AIAA SPACE Forum
12 - 14 Sep 2017, Orlando, FL
AIAA SPACE and Astronautics Forum and Exposition

AIAA 2017-5288

Hercules Single-Stage Reusable Vehicle supporting a Safe, Affordable, and Sustainable Human Lunar & Mars Campaign

D.R. Komar [1], Dr. Robert Moses[2],
NASA Langley Research Center, Hampton, VA, 23661

This paper presents a conceptual transportation architecture designed to support future lunar and Mars campaigns aimed at establishing a permanent and self-sustaining human presence beyond Earth orbit in the next half century, as a prelude to settlement and colonization, with NASA playing a major role. Initially designed to support a Mars campaign documented in NASA Langley's ISRU-to-the-Wall study[1], the Hercules Single-Stage Reusable Vehicle concept has evolved to become a space transportation system that sets a new standard for operational flexibility and safety. Referred to herein as the Hercules Transportation System, the modular and flexible transportation architecture allows a common system design that is configured to support planetary and interplanetary transport of cargo and crew between the Earth, the moon, and Mars. In addition, Hercules employs several key design features that enable full coverage aborts during both ascent and descent from either the moon or Mars. This paper presents an overview of the Hercules Transportation System and highlights the key design features and capabilities that enable a operationally flexible and safe space transportation system that supports future lunar and Mars campaigns.

Nomenclature

ACC6	=	Advanced Carbon-Carbon
ADS	=	Ascent/Descent System
ATLS	=	Abort/Terminal Landing System
ATO	=	Abort-to-Orbit
ATS	=	Abort-to-Surface
DSG	=	Deep Space Gateway
ECLSS	=	Environmental Control and Life Support System
EDL	=	Entry, Descent and Landing
EI	=	Entry Interface
EXAMINE	=	Exploration Architecture Model for In-Space and Earth-to-Orbit
EZ	=	Exploration Zone, defined as 50 km radius circle with the base located appx. in the center.
HCRV	=	Hercules Crew Rescue Vehicle
HIAD	=	Hypersonic Inflatable Aerodynamic Decelerator
HMTV	=	Hercules Mars Transfer Vehicle
HPDV	=	Hercules Payload Delivery Vehicle
HSRV	=	Hercules Single-Stage Reusable Vehicle
HTS	=	Hercules Transportation System
ISRU	=	In-Situ Resource Utilization
kg	=	kilograms
klbf	=	kilopounds-force
km/s	=	kilometers per second
kN	=	kilonewtons

[1] Aerospace Engineer, Vehicle Analysis Branch, MS 451, AIAA Senior Member.
[2] Atmospheric Flight and Entry Systems Branch, MS 489, AIAA Associate Fellow.

LCH_4	=	Liquid Methane
LCI	=	Layered Composite Insulation
LLO	=	Low-Lunar Orbit
LMO	=	Low-Mars Orbit
LOI	=	Lunar Orbit Insertion
LO_2	=	Liquid Oxygen
mt	=	metric tons
MCC	=	Mid-Course Correction
MLI	=	Multi-Layer Insulation
MOLA	=	Mars Orbiter Laser Altimeter, elevation model based on Mars Global Surveyor (MGS) data
m/s	=	meters per second
NASA	=	National Aeronautics and Space Administration
NRHO	=	Near Rectilinear Halo Orbit
OML	=	Outer moldline
PICA	=	Phenolic Impregnated Carbon Ablator
RCS	=	Reaction Control System
RPOD	=	Rendezvous, Proximity Operations, and Docking
RTEZ	=	Return to Exploration Zone
SLS	=	Space Launch System
SRP	=	Supersonic Retro-Propulsion
TCS	=	Thermal Control System
TDL	=	Terminal Descent and Landing
TPS	=	Thermal Protection System
V_∞	=	Excess Hyperbolic Speed relative to Departure or Arrival Body
V_{entry}	=	Inertial Velocity at Atmospheric Interface of Arrival Body
ΔV	=	Velocity Change due to Translational Propulsive Maneuvers
^oF	=	degrees Fahrenheit

I. Introduction

Over the past three years a team at NASA Langley Research Center has been developing a conceptual architecture to support the strategic goal of affordably establishing a permanent and self-sustaining settlement on Mars in the next half century, as a prelude to colonization, with NASA playing a major role. Employing both NASA's Space Launch System (SLS) and emerging commercial launch capabilities in this architecture, the Langley team found that utilizing reusable space transportation systems, leveraging Mars resources to the maximum extent through in-situ resource utilization (ISRU), and developing significant robotic capabilities for autonomous operations both on Mars surface and in Mars orbit are the keys to an affordable, sustainable campaign on the path towards permanent human settlement and eventual colonization.

Known as the ISRU-to-the-Wall study[1], the architecture utilizes a multi-phase, multi-decade campaign. Early phases emphasize technology development and demonstration, transportation system maturation, and autonomous surface and orbital systems operations, while follow-on phases focus on affordable growth of the base infrastructure and expansion of Mars industrial capabilities to enable the base to become self-sufficient.

The results of the campaign study illustrated the impact of ISRU, reusability, and automation on pioneering Mars. With current SLS launch assumptions of 2 per year with the potential for a third flight every other year, sustained presence on Mars requires a transition from Earth dependence to Earth independence. The study analyzed the surface and transportation architectures and compared campaigns that revealed the importance of ISRU and reusability, in particular, when applied to a lander. A reusable Mars lander eliminates the need to deliver a new descent and ascent stage with each cargo and crew delivery to Mars, substantially reducing the mass delivered from Earth over the course of a multi-decade campaign. As part of an evolvable transportation architecture, this investment is key to sustaining continuous human presence on Mars for the far term. The extensive use of the vast amounts of water and carbon dioxide abundant on Mars, to make and repair nearly everything on planet, reduces the logistics supply chain from Earth in order to support population growth at Mars. Reliable and autonomous systems, in conjunction with robotics, are required to enable ISRU architectures as systems must operate and maintain themselves while the crew is not present. By the time the first crew arrives, the systems have been operating, repaired, and studied for several years, with failure modes understood and solvable, which in the end, buys safety. Because Mars has abundant resources available to support human life, with the extraction and use of in-situ water being the fundamental Martian resource, this approach of extensive ISRU combined with a reusable lander, enables a sustained human presence at lower overall cost than would be possible otherwise.

Key attributes of the architecture include aggregation of Mars-bound systems and payloads at NASA's Deep Space Gateway (DSG)[2], reusable interplanetary transportation between cis-lunar space and Mars orbit, reusable ascent and landing systems that taxi cargo and crew between low Mars orbit (LMO) and Mars surface, and a Mars surface resource utilization system that produces, among other things, propellant for the reusable taxi. This taxi vehicle concept, named Hercules, is a single-stage, reusable vehicle designed to operate between LMO and the Mars surface base utilizing oxygen and methane propellants manufactured at the Mars base from Martian resources.

Initially designed to support this Mars architecture and campaign, the Hercules lander concept has evolved to become a space transportation system that sets a new standard for operational flexibility. Referred to herein as the Hercules Transportation System (HTS), the modular and flexible transportation architecture allows a common system design that is configured to support planetary and interplanetary transport of cargo and crew between the Earth, the moon, and Mars.

This paper emphasizes the design features and capabilities of the Hercules lander, now referred to as the Hercules Single-Stage Reusable Vehicle (HSRV). Described are the HSRV configurations that support the delivery of cargo or crew to the Martian surface and to the lunar surface and includes a summary of the nominal operations and the crew abort capabilities. Not covered in detail in this paper are the design features and capabilities for the interplanetary versions of the HTS, namely the Hercules Payload Delivery Vehicle (HPDV) and Hercules Mars Transfer Vehicle (HMTV) configurations. These will be discussed briefly but detailed in future papers.

II. Campaign and Mission Architecture Overview

Key attributes of the overall architecture and campaign that are consistent with the planned operations of the HTS are discussed. First it is assumed that the international space station, SLS, Orion, and DSG have all been developed and are operational. These capabilities are leveraged as part of the proposed architecture, although the affordability of the campaign is potentially improved by leveraging commercial services in place of the government owned and operated services (e.g. - SLS and Orion are used only as needed and commercial launch systems and capsules are used in some cases at lower cost). The campaign strategy proposed herein proposes the HTS as the next development step following the DSG.

A. Campaign Overview

The ISRU-to-the-Wall study proposed a four phase campaign conducted over multiple decades. The initial phase, *Prepare*, advances the technologies and builds systems to enable sustainable human exploration. This phase would also contain missions to local bodies (e.g. cis-lunar space) to develop, prove, and sustain needed capabilities. The *Found* phase would begin with the first human landing on Mars, establishing the initial human presence on Mars and emplacing the necessary hardware and infrastructure to sustain a human presence on Mars. The *Expand* phase increases the infrastructure on Mars to support a larger population and longer stays. Utilizing the infrastructure initially emplaced during the *Found* phase as well as additional capability to utilize in-situ resources, crew size can increase while reliance on Earth resources can decrease. Finally, the *Sustain* phase maintains a large human presence on Mars through extensive use of in-situ resources, automation, and reusability to explore and settle the planet[1].

The HTS supports all phases of this campaign strategy, including any lunar missions aimed at the economic expansion of the moon.

For the lunar campaign the HSRV is delivered by the SLS to the DSG, assumed to be located in a Near Rectilinear Halo Orbit (NRHO) around the moon[15]. Propellant resupply via commercial launch at the DSG allows the HSRV to demonstrate its reusability and other attributes (modularity, operational flexibility, safety, risk mitigation, etc…) at the moon to buy down risk for the future Mars campaign. The HSRV could also serve as the work horse for the setup of a lunar base not only necessary for Mars preparations but also for commercial ventures requiring repeatable and affordable sorties to the lunar surface.

For the Mars campaign the initial HSRV landings would be uncrewed, focusing on delivery of critical payloads for the surface infrastructure of a Mars base. This includes landing, autonomously deploying, and initiating power, thermal, habitation, mobility, in-situ resource acquisition and processing, and propellant production and storage infrastructure. The reusable HSRV (discussed in detail below), operating initially as an "expendable" lander, uses these early flights to build system maturity and reliability on its way to being human-rated. The expended lander hardware elements, designed to be modular and multi-functional, are re-purposed for use as part of the base infrastructure. Once a functioning base that is generating propellant is established, the campaign will transition into a build-up phase where reusable systems become operational so that the base can grow affordably. Between the initial interplanetary cargo delivery (using the HPDV) flights and HSRV flights flown in the *Prepare* phase, multiple aerocapture and entry, descent, and (precision) terminal landings will have occurred at Mars, significantly increasing the maturity and reliability of the vehicles prior to the first crew landing. First crewed missions in the *Found* phase begin when sufficient infrastructure is deployed and operating with sufficient maturity demonstrated to enable crewed landings with maximum safety provisions.

B. Transportation Architecture Overview

For early lunar missions, the HSRV can deliver up to 20 metric tons (mt) of cargo or a crew of 4 to the lunar surface from the DSG where Hercules is resupplied from Earth with cargo and propellants. In addition to providing large cargo and crew delivery capability to the moon to support the lunar campaign, these missions offer an opportunity to demonstrate key operational capabilities in the lunar environment that are needed to support the future Mars campaign. Among these is the abort capability where an orbital Hercules crew rescue vehicle (HCRV), based at the DSG and derived from re-purposed HSRV systems, is docked at the DSG and readied for any abort-to-orbit event that occurs during lunar descent or lunar ascent. In this case the HCRV is loaded at the DSG with Earth-supplied propellants and is capable of transferring from the DSG to low-lunar orbit (LLO) to rendezvous with the crew stranded in lunar orbit and return them safely to the DSG.

After some period of time supporting the lunar campaign and demonstrating the operational capabilities required for Mars, the initial phase of the Mars campaign starts. For interplanetary cargo transfers to Mars, the HPDV configuration delivers up to 40-60 mt of cargo to LMO using a minimum energy transfer with an aerocapture at Mars enabled by utilizing an ablative thermal protection system (TPS).

At Mars, an orbital node is proposed that functions as a key aggregation point for the architecture in Mars sphere-of-influence. This node, delivered by the HPDV, would be located in a 500 km circular LMO at an inclination that offers access to the selected base site. The node provides multiple docking ports and offers autonomous or semi-autonomous robotic in-space assembly and servicing capabilities intended to 1) construct and maintain the node; 2) facilitate capture, berth and dock of incoming vehicles; 3) facilitate transfers of payloads between the HPDV and the HSRV; and 4) facilitate propellant transfers from the HSRV to various vehicles at the node or to the node itself. The LMO node serves as a crew transfer port, both for arriving and departing crews. The LMO node itself will require some habitability to enable the crew rotations, but the node is not intended to support crew for extended periods. If long duration crew habitability is required at the node due to emergency circumstances, for example, the crew would occupy and depend on the docked vehicles to offer the functions to keep them safe and alive. It is expected that an orbital HCRV, based at the node and derived from re-purposed HSRV systems, is available in the event that an abort-to-orbit occurs during entry or ascent and the crew was unable to return all the way back to the node. The orbital HCRV would transfer to the stranded crew vehicle, rendezvous and dock with it and return the crew back to the node.

Due to its close proximity to Mars, the node would be gravitationally stabilized and require minimal propellant for orbital attitude and maintenance. Selection of LMO, specifically a circular LMO at 500 km, for these functions minimizes the ΔV requirements on the HSRV (relative to an ascent to longer period elliptical orbits). The systems arriving from Earth require the capability to access the LMO node. This places more performance burden on the interplanetary transportation systems. However, since the HSRV is reusable and is expected to operate between the surface base and the LMO node twice per Earth year, and the HPDV arrives at the node less frequently, perhaps only once per synodic cycle (2.2 Earth years), choice of LMO is favorable. Also, once the HSRV is fully operating in reusable mode fewer interplanetary missions delivering new HSRV's are needed, thus reducing the performance burden on the interplanetary transportation systems.

For interplanetary crew transfers to Mars, the HMTV configuration delivers a crew of 4 to LMO using a 90-120 day fast-transit transfer with an aerocapture at Mars. Fast-transit transfers between Earth and Mars significantly reduce the crew exposure to galactic cosmic radiation and zero-gravity relative to minimum energy transfers that typically range from 180-300 days duration. Both of these interplanetary configurations depart from the DSG where propellant delivered from Earth or extraterrestrial sources (potentially using commercial launch systems) is resupplied to the vehicle along with crew and/or cargo.

At Mars, the HSRV cargo configuration delivers 20 mt from LMO to the Mars surface base, refuels on the surface with Mars produced propellants, and returns to LMO delivering 5 mt of propellant to the node. The HSRV crewed configuration delivers 4 crew from Mars orbit to the surface base, refuels on the surface, and returns to the low-Mars orbit with 4 crew and 4 tons of propellant. The propellant delivered to the node is in addition to propellant retained on the HSRV to allow return to the Martian surface and can be aggregated at the node for use in refueling interplanetary transfer vehicles returning to Earth.

C. Mars Surface Architecture Overview

For conceptual design and planning purposes, a base surface site is selected that offers key resources to support the development of a Mars surface settlement. Called an Exploration Zone (EZ) with a diameter of 100-km, NASA has identified up to 50 candidate EZ sites on Mars[3] that may offer habitability and resources suitable for sustaining human presence as well as being valuable for scientific discovery. The first Human Landing Site Workshop[4] tapped into leading experts in exploration and science to evaluate regions of Mars as suitable for the first crewed settlement. Of those, Deuteronilus Mensae[5] scored well and was therefore selected as the surface site target (see Table 1) for this study. Key among the favorable attributes of Deuteronilus Mensea is that ice resides close to the surface and it is less susceptible to dust storms.

Table 1. HSRV Landing Site Target

Mars Landing Site	Deuteronilus Mensae
Mars Latitude	43.9º N
Mars Longitude	23.6º E
Elevation relative to MOLA	-3.7 km

Conceptual layout of the base at the surface site is shown in Figure 1. Relative placement of the various functional zones is driven by the desire to minimize risk to the base during launch and landing operations.

Surface habitation that is protected from galactic cosmic radiation is emplaced prior to the initial crew arrival. Several habitation concepts are being explored that minimize spacecraft volume by using inflatable, deployable structures and reduce landed mass by accessing in situ resources such as regolith and water.

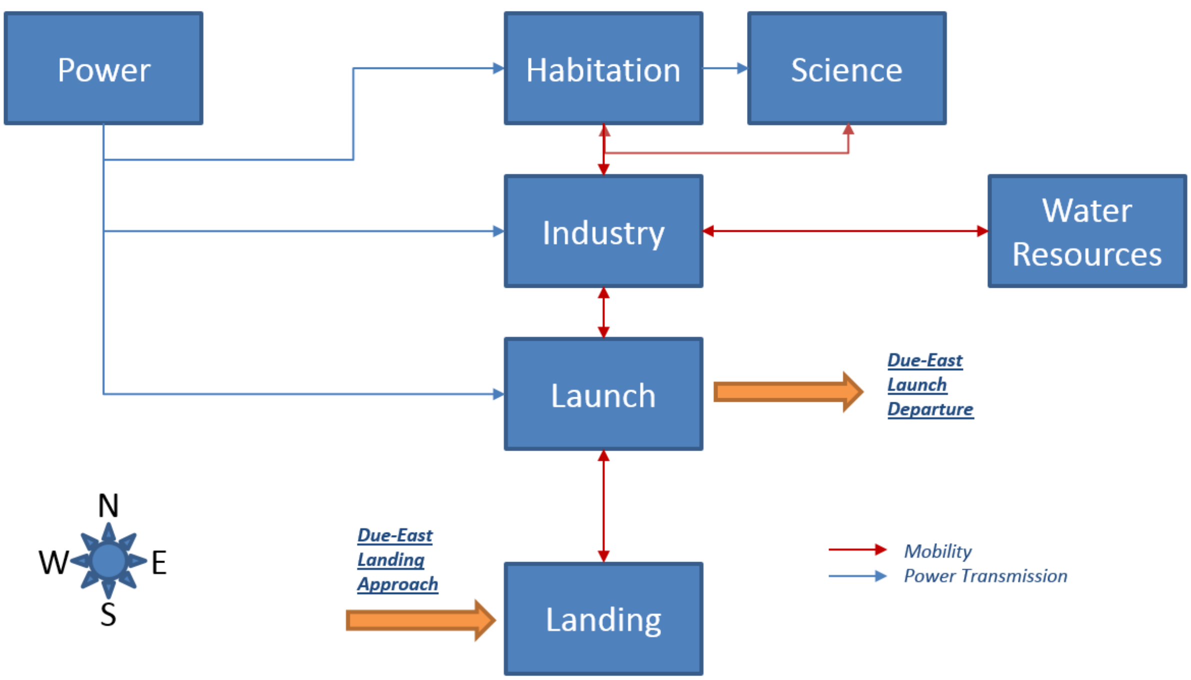

Figure 1. Base layout concept at Mars settlement.

The HSRV's nominal operations at Mars include loading propellants produced at Mars at the surface base, launching to the LMO node to pick up cargo or crew, and landing back at the surface base to offload the cargo. In terms of the Mars surface architecture, key infrastructure required to support the HSRV operations includes surface power systems, the propellant production plant, mobile cargo offloading equipment, a launch facility, a separate landing zone, and mobility equipment that can transport the HSRV from the landing zone to the launch facility. In addition, systems for autonomous inspection and maintenance are needed to verify the integrity of the vehicle prior to launch.

The propellant production plant is located in an industrial zone adjacent to the launch facility. The plant includes fixed and mobile infrastructure for resource acquisition (water and carbon dioxide), propellant manufacturing, liquefaction, propellant storage, and propellant transfer. The entire base, including the production plant, are supported by the power system infrastructure.

Offloading cargo from the HSRV payload bay requires some form of hoist or lift that can handle the 20 ton payload in Mars gravity. For the initial flights in the *Prepare* phase of the campaign the HSRV is not re-used; rather, the sections of the vehicle are re-purposed to support the base infrastructure and campaign. The HSRV design allows the sections to separate and be transported by the nose section for precise positioning. For example, the first flight may deliver a nuclear power system that is located in a remote zone relative to the other base zones (i.e. - habitation, industrial, launch, and landing zones). However, the ascent and descent sections of the HSRV are re-purposed to the industrial zone to serve as a propellant storage facility. Thus, the landing will initially target a landing location to position the tanks sections, then the nose and payload sections containing the power system separate and are relocated to the power zone using the nose section mobility system (described in Section III).

The launch zone is located such that the ascending HSRV does not fly over any of the base zones. Ideally, the launch zone has a prepared pad that mitigates the risks associated with surface ejecta due to rocket engine plumes at engine start for Mars ascent.

The landing zone is located 1-2 km south of the launch zone such that the arriving HSRV, coming in from the west, does not overfly the base. Despite expectations of a precision landing capability, the choice to have separate launch and landing zones is to minimize the risk to the base for a missed landing. This drives the need for mobility systems that can transport the HSRV from the landing to launch zones.

Given that the HSRV is reusable, mobile robotic systems that operate autonomously are needed to perform inspections and maintenance. Initial operations at the moon allow demonstration and development of autonomous robotic capabilities for servicing, maintenance, construction, etc…, where they can be operated semi-autonomously. Over time autonomy of the robotic systems is proven, thus buying down development and operational risk for the Mars campaign.

As a contingency during HSRV flight operations, an additional HCRV is based at the launch site, consisting of a space nose section from a previous re-purposed HSRV. Resupplied for every launch and entry event, this "surface" HCRV is on standby in the event of an abort-to-surface event by the HSRV. The HCRV is designed to have roundtrip hopping capability from the base to any point in the EZ – 50 km in any direction from the surface base – enabling crew rescue for any abort-to-surface event within the EZ.

Likewise, for lunar missions, a surface HCRV is docked at the base and readied for any abort-to-surface event that occurs during lunar descent or lunar ascent. In this case the HCRV is loaded at the base with Earth-supplied propellants and is capable of hopping from the base up to 100 km and back to retrieve the crew stranded on the surface. In this scenario, propellants are maintained on the HCRV for extended periods to ensure availability of the HCRV. The approach for this will be described in detail in a later section of the paper.

III. Hercules Transportation System

The HTS is a set of vehicle configurations derived from the common Hercules outer moldline (OML) that supports a range of missions between Earth, the moon and Mars. The primary variations include the following and are shown in Figure 2:

<u>Hercules Singe-Stage Reusable Vehicle (HSRV)</u>

- Mars Cargo
- Mars Crew
- Lunar Cargo
- Lunar Crew

<u>Hercules Payload Delivery Vehicle (HPDV)</u>

- Interplanetary Cargo

<u>Hercules Mars Transfer Vehicle (HMTV)</u>

- Interplanetary Crew

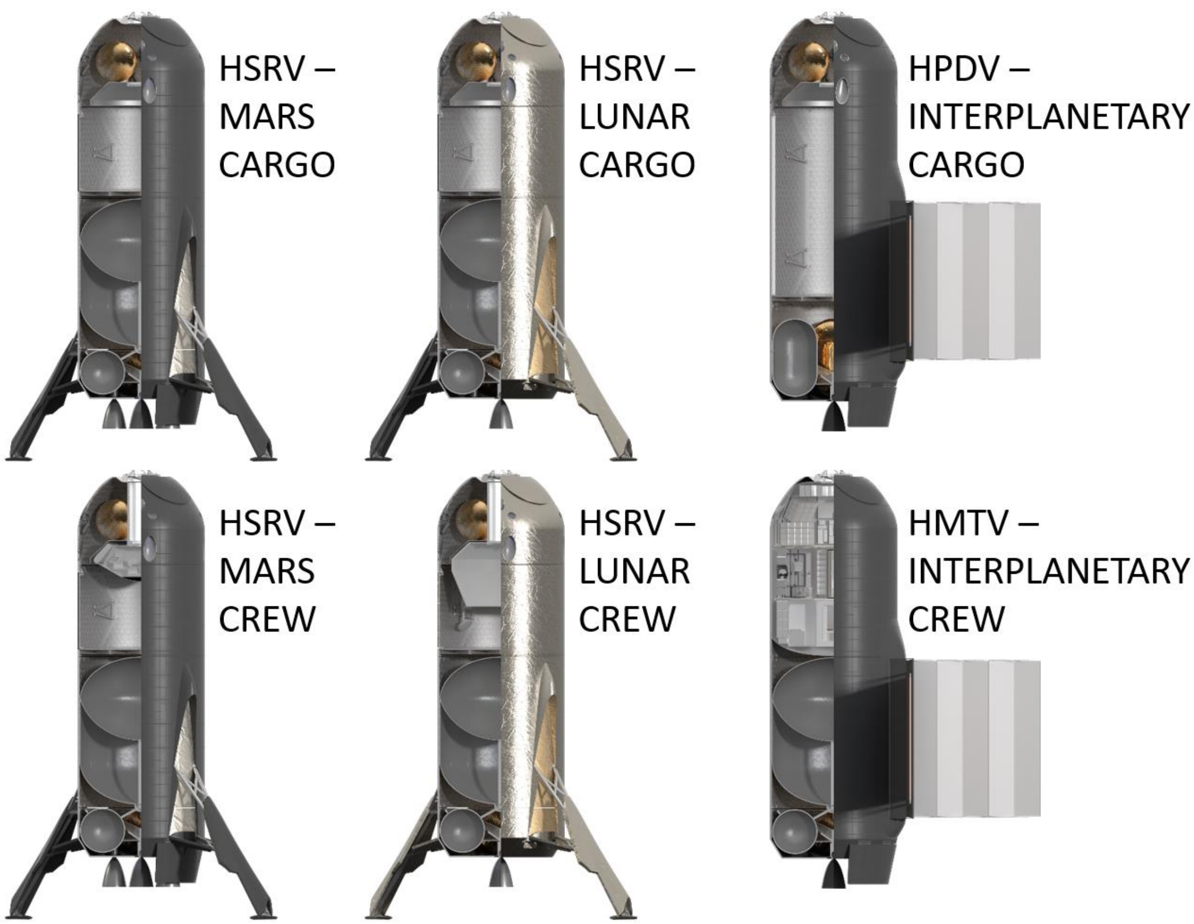

Figure 2. HTS Cargo and Crew Configurations supporting Lunar, Mars, and Interplanetary Missions.

The OML design for these options is based on the HSRV configuration designed for aerodynamic entry and launch at Mars. The HSRV is configured for vertical takeoff, mid lift-to-drag (mid-L/D) nose entry and vertical landing, supported by landing legs that extend from the vehicle's aft sections. The HSRV is 6.0 meters diameter and 19.2 meters long from the nose to the engine exit plane. Table 2 summarizes the aerodynamic design parameters for the HSRV.

Table 2. HSRV Aerodynamic Reference Data

OML Reference Diameter, m	5.99
Aerodynamic Reference Length, m	17.83
Aerodynamic Reference Area, m^2	28.18
Length-to-Diameter	2.98

One of the primary motives for configuring the HSRV as a mid-L/D, nose-entry vehicle is to allow it to package within SLS's 8.4 meter diameter fairing for Earth launch. Other key design considerations include the proper location of the center of gravity for both Mars entry and Mars ascent and the proximity of the payload to the surface (affects ease of payload offloading). Balancing these constraints and design attributes lead to the selected design for the Mars cargo and crew configurations.

The following subsections describe each of the configurations, focusing initially on the HSRV configurations for Mars and then discussing the lunar and interplanetary configurations relative to the HSRV in terms of its functionality and what is different, changed or modified in the design configuration.

A. HSRV – Mars Cargo and Crew Taxi

The HSRV concept is a multi-functional, modular, operationally flexible, single-stage, reusable vehicle designed to operate between LMO and the Mars surface base utilizing oxygen and methane propellants manufactured at the Mars base from Martian resources. Its primary function is cargo and crew transport between LMO and the Mars surface base.

The subsystem design and layout of the HSRV is driven by the desire to reduce or eliminate key programmatic and technical risks and to maximize crew safety. Implementation of this design philosophy results in the majority of the initial *Prepare* phase of the campaign being uncrewed and relying heavily on autonomous robotic systems. Items addressed through the HSRV configuration design include:

1) SLS Launch Shroud Packaging – fits within the baselined 8.4 meter shroud;
2) Backshell Heating – design protects payloads from heating with an aeroshell enclosure;
3) Surface/Plume Interaction – surface ejecta risks to vehicles and surface systems minimized;
4) Engine Deep-Throttling – limit the required level of throttling for engines systems to 50% for landing;
5) Engine Start Criticality – provide contingency abort option for failed retro-propulsion engine start;
6) Crew Abort/Recovery – enables crew abort and safe recovery during all phases of HSRV operation.

A common design choice used for all sections of the Mars HSRV are the primary structure and entry heat shield. All primary structures are a carbon-fiber composite material with a 350 ºF temperature limit. For the heat shield a durable and reusable advanced carbon-carbon (ACC6) hot structure TPS is used. Each TPS panel is mechanically attached to the composite structure. Each panel consists of an outer ACC6 layer with layers of opacifed fibrous insulation (OFI) and Nextel blankets providing the resistance to aerodynamic heating[6]. A common panel thickness is used that is sized for the worst case heating location on the HSRV. Nominally a panel consists of 0.05 inch thick ACC6 over 0.45 inch thick OFI, with a 0.02 inch thick layer of Nextel. Due to manufacturing limitations, the maximum panel size (length and width) is limited to one meter. The majority of the vehicle's cylindrical acreage is covered by these common panels, while custom sizes are required in closeouts and in the nose section. This approach, using a common panel thickness over the complete vehicle (including the backshell), minimizes the number of unique TPS panels required. For aerodynamic entry from LMO, maximum external wall temperatures approach 2400 ºF, well below the 2900 ºF maximum reusable temperature limit for the ACC6 hot structure[7].

Figure 3. HSRV landing on Mars using the nose section mounted ATLS engines.

A key design choice for the HSRV is to utilize two sets of main propulsion systems, with five (5) ascent/descent system (ADS) engines on the vehicle base and eight (8) abort/terminal landing system (ATLS) rocket engines in the nose section. The unique positioning of the ATLS rocket engines in the nose section gives the HSRV a high degree of operational flexibility and a means to address the risk and safety issues. Nominally, the ATLS rocket engines are used for terminal landing of the HSRV, as illustrated in Figure 3. This packaging approach reduces the hazards to the surface infrastructure due to surface ejecta blast from the rocket plumes and eliminates the need for deep throttling of the base-mounted descent engines. Landers using a single engine system for retro-propulsion and terminal landing typically require the engine throttle to 10-20% maximum thrust. By using a secondary system for terminal landing, both the ADS and ATLS engines require only 50% maximum throttle. Also, in a contingency, the ATLS rocket engines serve as an abort system that separates the nose section during an unlikely catastrophic vehicle failure during ascent or entry.

All of the configurations employ a common propellant architecture using liquid oxygen (LO_2) and liquid methane (LCH_4) for all propulsion systems including the ADS, the ATLS, and the reaction control system (RCS).

The systems are interconnected and allow propellants to be pumped from tank to tank, forward or aft, as needed. Also, all tank pressurization gases are derived from the main propellants, thus eliminating the need for helium which is not available in sufficient quantities on Mars to support the HSRV.

As illustrated in Figure 4, a nominal mission profile starts with the loading of Mars produced propellants onto the HSRV at the launch site. Once loaded, the HSRV ascends to LMO, then rendezvous and docks with the LMO node. On-orbit operations include transferring propellants from the HSRV to the node and either transferring cargo from the node to the HSRV, or rotating a crew complement. The HSRV then undocks, separates and positions itself for de-orbit. Following the de-orbit burn, the HSRV performs an aeroentry, transitions to supersonic retro-propulsion using the ADS engines on the vehicle base for terminal descent, then transitions to the ATLS engines in the nose section for terminal landing. While docked to the LMO node, power and consumables are provided by the node, thus the HSRV is designed for only 12 crew-days of operation.

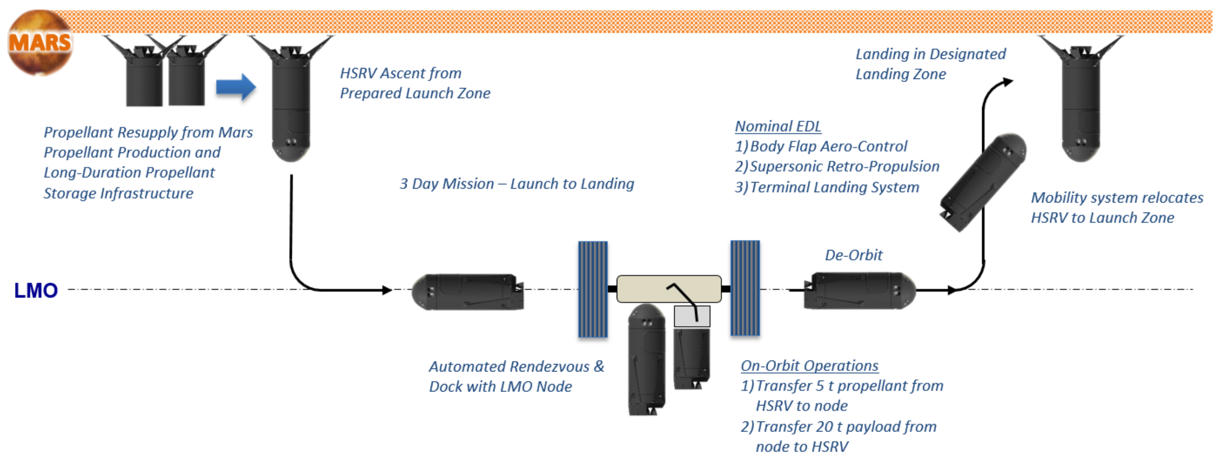

Figure 4. HSRV – Mars Concept of Operations.

In the cargo mode, the HSRV is designed to deliver 20 mt payload from LMO to the surface and then ascend back to LMO using propellant produced at the base without payload but carrying up to 5 mt of additional propellant that is used to resupply assets aggregated at the LMO node. In the crew delivery mode a 5.5 mt crew capsule supporting 4 crew is included in the nose section and is used for both ascent and entry, and up to 4 mt of propellant produced on Mars is resupplied to the node. The crew capsule is designed to be separable from the nose section and is used in the event of a catastrophic vehicle failure during Mars ascent or entry to safely recover the crew by abort-to-surface or abort-to-orbit.

The vehicle is divided into 4 sections:

1) Nose Section;
2) Payload Section;
3) Ascent/Descent Section.

Each of these sections are separable and serve multiple functions in the architecture. For early demonstration and base infrastructure buildup flights in the *Prepare* phase, the HSRV is "expendable", with the hardware sections re-purposed to operate as part of the base infrastructure. Once the Mars base infrastructure required to enable reusability of the HSRV is deployed and operating with sufficient maturity, the HSRV's are operated in its intended reusable mode and these sections no longer separate.

The following subsections describe the design of each of the HSRV sections along with the key capabilities, including nominal, contingency and re-purposed operations, enabled through the modular design.

Nose Section

The nose section design for both the cargo or crew variants includes structures and mechanisms; an external TPS; the ATLS tanks, feed and engines; the RCS; a power generation and tank pressurization system; and the vehicle avionics systems.

The nose layout is a 60 degree sphere-cone that transitions to a cylindrical shell. A retractable door, located at the tip of the spherical section, exposes a standardized mechanical docking system[8] for docking to the DSG, the

LMO node, or to other vehicles. The base of the nose section includes a conical structural adapter that supports the ATLS tanks. For the crewed configuration only, a crew capsule, suspended from below, is mated to this adapter connecting it to a pressurized tunnel, linking it to the docking port, that enables the crew to egress/ingress the capsule and the HSRV in zero-g. For the cargo configuration the capsule and tunnel are not required.

Four ATLS propellant tanks (two each LO_2 and LCH_4) are packaged in the nose. These spherical tanks, each 2 meters in diameter, are composite structure and store propellants at 500 psia. These tanks feed propellants to both the ATLS engines, scarfed into the cylindrical shell structure, and the RCS thrusters, scarfed into the conical section of the sphere-cone. Eight pressure-fed ATLS engines are installed in the nose section (four sets of two engines in redundant pairs) oriented 90 degrees apart and installed at a 30 degree cant angle relative to the vertical providing retro-propulsion for terminal landing, while twelve RCS thrusters (four sets of three thrusters) provide thrust for on-orbit, de-orbit, and entry attitude control. Specification of the ATLS engine and RCS thruster design parameters is shown in Table 3.

The nose section includes two power generation and tank pressurization systems that support the entire vehicle during ascent, on-orbit, and EDL. Each system uses an internal combustion engine (ICE) burning gaseous oxygen and gaseous methane drawn from the ATLS tanks, each producing up to 40 kW of peak power (at 100% ICE throttle) for short duration peak electrical loads. At idle, each ICE continuously provides 3 kW of electric power for the vehicle, consuming ullage and boiloff gases at low rates. The ICE is also used to generate pressurization gases for the ATLS and ADS tanks, on demand, from the liquid propellants using the ICE cooling loop to heat and vaporize the liquid propellants. This system is similar in form and function to the Integrated Vehicle Fluids (IVF) system in development at United Launch Alliance[9,10] but using oxygen and methane rather than oxygen and hydrogen.

The nose section of the HSRV is itself a multi-functional vehicle designed to provide operational flexibility. The nose section is essentially a separable spacecraft that serves multiple functions. For the cargo configuration, during the demonstration and infrastructure buildup missions of the *Prepare* phases, the nose section is used as a surface mobility system to relocate and/or precisely position payloads by lifting and flying them from one location to another. For the crewed configuration the nose section of the HSRV contains a 5.5 mt capsule (also separable) that serves as the cockpit for both EDL and ascent. When re-purposed on Mars surface and resupplied with Mars propellants, the nose section is also used as a crew rescue vehicle that hops from the base to recover the crew from an abort-to-surface within the EZ, or as an exploration hopper vehicle within the EZ able to access locales that are inaccessible with surface roving vehicles.

As noted above, the modular construction of the HSRV allows the nose section and crew capsule to be extracted to safety at all points during Mars entry or ascent and recovered by the HCRV systems re-purposed from the separable nose design. For abort-to-surface scenarios the crew capsule utilizes a combination of recovery system technologies including a hypersonic inflatable aerodynamic decelerator (HIAD), a supersonic parachute, solid propellant retro-rockets, and airbags or crushable materials to land safely. No other Mars launch or landing architecture known to the authors has this capability.

Payload Section

The payload section is a cylindrical composite structure that contains up to 20 mt of cargo. The section is 4 meters tall with a 5.9 meter inner diameter. Door clearance is 3.75 x 5.25 meters.

In the initial demonstration and infrastructure buildup phase (*Prepare*), the payload section can be separated and precisely positioned when using nose section capabilities. This is particularly useful for re-locating and positioning a nuclear surface power system. This also enables ease-of-offload for much of the initial infrastructure including the delivery of key payload offloading and ground mobility systems required for the reusable Hercules used later, as illustrated in Figure 5. Options for this offloading system include delivery of a mobile lift vehicle with

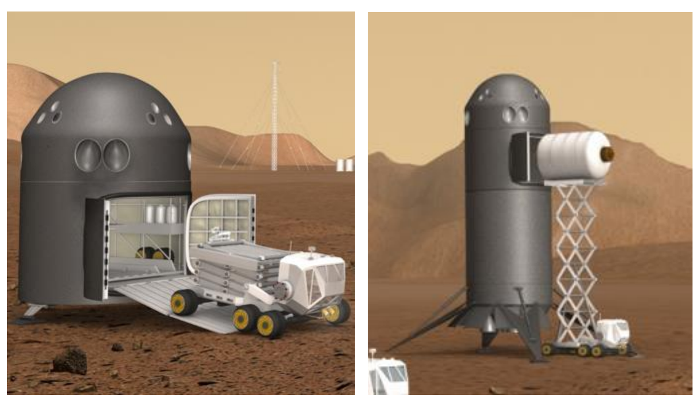

Figure 5. HSRV payload operations illustrating "expendable" scenario where payload bay is separated using nose section, and reusable scenario where mobility equipment is available for cargo offload.

a large scissors jack for lowering payloads to the surface; or a payload section mounted crane/hoist to lower payloads or crew.

Additional options under consideration for the infrastructure buildup phase include either converting the expended payload sections to habitable volumes or outfitting them to serve as surface habitats.

Ascent/Descent Section

The ascent/descent section includes the ascent propellant tank and feed system; a descent propellant tank and feed system; the ADS rocket engines; an aft engine bay with thrust structure; the body flap and actuation system; and the landing legs.

The ascent tank is a common bulkhead (CBH) design integral with the HSRV OML, designed to carry not just internal pressure but external compression and bending loads imposed on the HSRV during ground and flight operations. From Earth the ascent tank is launched empty, thus Earth launch loads do not drive the ascent tank design. The composite tank is 7.25 meters tall with a 5.9 meter inner diameter storing LCH_4 in the forward tank and LO_2 in the aft tank, both at 30 psia. Propellants are loaded and stored at normal boiling point conditions. Since the ascent tank is delivered to Mars empty a layered composite insulation (LCI) is used to provide resistance to heating during the storage of propellants on Mars surface. LCI is preferred for soft vacuum applications such as on Mars surface where atmospheric pressure is between 4.5-6 torr[11]. In addition, a system of broad area cooling tubes are installed between the tank outer wall and the LCI but are not used in flight. Rather, this system is connected to a ground system that provides the cryocooling needed for long-duration storage on Mars surface, thus minimizing the mass impact on the HSRV for long-duration storage hardware (cryocoolers, power system, and radiator system).

The descent tanks store propellant for the Mars atmospheric descent phase. Choice to use dedicated tanks for descent, as opposed to using the large ascent tank to store both ascent and descent propellants, is to reduce the risk of failed engine start during the critical supersonic retro-propulsion (SRP) engine ignition event by providing a smaller set of tanks that are full. In contrast, a single tank system designed for ascent and descent would be approximately 5-10% filled at SRP initiation. Given the dynamics of the vehicle at that point in flight (i.e. – the vehicle is re-orienting and experiencing external acceleration loads due to atmospheric drag) and the time-criticality of the SRP event, the risk that the propellants would not be properly “settled” over the tank outlet, ensuring a solid slug of fluid is available to the engines for start, is rather high. Thus, the choice to use a separate ascent and descent tanksets was baselined in the design.

The descent tank system includes four spherical tanks, two each for the LCH_4 and LO_2. These composite tanks are 1.8 meters diameter and store propellants at 30 psia. Like the ascent tanks, propellants are stored at normal boiling point. Since the descent tanks remain full during orbital flight phases, multi-layer insulation (MLI) blanket are used. Like the ascent tank design, broad area cooling tubes are mounted on the tank beneath the MLI, and a separate cryocooling system can be connected to the broad area cooling system to provide long-duration storage capability. For the interplanetary transfer phases, for example, a cryocooling system bookkept as part of the payload provides the systems necessary to ensure descent tank propellants are properly conditioned. On Mars surface a ground system provides the cryocooling functionality.

The ascent and descent tanks, along with the ATLS tanks, are interconnected to provide an additional degree of operational flexibility. Propellants can be transferred from tank to tank to provide some center of gravity control, but also to allow propellant scavenging, circulation, thermal conditioning, and to move propellants for specific maneuvers.

The ADS includes five LO_2/LCH_4 pump-fed (gas generator cycle) rocket engines, each delivering ~55 klbf (~245 kN) at a minimum specific impulse of 360 seconds. These engines are sized for 2.5 Earth g’s max during SRP assuming 70% throttle. Sizing to this criteria ensures adequate propellant and thrust reserves for precision landing. Table 3 highlights the ADS engine design parameters.

The descent tanks and engines are mounted in the aft bay. The aft bay, made of composite materials, support the descent tanks and include the ADS engine thrust structure. Thrust loads are transferred from the engine thrust chamber mount through this structure to the OML of the aft bay.

The body flap aerosurfaces are mounted on the external surface of the aft bay. These flaps are wrapped nearly 180 degrees around the windward side of the vehicle and are the primary system for trim and down-range control during entry. The flaps are installed on the aft portion of the descent section in order to maximize the moment arm of the flaps, but also to protect the engines from entry heating similar to the Space Shuttle orbiter. The electromechanical actuation system, powered by the ICE, provides the flap deflection control.

Finally, four deployable/retractable landing legs are used at landing. The legs are mounted on the exterior of the OML. When retracted, a stabilizer mounted on the landing leg strut provides aerodynamic stability during atmospheric flight. An electromechanical actuation system, powered by the ICE, provides the means to deploy and retract the landing legs.

In the initial demonstration and infrastructure buildup phase (*Prepare*), the ascent/descent section is re-positioned from the landing zone (using surface mobility systems) and re-purposed as a long-duration propellant storage facility as part of the in-situ propellant production infrastructure. This is illustrated in Figure 6. Alternative ideas for re-purposing include replacing the ascent tank system with a habitat "shell" for use on the surface. Additional subsystems and logistics delivered separately could be assembled with the habitat to form a fully-functional surface habitat.

In the later campaign phases (*Found*, *Expand*, *and Sustain*) when the HSRV is fully reusable, the ascent/descent section is loaded with propellant manufactured at the base from Mars resources just prior to flight. This "load-and-go" resupply strategy places the burden for long-duration thermal management of cryogenic propellants on the ground system infrastructure.

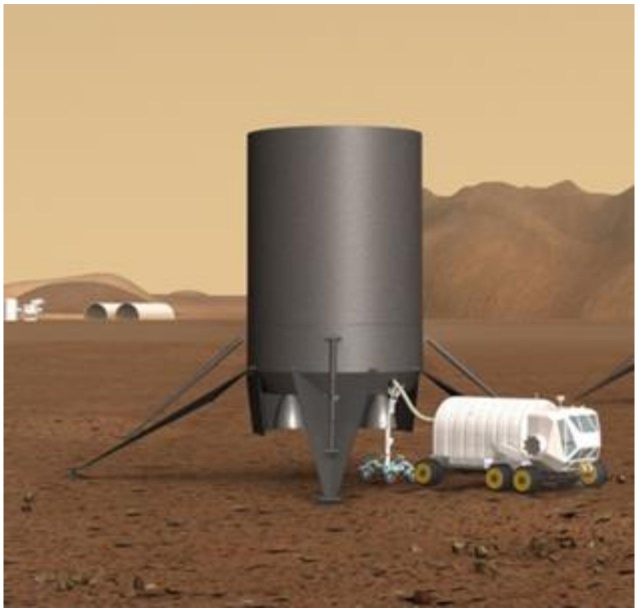

Figure 6. HSRV ascent/descent section repurposed on Mars as a propellant storage facility in support of the in-situ propellant production infrastructure.

B. HSRV – Lunar Cargo and Crew Taxi

The HSRV configurations supporting lunar operations utilize the DSG as the primary aggregation node. Once an HSRV is delivered to the DSG using the SLS, the HSRV is then resupplied with payload and propellants from Earth using a combination of SLS and commercial flights. (Note: This assumes that the capability to store and distribute LO_2 and LCH_4 propellant is added to the DSG as an evolution of the system as currently envisioned.) As illustrated in Figure 7, once readied for the lunar landing mission, the HSRV undocks and departs the DSG, transferring first to low-lunar orbit (LLO), then deorbits and descends to the landing site. If the HSRV remains on the surface longer

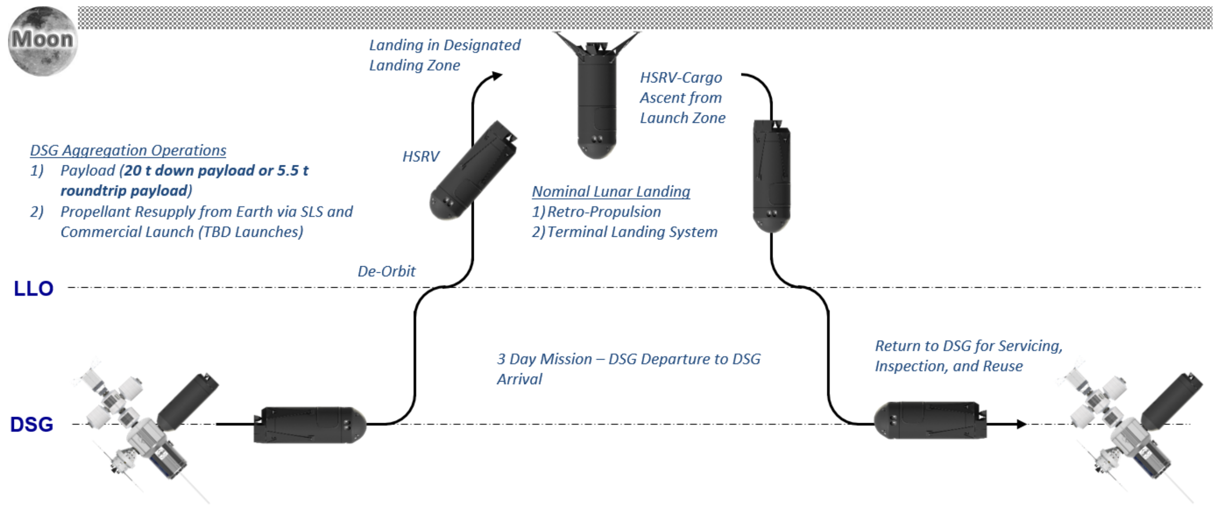

Figure 7. HSRV – Lunar Concept of Operations.

than a day or two, the HSRV requires power from the lunar base. Cargo is offloaded and/or crew is transferred prior to HSRV ascent. Ascent from the lunar base first targets LLO, then from LLO the HSRV transfers back to the DSG, performing rendezvous and docking. While aggregating at the DSG the HSRV is inspected, serviced, and prepared for reuse, either autonomously or with the aid of crew based at the DSG.

Relative to the HSRV – Mars configurations, the HSRV – Lunar design requires just one of the five ADS rocket engines and associated thrust structure. Since the recurring HSRV – Lunar mission operates solely in the vacuum environment between the DSG and the lunar surface, the ACC6 TPS system is not required, nor is the body flap or actuation system. MLI is needed to resist heat leaks into the propellant tanks, including the ascent tank system, during the mission. While at the DSG the HSRV is resupplied with propellants over a long period, thus a long duration storage solution is required. Thus, the HSRV design for the moon, like the Mars configuration, requires broad area cooling tubes installed between the MLI and the tank structure to intercept heat leaks while at the DSG. This assumes the DSG provides power and heat rejection capabilities along with a cryocooling system that can interface with the HSRV tank system.

For the crewed configuration, the capsule is replaced with a hopper habitat planned for use on Mars in the HCRV. This habitat is oriented such that the crew is standing during launch and landing and offers views of the surface to allow crew to fly the vehicle manually.

C. HPDV – Interplanetary Cargo

The primary function of the HPDV is interplanetary payload delivery from the DSG to LMO using a minimum-energy transfer, with Mars arrival V_∞ of 3.8 km/s (equivalent to inertial entry velocity at Mars atmospheric interface V_{entry} of 6.2 km/s). The HPDV configuration design, discussed below, offers large payload volume and increased delivery capability relative to the HSRV. Specifically, up to 60 mt of cargo (either a monolithic payload for the LMO node or three 20 mt pallets destined for Mars surface) are packaged in an extended payload bay that replaces the ascent tank.

Alternatively, the HPDV potentially offers the following functional options for the campaign:

- Utilize the HPDV to deliver 20 mt payloads to Mars surface that do not fit within the HSRV payload volume. For this option the HPDV, unlike the HSRV, cannot return to LMO from Mars surface, thus the HPDV sections would be re-purposed following landing.
- Utilize the HPDV to return nearly 10 mt of cargo to Earth if resupplied at the LMO node with Mars propellant.
- Utilize the HPDV to demonstrate Earth-to-Mars fast-transits with aerocapture at Mars. For this option up to 10 mt of cargo is delivered from the DSG to LMO node with Mars arrival V_∞ of 6.9 km/s (equivalent to inertial entry velocity at Mars atmospheric interface V_{entry} of 8.5 km/s).

Key configuration differences relative to the HSRV include the extended payload bay (replacing the ascent tank), and the descent tanks are stretched to provide the requisite performance for a minimum energy trans-Mars insertion (TMI) with the 60 mt payload.

One ADS engine is required for all functional operations except Mars landing. If the HPDV is expected to land large volume payloads, five ADS engines are installed and used for SRP.

Power and thermal control services are packaged in a backpack-like fairing located outside of the OML, covered by TPS (see Figure 2). The fairing doors open following DSG departure and close prior to Mars arrival. This system provides power generation (via deployable/retractable solar arrays) and heat acquisition and rejection (via deployable/retractable radiators) during the 180-300 day minimum-energy interplanetary coast, reducing the burden on the ICE system.

Since the heating environment for aerocapture exceeds the capability of the ACC6 TPS, the hot structure heat shield is replaced with a phenolic impregnated carbon ablator (PICA) designed for fast-transit aerocapture arrival. This ablative TPS system presently offers a single use, so for early flights the HPDV is re-purposed as part of the either the LMO node or on the surface. However, developing a reusable TPS system that can withstand the heating environment for the fast-transit aerocapture arrival is highly desirable. While at the node, the PICA heat shield is autonomously inspected and evaluated for multi-use capability.

Re-purposing options include:

- Utilize the nose section as the HCRV based at the LMO node, retrieving stranded crew in abort-to-orbit scenarios.

- Utilize the complete HPDV at the LMO node to provide power and thermal services, storage tanks for Mars propellants delivered by the HSRV, and volume for logistics storage.
- Utilize the HPDV to conduct Mars entry flight demonstrations.

D. HMTV – Interplanetary Crew Taxi

The HMTV is designed for fast-transit interplanetary crew transfer from the DSG to LMO using aerocapture at Mars arrival.

Key design differences relative to the HSRV include:

- Replacing the nose and payload sections with a transit habitat that supports a crew of four for the 90-120 day fast-transit transfer (see Figure 2).
- Adding the power and thermal control services backpack, similar to the HPDV, to provide power generation and heat rejection during the interplanetary coast.
- Replacing the ACC6 hot structure with PICA system designed for fast-transit aerocapture arrival.

Risk and safety for crew aerocapture is matured throughout the campaign, with several aerocapture flight demonstrations, at both minimum energy and fast transit arrival velocities, occurring prior to the first crewed flight to Mars.

IV. HSRV Nominal and Abort Performance and Sizing

The flight performance and sizing for the HSRV Mars and lunar configurations for both the nominal mission and for the various abort scenarios is presented in this section. Subsections discussing nominal flight performance highlight the reference trajectory assumptions and present results illustrating the variation in design ΔV to key vehicle sizing parameters. Subsections on sizing present the resulting dry and propellant masses of the as-sized vehicle, but detailed assumptions are limited to those key to understanding the HSRV abort system capabilities. Finally, subsections on abort capabilities briefly highlight the HSRV's abort capabilities for the complete range of ascent and EDL flight operations. Flight performance for each scenario and capsule design details are not discussed herein, however. Instead, these details are planned for a future paper focusing on the HSRV's unique abort system capabilities.

A. HSRV – Mars

This section discusses the nominal and abort performance and sizing of the HSRV supporting the Mars cargo and crew configurations.

<u>*Nominal Ascent Performance*</u>

A sensitivity study to assess the ascent performance of the HSRV from the Mars surface site is performed using a reference trajectory model in the Program to Optimize Simulated Trajectories (POST2)[12]. Accounting for the actual elevation of the Deuteronilus Mensae site (3.7 km below MOLA) and ascending to a 100 km by 250 km insertion orbit inclined 43.9 degrees relative to Mars equator, the variation in ascent ΔV as the initial vehicle thrust-to-mass varies shows the optimal ΔV exists around 0.75 Earth g's. Since the engine thrust is determined based on the entry trajectory and the ascent launch mass from Mars surface varies depending on whether it is a crew or cargo launch, the curve illustrated in Figure 8 is used in the sizing process to ensure sufficient propellant is available for either case.

Additional maneuvers are required during ascent to rendezvous with the LMO node. A 92.5 m/s burn transfers the HSRV from the insertion orbit to an intermediate 250 km by 500 km orbit, then a 55.4 m/s burn circularizes the vehicle at 500 km. An additional 25 m/s is reserved for phasing, rendezvous, proximity operations, and docking.

<u>*Nominal EDL Performance*</u>

In order to understand the sizing influence of ballistic coefficient on the key ΔV's for EDL, a POST2 reference trajectory is used. In this reference trajectory the de-orbit ΔV is determined based on targeting a 2.5 Earth g's maximum deceleration during entry. Three bank maneuvers are modeled, followed by a heading alignment phase that targets the landing site. Transition to SRP-only mode (using the ADS engines) assumes a 5 second delay where the HSRV re-orients from the 55 degree entry angle-of-attack to 180 degrees (i.e. – engine thrust aligned with the velocity vector). During this transition and continuing until terminal landing, a vacuum condition is assumed, thus no deceleration due to drag is accounted for. The ADS engines are sized during this phase, targeting a maximum deceleration of 2.5 Earth g's. The SRP-only mode decelerates the HSRV, targeting a point 100 meters above the landing site. At this point the horizontal velocity is nearly zero, but the vehicle is falling vertically. A two second transition from SRP-to-ATLS engines is assumed. Once the SRP engines are off, the ATLS decelerates the HSRV from about 50 m/s vertical velocity and lands at 2.5 m/s.

Figure 9 illustrates a trend where both SRP-only ΔV and de-orbit ΔV increase as ballistic coefficient increases. Based on these curves, the design allocation for SRP-only ΔV is 575 m/s, while the de-orbit ΔV allocated for sizing is 210 m/s. In addition, Table 5 and 6 includes the ΔV's allocated for other maneuver events used in the sizing.

HSRV – Mars Cargo and Crew Sizing

Sizing of the HSRV is performed using the Exploration Architecture Model for In-Space and Earth-to-Orbit (EXAMINE), a NASA-Langley developed framework used for conceptual level sizing[13].

Table 4 shows a breakdown of dry masses for the as-sized HSRV cargo and crew configurations. The primary difference between the cargo and crew dry mass is that the crew does not require a 400 kg adapter required by the cargo configuration to support the 20 mt payload.

Tables 5 and 6 breakdown the mission events, highlighting the propellant mass usage over the mission profile for both cargo and crew configurations. The ΔV's shown in Tables 5 and 6 are based on the ascent and EDL reference trajectory performance discussed above.

Table 7 summarizes the vehicle state at ascent and entry conditions, highlighting the propellant inventory for the various propulsion subsystems for both the cargo and crew configurations. In addition to drawing attention to the propellant inventories for the ADS and ATLS, Table 7 shows a breakdown of the abort separation for both the crew ascent and entry states. Included in the abort separation mass is the predicted dry mass of the nose section, the cargo that is separated along with the nose, the unusable propellant in the ATLS tanks, and the amount of usable propellant in the ATLS tanks that is used to support the abort scenarios.

An important difference between the cargo and crew cases are that the cargo ascent propellant residuals are vented once the HSRV reaches the LMO node, but for the crew configuration the ascent residuals are scavenged and pumped to the ATLS tanks. This enables the maximum amount of propellant in the ATLS tanks for entry. These propellants are used nominally for terminal landing and payload positioning, as previously discussed. However, in a contingency these propellants are available for use in an abort situation, either abort-to-orbit (ATO) or abort-to-surface (ATS). During ascent, 1.3 km/s is available to support ascent ATO or ATS, while during entry over 1.0 km/s is available for entry ATO or ATS.

HSRV – Mars Crew Capsule Sizing

For trajectory design purposes allocated masses include 5 mt for the capsule, 0.5 mt for the crew, and 0.25 mt for samples (used only for ascent abort scenarios). Functional subsystems required for the capsule habitability include primary structure, ingress and egress hatches, a pressurized tunnel, ECLSS and crew provisions to support 4 crew for 3 days, crew seats, a TCS acquiring and rejecting crew and avionics waste heat, contingency batteries that are used only when the capsule separates from the nose section in abort situations, and the avionics and crew control systems. Recovery systems, packaged external to the capsule OML, include a 10 meter HIAD and deployment system, a 20 meter diameter supersonic parachute and deployment system, solid retro-propulsion rockets, and either an airbag or crushable material for absorbing the landing loads. In addition, TPS covering the deployed HIAD and capsule nose cap are required.

Contingency Ascent Abort Capabilities

Four ascent abort scenarios were examined: 1) ATS targeting a return to exploration zone (RTEZ) using the ATLS propulsion system only; 2) ATS targeting the RTEZ using the capsule aeroentry and landing systems; 3) ATS targeting a common downrange location approximately 500 km east of the launch site using the capsule aeroentry and landing capabilities; and 4) ATO using the ATLS propulsion capabilities only. Figure 10 illustrates the approximate trajectory times each of these abort scenarios are possible relative to the nominal ascent trajectory.

Contingency EDL Abort Capabilities

As shown in Figure 11, five entry abort scenarios were examined: 1) ATO using the ATLS propulsion capabilities only; 2) ATO minimal safe orbit, defined as 150 km circular, using the ATLS propulsion capabilities only; 3) ATS targeting the RTEZ using the capsule aeroentry and landing systems; 4) ATS targeting the RTEZ using the capsule parachute and landing systems only (i.e. – no HIAD deployment required); and 5) ATS targeting the RTEZ using only the nose section propulsive capabilities.

B. HSRV – Lunar

This section discusses the nominal and abort performance for the HSRV supporting the lunar cargo and crew configurations. The HSRV is delivered to trans-lunar insertion (TLI) by the SLS. Since the SLS payload delivery capability to TLI is approximately 45 mt, the HSRV ascent tanks are empty and the descent and ATLS tanks are partially loaded with enough propellant for lunar orbit insertion (LOI) and rendezvous with the DSG. Once at the DSG the operations concept supporting lunar missions follows that illustrated in Figure 7.

Nominal Performance Data

Performance requirements needed for determining the HSRV propellant requirements to support lunar missions are derived from various sources.

With the DSG is located in a NRHO, the LOI ΔV of 429 m/s includes a 178 m/s lunar flyby burn, a 251 m/s insertion burn, and assumes the total TLI-to-NRHO transfer time is 5 days. Orbit transfer between the DSG and LLO is 730 m/s assuming a 0.5 day transfer[14].

Terminal descent and landing (TDL) from the 100 km circular LLO begins with a de-orbit burn that targets a 15 km by 100 km initial descent transfer orbit. Following coast, the ADS engine restarts for the braking phase that steers the vehicle toward the landing site. The engines continue to operate through a pitch-up phase that reorients the vehicle for visibility, followed by an approach phase where the vehicle maintains a constant altitude. TDL ends with the terminal landing phase where the vehicle descends slowly to the landing site. During this final phase, additional performance is allocated to enable vehicle re-designation to avoid obstacles during landing[15]. To support sizing, a total ΔV allocated for TDL is 2,200 m/s, with the final 50 m/s allocated to the ATLS engines. A reference trajectory of the TDL, constructed in POST2, provides the basis for conducting descent abort studies.

Likewise, a reference trajectory of the lunar ascent is used to support ascent abort studies. Three key phases are used in the trajectory: launch, pitch over, and pitch control. The launch phase starts with liftoff and continues to rise vertically until 100 m. The vehicle then starts to pitch over and follows a gravity turn. The pitch control phase optimizes the pitch rates of the vehicle to optimally target insertion into lunar orbit[16]. For sizing purposes, a total ΔV allocated for ascent is 2,000 m/s that includes additional capabilities for phasing and rendezvous in LLO.

HSRV – Lunar Cargo and Crew Sizing

As shown in Table 8, mass savings for the lunar configurations relative to the Mars configurations is about 4.5 mt. This results from eliminating the TPS, four ADS engines, feed systems, and associated support/thrust structures, the body flap and actuation system, and growth allowance for these subsystems.

Tables 9 and 10 breakdown the mission events, highlighting the propellant mass usage over the mission profile for both cargo and crew configurations. The ΔV's shown in Tables 9 and 10 are based on the descent and ascent reference trajectory performance discussed above.

Table 11 summarizes the vehicle state at TLI (initial delivery of HSRV to the DSG), at the DSG prior to propellant resupply, at lunar descent, and at lunar ascent. This table highlights the propellant inventory for the various propulsion subsystems for both the cargo and crew configurations. In addition to drawing attention to the propellant inventories for the ADS and ATLS, Table 11 shows a breakdown of the abort separation for both the crew descent and ascent states. Included in the abort separation mass is the predicted dry mass of the nose section, the cargo that is separated along with the nose, the unusable propellant in the ATLS tanks, and the amount of usable propellant in the ATLS tanks that is used to support the abort scenarios.

Contingency Descent Abort Capabilities

As shown in Figure 12, two lunar descent abort scenarios were examined: 1) ATO using the ATLS propulsion capabilities only; 2) ATS targeting the RTEZ using the ATLS propulsion capabilities only.

For the ATO scenario, the nose section separates and accelerates, targeting the 100 km circular LLO. The DSG-based HCRV then departs the DSG to rendezvous with the crew in LLO and return them to the DSG.

For the ATS scenario, the nose section separates and accelerates to target the EZ with a soft, propulsive landing using the ATLS. The HCRV based at the lunar site hops to the landing site to retrieve the crew and return them to the lunar base.

Contingency Ascent Abort Capabilities

Figure 13 illustrates the two lunar ascent abort scenarios: 1) ATS targeting the RTEZ using the ATLS propulsion capabilities only; and 2) ATO using the ATLS propulsion capabilities only.

For the ATS scenario, the nose section separates and accelerates to target the EZ with a soft, propulsive landing using the ATLS. The HCRV based at the lunar site hops to the landing site to retrieve the crew and return them to the lunar base.

For the ATO scenario, the nose section separates and accelerates, targeting the 100 km circular LLO. The DSG-based HCRV then departs the DSG to rendezvous with the crew in LLO and return them to the DSG.

V. Conclusions

The Hercules Transportation System concept presented in this paper offers a high degree of functionality and operational flexibility in support of both lunar and Mars campaigns, providing a common, evolvable vehicle architecture that initially supports lunar missions and ultimately supports Mars missions with interplanetary and Mars planetary transportation capabilities. Extending the configuration design and philosophies of HSRV to the interplanetary transportation systems, Mars orbital node and to support early lunar campaign is good application of commonality, resulting in further risk reduction, affordability and sustainability. In addition, the HSRV crewed configurations offer a unique, unprecedented abort capability for both the moon and Mars. As part of an evolvable transportation architecture, this investment is key to enabling the safe, affordable, and sustainable expansion of human being's beyond Earth orbit and ultimately establishing a continuous human presence on Mars.

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[16]Whitley, R., and Martinez, R.., "Options for Staging Orbits in Cis-Lunar Space," Presented at the 2016 IEEE Aerospace Conference, Big Sky, MT, 2016.

Table 3. Engine Design Parameters.

	ADS Engines	*ATLS Engines*	*RCS Thrusters*
Number of Engines - Propellant	5 - O_2/CH_4	8 - O_2/CH_4	12 - O_2/CH_4
Sizing Condition	2.5 Earth g's Deceleration during SRP	6.0 Mars g's Separation during Abort	Entry Attitude Control
Feed Type	Pump-Fed (Gas Generator)	Pressure-Fed	Pressure-Fed
Installation	Vehicle Base	Nose Section with 30º Cant Angle	Nose Section 4 –X thrusters 2 +Y thrusters 2 –Y thrusters 2 +Z thrusters 2 –Z thrusters
Thrust per Engine	55,546 lbf (246.7 kN)	13,409 lbf (59.7 kN)	500 lbf (2.2 kN)
Chamber Pressure	2,000 psia (13.8 MPa)	400 psia (2.75 MPa)	400 psia (2.75 MPa)
Mixture Ratio	3.5	3.0	3.0
Area Ratio	100	75	200
Engine Thrust-to-Weight	100	100	25
Vacuum Specific Impulse	364.7 sec	353.9 sec	365.6 sec
Mars Surface Specific Impulse	363.4 sec	349.0 sec	352.5 sec
Design Specific Impulse	360.0 sec	302.3 sec	330.0 sec
Engine Length	6.74 ft (2.055 m)	6.57 ft (2.00 m)	1.92 ft (0.58 m)
Engine Exit Diameter	3.53 ft (1.076 m)	3.42 ft (1.04 m)	1.06 ft (0.32 m)

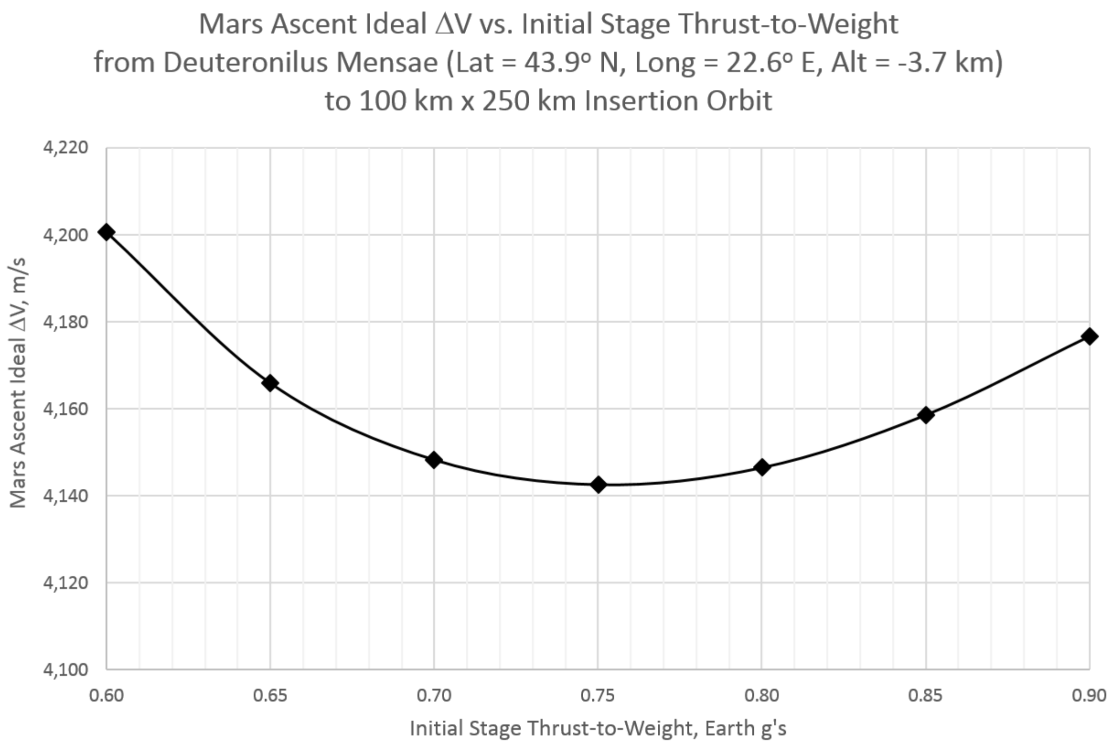

Figure 8. Mars Ascent Performance.

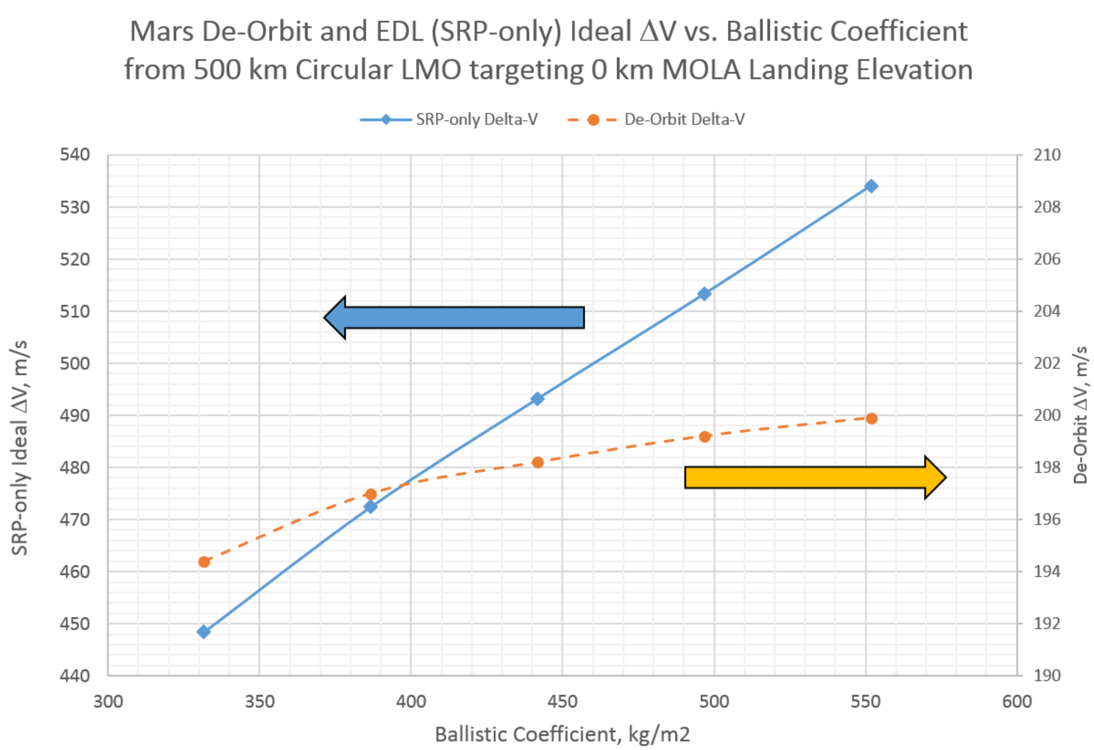

Figure 9. Mars EDL Performance.

Table 4. Dry Mass Summary for HSRV – Mars Configurations.

Mass Summary	***Cargo***	***Crew***
Structures	6,101	5,701
Nose Section	1,642	1,642
Payload Section	979	579
Ascent Section	1,438	1,438
Descent Section	1,319	1,319
Secondary Structure	461	461
Body Flap & Actuation	262	262
Thermal Protection	2,080	2,080
Landing Legs & Actuation	1,036	1,036
Ascent Propellant Tank	1,936	1,936
Descent Propellant Tanks	318	318
ADS Propellant Feed	474	474
ADS Engines	1,383	1,383
ATLS Propellant Tanks	725	725
ATLS Propellant Feed	543	543
ATLS Engines	511	511
RCS Thrusters	114	114
IVF Power Generation	209	209
Batteries	116	116
Power Management & Distribution	225	225
Avionics	300	300
Basic Dry Mass	**16,082**	**15,682**
Growth/Margin	**3,216**	**3,216**
Predicted Dry Mass	**19,298**	**18,898**

Table 5. Event Mass Tracking for HSRV – Mars Cargo

Event	Isp, sec	ΔV, m/s	Initial Mass, kg	Final Mass, kg	Propellant Mass, kg [1]	Payload Mass, kg
Ascent to 100 x 250 km ➢ *ADS + RCS*	360 + 330	4,181 + 10	161,152	47,573	113,579	0
Transfer to 250 x 500 km ➢ *ADS*	360	92.5	47,573	46,213	1,360	0
Transfer to 500 km Circ ➢ *ADS*	360	55.4	46,213	45,028	1,185	0
Orbit Maintenance & Dock ➢ *RCS*	330	25	45,028	44,223	805	0
Propellant Resupply to Node ➢ *ADS*	---	---	44,223	35,151	9,072	0
Orbit Maintenance & Undock ➢ *RCS*	330	25	55,151	54,727	424	20,000
Deorbit from 500 km Circ ➢ *RCS*	330	210	54,727	51,269	3,439	20,000
Entry Control Maneuvers ➢ *RCS*	330	30	51,269	50,796	473	20,000
Control for SRP Initiation ➢ *RCS*	330	30	50,796	50,308	469	20,000
Terminal Descent ➢ *ADS*	360	575	50,308	42,747	7,561	20,000
Terminal Transition ➢ *ADS + ATLS*	360 + 302.3	50 + 10	42,747	42,002	745	20,000
Terminal Landing ➢ *ATLS*	302.3	40	42,002	41,436	563	20,000

[1] Propellant mass includes power consumables and vented propellants.

Table 6. Event Mass Tracking for HSRV – Mars Crew

Event	Isp, sec	ΔV, m/s	Initial Mass, kg	Final Mass, kg	Propellant Mass, kg [1]	Payload Mass, kg
Ascent to 100 x 250 km ➢ *ADS + RCS*	360 + 330	4,186 + 10	162,819	48,013	114,806	5,750
Transfer to 250 x 500 km ➢ *ADS*	360	92.5	48,013	46,641	1,372	5,750
Transfer to 500 km Circ ➢ *ADS*	360	55.4	46,641	45,448	1,193	5,750
Orbit Maintenance & Dock ➢ *RCS*	330	25	45,448	44,390	1,058	5,750
Propellant Resupply to Node ➢ *ADS*	---	---	44,390	40,192	4,198	5,500
Orbit Maintenance & Undock ➢ *RCS*	330	25	40,192	39,882	309	5,500
Deorbit from 500 km Circ ➢ *RCS*	330	210	39,882	37,358	2,525	5,500
Entry Control Maneuvers ➢ *RCS*	330	30	37,358	37,013	345	5,500
Control for SRP Initiation ➢ *RCS*	330	30	37,013	36,652	361	5,500
Terminal Descent ➢ *ADS*	360	575	36,652	31,143	5,509	5,500
Terminal Transition ➢ *ADS + ATLS*	360 + 302.3	50 + 10	31,143	30,601	543	5,500
Terminal Landing ➢ *ATLS*	302.3	40	30,601	30,190	410	5,500

[1] Propellant mass includes power consumables and vented propellants.

Table 7. Propellant Inventory for Mars HSRV.

Propellant Mass Inventory, kg	HSRV-Cargo		HSRV-Crew	
	Ascent	Entry	Ascent	Entry
Predicted	**19,298**	**19,298**	**18,898**	**18,898**
Ascent Tanks	**121,714**	**0**	**121,714**	**3,651**
CH_4	27,048	0	27,048	811
Usable	24,855	0	24,855	0
Node Resupply	1,111	0	1,111	0
Reserves	270	0	270	0
Residuals	811	0	811	811
O_2	94,667	0	94,667	2,840
Usable	86,991	0	86,991	0
Node Resupply	3,889	0	3,889	0
Reserves	947	0	947	0
Residuals	2,840	0	2,840	2,840
Descent Tanks	**10,501**	**9,165**	**8,286**	**6,949**
CH_4	3,020	2,129	2,528	1,637
Usable	1,814	1,814	1,322	1,322
Reserves	60	60	60	60
Residuals	91	91	91	91
Tank Pressurant	139	139	139	139
Power Consumables	917	25	917	25
O_2	7,481	7,036	5,758	5,312
Usable	6,349	6,349	4,625	4,625
Reserves	150	150	15	150
Residuals	224	224	224	224
Tank Pressurant	300	300	300	300
Power Consumables	458	13	458	13
ATLS Tanks	**9,638**	**2,807**	**8,171**	**2,359**
CH_4	2,433	725	2,066	613
Usable	2,120	412	1,753	300
Reserves	49	49	49	49
Residuals	73	73	73	73
Pressurant	191	191	191	191
O_2	7,205	2,082	6,105	1,746
Usable	6,360	1,237	5,260	901
Reserves	144	144	144	144
Residuals	216	216	216	216
Pressurant	485	485	485	485
Propellant Mass	**141,854**	**11,971**	**138,171**	**12,959**
Stage Mass	**161,152**	**31,269**	**157,069**	**31,858**
Usable Propellant Mass	133,488	9,811	129,805	7,148
Inert Mass	27,663	21,458	27,263	24,709
Payload	**0**	**20,000**	**5,750**	**5,500**
Gross Mass	**161,152**	**51,269**	**162,819**	**37,358**
Abort Separation Mass	**n/a**	**n/a**	**19,930**	**17,520**
Nose Section Predicted			6,010	6,010
Separated Cargo			5,750	5,500
ATLS Unusable Propellant			965	965
ATLS Usable Propellant			7,206	5,045
Abort ΔV Available, m/s	**n/a**	**n/a**	**1,330**	**1,007**

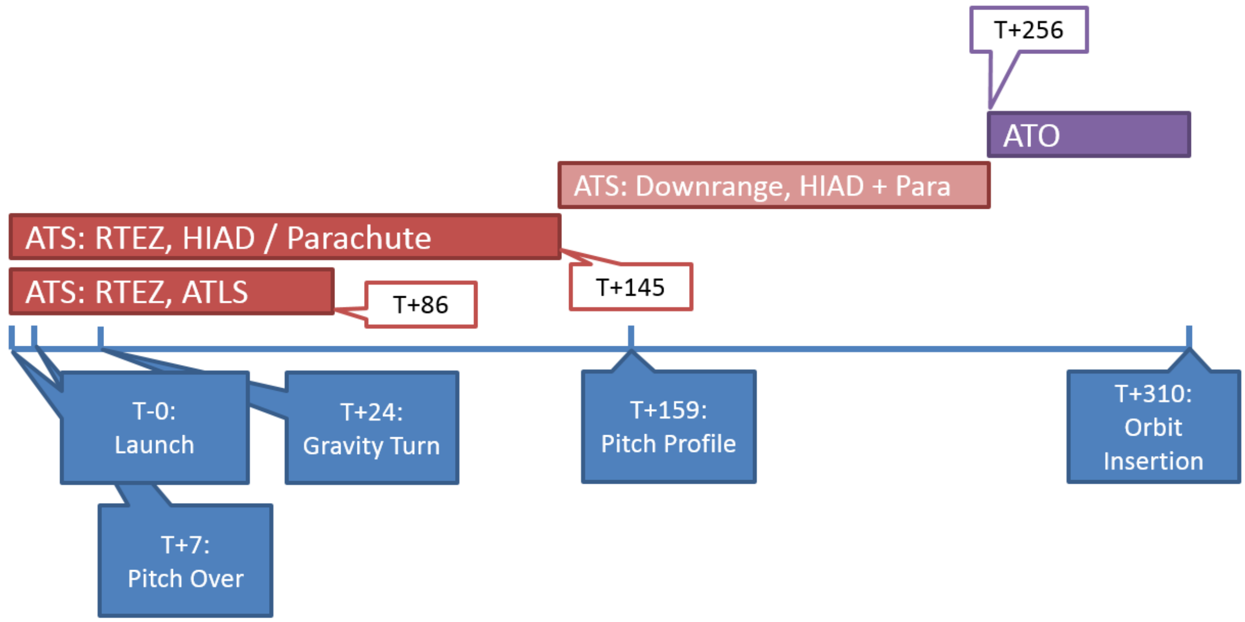

Figure 10. Mars Ascent Abort Capability

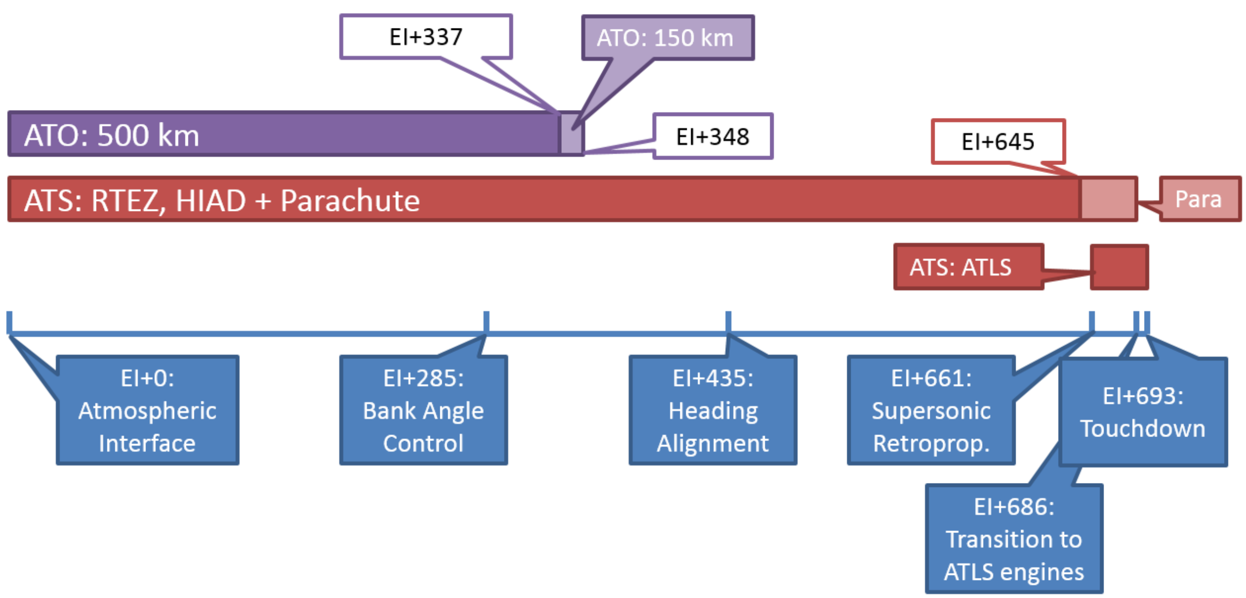

Figure 11. Mars EDL Abort Capability

Table 8. Dry Mass Summary for HSRV – Lunar Configurations.

Mass Summary	***Cargo***	***Crew***
Structures	5,488	5,088
Nose Section	1,642	1,642
Payload Section	979	579
Ascent Section	1,438	1,438
Descent Section	968	968
Secondary Structure	461	461
Body Flap & Actuation	0	0
Thermal Protection	0	0
Landing Legs & Actuation	1,036	1,036
Ascent Propellant Tank	1,936	1,936
Descent Propellant Tanks	318	318
ADS Propellant Feed	418	418
ADS Engines	277	277
ATLS Propellant Tanks	725	725
ATLS Propellant Feed	543	543
ATLS Engines	511	511
RCS Thrusters	114	114
IVF Power Generation	209	209
Batteries	116	116
Power Management & Distribution	225	225
Avionics	300	300
Basic Dry Mass	**12,227**	**11,827**
Growth/Margin	**2,445**	**2,445**
Predicted Dry Mass	**14,672**	**14,272**

Table 9. Event Mass Tracking for HSRV – Lunar Cargo

Event	Isp, sec	ΔV, m/s	Initial Mass, kg	Final Mass, kg	Propellant Mass, kg	Payload Mass, kg
Earth-to-DSG MCC ➢ *RCS*	330	30	45,000	44,585	415	20,000
DSG Insertion ➢ *ADS*	360	429	44,585	39,483	5,102	20,000
DSG RPOD ➢ *RCS*	330	30	39,483	39,119	364	20,000
HSRV Propellant Resupply @ DSG	---	---	39,119	165,522	-126,403	20,000
DSG-to-LLO Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	165,522	134,347	31,175	20,000
LLO-to-Lunar Surface TDL ➢ *ADS + RCS + ATLS*	360 + 330 + 302.5	2,150 + 5 + 50	134,347	50,617	63,731	20,000
Lunar Surface-to-LLO Ascent ➢ *ADS + RCS*	360 + 330	2,000 + 5	50,617	28,647	21,970	0
LLO-to-DSG Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	28,647	23,252	5,395	0
DSG RPOD ➢ *RCS*	330	30	23,252	23,037	215	0
HSRV Propellant Resupply @ DSG	---	---	23,037	165,522	-122,485	20,000
DSG-to-LLO Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	165,522	134,347	31,175	20,000
LLO-to-Lunar Surface TDL ➢ *ADS + RCS + ATLS*	360 + 330 + 302.5	2,150 + 5 + 50	134,347	50,617	63,731	20,000
Lunar Surface-to-LLO Ascent ➢ *ADS + RCS*	360 + 330	2,000 + 5	50,617	28,647	21,970	0
LLO-to-DSG Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	28,647	23,252	5,395	0
DSG RPOD ➢ *RCS*	330	30	23,252	23,037	215	0

Table 10. Event Mass Tracking for HSRV – Lunar Crew

Event	Isp, sec	ΔV, m/s	Initial Mass, kg	Final Mass, kg	Propellant Mass, kg	Payload Mass, kg
Earth-to-DSG MCC ➢ *RCS*	330	30	45,000	44,585	415	5,000
DSG Insertion ➢ *ADS*	360	429	44,585	39,483	5,102	5,000
DSG RPOD ➢ *RCS*	330	30	39,483	39,119	364	5,000
HSRV Propellant Resupply @ DSG	---	---	39,119	145,610	-105,991	5,000
DSG-to-LLO Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	145,610	118,186	27,424	5,500
LLO-to-Lunar Surface TDL ➢ *ADS + RCS + ATLS*	360 + 330 + 302.5	2,150 + 5 + 50	118,186	62,372	56,064	5,500
Lunar Surface-to-LLO Ascent ➢ *ADS + RCS*	360 + 330	2,000 + 5	62,372	35,300	27,072	5,750
LLO-to-DSG Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	35,300	28,651	6,648	5,750
DSG RPOD ➢ *RCS*	330	30	28,651	28,387	264	5,750
HSRV Propellant Resupply @ DSG	---	---	27,637	145,610	-117,473	5,000
DSG-to-LLO Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	145,610	118,186	27,424	5,500
LLO-to-Lunar Surface TDL ➢ *ADS + RCS + ATLS*	360 + 330 + 302.5	2,150 + 5 + 50	118,186	62,372	56,064	5,500
Lunar Surface-to-LLO Ascent ➢ *ADS + RCS*	360 + 330	2,000 + 5	62,372	35,300	27,072	5,750
LLO-to-DSG Transfer ➢ *ADS + RCS*	360 + 330	730 + 5	35,300	28,651	6,648	5,750
DSG RPOD ➢ *RCS*	330	30	28,651	18,223	264	5,750

Table 11. Propellant inventory for Lunar HSRV.

Propellant Mass Inventory, kg	HSRV-Cargo				HSRV-Crew			
	TLI	Node R/S	Descent	Ascent	TLI	Node R/S	Descent	Ascent
Predicted	**14,672**	**14,672**	**14,672**	**14,672**	**14,272**	**14,272**	**14,272**	**14,272**
Ascent Tanks	**0**	**110,711**	**79,791**	**18,515**	**5,589**	**105,699**	**78,275**	**22,211**
CH_4	0	24,602	17,731	4,114	1,242	23,253	17,159	4,700
Usable	0	23,520	16,650	3,033	160	22,171	16,077	3,618
Node Resupply	0	0	0	0	0	0	0	0
Reserves	0	270	270	270	270	270	270	270
Residuals	0	811	811	811	811	811	811	811
O_2	0	86,108	62,060	14,401	4,347	82,446	61,116	17,511
Usable	0	82,322	58,273	10,614	560	78,659	57,329	13,724
Node Resupply	0	0	0	0	0	0	0	0
Reserves	0	947	947	947	947	947	947	947
Residuals	0	2,840	2,840	2,840	2,840	2,840	2,840	2,840
Descent Tanks	**7,440**	**10,501**	**10,501**	**10,501**	**10,501**	**10,501**	**10,501**	**10,501**
CH_4	2,340	3,020	3,020	3,020	3,020	3,020	3,020	3,020
Usable	1,134	1,814	1,814	1,814	1,814	1,814	1,814	1,814
Reserves	60	60	60	60	60	60	60	60
Residuals	91	91	91	91	91	91	91	91
Tank Pressurant	139	139	139	139	139	139	139	139
Power Consumables	917	917	917	917	917	917	917	917
O_2	5,100	7,481	7,481	7,481	7,481	7,481	7,481	7,481
Usable	3,968	6,349	6,349	6,349	6,349	6,349	6,349	6,349
Reserves	150	150	150	150	150	150	150	150
Residuals	224	224	224	224	224	224	224	224
Tank Pressurant	300	300	300	300	300	300	300	300
Power Consumables	458	458	458	458	458	458	458	458
ATLS Tanks	**2,888**	**9,638**	**9,383**	**6,929**	**9,638**	**9,638**	**9,638**	**9,638**
CH_4	745	2,433	2,369	1,755	2,433	2,433	2,433	2,433
Usable	433	2,120	2,056	1,443	2,120	2,120	2,120	2,120
Reserves	49	49	49	49	49	49	49	49
Residuals	73	73	73	73	73	73	73	73
Pressurant	191	191	191	191	191	191	191	191
O_2	2,143	7,205	7,014	5,173	7,205	7,205	7,205	7,205
Usable	1,298	6,360	6,169	4,328	6,360	6,360	6,360	6,360
Reserves	144	144	144	144	144	144	144	144
Residuals	216	216	216	216	216	216	216	216
Pressurant	485	485	485	485	485	485	485	485
Propellant Mass	**10,328**	**130,522**	**99,675**	**35,945**	**25,728**	**125,838**	**98,414**	**42,350**
Stage Mass	**25,000**	**145,522**	**114,347**	**50,617**	**40,000**	**140,110**	**112,686**	**56,622**
Usable Propellant Mass	6,832	122,485	91,310	27,580	17,363	117,473	90,049	33,985
Inert Mass	18,168	23,037	23,037	23,037	22,637	22,637	22,637	22,637
Payload	**20,000**	**20,000**	**20,000**	**0**	**5,000**	**5,500**	**5,500**	**5,750**
Gross Mass	**45,000**	**165,522**	**134,347**	**50,617**	**45,000**	**145,610**	**118,186**	**62,372**
Abort Separation Mass	**n/a**	**n/a**	**n/a**	**n/a**	**n/a**	**n/a**	**20,553**	**20,803**
Nose Section Predicted							5,415	5,415
Separated Cargo							5,500	5,750
ATLS Unusable Propellant							965	965
ATLS Usable Propellant							8,673	8,673
Abort ΔV Available, m/s	**n/a**	**n/a**	**n/a**	**n/a**	**n/a**	**n/a**	**1,625**	**1,599**

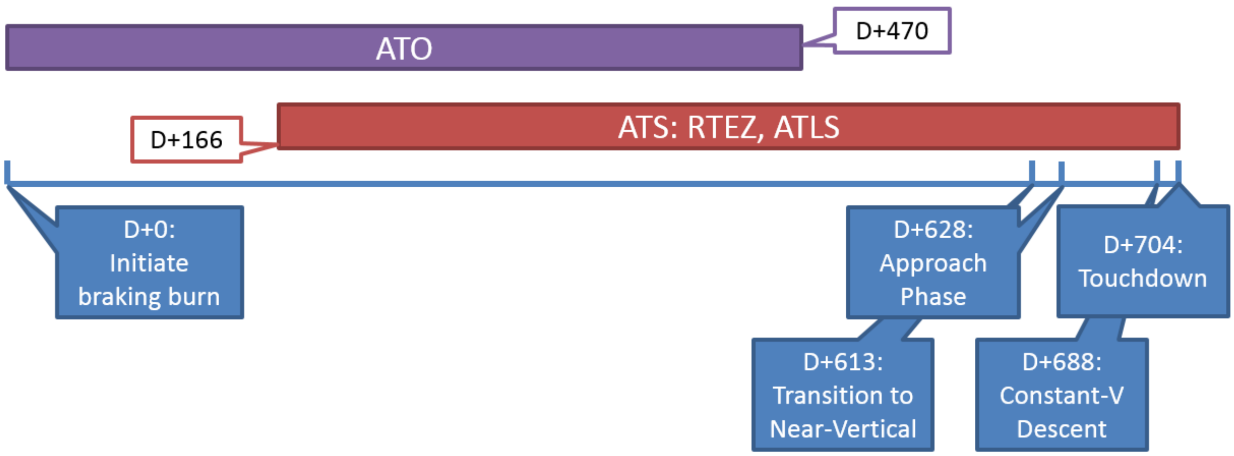

Figure 12. Lunar Descent Abort Capability

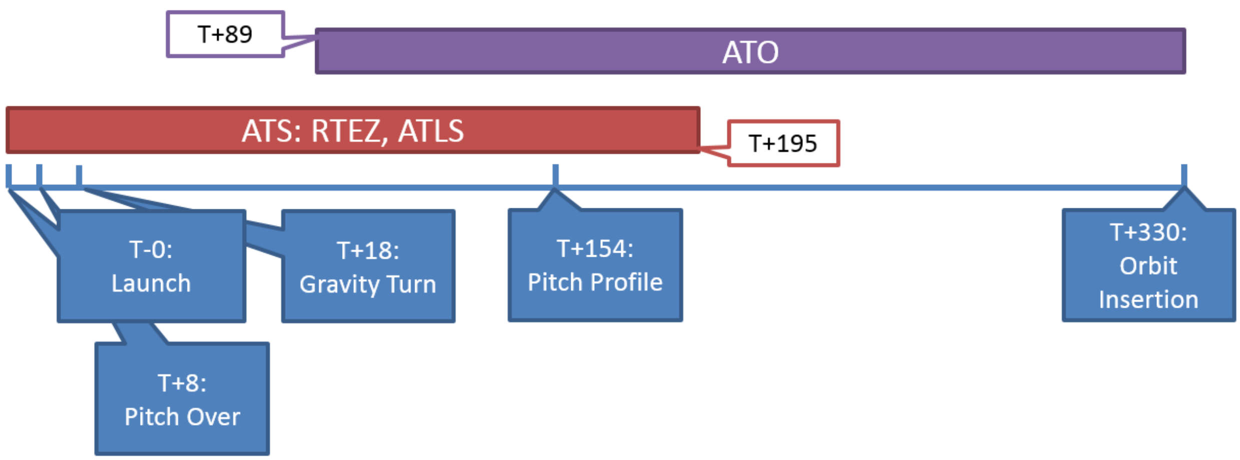

Figure 13. Lunar Ascent Abort Capability