space frame. The mounted hardware includes components for communications and tracking; control and data handling; guidance, navigation, and control; electrical power; thermal management, and science.

The Ti-6Al-4V ELI has a yield strength of 869 MPa (126 ksi) and an ultimate strength of 924 MPa (134 ksi) per the MMPDS (Ref. 21). Safety factors are 1.25 on the yield strength and 1.4 on the ultimate strength per NASA-STD-5001 (1996) (Ref. 22), for a protoflight design. The resulting allowable stress is 660 MPa (95.7 ksi) limited by the ultimate stress. The Ti alloy was selected due to its high strength and desired low thermal conductivity from ambient to near zero absolute temperatures. The thermal conductivity is 7.3 W/(m-K) (51 Btu-in/(hr-ft^2-°F)). The material and bus architecture provides a technology readiness level (TRL) of six as per Mankins (1995) (Ref. 23).

The main bus consists of square tubular members with 51 mm (2.0 in.) sides and a wall thickness of 0.8 mm (0.030 in.). The legs use 64 mm (2.5 in.) OD tubing with a wall thickness of 0.8 mm (0.030 in.).

The landing gear radius relative to the center of gravity (CG) height is 1.15 times the CG height. That is comparable to the Apollo Lunar Module and proposed Orion Lunar Lander as reported by Epps (2006) (Ref. 24).

The MEL for the structures and mechanisms of the Triton Lander is shown in Table 4.13.

TABLE 4.13—TRITON LANDER STRUCTURES AND MECHANISMS MEL

WBS number	Description Case 1 CD-2015-127	Quantity	Unit mass, kg	Basic mass, kg	Growth, %	Growth, kg	Total mass, kg
06.1.11	Structures and Mechanisms			59.51	18.0	10.71	70.22
06.1.11.a	Structures			53.04	18.0	9.55	62.58
06.1.11.a.a	Primary Structures			51.72	18.0	9.31	61.03
06.1.11.a.a.a	Tank Support Ring	1.00	11.22	11.22	18.0	2.02	13.24
06.1.11.a.a.b	Upper Bus	1.00	28.10	28.10	18.0	5.06	33.15
06.1.11.a.a.e	Long Legs	3.00	1.21	3.64	18.0	0.66	4.30
06.1.11.a.a.f	Short Legs	6.00	1.02	6.13	18.0	1.10	7.23
06.1.11.a.a.g	Landing Pad	3.00	0.88	2.63	18.0	0.47	3.11
06.1.11.a.b	Secondary Structures			1.32	18.0	0.24	1.56
06.1.11.a.b.a	LIDAR Mount	1.00	0.64	0.64	18.0	0.11	0.75
06.1.11.a.b.b	Antenna Mount	1.00	0.36	0.36	18.0	0.07	0.43
06.1.11.a.b.c	Camera Mount	1.00	0.32	0.32	18.0	0.06	0.38
06.1.11.b	Mechanisms			6.47	18.0	1.16	7.64
06.1.11.b.f	Installations			6.47	18.0	1.16	7.64
06.1.11.b.f.a	AD&C	1.00	0.58	0.58	18.0	0.11	0.69
06.1.11.b.f.b	CD&H	1.00	0.59	0.59	18.0	0.11	0.69
06.1.11.b.f.c	Comm. & Tracking	1.00	0.17	0.17	18.0	0.03	0.20
06.1.11.b.f.d	Electrical Power	1.00	1.58	1.58	18.0	0.28	1.86
06.1.11.b.f.e	Thermal	1.00	0.85	0.85	18.0	0.15	1.00
06.1.11.b.f.f	Propulsion Chem	1.00	2.70	2.70	18.0	0.49	3.19

4.8.4 Structures and Mechanisms Risk

Risks for the structures includes excessive g loads, a potential impact with a foreign object during flight, a harsh landing, insufficient stiffness in the bus may cause too much deformation, vibrations, or fracture of sections of the support structure which may affect the performance of the S/C and its instrumentation. Consequences include lower performance from mounted hardware to loss of mission. The likelihood is 3 with consequences as follows

- Cost: 3
- Schedule: 4
- Performance: 4
- Safety: 1

As a mitigation step the structure is to be designed to NASA standards to withstand expected g loads, a given impact, and to have sufficient stiffness and damping to minimize issues with vibrations. Ground transport and mission trajectories are to be planned to minimize the probability of excessive loads and impact with foreign objects or too much approach velocity for landing.

4.8.5 Structures and Mechanisms Trades

A Ti alloy was chosen for the all of the structural components of the Lander bus. Titanium was chosen because of its high strength and low thermal conductivity. An Al alloy can meet the structural mechanical requirements of the Lander bus but not the necessary low thermal conductivity requirement. No other trades were discussed at this time.

4.8.6 Structures and Mechanisms Analytical Methods

Preliminary structural analysis and modeling was performed using given launch and landing loads and the dimensions of the proposed S/C bus structure. Analytical methods utilizing a spreadsheet tool were employed to analyze the bus. A maximum axial load of approximately 3 g is anticipated on the S/C upon landing on Triton's surface.

A simple analysis was performed on the bus vertical members of the Lander bus. It was assumed that the members support 100 kg and 3 g acceleration is applied due to the landing. The resulting axial stress is 3 MPa (470 psi). With an allowable stress of 660 MPa (95.7 ksi) the resulting margin is 219. The load is assumed to be shared equally among the vertical members.

The landing legs do use a spring/damper system to absorb the impact energy during landing. Limiting the peak acceleration of the bus to 3 g requires a displacement of 17 cm (0.7 in.) with an approach velocity of 1 m/s (39 in./s) and a constant deceleration. With an approximate bus mass of 500 kg (1100 lb) and the assumption that one leg may take the brunt of the impact the resulting stress in the main tubular member would be 420 MPa (61 ksi). The resulting margin is 0.57.

An additional installation mass was added for each subsystem in the mechanisms section of the structures subsystem. These installations were modeled using 4% of the CBE dry mass of each of the subsystems. The 4% magnitude for an initial estimate compares well with values reported by Heineman (1994) (Ref. 25), for various manned systems. This is to account for attachments, bolts, screws, and other mechanisms necessary to attach the subsystem elements to the bus structure, and not book kept in the individual subsystems. An 18% growth margin was applied to the resulting installation mass. These margins are placed onto the subsystem elements prior to the additional margin that was added in order to reach the 30% MGA required on the dry mass elements.

4.8.7 Structures and Mechanisms Recommendations

For a complete design, a finite element analysis (FEA) should be conducted to provide a high fidelity model of the structure. The FEA results would determine stresses, displacements, modal frequencies for vibrations, the structural response due to static forces and forced vibrations from various sources, and buckling loads. The FEA results would aid in keeping natural frequencies away from the operating frequencies of the mounted hardware, and help determine the damping requirements.

More advanced material systems and architectures may be applied for greater mass reduction. Greater use of orthogrid, and/or isogrid panels may be utilized.

5.0 Triton Hopper Cost Estimate

5.1 Ground Rules and Assumptions

The following ground rules and assumptions apply to the S/C cost estimates

- **The scope of the estimates is the design and development and flight hardware for the Hopper only**. The orbiter, SEP stage, aerocapture system, and Triton capture and descent systems are ***not*** included. In addition, the following costs are not included
    - Launch services and delivery
    - NEPA and Nuclear Safety Launch Approval
    - Mission operations and data analysis
    - Mission level costs (PM, SMA, etc.)
    - ASRG development costs (assumed to be developed by an ASRG Program)

- Propellant costs
- Science and science instruments (other than the robotic arm and cryopump)
- Any necessary technology development up to TRL 6
- Reserves

- Other assumptions
  - All hardware is assumed to be contractor responsibility (no GFE)
  - Protoflight development of the Hopper
  - Estimates assume the Hopper is contracted to a major aerospace firm; 10% fee for a cost-reimbursable contract is included
  - All system integration internal to the Hopper is assumed to occur at the manufacturer's facilities
  - Quantitative risk analysis was performed using a Monte Carlo simulation driven by input parameter uncertainty and error statistics of the CERs
  - Costs presented are mode values (approximately 35th percentile), in constant FY16 dollars
  - Coefficient of Variance (Standard Deviation/Mean) of the estimate is approximately 45%

5.2 Estimating Methodology

The Hopper estimate was developed using a Microsoft™ Excel®-based parametric cost model created for this study. The model uses an approach similar to NAFCOM, i.e., the subsystem hardware and software elements of a product-oriented WBS were estimated primarily using parametric CERs (cost estimating relationships) and the sum of the subsystem costs were used to estimate system integration costs. The primary input to the vehicle cost model was the MEL developed by the COMPASS team for this study; most of the CERs use mass as at least one of the independent parameters. The WBS in the MEL is mapped to the cost estimate WBS, which in several cases rolls up multiple elements to create elements consistent with the most applicable CERs. Most CERs were developed in-house using "ZMPE" (zero-bias, minimum percent error) or "MUPE" (minimum unbiased percentage error) regression analysis, and are based on as many relevant data points as available so the standard errors used to develop the risk model have a strong statistical basis.

5.3 Hopper Cost Estimates

The development (TRL 6 and above) and flight cost of the Triton Hopper are less than $250M, as shown in Table 5.1. The Hopper cost curve is show in Figure 5.1. While operations, complete science instrument package, and costs of the Triton capture and descent stages need to be included, a price tag of a $250M seems reasonable to be part of a Flagship Neptune orbiter mission.

Note: the following abbreviations are used for the systems integration elements: Integration, Assembly and Check-out (IACO), Systems Test Operations (STO), Ground Support Equipment Hardware (GSE), Systems Engineering and Integration (SE&I), Program Management (PM) and Launch and Orbital Operations Support (LOOS). For more detailed explanations of integration costs, see Section 5.4.

TABLE 5.1—HOPPER ESTIMATE, CONSTANT FY16 $M.

WBS/Description		DDT&E Total (FY16$M)	Flight HW Total (FY16$M)	DD+FH Total (FY16$M)
06.1.1	Science Payload[a]	3	1	4
06.1.2	AD&C	14	8	21
06.1.3	C&DH	32	8	40
06.1.4	Communications and Tracking	4	1	5
06.1.5	Electrical Power Subsystem	2	23	25
06.1.6	Thermal Control (Non-Propellant)	3	2	5
06.1.7	Propulsion	13	6	19
06.1.11	Structures and Mechanisms	6	5	11
	Subsystem Subtotal	78	53	131
Systems Integration				
IACO and STO		11	5	16
GSE hardware		13	0	13
SE&I		14	15	28
PM		10	9	19
LOOS		11	0	11
	Spacecraft total	137	82	219
	Fee (10%)			22
	Prime total			241

[a]Science Payload only includes robotic arm and cryopump

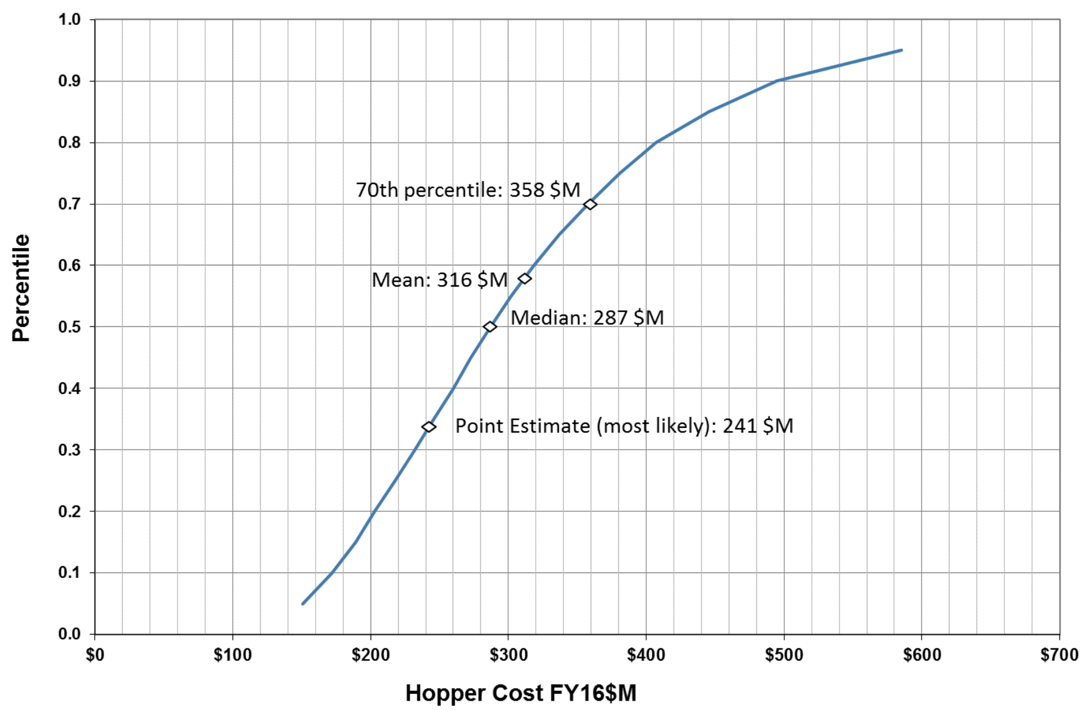


Figure 5.1—Hopper Cost Curve, Constant FY16 $M.

5.4 Definitions

5.4.1 Integration, Assembly and Checkout (IACO)

The IACO element contains all labor and material required to physically integrate (assemble) the various subsystems into a total system. Final assembly, including attachment, and the design and manufacture of installation hardware, final factory acceptance operations, packaging/crating, and shipment are included. IACO charged to DDT&E represents those costs incurred for the integration, assembly, and checkout of major test articles. IACO charged to the flight unit includes those same functions applied to the actual flight unit.

This item excludes the engineering effort required to establish the integration, assembly, and checkout procedures necessary for this effort. These engineering efforts are covered under systems engineering and integration.

5.4.2 System Test Operations (STO)

The STO element includes development testing and the test effort and test materials required for qualification and physical integration of all test and qualification units. Also included is the design and fabrication of test fixtures.

Specifically included are tests on all System Test Hardware (STH) to determine operational characteristics and compatibility with the overall system and its intended operational parameters. Such tests include operational tests, design verification tests, and reliability tests. Also included are the tests on systems and integrated systems to verify acceptability for required mission performance. These tests are conducted on hardware that has been produced, inspected, and assembled by established methods meeting all final design requirements. Further, system compatibility tests are included, as well as, functions associated with test planning and scheduling, data reduction, and report preparation.

5.4.3 Ground Support Equipment (GSE)

Functional elements associated with GSE include the labor and materials required to design, develop, manufacture, procure, assemble, test, checkout, and deliver the equipment necessary for system level final assembly and checkout. Specifically, the equipment utilized for integrated and/or electrical checkout, handling and protection, transportation, and calibration, and items such as component conversion kits, work stands, equipment racks, trailers, staging cryogenic equipment, and many other miscellaneous types of equipment are included.

Specifically excluded is the equipment designed to support only the mission operational phase.

5.4.4 Systems Engineering and Integration (SE&I)

The functions included in the SE&I element encompass: (1) the system engineering effort to transform an operational need into a description of system requirements and/or a preferred system configuration; (2) the logistics engineering effort to define, optimize, and integrate logistics support considerations to ensure the development and production of a supportable and cost effective system; and (3) the planning, monitoring, measuring, evaluating, and directing of the overall technical program. Specific functions include those for control and direction of engineering activities, cost/performance trade-offs, engineering change support and planning studies, technology utilization, and the engineering required for safety, reliability, and quality control and assurance. Also included is the effort for system optimization, configuration requirements analyses, and the submittal and maintenance of Interface Control Documents (ICDs).

Excluded from the SE&I element are those functions which are identifiable to subsystem SE&I.

5.4.5 Program Management (PM)

Elements included in the PM function consist of the effort and material required for the fundamental management direction and decision making to ensure that a product is developed, produced, and delivered.

Specifically included are direct charges for program administration, planning and control, scheduling and budgeting, contracts administration, and the management functions associated with engineering, manufacturing, support, quality assurance, configuration and project control, and documentation.

The PM element sums all of the effort required for planning, organizing, directing, coordinating, and controlling the project to help ensure that overall objectives are accomplished. This element also includes the effort required to coordinate, gather, and disseminate information.

Excluded from the PM element are those functions commonly charged to subsystem level activities.

5.4.6 Launch and Orbital Operations Support (LOOS)

This category includes the effort associated with pre-launch planning, launch and ascent, and initial on-orbit operations. The pre-launch activities include bus and payload preparation, as well as interface activities with the LV.

The launch and ascent period includes final assembly, checkout, and fueling, lift-off, telemetry, pre-launch, telemetry, tracking and command, recovery operations, and post-processing of lift-off data. Support during the mission includes drive planning and science operation, attitude and orbit control, support of on-orbit testing, routine monitoring and fault detection of space vehicle subsystem functions, and support of anomaly investigation and correction. This period ends when the newly deployed satellite is turned over to the operational user, typically after a period of 30 days.

6.0 Conclusions and Phase 2 Study Plans

The lessons learned from the Triton Hopper concept can be utilized on other bodies, whether icy or not. By the use of cryopumps grounded to the cold surface even thin atmospheres can be pumped to create propellants. Ices are even easier to collect and process given the right tools.

6.1 Benefits to NASA

The Triton Hopper addresses NASA's strategic goals 2, 3, and 6 by exploring the Triton moon and confirm that it is a KBO—a remnant of the formation of the solar system. It will directly target Objective 1.5 of the NASA Strategic plan, "*ascertain the content, origin, and evolution of the solar system and the potential for life elsewhere*," (Ref. 8) by direct examination of a KBO, with in situ characterization of primordial material from the early solar system unaltered by thermal processes, including organic deposits that may lead to clues to the origin of life.

The Triton Hoppers design will spawn a family of semi-autonomous Hoppers propelled by vaporizing gathered surface samples which could be extended to other icy surfaced objects in our solar system, and would capture the imaginations of educators and students by sharing with them exploration of a completely new environment on a foreign world. Triton Hopper also addresses the NASA technology areas of Space Power and Energy Storage, Robotics and Autonomous Systems, Communications and Navigation Systems, Science Instruments and Sensors, Materials, and Thermal Management Systems.

By addressing the challenges of autonomous hopping exploration in a cold outer solar system environment, Triton Hopper serves as a pathfinder for other exotic future exploration of the surfaces of other parts of the solar system. Exploration of Triton was selected as a candidate mission because of the challenge of the operation and the great scientific potential of Triton exploration, but once developed, icy moons and ice-containing dwarf planets are spread out across the outer solar system, and the concept could be used for exploration of many such bodies, ranging from the ice-surfaced moons of Jupiter and Saturn, the binary system of Pluto and Charon, to the trans-Neptunian objects, the Kuiper belt and beyond.

6.2 Phase II Study Plans

The very capable Triton Hopper point design based on the illuminating trades made during the study begs for better definition in a phase 2. Several options uncovered promise even more capability and should be investigated. New studies based on new observations of Triton should be utilized to refine the propellant collection techniques. More detailed modeling of this collection process, along with proof-of-concept testing would easily flush out a technology development path.

The combination of the thin atmosphere, extreme low temperatures and the variable surface provide both opportunities and challenges for the in situ collection and use of the volatile gases for propellants. Specific risks that must be addressed in phase II can be summarized as follows.

Mission: The main risks for the mission are delivery to Triton and safe hopping. The challenge of delivery to Triton in a timely manner (<15 years) and supporting it with a Neptune orbiter is key to the success of the Triton Hopper. Some combination of the following advanced technologies: nuclear electric or thermal propulsion, SEP, aerocapture, and radioisotope power are required to deliver the Triton Hopper to Triton's surface. Hopping many times safely is another challenge that will need to be addressed by up and coming collision avoidance techniques and high resolution mapping of the potential Triton landing sites by an orbiter.

Propellant Collection: Various methods of propellant collection were envisioned with a primary method of cryopumping and a backup method of scooping. The cryopump's main risk is an effective, continuous conductive path to the surface without which the 50 W of waste heat cannot be disposed. Given a fluffy or discontinuous surface this may be a challenge. The alternate use of an arm/scoop requires enough downward force on the scoop in low gravity and the presence of nitrogen at the landing site. The scoop method can also introduce contaminates (other volatiles, dust, and even rocks) into the propellant tank which will need to be filtered to eliminate the risk of impacting the tank door/seal or the propulsion valves. Both approaches can be refined with better modeling and some basic testing with frozen nitrogen.

Propulsion Performance: The baseline cold gas method has the risk of not being able to isolate and heat the tank sufficiently to produce 2000 psi/300 K propellant. Again better modeling and testing with frozen nitrogen should be able to retire that risk. Higher performance thrusters using Triton volatiles could be developed given sufficient thermal storage mass to heat the propellant during the hop – the mass of this system can be traded against the relatively heavy, high-pressure propellant tank. Better specific impulses can quadruple hop distances by doubling specific impulses (increasing from 50 to 100 s can increase the hop from 5 to 20 km.)

Appendix A.—Alternate Triton Hopper Delivery Mission

A.1 Hopper Delivery—Direct to Triton Scenario

This scenario starts with an assumed ELV launch to one of three possible delivery orbits. The first orbit is a correctly sized ~120k- by 330k-km orbit, centered at Neptune. The second possibility for a delivery orbit is an ~120k- by 1.5M-km orbit. From this orbit, an aero-braking maneuver is performed to lower the apogee of the orbit to make it 120k- by 330k-km. This scenario then reduces to the previous one. The 120k- by 330k-km around Neptune translates into a stable ~247- by 1281-km orbit centered at Triton from which the Hopper vehicle performs de-orbit, descend and landing. The third possibility is that of a delivery orbit at Triton that is ~247- by 1281-km. The same de-orbit, descend and landing scenario is then used. Table A.1 shows a summary of the three possible delivery orbits for the complete scenario.

The delivered vehicle stack consists of the Hopper vehicle and two solid rocket motor stages. De-orbit, descend and landing are as follows. Starting in the 2038 timeframe in the Triton orbit of ~247- by 1281-km, a Star 12GV solid rocket stage performs a de-orbit burn. The de-orbit burn is followed by a coast along a sub-orbital arc (i.e., a fall towards the surface) after which a Star 26 solid rocket stage performs a slowdown burn to reduce velocity. Finally, the Hopper uses preloaded propellant to perform a controlled landing on Triton's surface. For this scenario, an earlier definition of the Hopper propulsion system was used where the Cold Gas Rocket operates at an initial tank pressure of 3,500 psia and the thrust level is 764 N. The Hopper itself has a 300 kg wet mass and a 100 kg propellant limit.

Table A.2 shows a summary of de-orbit, descend and landing scenario.

TABLE A.1—SUMMARY OF DELIVERY ORBITS FOR COMPLETE SCENARIO

Triton Hopper in the 2038 timeframe: Earth-Jupiter-Neptune (or Triton direct) capture										
Case	Departure		Arrival							
	Date	Earth ΔV, km/s	Date	Capture ΔV, km/s	Hp, km	Ha, km	Inc, deg	Raan, deg	Argp, deg	Notes
Neptune correct	02-16-2019 at 03:58:06.31	6.6482	06-28-2038 at 21:41:58.30	5.7639	119,901.60	329,995.00	129.8120	–134.9541	322.6060	Neptune based
Neptune aerobrake	02-16-2019 at 04:51:00.06	6.6465	06-28-2038 at 21:41:58.30	5.1817	119,901.60	14,975,236.00	129.8120	–134.9541	322.6060	Neptune based
Triton direct	02-16-2019 at 03:50:56.88	6.6484	06-28-2038 at 21:41:58.30	4.7151	247.46	1,281.00	118.2357	–22.7934	33.8952	Triton based

Note: Orbital reference frame is: *body-centered ecliptic J2000*

TABLE A.2—SUMMARY OF THE DE-ORBIT, DESCEND AND LANDING IN THE COMPLETE SCENARIO

Triton Hopper in the 2038 timeframe - Descend and Land from 247- by 1281-km (altitude) Triton orbit; gets Hopper to the surface															
Stage	Final landing via Hopper's Cold Gas Rocket: I_{sp} from CGR nitrogen Model M_0= 300 kg (for Hopper vehicle)														Notes
	Initial (max) thrust, N	Initial (max) thrust, lbf	Initial T/W, Earth	Initial T/W, Triton	Hp, km	Ha, km	Altitude, km	Mission elapsed time[a], s	Initial mass, kg	Final mass, kg	Prop used, kg	ΔV[a], m/s	Final velocity, m/s	Final T/W, Triton g	
1x Star 12GV	6,472.1600	1,454.9994	1.0678	13.4426	247.46	1,281.00	1,280.8683	7,906.8518	618.0566	584.8708	33.1858	62.3973	563.0261	14.2053	De-orbit
Coast	0.0000	0.0000	0.0000	0.0000	-183.87	1,285.58	1,281.2097	14,115.1770	576.1608	576.1608	0.0000	62.3973	1,156.1136	0.0000	Fall
1x Star 26	33,361.6620	7,500.0000	5.9045	74.3304	-1,336.49	2.29	2.2895	14,134.1770	576.1608	337.6482	238.5126	948.8718	148.9547	126.8370	Slow down
Hopper	761.0907	171.0996	0.2587	3.2567	-1,352.44	0.00	0.0000	14,238.5150	300.0000	217.2131	82.7870	1,132.2100	1.0000	3.4283	[b]Land

[a] Cumulative amount
[b]Landing velocity limit of 1 m/s; no landing g-limit

Table A.3 shows an overall ΔV summary for the three delivery options.

TABLE A.3—ΔV SUMMARY OF THE THREE DELIVERY OPTIONS

Triton Hopper in the 2038 timeframe: ΔV summary for complete scenario							
Case	Departure		Arrival				Notes
	Date	Earth ΔV, km/s	Date	Capture ΔV, km/s	Landing ΔV, km/s	Total ΔV	
Neptune correct	02-16-2019 at 03:58:06.31	6.6482	06-28-2038 at 21:41:58.30	5.7639	1.1322	6.8961	Launch to Neptune orbit and perform landing on Triton
Neptune aerobrake	02-16-2019 at 04:51:00.06	6.6465	06-28-2038 at 21:41:58.30	5.1817	1.1322	6.3139	Launch to Neptune orbit, do aerobrake (free) and perform landing on Triton
Triton direct	02-16-2019 at 03:50:56.88	6.6484	06-28-2038 at 21:41:58.30	4.7151	1.1322	5.8473	Launch to Triton orbit and perform landing on Triton

Appendix B.—Study Participants

Triton Hopper Team			
Subsystem	Name	Center	Email
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