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Lunar and Mars Ascent and Descent/Entry Crew Abort Modes for the Hercules Single-Stage Reusable Vehicle

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June 2018

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LIST OF FIGURES

ACRONYMS

ACC6	advanced carbon-carbon
ADS	ascent/descent System
AOA	angle-of-attack
ATLS	abort/terminal landing system
ATO	abort-to-orbit
ATS	abort-to-surface
CH4	methane
DRLZ	down range landing zone
DSG	Deep Space Gateway
EDL	entry, descent and landing
EI	entry interface
EVA	extra-vehicular activity
EXAMINE	Exploration Architecture Model for In-Space and Earth-to-Orbit
EZ	exploration zone
HCRV	Hercules crew rescue vehicle
HIAD	hypersonic inflatable aerodynamic decelerator
HSRV	Hercules single-stage reusable vehicle
ISRU	in-situ resourse utilization
LMO	low Mars orbit
LLO	low Lunar orbit
MER	Mars Exploration Rover
Mid-L/D	mid lift-to-drag ratio
MOLA	Mars Orbiter Laser Altimeter
MSL	Mars Science Laboratory
NASA	National Aeronautics and Space Administration
O2	oxygen
POST2	Program to Optimize Simulated Trajectories II
RTEZ	return to exploration zone
SRP	supersonic retro-propulsion
TPS	thermal protection system
T/W	thrust-to-weight ratio
km	kilometer
m/s	meters per second
^{o}	degrees
kN	kilo-Newton
ΔV	delta-velocity

1 Abstract

The Hercules single-stage reusable vehicle is designed to support crewed missions to both the lunar surface and Mars surface. The design maximizes crew safety by providing full coverage crew abort capability during ascent and entry/descent, either through abort-to-surface or abort-to-orbit. This paper outlines each of the abort modes and discuss the Hercules vehicle design along with the base and orbital infrastructure required to enable the full coverage abort capability. For each abort mode, trajectory simulations are flown that illustrate the requisite design capabilities and highlight the sensitivity to key design variables.

2 Introduction

In May 2014, the Pioneering Space document presented a new strategic approach for humans on Mars [1]. This approach emphasized building toward a permanent presence of humanity beyond the Earth surface. In response, NASA Langley Research Center technical leadership held a day-long blue sky meeting in July 2014 to kick off an architecture study to analyze the potential benefit of in-situ resource utilization (ISRU), automation and reusability to enable a sustained human presence on Mars surface. The results of this study, known as the "ISRU-to-the-Wall" study, are intended to inform NASA's human Mars exploration conversations [2].

A key outcome of this effort was the conceptual design of a single-stage reusable vehicle called Hercules. Designed to operate between low Mars orbit (LMO) and a Mars surface base utilizing oxygen and methane propellants manufactured in-situ from Martian resources, the Hercules single-stage, reusable vehicle (HSRV) concept supports this alternative human spaceflight strategy. The goal of this campaign strategy is to affordably establish a permanent and self-sustaining settlement on Mars in the next half century, as a prelude to colonization, with NASA playing a major role. This strategy postpones early human landings on Mars until key technologies and systems are demonstrated and matured, and a significant amount of infrastructure is established at Mars to safely support humans.

Since the 2014 study, the Hercules vehicle concept has evolved into an integrated transportation system that sets a new standard for operational flexibility and crew safety. Its modular and reconfigurable architecture allows a common system design to support planetary and interplanetary transport of cargo and crew between the Earth, the moon, and Mars [3].

One of the unique features of the HSRV concept is that it is designed to provide abort-to-surface (ATS) and abort-to-orbit (ATO) capability at the Moon and at Mars for both ascent and entry/descent, significantly improving crew safety for human missions.

Abort capabilities are a desired attribute for crewed transportation systems, including launch and landing vehicles. For current Earth launch systems, NASA's human rating requirements [4] require that the space system shall provide abort capability from the launch pad until Earth-orbit insertion to protect against a complete loss of ascent thrust/propulsion or loss of attitude or flight path control. For Earth entry systems, however, no requirement exists to separate and safely recover the crew from a failed entry vehicle during flight.

For past lunar mission strategies, including both the Apollo lunar excursion module [5] and the Altair lunar lander from NASA's Constellation program [6], an expendable two-stage vehicle was utilized that enabled the crew to abort-to-orbit during a failed landing attempt. This approach is consistent with present human rating requirements for lunar descent. However, no capability for abort during lunar ascent was employed for Apollo or Altair, nor is required per the current requirements.

For the many past human Mars mission architecture study efforts, none have considered safe crew abort and recovery options for Mars ascent or entry, descent and landing (EDL) [7][8].

As chronicled in Air & Space Magazine's The Mars Dilemma, Apollo 17 astronaut Jack Schmitt explained to the nation's experts on Mars EDL technology what a Mars landing would look like from an astronaut's point of view [9].

> "An unexpected issue surfaced when Jack commented, "To do this landing safely, of course, we need the ability for the astronauts at any time to hit the abort button, wave off the landing approach, and go back into outer space.""

In response to Jack's comments, the EDL design community leaned on its vast experience with robotic-class Mars landers.

> "With Mars landers, aborting raises a much bigger problem than landing. All of our Mars landers and rovers, as they approached the planet, shed an assemblage of intricate hardware and components, leaving a trail of debris on the way to the surface. We assumed that a human mission would do the same. How could it do all of those transformations and undressings, while still being able to turn around to fly back up into outer space in an emergency?"

Hercules answers the question by turning its assumptions on its head: don't undress for the normal landing – keep the transformations in reserve for emergency situations only.

The HSRV concept presented in this paper attempts to maximize the abort coverage for all phases of Mars ascent and entry, as well as lunar descent and ascent, exploring both ATO and ATS scenarios. The objective of this paper is to summarize the HSRV design, highlighting the crew abort and recovery capabilities enabled for both the moon and Mars.

3 HSRV Concept Overview

The Hercules concept is a multi-functional, modular, operationally flexible, single-stage, reusable vehicle initially designed to transport cargo and crew between LMO and the Mars surface base. Recent design efforts, as discussed in reference 3, evolved the capabilities to support cargo and crewed lunar landing missions. Like the crewed HSRV supporting Mars, the crewed HSRV supporting lunar landings offers crew abort and recovery capabilities.

This section describes the concept design for the HSRV supporting crew missions at both the moon and Mars. In addition, the modular systems derived from the base HSRV that support the crew abort and recovery operations are described. This includes the separable nose section for both the Mars and lunar configurations, the separable crew capsule supporting the Mars configuration, and the crew habitat supporting the lunar configuration. The crew habitat, used in conjunction with a separable nose section, provides a crew rescue vehicle function on the surface and in-orbit around both Mars and the moon. These systems, operating together and in concert with a campaign buildup strategy that emphasizes pre-emplacement of key surface and orbital infrastructure, attempt to maximize crew safety.

3.1 HSRV – Mars Crew Configuration

As illustrated in Figure 1 and Figure 2, the HSRV is configured for vertical takeoff, mid lift-to-drag (mid-L/D) nose entry and vertical landing, supported by deployable/retractable landing legs that extend at the vehicle's base. The HSRV is designed with three sections: the nose section, the payload section and the ascent/descent section. Each of these sections are separable and serve multiple functions in the architecture. For early demonstration and base infrastructure buildup flights the HSRV is "expendable", with the hardware sections re-purposed to operate as part of the base infrastructure. Once the Mars surface and orbital 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.

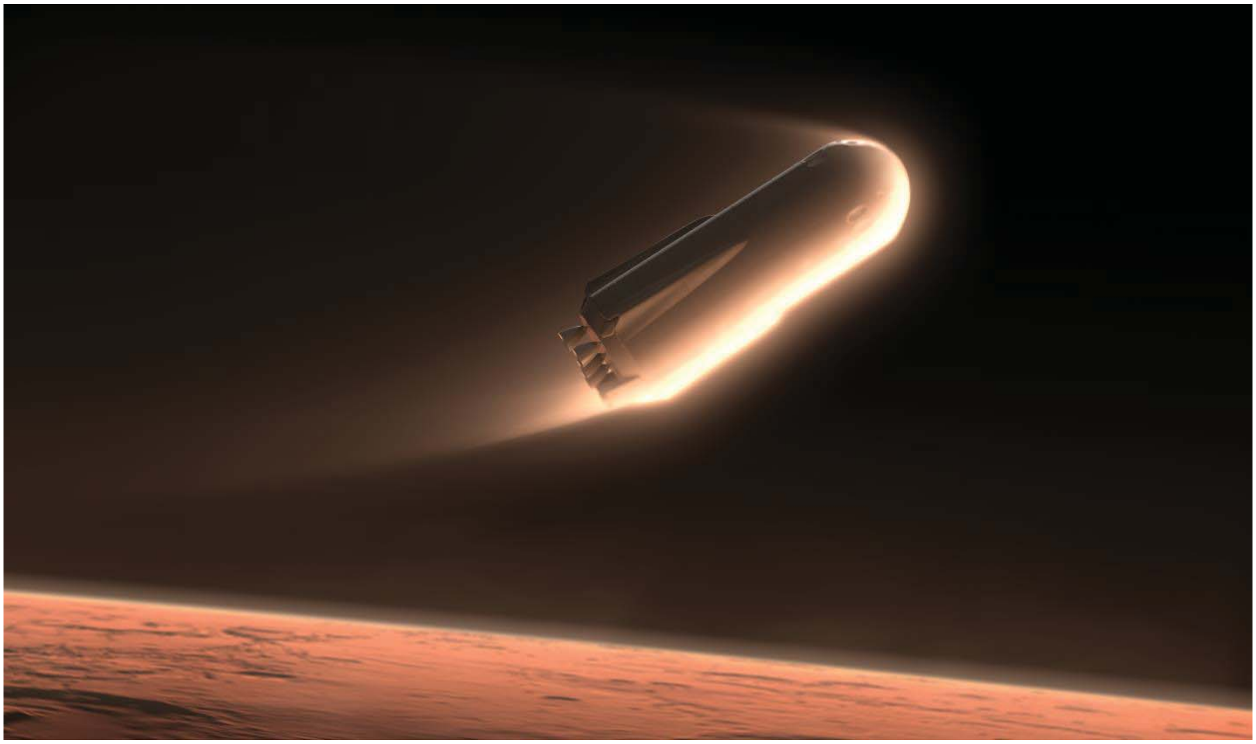
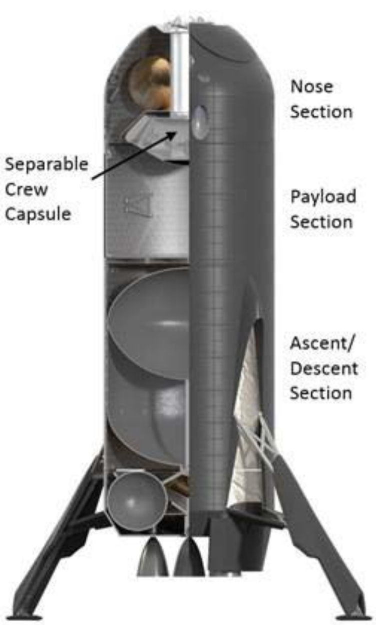


Figure 1. HSRV during Mars entry.

HSRV – Mars Crew Configuration



Design Features
- Primary Structure: composite; sections designed to separate
- TPS: mechanically-attached ACC6 hot structure with opacifed fibrous insulation
- Propellant Architecture: liquid oxygen and liquid methane (O_2/CH_4) common across complete system; interconnected propulsion systems

Nose Section
- Abort/Terminal Landing System (ATLS)
  - CH_4 Tanks: 2 x 2 m diameter at 500 psia
  - O_2 Tanks: 2 x 2 m diameter at 500 psia
  - Engines: 8 pressure-fed engines installed with 30° cant angle delivering ~60 kN each at min effective Isp 300 sec.
  - RCS Thrusters: 12 pressure-fed thrusters
- Power Generation: 2 internal combustion engines burning O_2/CH_4 delivering 3 kWe at idle, 40 kWe at max throttle
- Capsule Adapter: supports 5.5 t separable crew capsule
- Crew Support: standard docking system; pressurized tunnel to capsule

Payload Section
- Crew Capsule supporting 4 crew for 3 days
- Available Volume: 109 m^3
- Door Clearance: 4.5 m wide x 3.8 m tall

Ascent/Descent Section
- Ascent Tank: common-bulkhead storing O_2/CH_4 at 30 psia
- Descent Tanks: dedicated for terminal descent propellant
  - CH_4 Tanks: 2 x 2 m diameter at 30 psia
  - O_2 Tanks: 2 x 2 m diameter at 30 psia
- Ascent/Descent Engines: 5 pump-fed engines installed at vehicle base delivering ~245 kN each at min effective Isp of 360 sec
- Body Flap & Actuation: primary control for atmospheric flight
- Landing Legs: deployable/retractable

Subsystem	Ascent Mass (kg)	Entry Mass (kg)
Predicted Mass	**18,898**	**18,898**
Structures	5,701	5,701
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
Power & Avionics	850	850
Growth/Margin	3,216	3,216
Propellant Mass	**138,171**	**12,959**
Ascent Tank	121,714	0
Descent Tanks	8,286	6,949
ATLS Tanks	8,171	6,010
Payload Mass	**5,750**	**5,500**
Cargo	5,000	5,000
Crew & Suits	500	500
Samples	250	0
TOTALS	**162,819**	**37,358**

Figure 2. Summary of the HSRV configuration used for Mars crewed flights.

The all-composite structure of the HSRV is protected by a mechanically-attached advanced carbon-carbon (ACC6) hot structure thermal protection system (TPS) that is durable and reusable, designed to withstand the aerodynamic heating of an entry from LMO [10].

The HSRV utilizes only two propellants for all propulsion, power and pressurization needs: liquid oxygen (O2) and liquid methane (CH4). Propellants are stored in three separate tank storage systems: the abort/terminal landing system (ATLS) packaged in the vehicle's nose section; and the ascent and descent tank systems packaged in the ascent/descent section. These propellant systems feed two separate engine systems: the ATLS engines and RCS thrusters, packaged in the nose; and the ascent/descent system (ADS) engines, packaged on the base of the vehicle.

A key design feature of the HSRV is the unique positioning of the ATLS rocket engines in the nose section (four sets of two engines canted 30º outboard from the vertical). Nominally, the ATLS engines are used for terminal landing of the HSRV. 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. In addition, the nose section can be re-purposed to provide precise payload positioning on the surface (as illustrated in Figure 3), crew rescue recovery operations, and as an exploration hopper vehicle within the exploration zone (EZ) able to access locales that are inaccessible with roving vehicles. In a contingency, however, the ATLS engines serve as an abort system that separates the nose section during an unlikely catastrophic vehicle failure during ascent or entry.

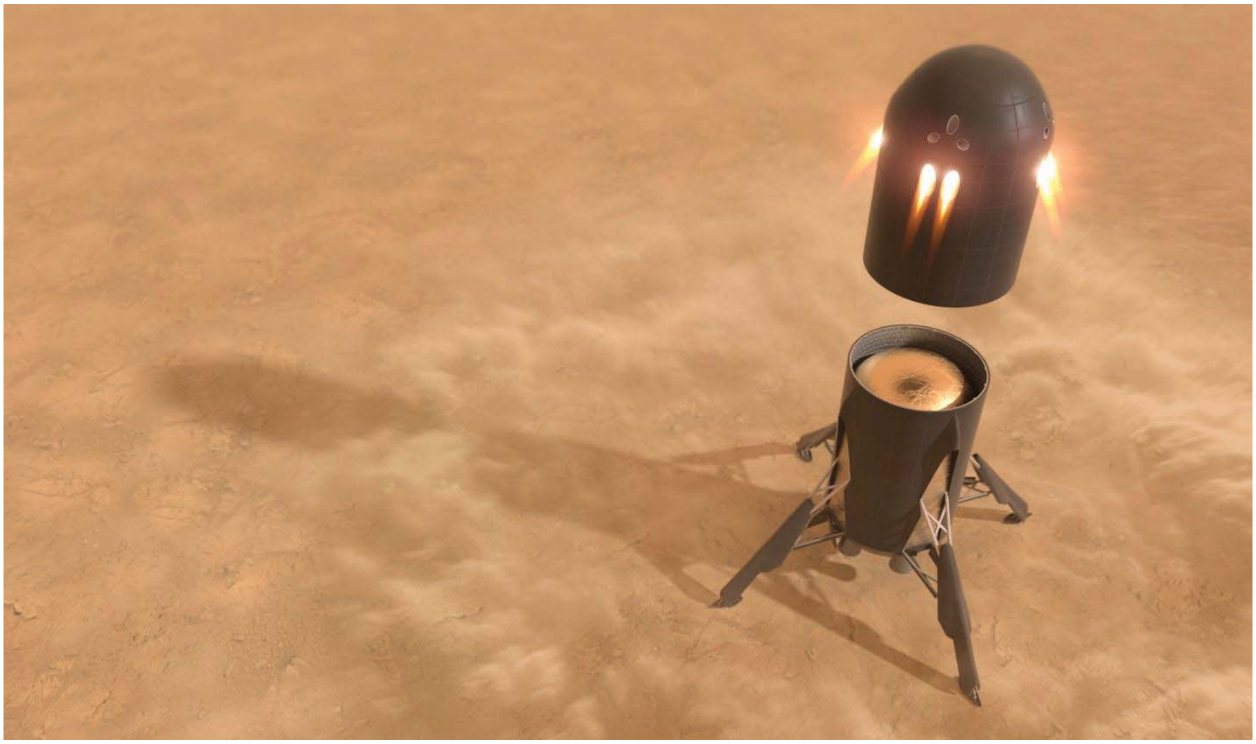


Figure 3. HSRV landing on Mars uses the ATLS for precise payload positioning.

The HSRV supporting crewed flights at Mars uses a separable crew capsule as it primary habitat for the nominal mission. As shown in Figure 4, the nominal recurring concept of operations has the HSRV resupply on Mars surface with in-situ propellants. A typical three day crew rotation mission starts with ascent to a 100 kilometer (km) by 250 km insertion orbit and a two-burn transfer to a 500 km circular LMO. Following an automated rendezvous and docking with a LMO node, on-orbit operations commence that includes crew rotation and propellant transfer to the vehicle stationed at the node (e.g. – a Hercules Mars Transit Vehicle). After undocking and phasing to prepare for entry, the HSRV with its new crew deorbits and performs an aeroassisted entry using the body flap as the primary aerodynamic control system. At about Mach 2.5 the HSRV reorients for its supersonic retro-propulsion (SRP) phase where the base-mounted ADS engines ignite to decelerate the vehicle to subsonic velocities. At about Mach 0.5 the ATLS engines ignite, the ADS engines shutdown, and the HSRV completes terminal landing using just four of the eight ATLS engines (enabling engine-out for each pair of the pressure-fed engines).

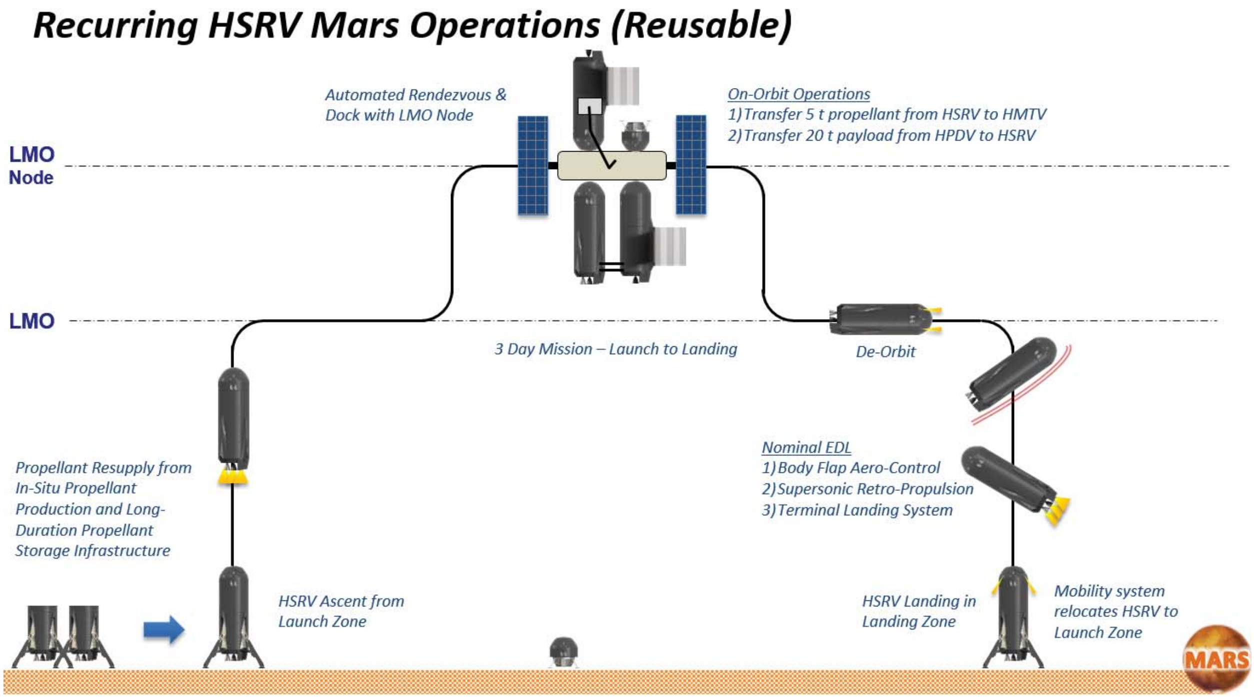


Figure 4. Concept of operations for the HSRV at Mars.

3.2 Mars Crew Abort Configuration

As noted in the discussion of the baseline HSRV, the unique positioning of the ATLS engines, the modular design of the HSRV, and the use of a separable nose section and separable crew capsule enable safe crew abort and recovery over the end-to-end flight profile for Mars. Depending on the flight regime, either an ATO or an ATS is possible for both Mars ascent and EDL. These systems, however, are only separated in support of an abort contingency.

When an abort event occurs, the nose section separates (along with the crew capsule) from the failed vehicle using the ATLS engines. The ATLS engines are sized for the Mars ascent abort to provide six Mars g's acceleration for separating the nose section and crew capsule. This results in each of the eight engines delivering ~60 kilo-Newton (kN) thrust.

The crew capsule for the HSRV supporting the Mars campaign provides the primary habitation for crew for the nominal Mars EDL and Mars ascent. The 3.5 meter diameter capsule offers 8.4 cubic meters of pressurized volume to support four crew for three days of flight between Mars surface and the LMO node.

Key capsule systems for nominal operations include: 1) an all composite integral pressure vessel structure with two hatches – one for nominal ingress/egress through the payload bay, and one on top for ingress/egress through the nose section tunnel and port that docks to the LMO node; 2) a jettisonable nose cap with ablative TPS; 3) avionics and cabling for command, control and data handling, guidance and navigation flight controls, and communications; 4) open-loop life support systems; and 5) crew recumbent seats and crew provisions. Nominally, power and thermal control are provided by the HSRV. Note the jettisonable nose cap is only jettisoned in support of Mars ATS scenarios.

Additional subsystems required to support abort and recovery include: 1) emergency lithium-ion batteries; 2) emergency heat rejection systems; 3) extra-vehicular activity (EVA) systems; and 4) capsule EDL systems required for ATS scenarios.

The capsule EDL systems include a hypersonic inflatable aerodynamic decelerator (HIAD), supersonic parachutes, solid propellant retro-rockets, and crushable energy absorbers.

The HIAD is 10 meters diameter and includes an inflation system and flexible TPS [11]. The HIAD is packaged around the perimeter of the capsule's conical backshell. Once inflated the HIAD decelerates the capsule down to supersonic flight condition conducive for supersonic parachute deployment. Once deployed and used, the HIAD is easily jettisoned prior to parachute deployment.

The supersonic parachute is a 20 meter diameter disk-band-gap design consistent with the design of the Mars Science Laboratory (MSL) parachute system [12]. Components include the canopy, suspension lines, and deployment mortar. These are packaged on the exterior of the capsule's conical backshell. Once deployed, the parachute decelerates the capsule to Mars terminal velocity, approximately 140 meters per second (m/s).

The baseline retro-rocket system includes sixteen STAR 8 solid rockets manufactured by Orbital ATK [13]. The STAR 8 rockets were used as the rocket assisted deceleration motor for the Mars Exploration Rover (MER) program. Each motor operates for ~4.5 seconds with a specific impulse around 274 seconds, delivering up to 1,740 pound-force thrust each to decelerate the capsule from Mars terminal velocity to ~2.5 m/s. These motors are packaged on the conical backshell of the capsule and fire through the rigid heatshield. Heaters are required to maintain the solid rockets within their design limits.

The crushable energy absorber is used for final landing of the aborted capsule, absorbing the landing energy with a kevlar honeycomb structure deployed from the capsule nose [14]. The crushable structure folds and is efficiently packaged in the rigid nose cap which jettisons prior to capsule landing.

Mars Crew Abort Configuration

Mars Abort	Ascent Mass (kg)	EDL Mass (kg)
Predicted Mass	**6,010**	**6,010**
Structures	1,569	1,569
Thermal Protection	495	495
Landing Legs & Actuation	200	200
ATLS Propellant Tanks	725	725
ATLS Propellant Feed	543	543
ATLS Engines	511	511
RCS Thrusters	114	114
Power & Avionics	850	850
Growth/Margin	1,002	1,002
Payload Mass	**5,750**	**5,500**
Cargo (Crew Capsule)	5,000	5,000
Crew & Suits	500	500
Samples	250	0
ATLS Propellant Mass	**8,171**	**6,010**
Usable CH_4	1,802	1,161
Residuals/Press CH_4	264	264
Usable O_2	5,404	3,885
Residuals/Press O_2	701	701
Abort Separation Mass	**19,930**	**17,520**
Abort ΔV Available (m/s)	**1,330**	**1,007**

Crew Capsule	Mass (kg)
Predicted Mass	**3,744**
Structures	1,080
HIAD + Thermal Protection	467
Supersonic Parachute	116
Retro-Rocket	103
Crushable Energy Absorber	61
Avionics	241
Power & Thermal Control Systems	759
Life Support Systems	230
Crew & EVA Systems	63
Growth/Margin	624
Crew & Provisions Mass	**1,100**
Crew + Suits	500
Crew Provisions	125
EVA Equipment & Consumables	400
ECLSS Consumables	75
Propellant Mass	**246**
Retro-Rocket Propellant	246
Capsule Gross Mass	**5,090**
Design Reserve Mass	**410**
Capsule Reference Mass	**5,500**

Figure 5. Separable HSRV nose section with separable crew capsule supporting Mars abort scenarios.

Figure 5 summarizes the crew abort configuration for Mars, including the masses of the nose section, crew capsule, and the total aborted mass at both ascent and entry. The crew capsule mass of 5,000 kg is included in the total mass breakdown for both the ascent and EDL conditions. Note that this includes 410 kg of unallocated reserve mass in addition to the 624 kg of mass growth allowance.

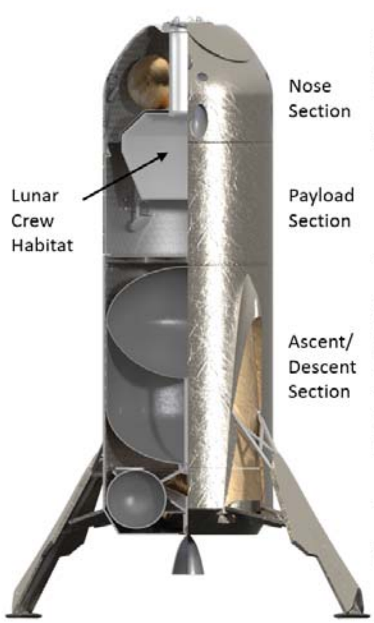
For ascent aborts, the total separation mass of 19,930 kg includes 7,206 kg of usable ATLS propellant, yielding 1,300 m/s of available ΔV for abort separation and manuevering, with more than 52 seconds of full thrust burn time. Likewise, for the EDL abort case, the total separation mass of 17,520 kg includes 5,045 kg of usable propellant for up to 1,007 m/s for safe abort, with more than 36 seconds of full thrust burn time.

3.3 HSRV – Lunar Crew Configuration

The HSRV supporting the lunar campaign is derived from the HSRV supporting the Mars campaign. This common design strategy allows a significant amount of the HSRV capability required for Mars to be demonstrated supporting the lunar campaign.

Figure 6 summarizes the HSRV configuration supporting the crewed lunar missions. Relative to the Mars configuration, significant differences include: 1) the ACC6 TPS, the body flap and flap actuation system are not required; 2) only one of the five ADS engines is required to support lunar landings; and 3) the crew capsule is replaced with a crew hopper habitat. In total, the dry mass of the lunar configuration is ~4,500 kg less than the Mars configuration. To date a conceptual design for the crew hopper habitat is not complete, thus for purposes of conducting lunar abort and recovery analysis 5,000 kg is assumed/allocated.

HSRV – Lunar Crew Configuration



Design Features
- Primary Structure: composite; sections designed to separate
- TPS: not required
- Propellant Architecture: liquid oxygen and liquid methane (O_2/CH_4) common across complete system; interconnected propulsion systems

Nose Section
- Abort/Terminal Landing System (ATLS)
  - CH_4 Tanks: 2 x 2 m diameter at 500 psia
  - O_2 Tanks: 2 x 2 m diameter at 500 psia
  - Engines: 8 pressure-fed engines installed with 30° cant angle delivering ~60 kN each at min effective Isp 300 sec.
  - RCS Thrusters: 12 pressure-fed thrusters
- Power Generation: 2 internal combustion engines burning O_2/CH_4 delivering 3 kWe at idle, 40 kWe at max throttle
- Habitat Adapter: supports 5.5 t crew hopper habitat
- Crew Support: standard docking system; pressurized tunnel to habitat

Payload Section
- Crew hopper habitat supporting 4 crew for 3 days
- Available Volume: 109 m^3
- Door Clearance: 4.5 m wide x 3.8 m tall

Ascent/Descent Section
- Ascent Tank: common-bulkhead storing O_2/CH_4 at 30 psia
- Descent Tanks: dedicated for terminal descent propellant
  - CH_4 Tanks: 2 x 2 m diameter at 30 psia
  - O_2 Tanks: 2 x 2 m diameter at 30 psia
- Ascent/Descent Engines: 1 pump-fed engines installed at vehicle base delivering ~245 kN at min effective Isp of 360 sec
- Body Flap & Actuation: primary control for atmospheric flight
- Landing Legs: deployable/retractable

Subsystem	Descent Mass (kg)	Ascent Mass (kg)
Predicted Mass	**14,272**	**14,272**
Structures	5,088	5,088
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
Power & Avionics	850	850
Growth/Margin	2,445	2,445
Propellant Mass	**125,838**	**42,350**
Ascent Tank	105,699	22,211
Descent Tanks	10,501	10,501
ATLS Tanks	9,638	9,638
Payload Mass	**5,500**	**5,750**
Cargo (Crew Habitat)	5,000	5,000
Crew & Suits	500	500
Samples	0	250
TOTALS	**145,610**	**62,372**

Figure 6. Summary of the HSRV configuration used for lunar crewed flights.

As shown in Figure 7, the HSRV configurations supporting lunar operations utilizes a Deep Space Gateway (DSG) in the lunar vicinity as the primary aggregation node in Earth's sphere of influence (as illustrated in Figure 8). Once an HSRV is delivered to the DSG, the HSRV is resupplied with payload and propellants from Earth using commercial flights. 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 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.

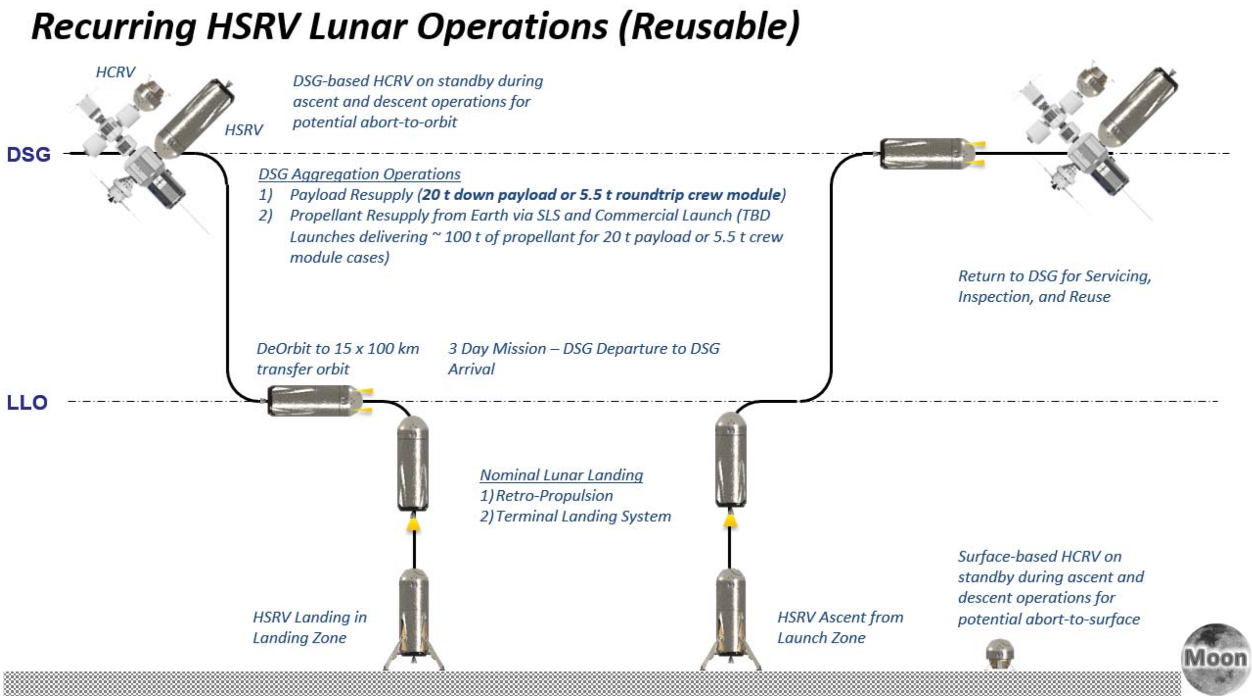


Figure 7. Concept of operations for the HSRV at the Moon.

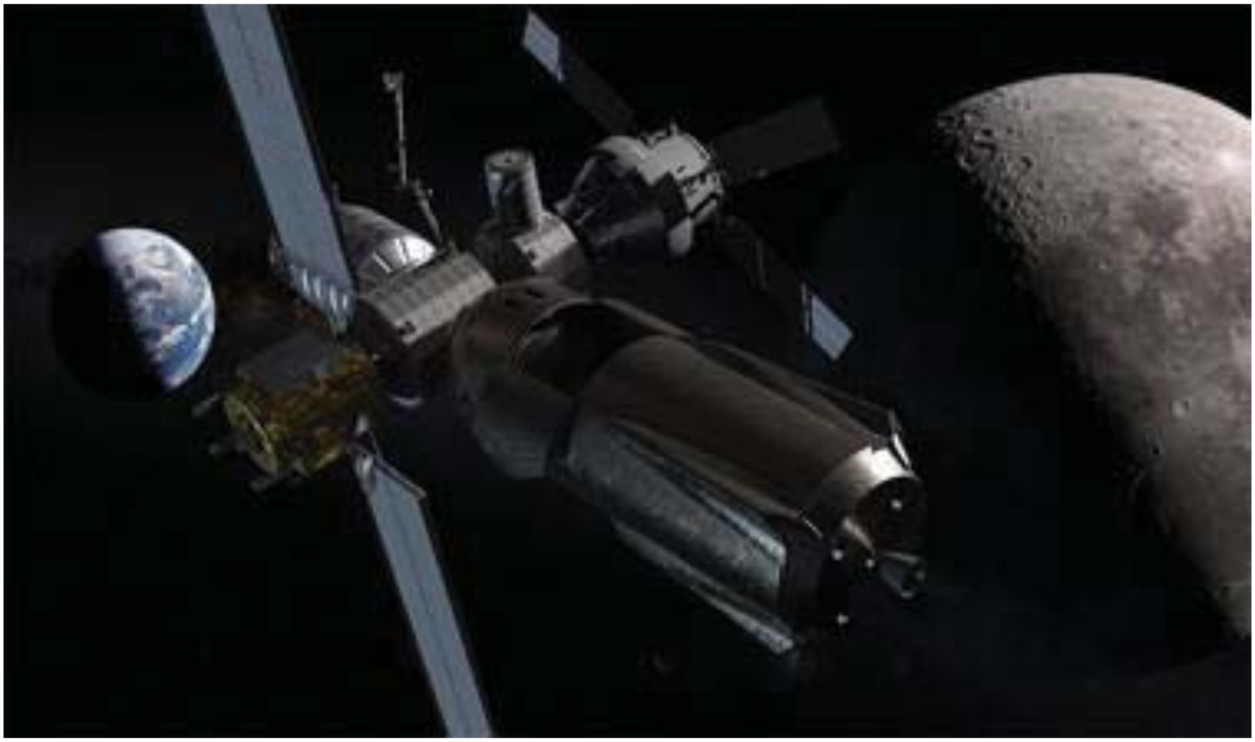


Figure 8. HSRV docked at the Deep Space Gateway.

Use of the DSG in this manner assumes the DSG functionality is evolved relative to the capability currently envisioned to enable: 1) additional systems to dock, including the HSRV, a Hercules Crew Recovery Vehicle (HCRV), and Earth resupply systems or vehicles; 2) O2 and CH4 propellant resupply from Earth supplied systems to HSRV and HCRV; 3) payload transfer from Earth supplied systems to the cargo HSRV; and 4) crew and logistics transfer from Earth supplied systems to crewed HSRV.

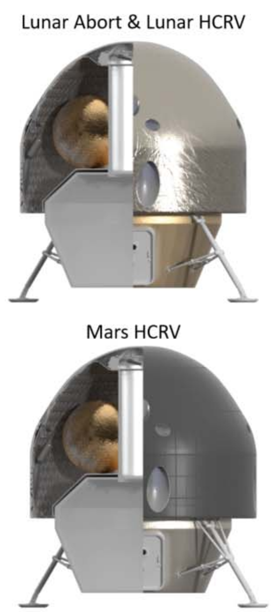
In addition, during one of the initial cargo missions to the lunar surface the HSRV re-purposes the nose section and crew hopper habitat to provide crew recovery. This surface-based HCRV supports ATS scenarios. This assumes the lunar base provides power and cryo-conditioning systems to maintain the HCRV on the surface.

3.4 Lunar Crew Abort and HCRV Configurations

Figure 9 summarizes the lunar crew abort configuration and the HCRV configurations for both the moon and Mars. The design goal is to enable crew abort for lunar operations. This includes: 1) ATO (during descent or ascent) where the lunar crew abort configuration re-enters LLO and the crew are recovered by a DSG-based HCRV; and 2) ATS (during descent or ascent) where the lunar crew abort configuration safely lands within the 50 km EZ around the lunar base and the crew are recovered by the surface HCRV.

Lunar Crew Abort & HCRV Configurations

Lunar Abort	Descent Mass (kg)	Ascent Mass (kg)
Nose Section Mass	**5,415**	**5,415**
Structures	1,569	1,569
Thermal Protection	0	0
Landing Legs & Actuation	200	200
ATLS Propellant Tanks	725	725
ATLS Propellant Feed	543	543
ATLS Engines	511	511
RCS Thrusters	114	114
Power & Avionics	850	850
Growth/Margin	903	903
Payload Mass	**5,500**	**5,750**
Cargo (Crew Habitat)	5,000	5,000
Crew & Suits	500	500
Samples	0	250
ATLS Propellant Mass	**9,638**	**7,479**
Usable CH_4	2,120	1,580
Reserve CH_4	49	49
Residuals/Press CH_4	264	264
Usable O_2	6,360	4,741
Reserve O_2	144	144
Residuals/Press O_2	701	701
Abort Separation Mass	**20,553**	**18,644**
Abort ΔV Available (m/s)	**1,625**	**1,274**



HCRV	Lunar Mass (kg)	Mars Mass (kg)
Nose Section Mass	**4,963**	**5,557**
Structures	1,192	1,192
Thermal Protection	0	495
Landing Legs & Actuation	200	200
ATLS Propellant Tanks	725	725
ATLS Propellant Feed	543	543
ATLS Engines	511	511
RCS Thrusters	114	114
Power & Avionics	850	850
Growth/Margin	827	926
Payload Mass	**5,500**	**5,500**
Hopper Habitat	5,000	5,000
Crew & Suits	500	500
ATLS Propellant Mass	**12,444**	**12,444**
Usable CH_4	2,648	2,648
Reserve CH_4	67	67
Residuals/Press CH_4	646	646
Usable O_2	7,875	7,875
Reserve O_2	182	182
Residuals/Press O_2	1,026	1,026
Crew Hopper Mass	**22,906**	**23,500**
Hopper ΔV Available (m/s)	**1,809**	**1,747**
Roundtrip Range (km)	**125**	**51**

Figure 9. Summary of HSRV nose section with crew habitat supporting lunar abort scenarios and crew rescue.

For descent aborts, the total separation mass of 20,553 kg includes 8,673 kg of usable plus reserve ATLS propellant, yielding 1,625 m/s of available ΔV for abort separation and manuevering, with over 63 seconds of full thrust burn time. Likewise, for the ascent abort case, the total separation mass of 18,644 kg includes 6,514 kg of usable plus reserve propellant for up to 1,274 m/s for a safe abort, with more than 47 seconds of full thrust burn time. Note, for the lunar crew abort configuration that the available ΔV for aborts is greater than 70% of the ΔV necessary for lunar descent (from LLO) or lunar ascent (to LLO).

The crew hopper habitat, used in conjunction with a separable nose section, provides the HCRV function on the surface and in-orbit around both Mars (as shown in Figure 10) and the moon. The lunar supported HCRV's do not have the ACC6 TPS, thus the nose section dry mass and total HCRV mass is 495 kg lighter for the lunar configuration. The ATLS tanks are sized for this hopper functional capability to ensure the HCRV has a minimum of 50 km roundtrip range on Mars. Thus, the fully loaded ATLS tanks on the Mars HCRV deliver 1,747 m/s ΔV and a roundtrip hopper range of 51 km (on Mars), while the lunar HCRV delivers 1,809 m/s and 125 km roundtrip range (on the moon).

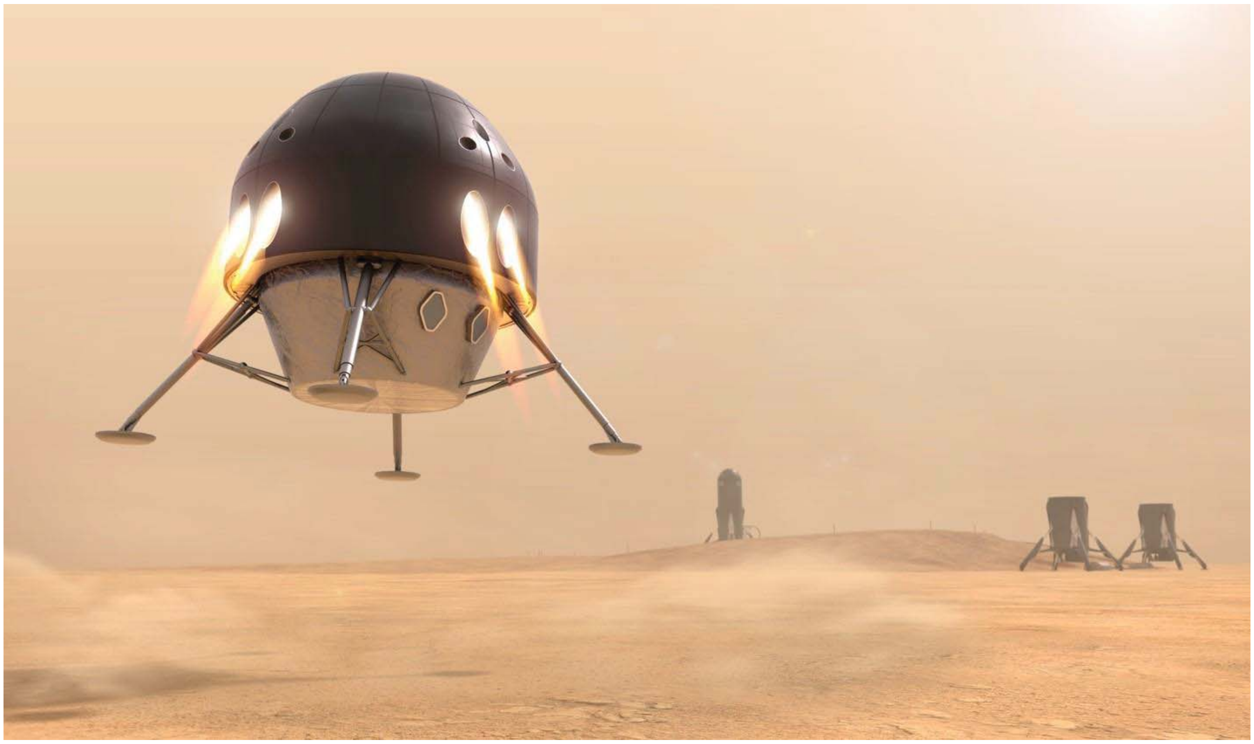


Figure 10. HCRV performing hopping manuever on Mars surface.

4 HSRV Abort Analysis Overview

4.1 Abort Analysis Objectives

The HSRV is designed with unique capabilities to enable a high degree of operational flexibility and crew safety. However, an analysis of the abort flight performance relative to the reference trajectory for each application is needed to verify and validate the concept. The primary objective of this study is to complete this concept verification and validation for each of the HSRV's four missions: 1) Mars ascent; 2) Mars EDL; 3) lunar descent; and 4) lunar ascent.

For each of these missions, three key tasks were completed: 1) define abort scenarios, both ATO and ATS, for each HSRV mission; 2) assess each abort scenario through flight performance analysis; and 3) identify the limiting cases that define key boundaries and breakpoints for each scenario.

All of the trajectory analysis discussed in this paper, including the reference and abort trajectories, were performed using the Program to Optimize Simulated Trajectories II (POST2) [15].

4.2 Abort Analysis Assumptions

For all mission scenarios:

- The initial (nose section plus crew capsule/habitat) separation mass and the available/usable ATLS propellant mass are as shown in Figure 5 for Mars missions and Figure 9 for lunar missions.
- The ATLS thrust and specific impulse values are as shown in Figure 2 for Mars missions and Figure 6 for lunar missions.
- At separation the ATLS performs a minimum six second burn (all eight engines at full thrust) up or forward to clear the nose section from the failed vehicle.

For lunar mission scenarios:

- The EZ is centered on Shackleton crater near the moon's South Pole (89.9° South latitude and 0.0° East longitude) and is assumed to extend radially for 50 km.

For Mars mission scenarios:

- The EZ is centered on the Deuteronilus Mensae site (assumed in original study [3] for assessment purposes) located at 43.9° North latitude and 22.6° East longitude and is assumed to extend radially for 50 km. The site's ground surface altitude is 3.7 km below the reference surface areoid as mapped by the Mars Orbiter Laser Altimeter (MOLA) experiment.
- The separated nose section can aerodynamically or propulsively reorient to control the post-separation flight path, assuming a drag coefficient of 1.3 at all angle-of-attack (AOA) and an aerodynamic reference area defined by the nose section diameter.
- The crew capsule may (dependent on the HSRV orientation at nose section separation) require 180° flip (using an aerodynamic trim tab) when separating from the nose section to orient for HIAD or parachute deployment.
- The crew capsule (prior to HIAD deployment) assumes a drag coefficient of 1.3, an aerodynamic reference area defined by the capsule diameter, is restricted to 0° AOA, and flies an uncontrolled ballistic trajectory.
- The crew capsule's HIAD deploys in 10 seconds, transforming linearly from the capsule diameter to the HIAD diameter, smoothly increasing the aerodynamic reference area. The deployed HIAD assumes the aerodynamic characteristics of a 60° sphere-cone.
- The crew capsule with the HIAD deployed is restricted to 0° AOA and flies an uncontrolled ballistic trajectory.
- The crew capsule jettisons the deployed HIAD prior to parachute deployment. The assumed jettison mass is 100 kg, although the actual HIAD jettison mass is closer to 500 kg.
- The parachute cannot deploy unless in the parachute deployment box, defined in reference [12] as between Mach 1.1-2.1, in the dynamic pressure range of 250-1,250 Pascal, a minimum of five kilometer altitude above MOLA reference areoid, and descending (negative rate of change in altitude).
- The parachute deployment assumes the MSL deployment time (0.88 second mortar flight time, then fully open at 1.55 seconds) and Mach number versus drag coefficient dependence as implemented in the POST2 trajectory tool.
- The retrorocket thrust profile and specific impulse are as shown in reference [13].

5 Mars Ascent Abort

5.1 Mars Ascent Reference Trajectory

For the Mars ascent reference trajectory, launch was assumed to take place from Deuteronilus Mensae from the actual surface altitide, 3.7 km below the MOLA areoid. The ascent trajectory simulation includes models for the Martian atmosphere (Mars-GRAM 2005), Mars Gravity (75x75 gravity field) and vehicle aerodynamics. A liftoff mass of 136.8 metric tons is assumed and the O2/CH4 ADS engines provide a liftoff thrust-to-weight (T/W) ratio of ~0.8 Earth g's.

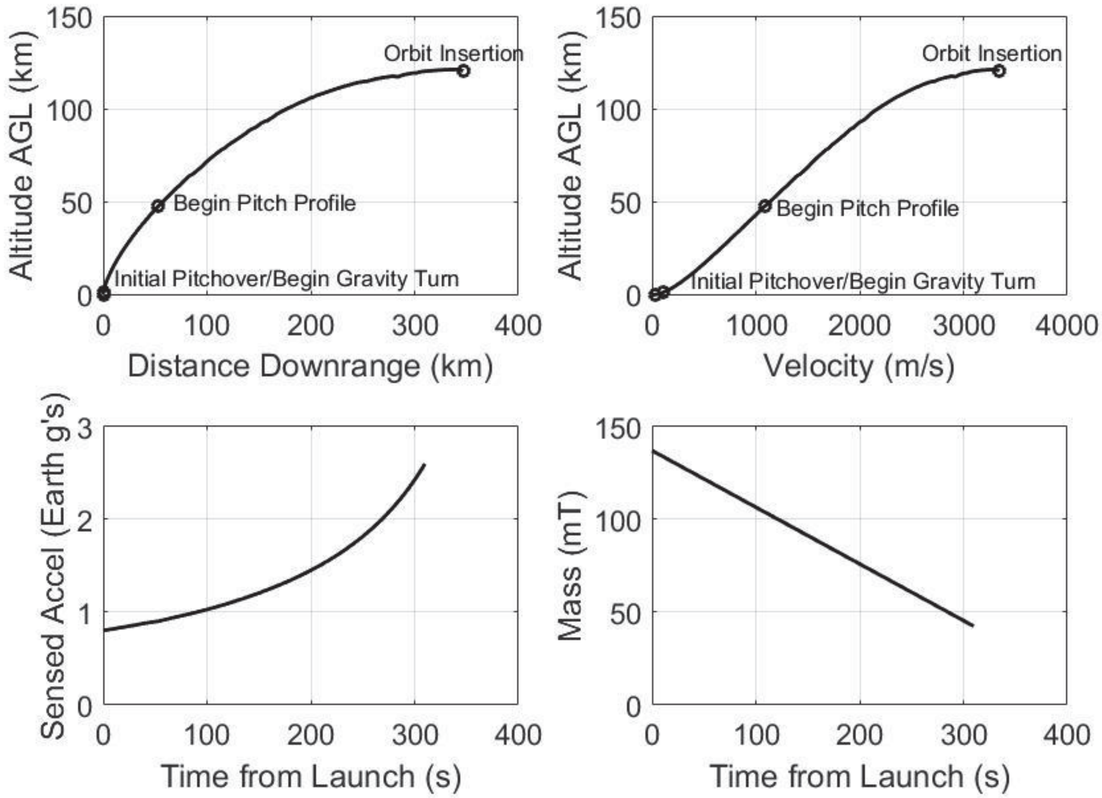


Figure 11. Mars ascent reference trajectory.

Figure 11 shows the sequence of events that occur during the ascent trajectory. After liftoff the HSRV rises vertically for 100 meters and then performs a pitch-over maneuver towards a due East heading. When the dynamic pressure increases to 100 Pascal, the HSRV transitions to 0º AOA gravity turn and holds this attitude through maximum dynamic pressure and beyond, until dynamic pressure drops below 100 Pascal. At this point the HSRV follows a pitch profile that maximizes the mass inserted into the 108 x 235 km target orbit. The entire ascent trajectory has a duration of 310 seconds and covers ~350 km in downrange.

5.2 Mars Ascent Abort Modes

Four ascent abort scenarios were defined and 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 landing zone (DRLZ) location approximately 600 km east of the launch site using the capsule aeroentry and landing capabilities; and 4) ATO using the ATLS propulsion capabilities only. Figure 12 illustrates the approximate trajectory times each of these abort scenarios are possible relative to the nominal ascent trajectory.

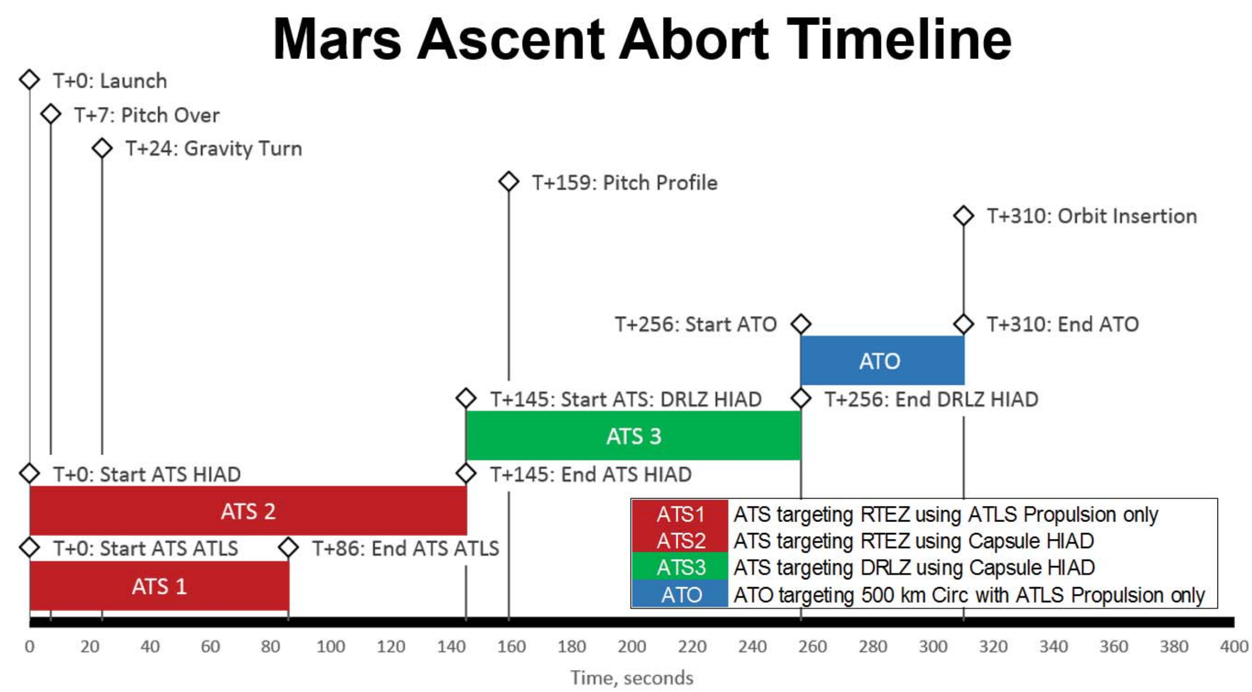


Figure 12. Mars ascent abort options.

Note, for the three ATS abort scenarios, since the reference trajectory begins at the actual ground level altitude (3.7 km below the MOLA reference areoid), the aborted trajectories also target landing at the actual ground level.

5.2.1 ATS targeting RTEZ using ALTS Propulsion only

Figure 13 illustrates the operations assumed for this abort scenario. If an abort event occurs in the first 86 seconds of ascent, the nose section (along with the crew capsule) can execute a purely propulsive abort and landing, returning the crew to the launch site itself or to within the 50 km EZ and within range of pickup by the surface-based HCRV. Figure 14 illustrates the limiting cases for the scenario.

On the launch pad and very early in flight (as illustrated by the T+1 second case), if an abort is initiated all eight ATLS engines ignite to separate the nose section from the failed vehicle. The abort trajectory assumes the ATLS engines operate for nine seconds and are shut down. The aborted vehicle coasts to about four kilometer altitude and begins to descend back to the launch site. The ATLS engines then re-ignite for a soft propulsive landing at the site.

As the HSRV approaches the T+86 seconds mark the nose section can separate with a six second separation burn, followed by a reorientation manuever to pitch the nose section over to 180º AOA. The ATLS engines continue to operate for up to an additional 15 seconds to boost the nose section back toward the site. The ATLS engines then shutdown, the nose section coasts and descends, and at the appropriate time the ATLS engines re-ignite for a soft propulsive landing. This case at T+86 seconds represents the latest time that this abort mode can return the nose section to the EZ.

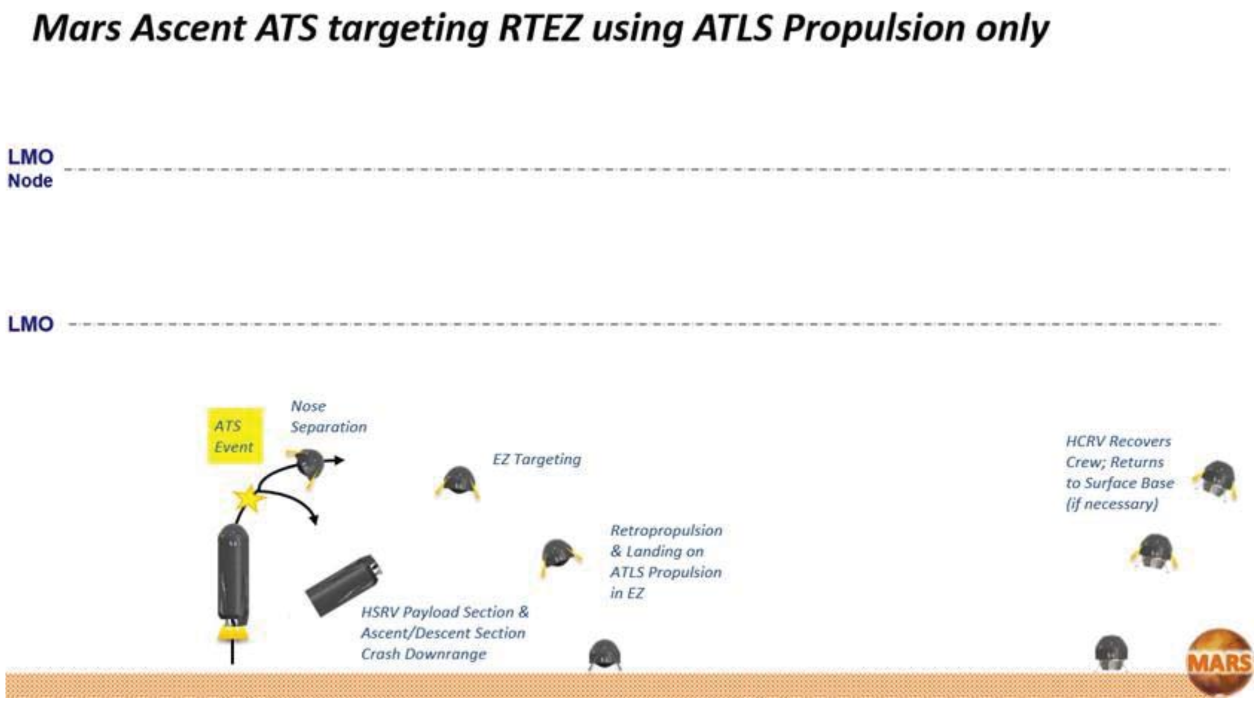


Figure 13. Concept of operations illustrating Mars ascent ATS targeting a return to exploration zone using the ATLS propulsion system only.

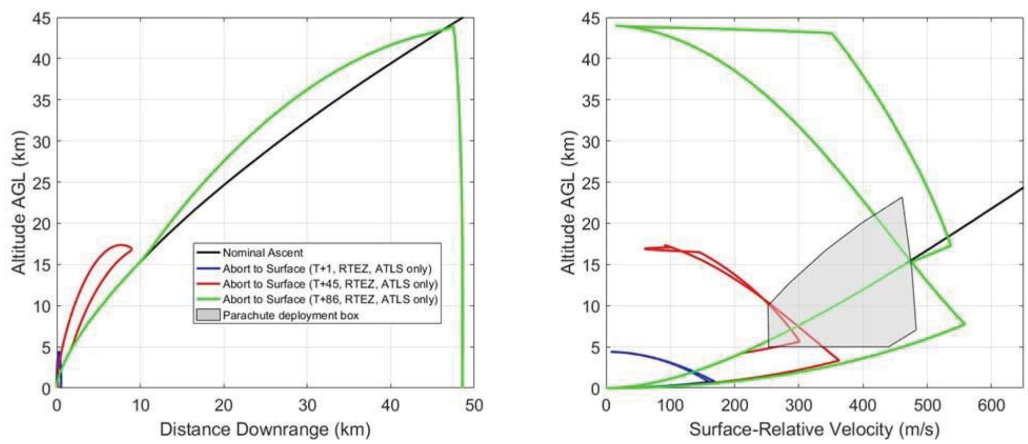


Figure 14. Trajectory plots illustrating Mars ascent ATS targeting a return to exploration zone (RTEZ) using the ATLS propulsion system only.

5.2.2 ATS targeting RTEZ using Capsule HIAD

Figure 15 illustrates the operations required for this abort scenario. It overlaps the ATS targeting RTEZ using ATLS Propulsion only scenario starting at T+0 but is possible to a later time in the ascent profile, around T+145 seconds. In this scenario the crew capsule separates from the nose section after the nose section plus capsule clear the abort hazard and propulsively target the EZ. The capsule then uses its EDL systems to safely land in the EZ and the HCRV recovers the crew and returns them to the base. An important consideration is whether the flight profile during aerodynamic deceleration pass through the defined deployment region for supersonic parachutes.

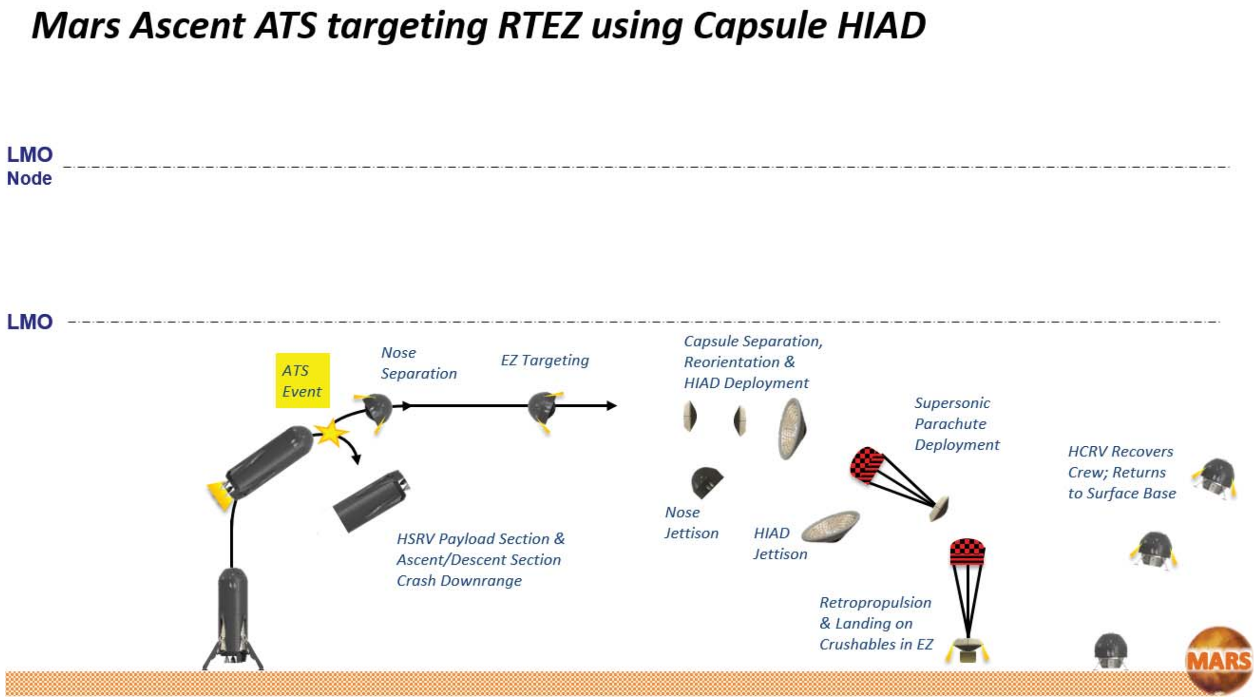


Figure 15. Concept of operation illustrating Mars ascent ATS targeting a RTEZ using the capsule aeroentry and landing systems.

As illustrated in Figure 16, the limiting cases for the scenario occur at T+1 and T+145 seconds.

On the pad or just after launch (as illustrated by the T+1 second case), the nose section ATLS engines perform a 25 second separation burn that delivers the separated nose section and capsule to about 6 km. The ATLS engines are shutdown and the vehicle coasts to about 25 km altitude. During this coast the crew capsule separates from the nose section and reorients for the descent aerodynamic coast. The capsule gains sufficient velocity to enter the parachute deploy box. Following parachute descent, the capsule lands utilizing 14 of the 16 retro-propulsion thrusters and deployment of a crushable device that absorbs the energy of terminal landing. In this case, the HIAD is not required.

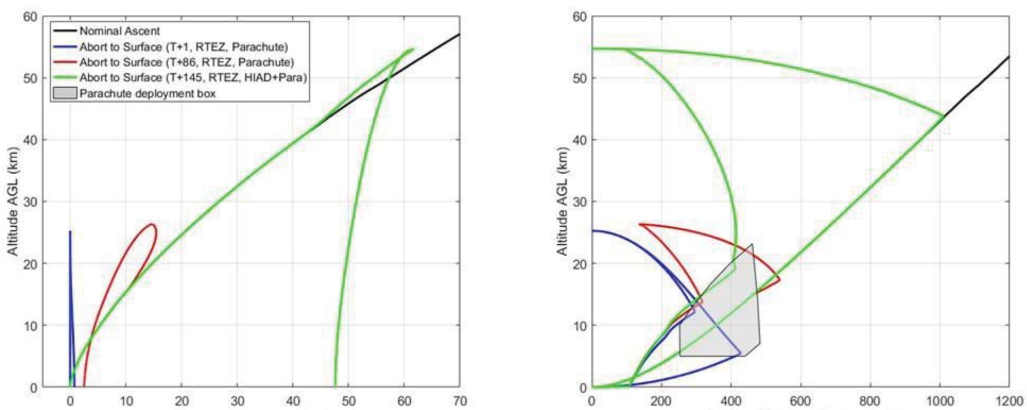


Figure 16. Trajectory plots illustrating Mars ascent ATS targeting a RTEZ using the capsule aeroentry and landing systems.

The T+145 second mark represents the latest time during ascent this abort scenario is possible. Following a 6 second ATLS engine separation burn, the nose section pitches to 180º AOA and the ATLS burns for another 40 seconds, boosting the nose section back toward the EZ. Once the ATLS engines consume all of its propellant the crew capsule separates from the nose section and deploys the HIAD after a 5 second delay. The HIAD inflates over the next 10 seconds and provides aerodynamc deceleration in the hypersonic flight regime. As the flight profile approaches the supersonic parachute deploy box, the HIAD is jettisoned and the parachute deployed. Following parachute descent, the capsule lands utilizing 14 of the 16 retro-propulsion thrusters and deployment of a crushable device that absorbs the energy of terminal landing. Because this scenario allows them to expend the full load of the nose segment's propellant, they can return to the boundary of the EZ from later in flight, up to T+145 seconds.

5.2.3 ATS targeting DRLZ using Capsule HIAD

Between T+145 and T+256 seconds, the vehicle is moving too quickly downrange to return to the EZ, but not quickly enough to ATO. During this window, the crew can execute a 'downrange abort', analogous to the Space Shuttle's Trans-Atlantic Landing abort mode [16]. As illustrated in Figure 17, a mobile habitat will drive itself autonomously to a downrange abort landing site just over 600 km to the east of the launch site prior to flight. That particular range is chosen because that is the shortest possible landing that can be executed if the crew must abort immediately prior to the ATO window opening (using the entire fuel capacity of the nose capsule to slow down before separating the crew module and deploying the HIAD). If the crew must abort shortly after the ATS targeting RTEZ using Capsule HIAD window closes, there is ample fuel capacity to boost the nose segment to land at the downrange abort landing site. The limiting cases for this scenario are shown in Figure 18.

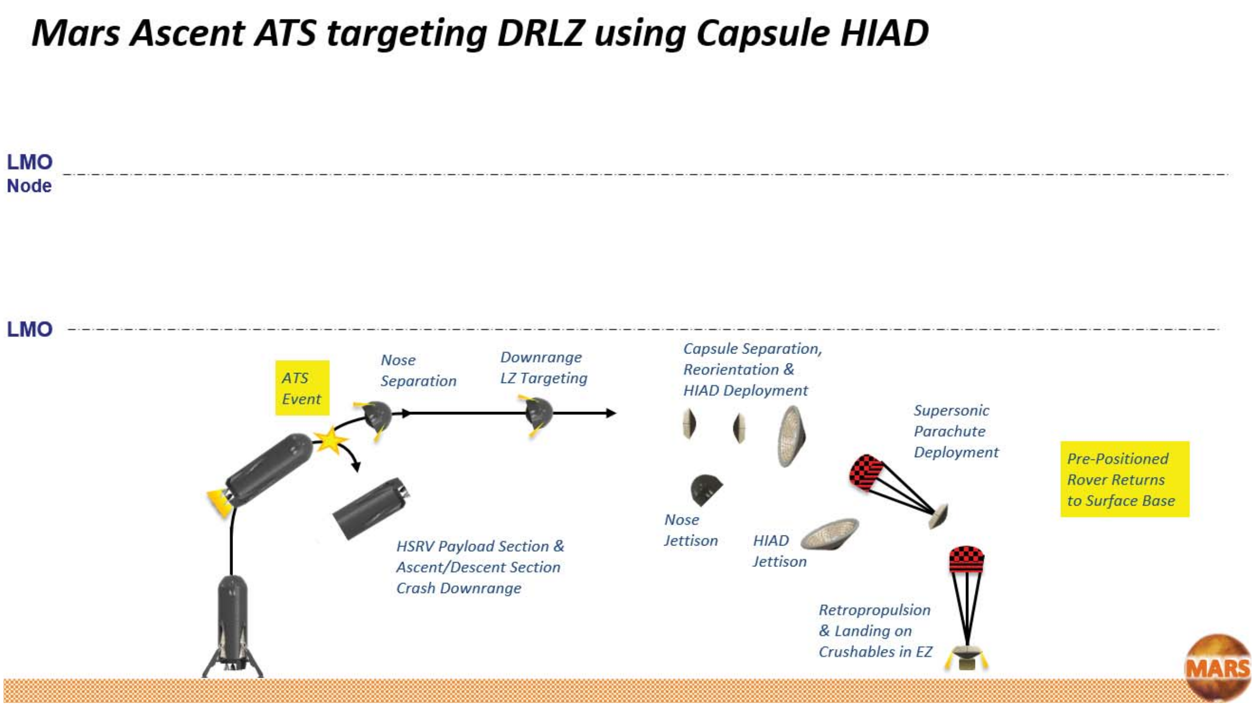


Figure 17. Concept of operation illustrating Mars ascent ATS targeting a common downrange location approximately 500 km east of the launch site using the capsule aeroentry and landing capabilities.

For an abort initiated at the T+145 mark, the nose section separates with a six second ATLS burn. The nose section then reorients to an AOA of 0º and continues burning the ATLS engines for an additional 19 seconds. The crew capsule then separates and the HIAD is deployed. As the capsule approaches the parachute deploy box, the HIAD is jettisoned and the parachute deployed. Terminal landing of the capsule is completed using 14 of the 16 retro-rocket thrusters and the crushable energy absorber.

The T+256 second mark represents the last moment an ATS targeting the DRLZ is possible. Following a six second ATLS separation burn the nose section reorients to a -60° pitch (to cancel out the vertical momentum and limit lofting), 270° yaw (to point back at the launch site) position and all remaining ATLS propellant is consumed. After the crew capsule separates, descent and landing at the DRLZ is completed using the HIAD, parachute, retrorockets and crushables.

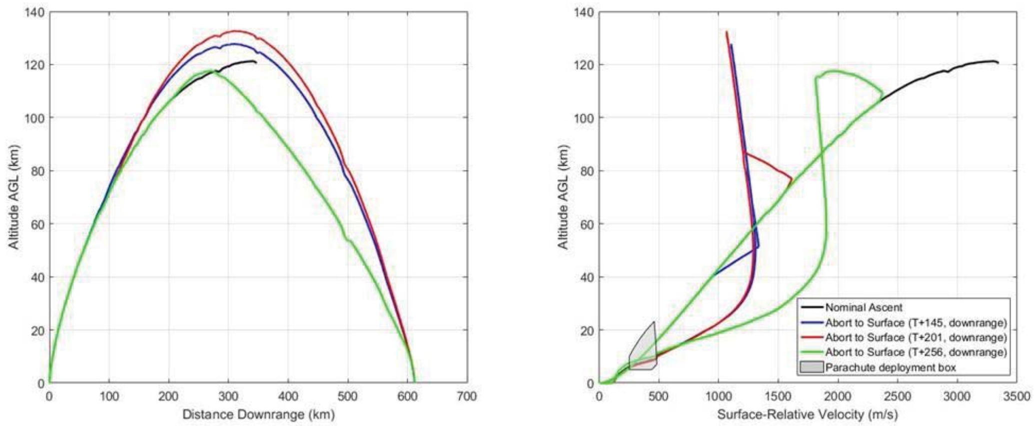


Figure 18. Trajectory plots illustrating Mars ascent ATS targeting a common downrange location approximately 500 km east of the launch site using the capsule aeroentry and landing capabilities.

5.2.4 ATO targeting 500 km Circular Orbit with ATLS Propulsion only

Beyond T+256 seconds the nose segment has enough fuel to complete the orbital insertion on its own. The T+256 second case places the aborted nose section and crew capsule in a 100 x 250 km orbit, where it will await retrieval by the HCRV based at the LMO node (as illustrated in Figure 19). For cases occuring near the end of ascent, such as the T+309 second case shown in Figure 20, the nose section and crew capsule are also inserted into a 100 x 250 km orbit. However, sufficient propellant remains available on-board to perform the rendezvous manuevers with the LMO node.

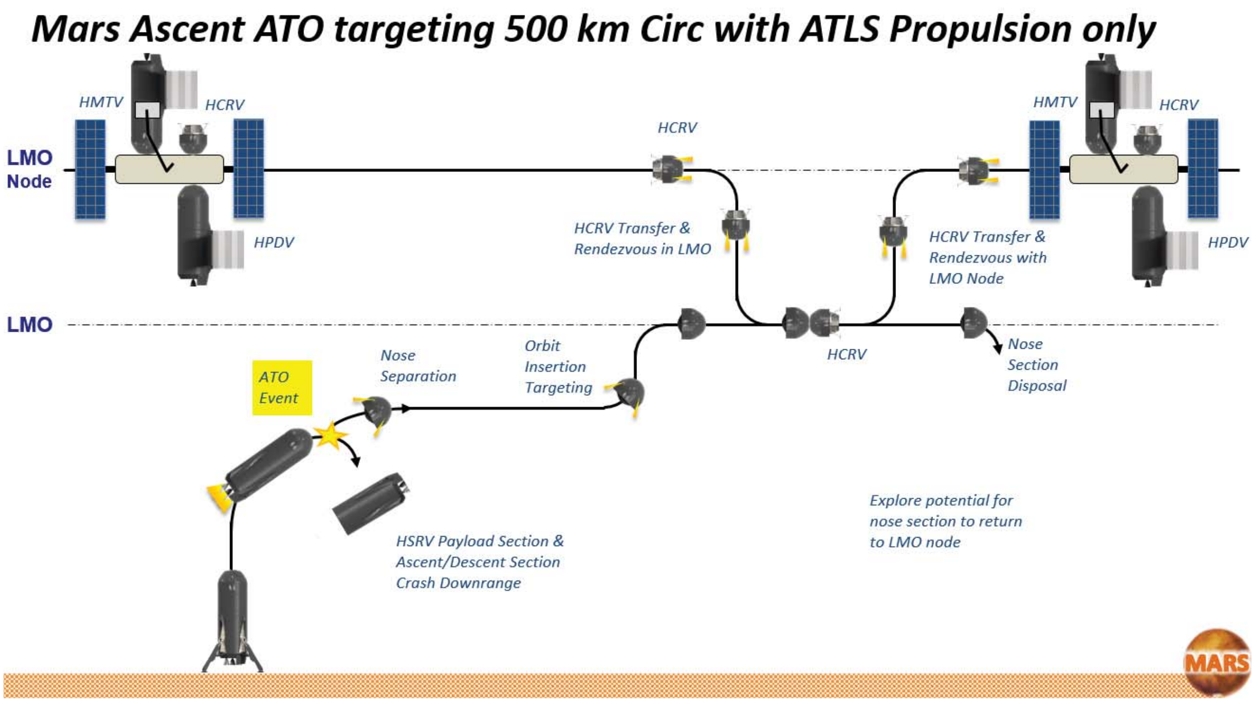


Figure 19. Concept of operation illustrating Mars ascent ATO using the ATLS propulsion capabilities only.

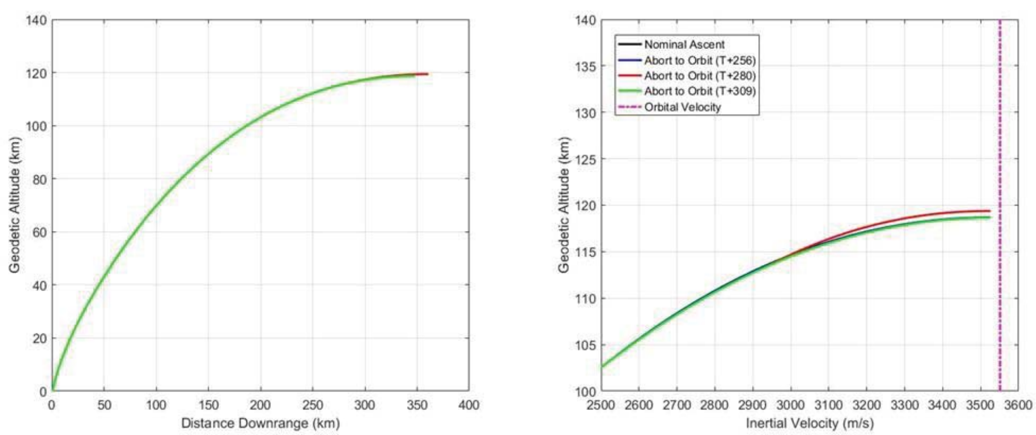


Figure 20. Trajectory plots illustrating Mars ascent ATO using the ATLS propulsion capabilities only.

6 Mars EDL Abort

6.1 Mars EDL Reference Trajectory

The Mars EDL reference trajectory begins with deorbit from a 500 km circular orbit with a 45° inclination, and targets a landing site at Deuteronilus Mensae. For conservatism, the reference trajectory targets a landing site that is at a surface elevation of zero kilometer relative to the Mars MOLA areoid definition (actual site elevation is -3.7 km).

The initial mass of the HSRV is assumed to be 30.9 t. Figure 21 shows the sequence of events that occur during the EDL trajectory. After a 149 m/s deorbit burn, the vehicle coasts for ~30 minutes until entry interface (EI), defined as the radial distance of 3,522 km from Mars center. At this point the HSRV enters the atmosphere with a velocity of 3,300 m/s and a flight path angle of -5.1º. At entry, the bank angle is 0º (lift-up) and the AOA is 55º. That AOA provides an L/D ratio between 0.4 and 0.5 and is held throughout entry, until SRP initiation.

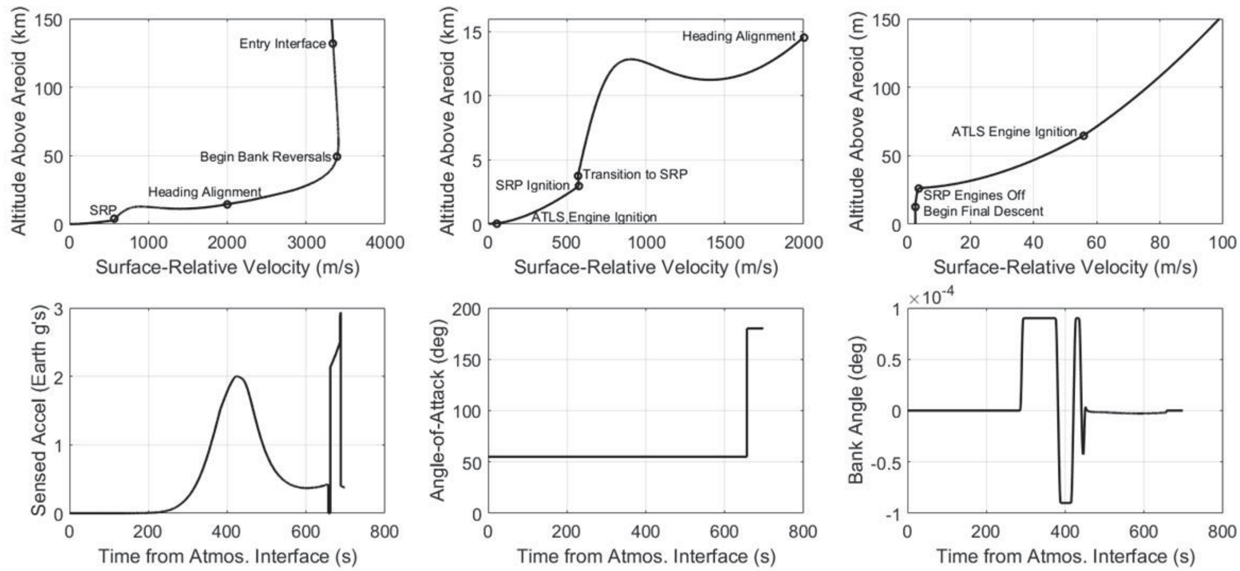


Figure 21. Mars EDL reference trajectory.

Approximately 285 seconds after EI, when the sensed axial acceleration has increased to 0.15 Earth g's, a series of bank reversals are performed for range control. During this phase the sensed acceleration peaks at 2.0 Earth g's. When the velocity decreases below 2,000 m/s, the bank reversals end and a heading alignment phase begins where the HSRV banks to steer the vehicle toward the target landing site, eventually reaching a zero bank (lift-up) orientation when the target heading is acquired.

At an altitude of ~4.6 km and velocity of ~570 m/s (Mach 2.5) the HSRV performs a five second reorientation to an engine-first attitude (180º AOA) in preparation for SRP. At SRP ignition, the ADS engines ignite to slow the vehicle. The ADS engines are sized for the SRP phase with an 80% throttle to limit deceleration to 2.5 Earth g's during the main SRP phase. For conservatism, aerodynamic drag is assumed to be zero during SRP. After ~25 seconds, the ATLS engines are ignited at an altitude of 65 meters and a velocity of 56 m/s. Over the next 2 seconds, the ADS engines are shut down, and the ATLS engines continue to burn for an additional 5 seconds until the velocity has been slowed to 2.5 m/s. At this point the final descent phase begins and the ATLS engines are throttled so that the HSRV descends at a constant 2.5 m/s descent rate until landing 5 seconds later.

The entire EDL trajectory, from deorbit to landing, has a duration of 2,600 seconds, with the final 700 seconds being the atmospheric entry, descent and landing.

6.2 Mars EDL Abort Modes

As shown in Figure 22, five entry abort scenarios were defined and 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.

Note, for the three ATS scenarios, since the reference Mars EDL trajectory targets landing at zero kilometer MOLA, the ATS trajectories also target this altitude. Since the entire EZ around Deuteronilus Mensae is below zero kilometer MOLA, this results in conservative estimates for the ATS scenarios.

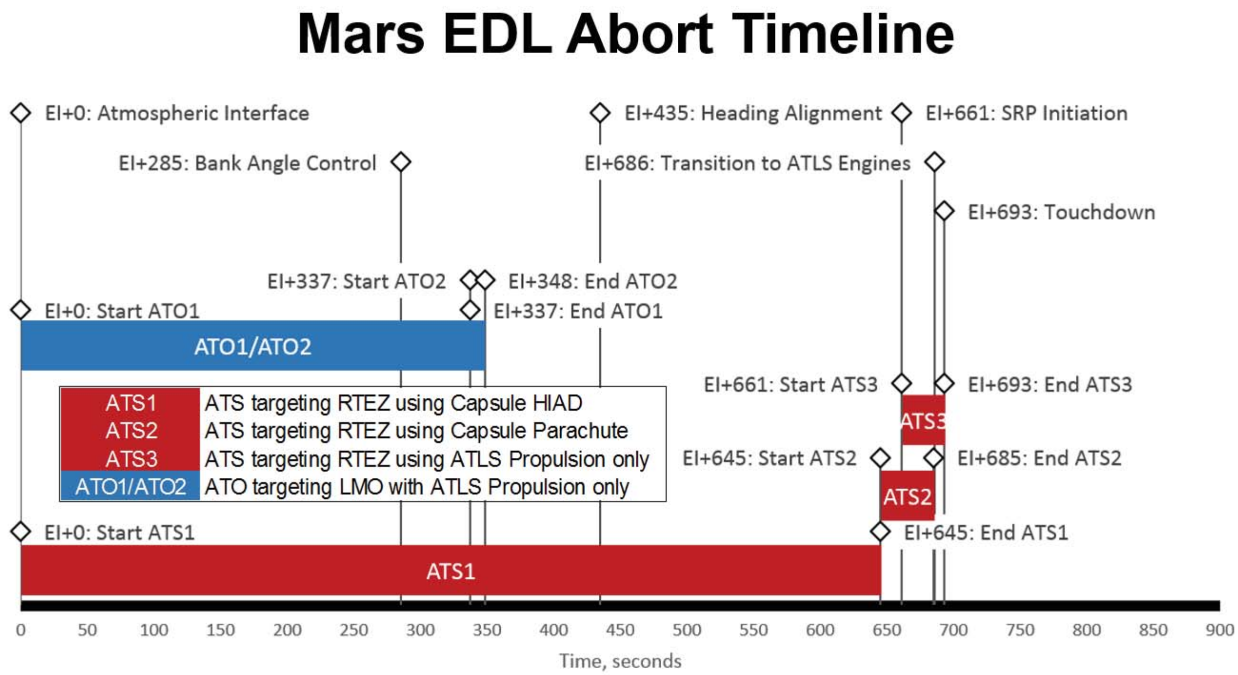


Figure 22. Mars EDL abort options.

6.2.1 ATO targeting LMO using ALTS Propulsion only

Figure 23 illustrates the operations assumed for this abort scenario, while Figure 24 illustrates the limiting cases.

If an abort event occurs after deorbit but prior to EI, the nose section can easily separate and return to LMO. Even after contact with the atmosphere ("EI+0"), the nose section of the vehicle can detach and execute an ATO, returning to the target 500 km orbit as late as EI+337, or to a 150 km minimum stable orbit as late as EI+348.

The altitude-velocity plot in Figure 24 shows why the ATO is limited to EI+348 sec. After EI+252 seconds, the vehicle begins to decelerate rapidly due to atmospheric drag. In order to return to orbit, the vehicle must push its way back up through the atmosphere and then make up any velocity losses when it reaches apoapsis, which becomes increasingly taxing the later that ATO is initiated.

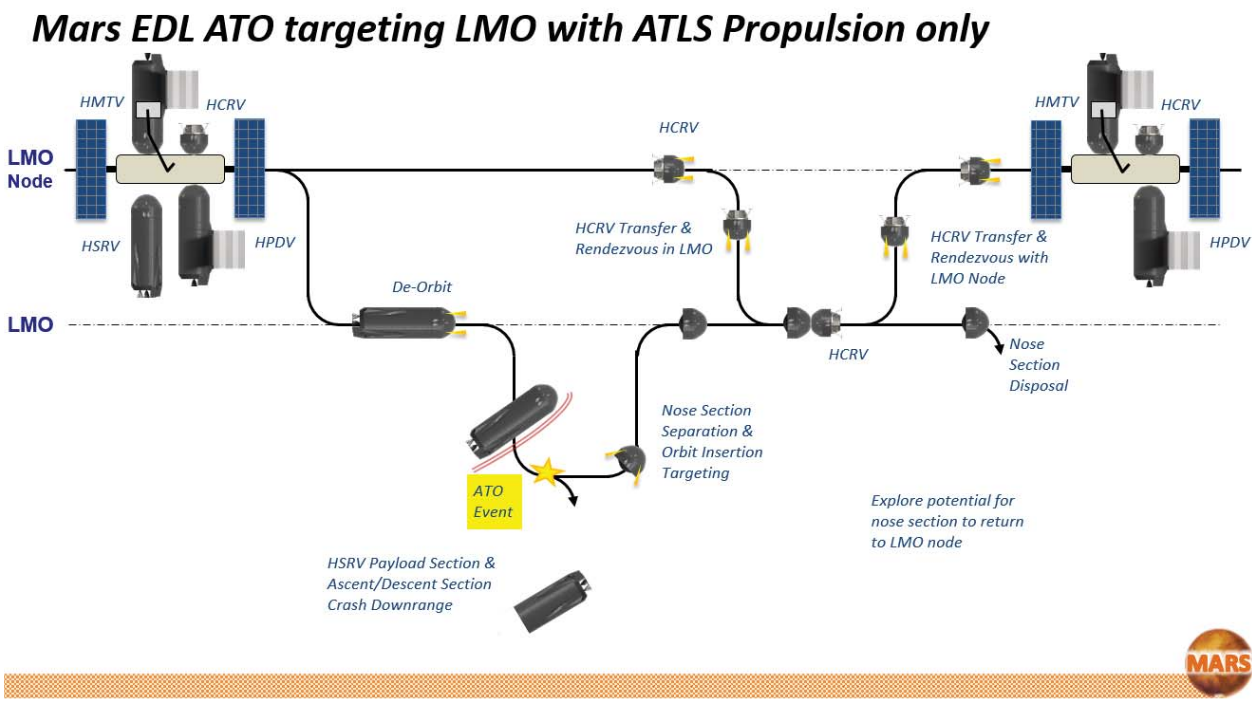


Figure 23. Concept of operation illustrating Mars EDL ATO using the ATLS propulsion capabilities only.

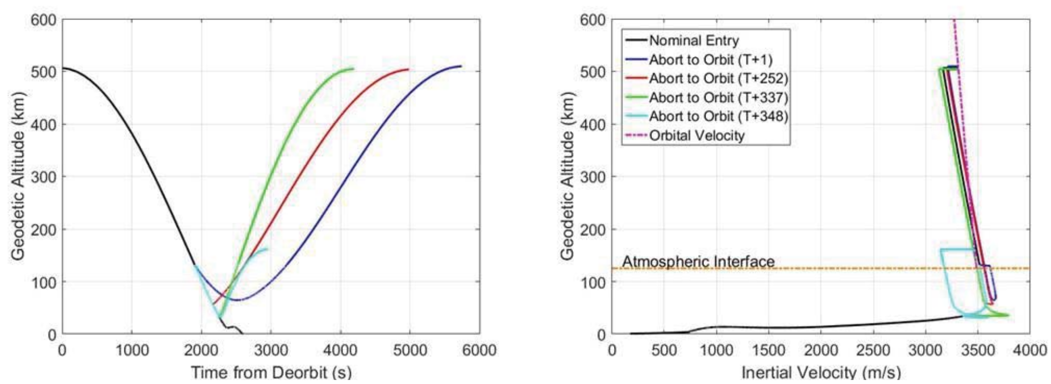


Figure 24. Trajectory plots illustrating Mars EDL ATO using the ATLS propulsion capabilities only.

6.2.2 ATS targeting RTEZ using Capsule HIAD

At any time during Mars aerodynamic entry an ATS is possible using the crew capsule's HIAD and supersonic parachute, illustrated in Figure 25. The nose section ATLS fires to pull the crew capsule away from the failed HSRV, then the crew capsule separates from the back of the nose section. After aerodynamically reorienting, the capsule deploys the HIAD, decelerating the capsule and targeting the parachute deploy box. As parachute deploy approaches, the HIAD is jettisoned, the supersonic parachutes deploy and decelerate the capsule to terminal velocity. As the capsule approaches the surface (within ~250 meters), a set of solid retrorockets fire to decelerate it from terminal velocity down to 2.5 m/s with the capsule positioned a few meters above the surface. The capsule lands at 2.5 m/s on the crushable energy absorber and the crew egresses through the top tunnel hatch. The HCRV based at the surface site hops to the aborted landing site to recover the crew and return them to the base.

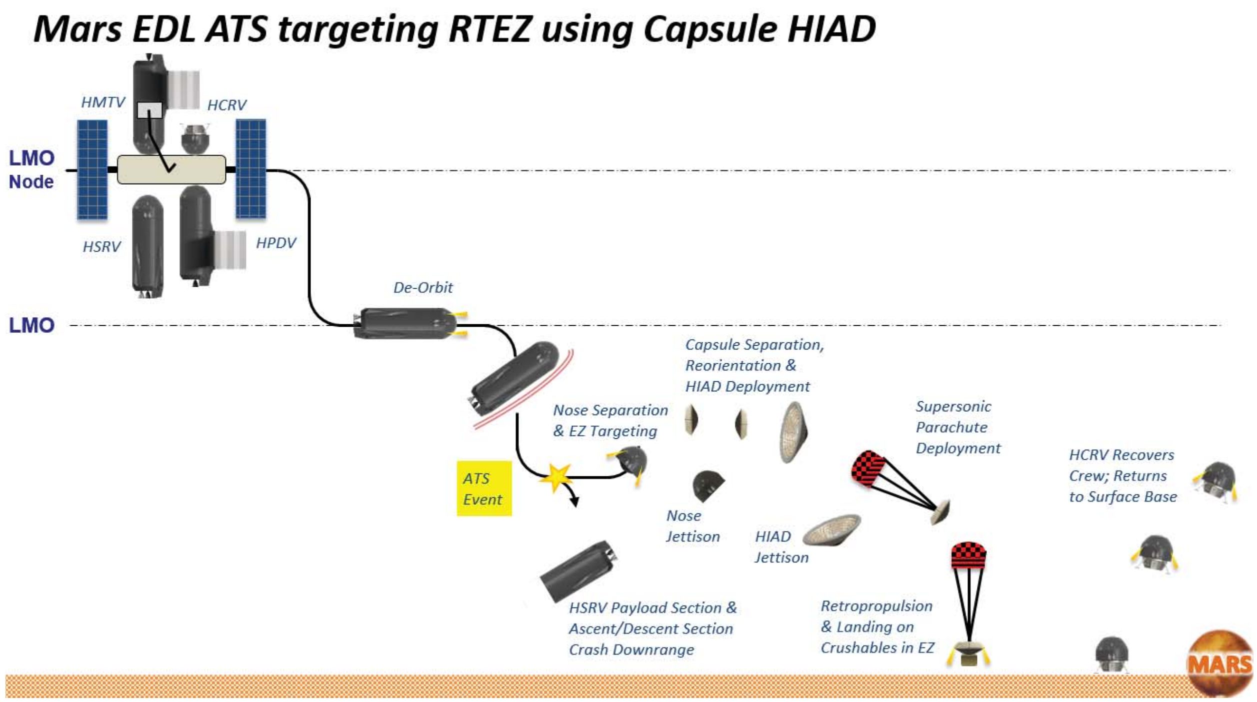


Figure 25. Concept of operation illustrating Mars EDL ATS targeting the RTEZ using the capsule aeroentry and landing systems.

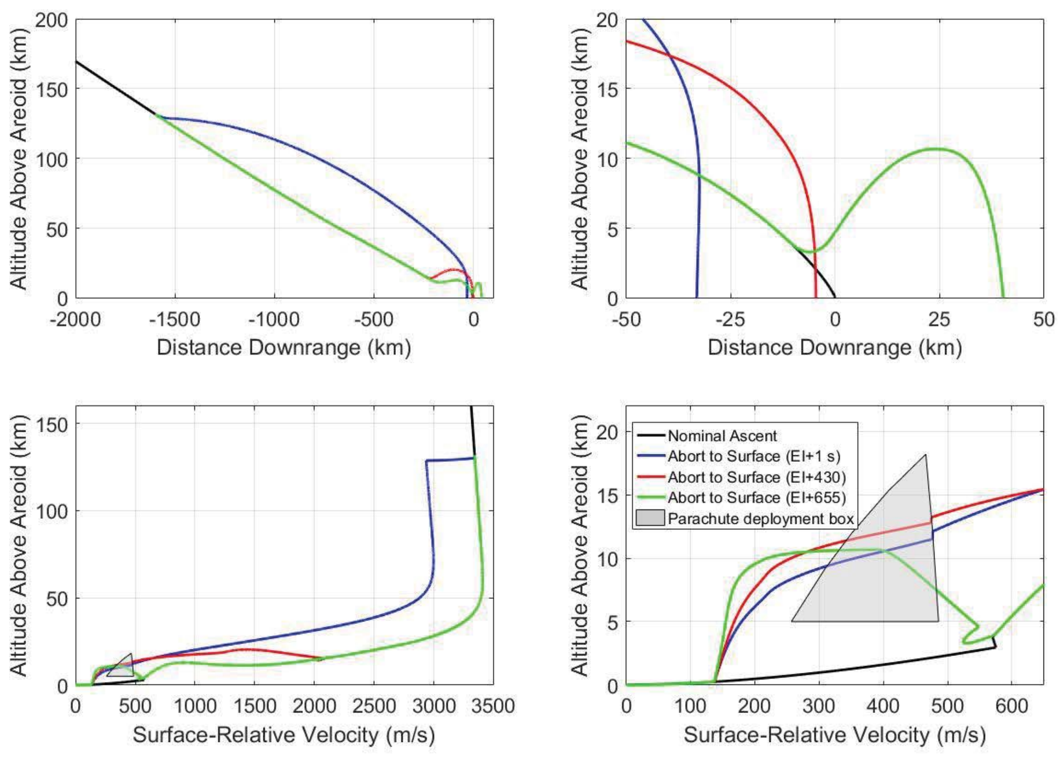


Figure 26. Trajectory plots illustrating Mars EDL ATS targeting the RTEZ using the capsule aeroentry and landing systems.

To target the EZ, the ATLS burn duration and direction varies, depending on when the abort occurs. For most of the descent, the engines fire to accelerate the capsule forward and prevent the crew from falling short of the EZ. However, for a very early abort (as illustrated in Figure 26 at EI+1 second), a retrograde burn is used to prevent overshooting the EZ. For a very late abort just prior to SRP initiation (as illustrated in Figure 26 at EI+655 seconds), the ATLS engines fire upwards to lift the capsule's trajectory, to gain enough altitude to deploy the parachutes safely. For this case the HIAD is not deployed at all as the vehicle is already close to the Mach 2.1 limit for parachute deployment.

The EI+430 second case illustrates a key pinch-point where the vehicle has to 'thread the needle' between accelerating forward to not fall short of the EZ and also getting in the parachute box (i.e. – the system cannot be too fast too low). However, optimizing the attitude of the separated nose section for the ATLS separation burn determines the necessary lofting that enables both constraints to be satisifed over the time range of this scenario.

Note that this scenario overlaps the *ATO targeting LMO with ATLS Propulsion only* scenario starting at EI+1 but is possible to a later time in the entry profile, around EI+685 seconds. This overlap suggests an excess capability in the design. It also represents an opportunity for mission architects to either pick the preferred abort strategy or perhaps consider eliminating the HCRV infrastructure based at the LMO node, for example. ATO is still possible without the HCRV based at the node, but the separated nose section would need to reserve sufficient propellant to get itself back to the node, thus the latest time for ATO would be much earlier. In addition, this strategy must consider the impact on Mars ascent ATO scenarios.

6.2.3 ATS targeting RTEZ using Capsule Parachute

During SRP, the crew still has the option of performing an aeroassist landing, with the nose section accelerating upwards to a safe altitude for parachute deployment. As shown in Figure 27, the HIAD is not deployed as the vehicle is already close to (or below) the Mach 2 limit of parachute deployment.

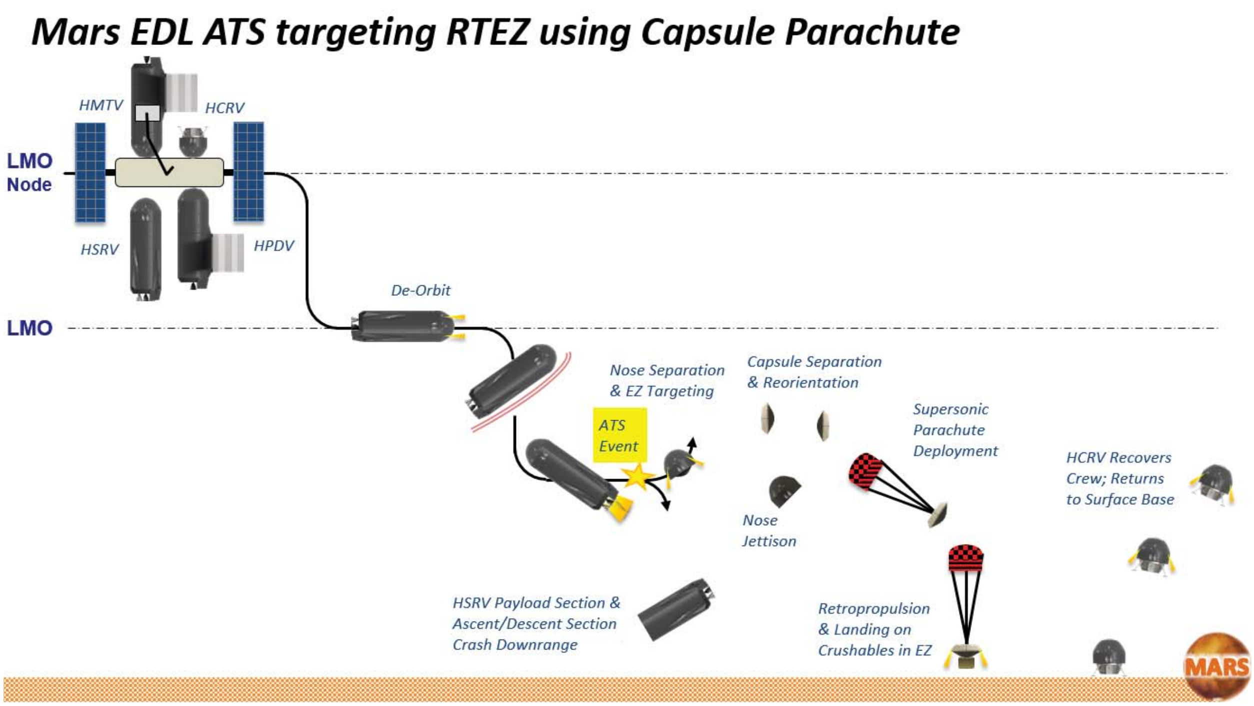


Figure 27. Concept of operation illustrating Mars EDL ATS targeting the RTEZ using the capsule parachute and landing systems only (i.e. – no HIAD deployment required).

Figure 28 illustrates the limiting cases for this scenario. At EI+662 seconds, just after SRP engines ignition, the nose section separates using the ATLS engines, gaining altitude and maintaining sufficient velocity for parachute deployment. The capsule then separates from the nose section and decelerates until within the parachute deployment box. Terminal landing employs retropropulsion and crushable energy absorbers and the crew is recovered by the HCRV based at the surface site.

At EI+685 seconds, a similar abort profile is followed except the ATLS engines must burn longer to increase the relative velocity and altitude for parachute deployment.

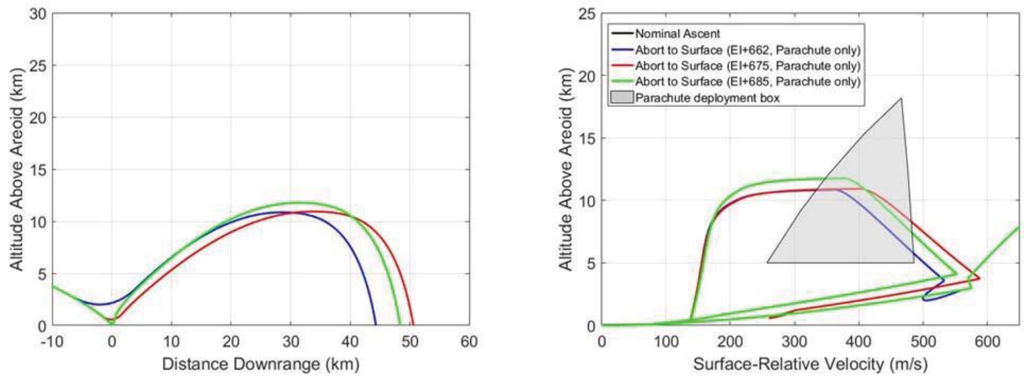


Figure 28. Trajectory plots illustrating Mars EDL ATS targeting the RTEZ using the capsule parachute and landing systems only (i.e. – no HIAD deployment required).

6.2.4 ATS targeting RTEZ using ATLS Propulsion only

After SRP initiation an alternative abort scenario is considered. Instead of separating the crew capsule from the nose section following the ATLS engine burn to accelerate the crew clear of the failed vehicle, the entire nose section simply executes a landing with the ATLS proplusion system. Figure 29 illustrates the operations associated with this abort mode.

As shown in Figure 30, this abort strategy is possible from SRP initiation (EI+662 seconds) down to just before the planned ignition of the ATLS engines for a nominal landing (EI+685 seconds). In all cases for this scenario the nose section lands within 2 km of the nominal landing target.

In contrast, for the ATS targeting RTEZ using Capsule Parachute scenario the crew capsule lands much further away from the nominal landing target (near the edge of the EZ). Give that these two scenarios overlap, mission designers can consider the pros and cons of each to establish a preferred abort strategy for events occuring after SRP initiation.

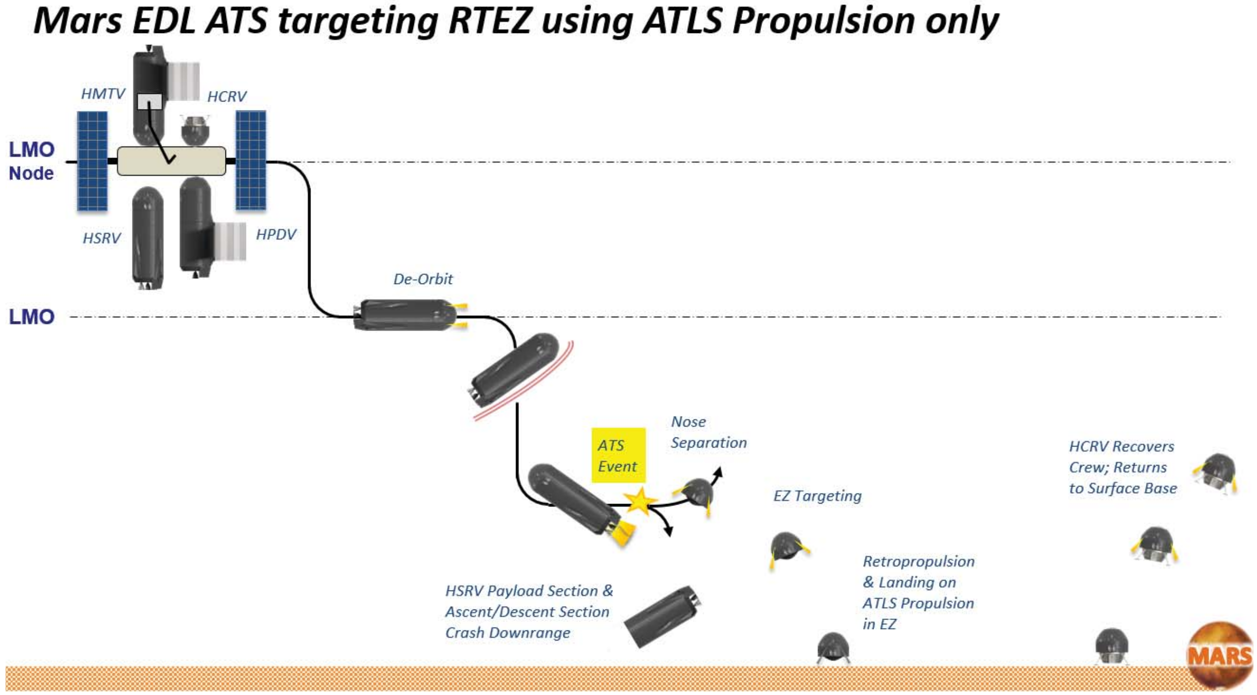


Figure 29. Concept of operation illustrating Mars EDL ATS targeting the RTEZ using only the nose section propulsive capabilities.

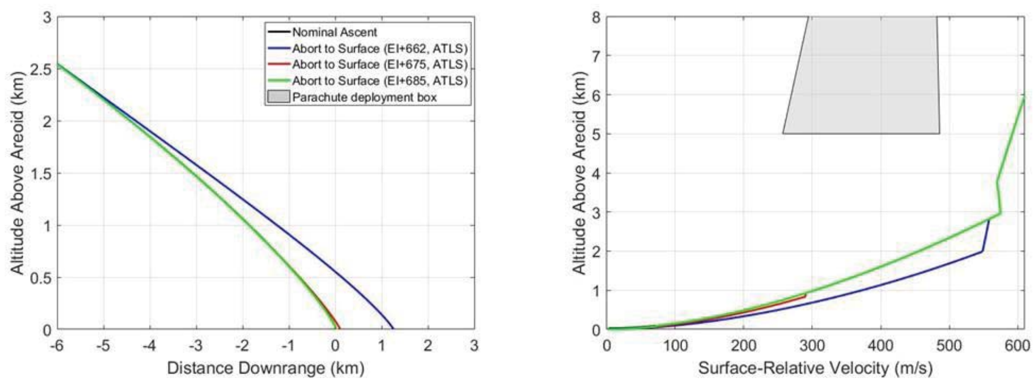


Figure 30. Trajectory plots illustrating Mars EDL ATS targeting the RTEZ using only the nose section propulsive capabilities.

7 Lunar Descent Abort

7.1 Lunar Descent Reference Trajectory

The Lunar descent reference trajectory begins with a 118.2 metric ton HSRV in a 100 km circular polar orbit. The sequence of events that occur during descent are shown in Figure 31.

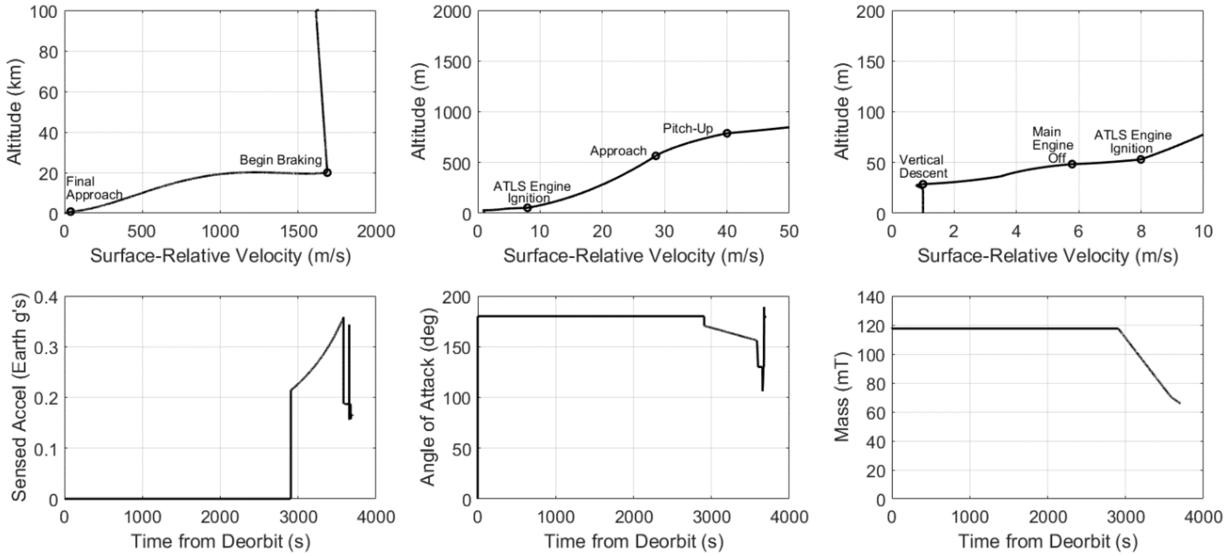


Figure 31. Lunar descent reference trajectory.

An initial deorbit burn places the HSRV into a 100 x 15.24 km intermediate orbit. At an altitude of 20 km, just prior to perilune, a braking maneuver is performed using the single ADS engine, resulting in an initial T/W of 0.214 Earth g's. When the engine is ignited the HSRV is at an AOA of 170°. The braking thrust is held for ~677 seconds until the velocity is reduced to 40 m/s and the altitude is 790 m. At this point the ADS engine is throttled to ~50% and a pitch-up maneuver is performed placing the HSRV in a nearly vertical attitude to facilitate terrain sensing. The approach attitude and thrust level are held for ~70 seconds until the velocity has been reduced to eight meters per second and the altitude is 50 meters above the surface. Four ATLS engines each at 50% throttle are ignited and once verified operating properly the ADS engines are turned off. The approach phase is completed after 19 sec operating the ATLS engines only. At 30 m altitude and falling at one meter per second, the ATLS engines continue for the final 30 sec vertical descent and landing phase, throttling slightly to maintain the one meter per second descent rate to landing. The entire lunar descent trajectory, from deorbit to landing, has a duration of ~3,700 seconds.

7.2 Lunar Descent Abort Modes

As shown in Figure 32, two lunar descent abort scenarios were defined and examined: 1) ATO targeting 100 km circular LLO using the ATLS propulsion capabilities only; and 2) ATS targeting the RTEZ using the ATLS propulsion capabilities only.

For the ATO scenario, as shown in Figure 33, 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, as shown in Figure 35, 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.

Between D+166 and D+470 seconds an abort event can be either ATO or ATS. This overlap suggests an excess capability in the design.

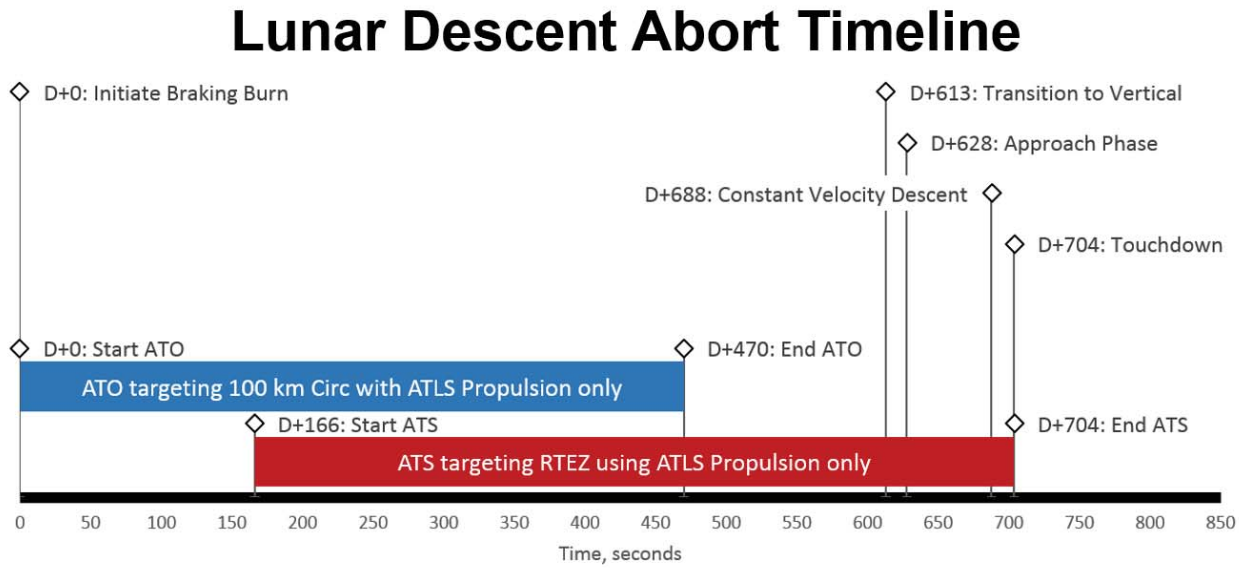


Figure 32. Lunar descent abort options.

7.2.1 ATO targeting 100 km Circular Orbit using ATLS Propulsion only

As shown in Figure 34, the limiting cases for the scenario occur at D+1 and D+470 seconds. For purposes of identifying abort times, "D+0" is the moment when the final braking burn initiates, taking the vehicle from its 15x100 km descent transfer orbit to the surface. If an abort is required after deorbit but prior to D+0 seconds, the nose section separates and instead of initiating the braking burn the nose section continues to coast to apolune of the transfer orbit. At apolune the nose section re-circularizes.

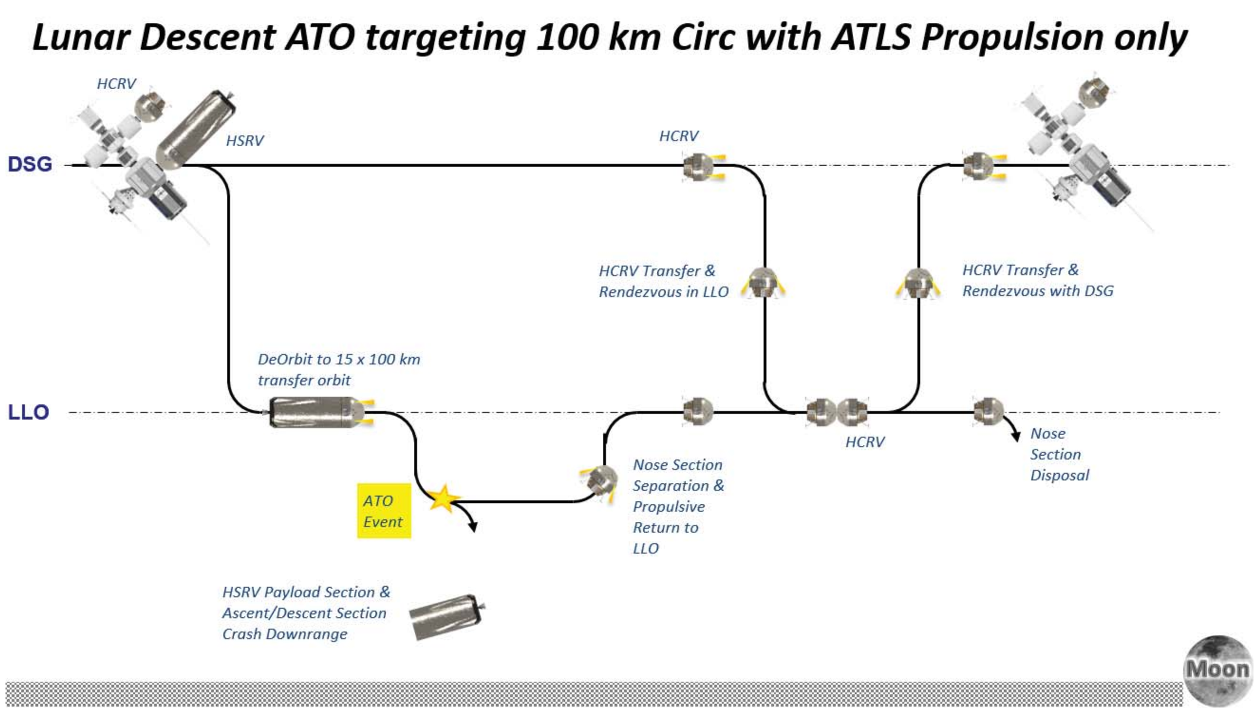


Figure 33. Concept of operation illustrating Lunar descent ATO using the ATLS propulsion capabilities only.

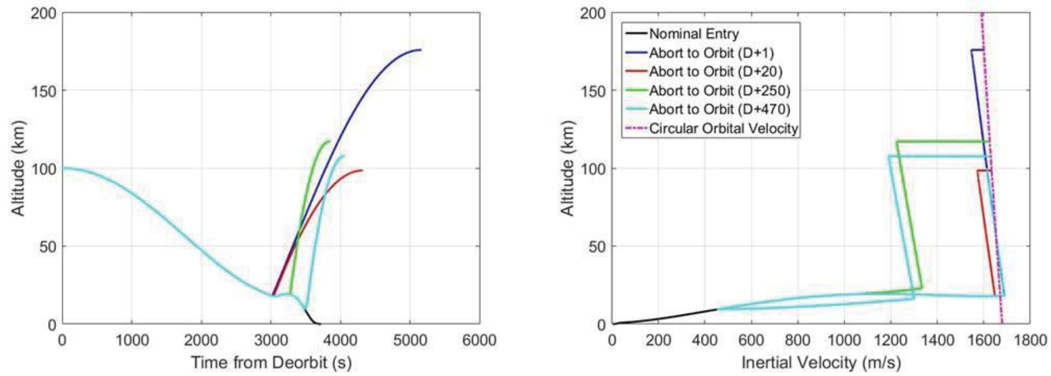


Figure 34. Trajectory plots illustrating Lunar descent ATO using the ATLS propulsion capabilities only.

For an abort initiated at D+1 seconds, the nose section performs the minimum six second separation burn using the ATLS engines. This results in the nose section overshooting the 100 km circular orbit. This is the case as late as D+20 seconds. However, since the goal of the ATO is to return the crew safely to the DSG, the higher energy orbit reduces the propellant requirement of the nose section to return itself to the DSG or for the HCRV based at the DSG to recover the crew.

The D+470 seconds case represents the latest an ATO is possible for lunar descent. Beyond D+470 seconds there is insufficient ΔV to reach 100 km circular orbit. ATO may be feasible even later than D+470 seconds, but there is still a trade to be made on how much fuel should be left over for rendezvous in LLO.

7.2.2 ATS targeting RTEZ using ATLS Propulsion only

As shown in Figure 36, the limiting cases for this scenario are D+166 and D+688 seconds. Prior to D+166 seconds, an abort event would require the nose section to return to orbit. For an abort event at D+166 seconds, the nose section propulsively separates with a six second full thrust burn. The nose section then reorients to 180^{o} AOA and continues full thrust for another 15 seconds to target the EZ, overshooting the site and landing just inside the 50 km EZ boundary. The later the abort event is triggered the less propellant is required and the closer to the site the aborted crew lands (see the D+470 case in Figure 36).

At D+688 the HSRV is in a completely vertical descent, so if an abort event occurs this late in the lunar descent the nose section simply separates and burns the ATLS for nine seconds, climbing vertically to gain sufficient altitude clear the abort hazard and to redesignate the nose section for a soft landing using the ATLS.

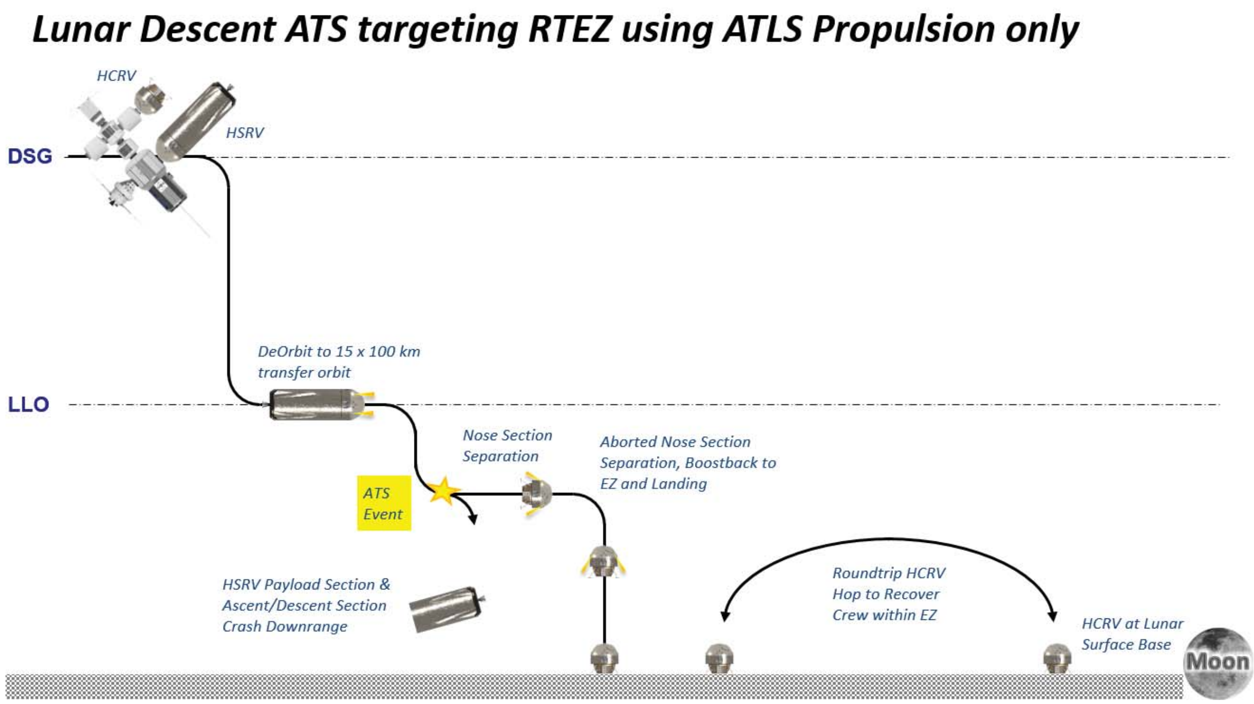


Figure 35. Concept of operation illustrating Lunar descent ATS targeting the RTEZ using the ATLS propulsion capabilities only.

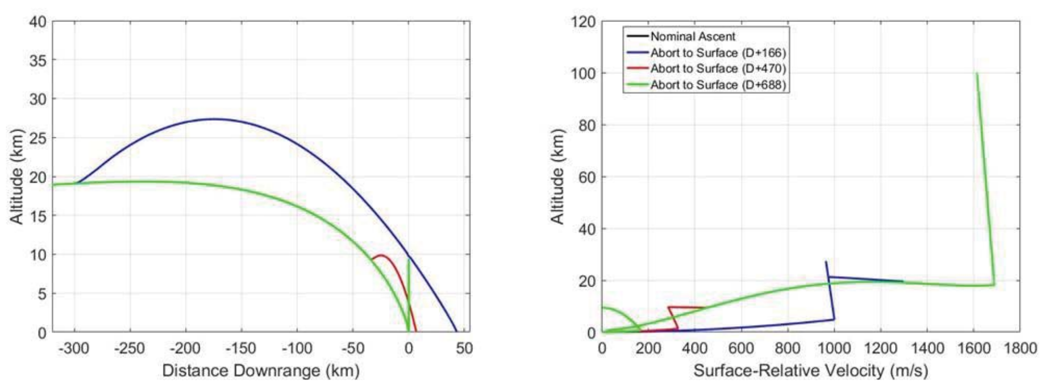


Figure 36. Trajectory plots illustrating Lunar descent ATS targeting the RTEZ using the ATLS propulsion capabilities only.

8 Lunar Ascent Abort

8.1 Lunar Ascent Reference Trajectory

For the lunar ascent reference trajectory, a liftoff mass of 62.4 metric tons is assumed. Thrust is provided by a single ADS engine, resulting in a liftoff T/W of ~0.41 Earth g's.

Figure 37 shows the sequence of events that occur during the lunar ascent trajectory. After liftoff the HSRV rises vertically for 100 meters and then performs a pitch-over maneuver. After 10 seconds the HSRV transitions to 0º

AOA and holds that attitude until the velocity reaches 620 m/s. At this point the HSRV follows a pitch profile that maximizes the mass inserted into the 15 km x 75 km target orbit. The entire ascent trajectory has a duration of 330 seconds and covers ~235 km in downrange. Additional manuevers, not simulated in the POST2 trajectory, target the 100 km circular LLO parking orbit.

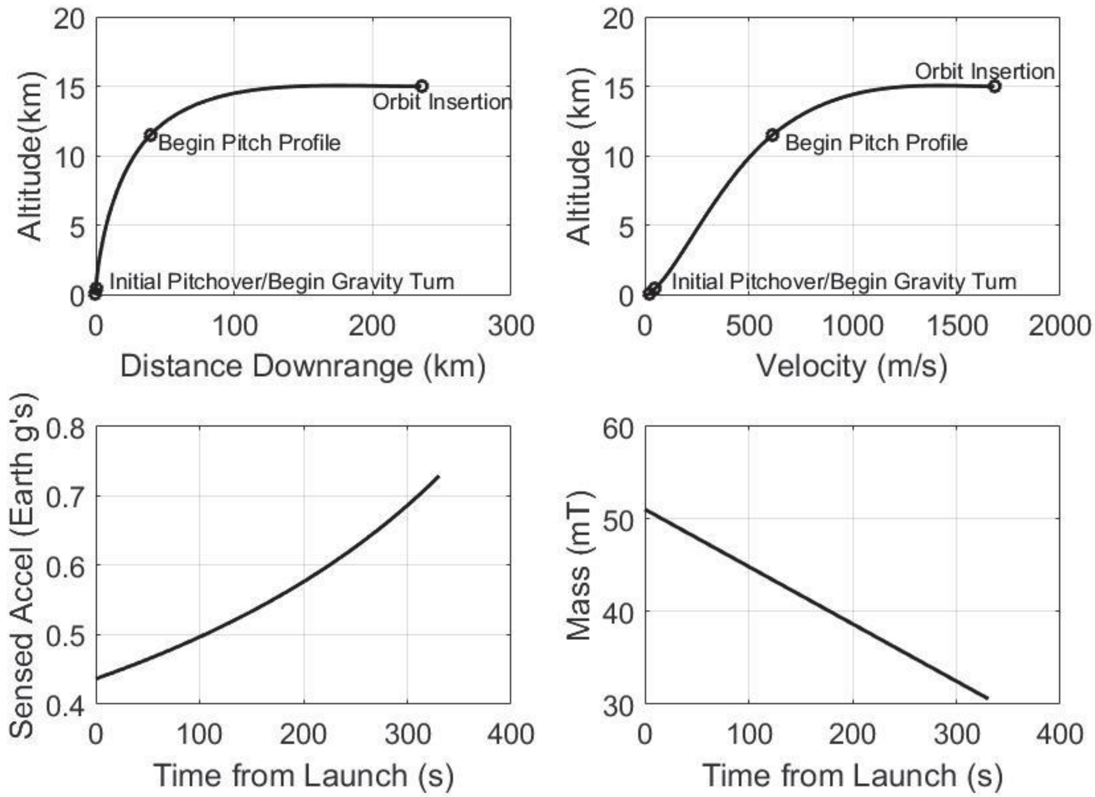


Figure 37. Lunar ascent reference trajectory.

8.2 Lunar Ascent Abort Modes

Figure 38 illustrates the two lunar ascent abort scenarios defined and examined: 1) ATS targeting the RTEZ using the ATLS propulsion capabilities only; and 2) ATO targeting 100 km circular LLO using the ATLS propulsion capabilities only.

For the ATS scenario, as illustrated in Figure 39, the nose section separates, reorients, and accelerates back toward the site, targeting 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, as shown in Figure 41, 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.

Because the nose section has 1,274 m/s of ΔV capability, and this is a much larger fraction of the total ascent ΔV budget (1,809 m/s), there is a large overlap between the ATS and ATO regions. Thus, there is no need for an intermediate DRLZ abort mode. In fact, between T+89 and T+195 seconds an abort event can be either ATO or ATS. This overlap suggests an excess capability in the design.

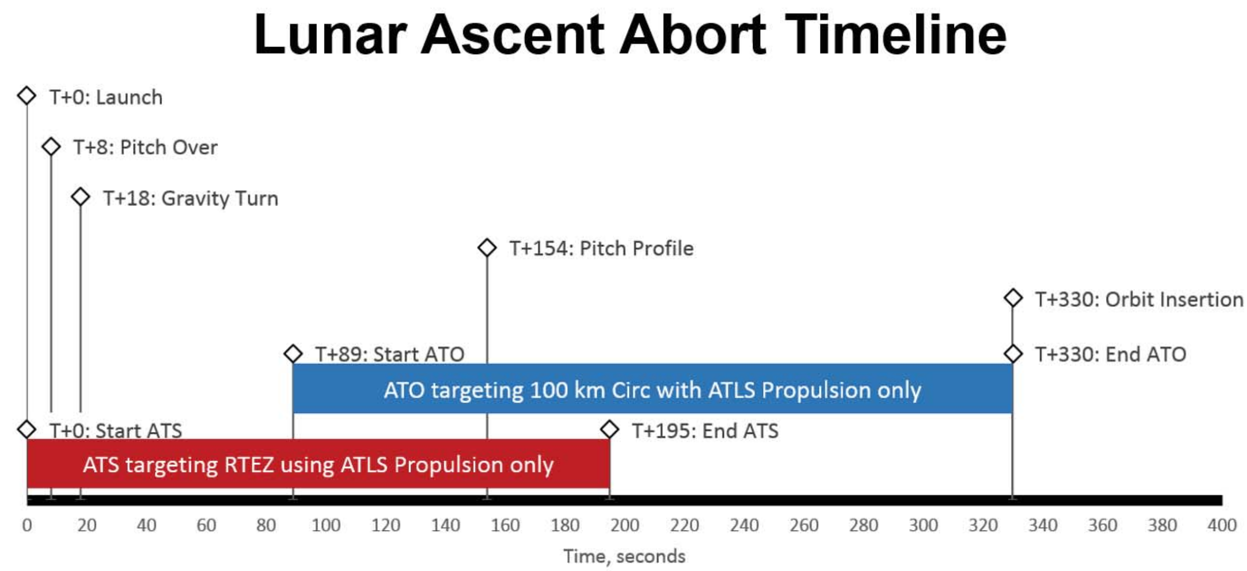


Figure 38. Lunar ascent abort options.

8.2.1 ATS targeting RTEZ using ATLS Propulsion only

The limitation for this scenario is the ability to RTEZ. If the EZ is bigger, the boundary moves later. This is a consideration given that the HCRV, operated on the lunar surface, has a roundtrip range of 125 km. However, preliminary analysis limited the radius of an EZ to 50 km to remain consistent with the approach taken for Mars.

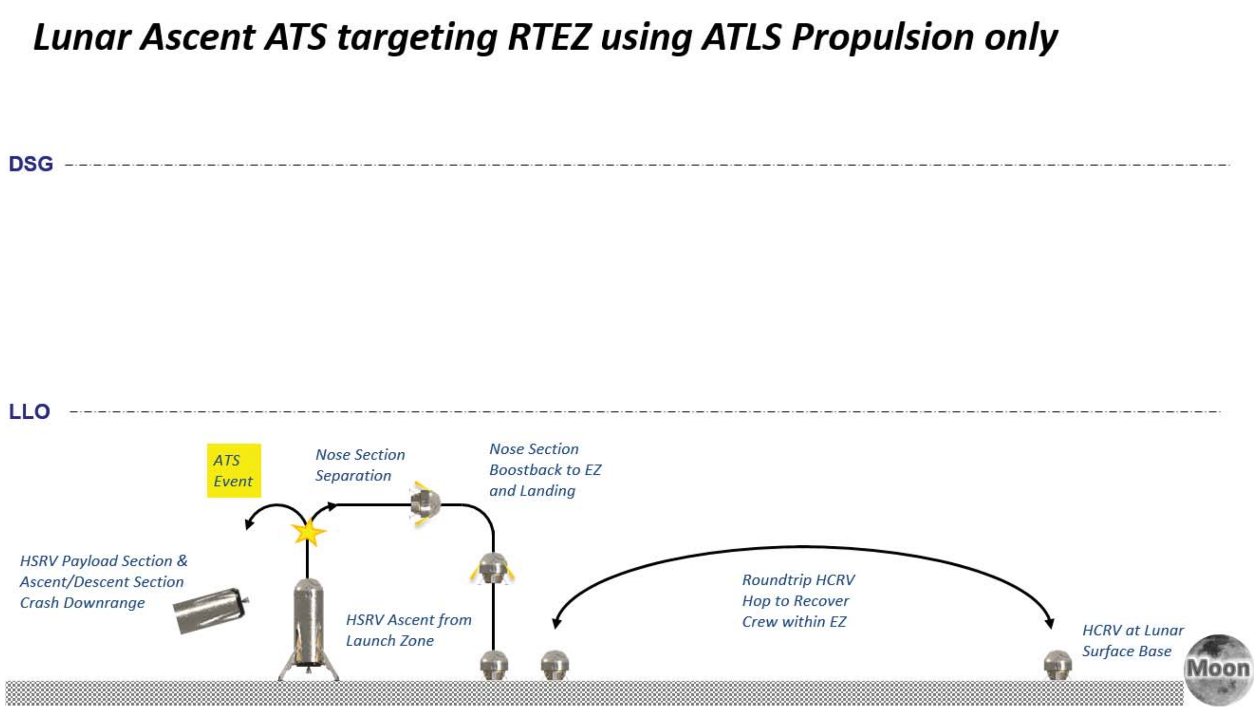


Figure 39. Concept of operation illustrating Lunar ascent ATS targeting the RTEZ using the ATLS propulsion capabilities only.

As shown in Figure 40, the T+1 second case represents an abort event occuring pre-launch or at one second after launch. In this scenario the nose section separates and burns vertically for nine seconds, gaining altitude and clearing the abort hazard. After redesignating to a safe landing location, the nose section falls vertically, re-ignites the ATLS and performs a soft landing.

Later in ascent, after the HSRV gains some altitude, velocity, and is accelerating downrange, an abort event must overcome this velocity moving away from the site. Propellant usage for these scenarios begins to accumulate quickly as the aborted system must decelerate from the abort velocity and then accelerate in the opposite direction to return to the site.

The T+89 second case is shown as it represent a point where either ATS or ATO could occur. Since it is early in the ascent, the preferred scenario from a propellant consumption standpoint is to ATS (as the ATO case at T+89 second, discussed below, consumes all available propellant to enter LLO).

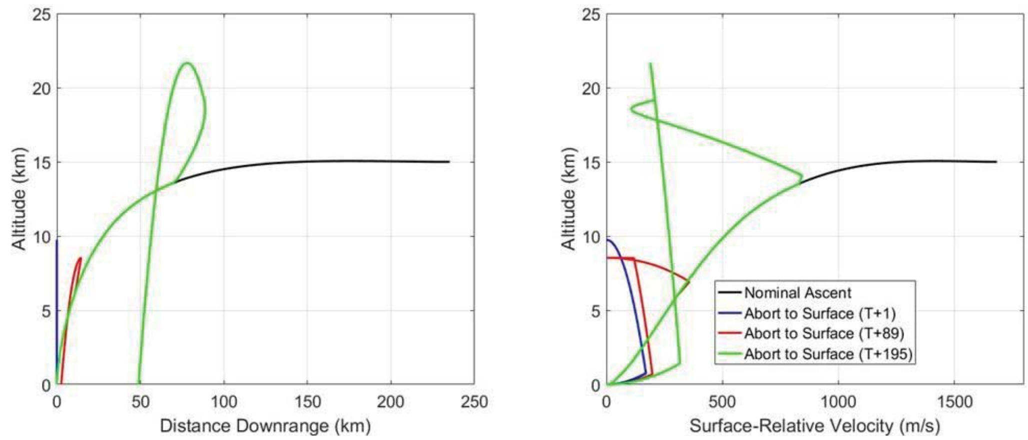


Figure 40. Trajectory plots illustrating Lunar ascent ATS targeting the RTEZ using the ATLS propulsion capabilities only.

The T+195 second case represents the latest time that enables the nose section to RTEZ, landing just inside the 50 km EZ. Following the six second separation burn, the ATLS operates an additional 42 seconds to overcome the ~800 m/s velocity moving away from the EZ and then accelerate back toward the EZ.

8.2.2 ATO targeting 100 km Circular Orbit using ATLS Propulsion only

Prior to T+89 the nose section has insufficient ΔV to achieve the 100 km circular orbit.

The T+89 second case, as shown in Figure 42, represents the earliest time an ATO is possible. The nose section separates and continues to accelerate in the direction of flight, consumable all propellant to enter LLO. In this case the DSG-based HCRV must rendezvous with the nose section in LLO to recover the crew and return them to the DSG.

Anytime after T+89 the nose section can achieve 100 km circular orbit.

At points closer to insertion, such as the T+329 second case, sufficient propellant is available for the separated nose section to reach the DSG, thus the DSG-based HCRV is not required to recover crew.

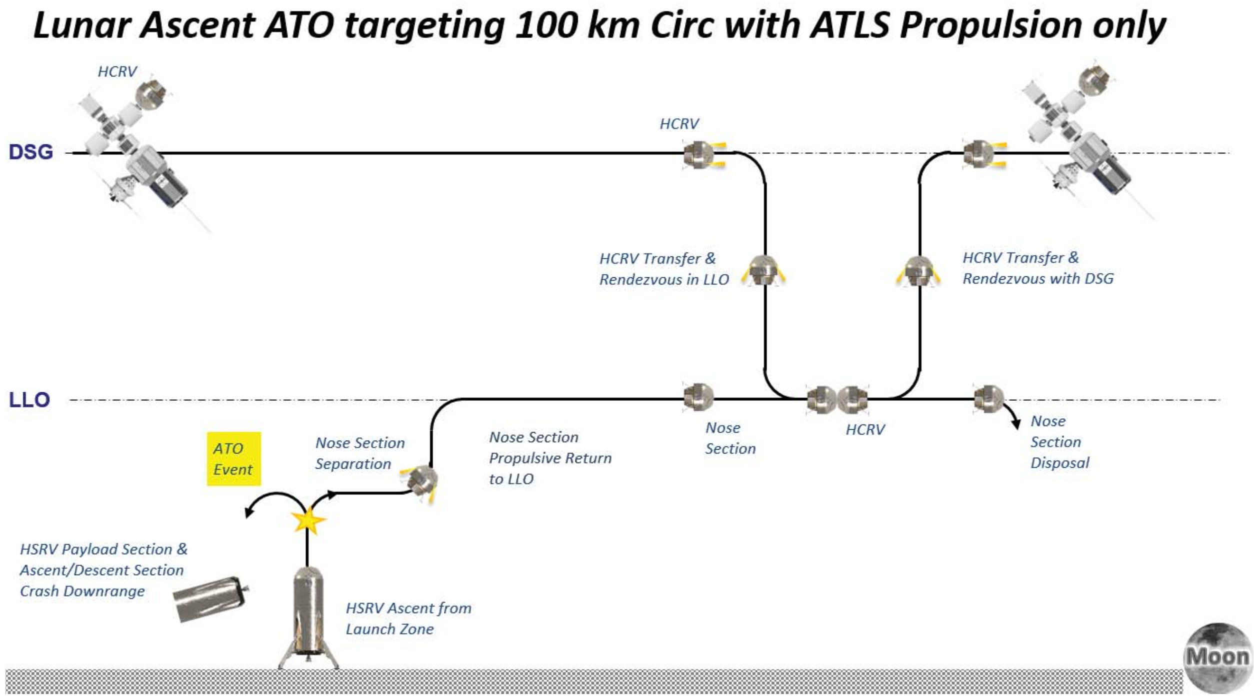


Figure 41. Concept of operation illustrating Lunar ascent ATO using the ATLS propulsion capabilities only.

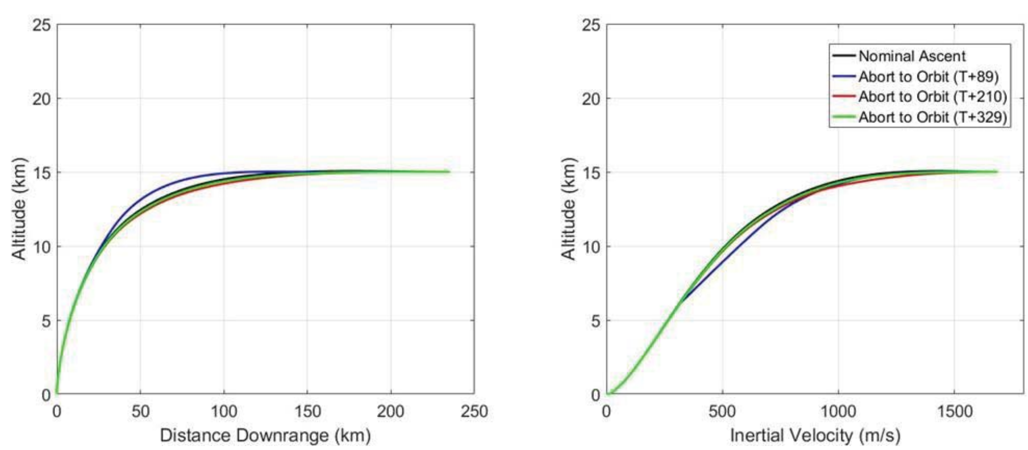


Figure 42. Trajectory plots illustrating Lunar ascent ATO using the ATLS propulsion capabilities only.

9 Summary and Next Steps

This paper presented a preliminary verification and validation of the HSRV's crew abort capability, addressing Mars ascent, Mars EDL, lunar descent and lunar ascent scenarios with either ATO or ATS capability. The design attributes that address operational flexibility and crew safety include the vehicle reusability; modular, separable, and re-purposable vehicle sections; multi-use auxilary propulsion system (i.e. – the ATLS); and separable crew capsule with aero-assist technologies to faciliate safe crew abort. The capabilities are unique to the HSRV design. No other Mars or lunar launch or landing architecture have these capabilities.

With the present assumptions on HSRV design and campaign strategy for Mars, 100% abort and crew recovery coverage is shown for Mars ascent and EDL. Significant overlap of ATO and ATS (for both ascent and EDL) suggest excess capability in the design and an opportunity to optimize ATLS propellant load requirements for Mars missions.

Likewise, 100% abort and crew recovery coverage is shown for lunar descent and ascent. Significant overlap of ATO and ATS (for both lunar ascent and descent) suggest excess capability in the design and an opportunity to optimize ATLS propellant load requirements for lunar missions.

If the HSRV mass properties and/or propulsive characteristic change (as is necessary as the HSRV concept design matures), the boundaries of various abort modes may shift, and the distance to the DRLZ used for Mars ascent may change, but all abort modes (and the 100% coverage feature) should still be valid.

Next steps:

1. Conduct a conceptual design of the crew hopper habitat used for the lunar HSRV and HCRV's.
2. Perform trade study on crew capsule hypersonic and supersonic aero-assist techniques including assessment of HIAD diameter covering a range of 7 to 16+ meters. A 7 meter HIAD requires less mass but increases parachute loads relative to the 10 meter baseline. A 16+ meters HIAD requires more mass but potentially eliminates need for supersonic parachute subsystem.
3. Perform trade study of crew capsule retro-propulsion including assessment of number, thrust, and burn time for solid rocket options including the reference STAR 8 motor and new designs. Also conduct a solid versus liquid monopropellant trade, investigating tradeoffs of mass and complexity against the controllability and long-term storability.
4. Conduct engineering design for the crew capsule, refining and maturing the atmospheric flight trajectories, the capsule subsystems, and suite of aero-assist technologies informed by trades.

10 Conclusions

The HSRV offers an unprecedented degree of operational flexibility and crew safety, including crew abort and recovery, on a common vehicle system designed to support reusable flights to the surface of Mars and the surface of the moon. A unique combination of design features enable the HSRV to offer these abort capabilities, highly desirable by astronauts, but not seriously studied by any Mars or lunar mission architecture until now.

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12 Biography

David R. Komar is an Integrated Architecture, Vehicle Concept & Mission Analysis Lead in the Vehicle Analysis Branch at NASA Langley Research Center with over 27 years experience in space exploration systems design, development, integration and operations. Mr. Komar is the lead concept designer for the Hercules Transportation System over the last 3 years. His technical interests include multi-disciplinary design of space exploration system concepts including advanced propulsion and aeroassist technologies. He has extensive experience developing tools, models and methods for performing conceptual design including the Exploration Architecture Model for In-Space and Earth-to-Orbit (EXAMINE) framework. Mr. Komar received a Bachelor of Science in Aerospace Engineering from the Illinois Institute of Technology in 1990 and a Master of Science in Engineering Management from the University of Central Florida in 1996.

Paul Tartabini is a Flight Mechanics Technical Discipline Lead at NASA's Langley Research Center with over 20 years of experience in flight trajectory analysis and optimization. His technical interests include trajectory optimization and design, end-to-end trajectory simulation, stage separation modeling, and abort system design and analysis. He has extensive experience using the Program to Optimize Simulated Trajectories (POST) to model both three and six degree of freedom trajectories. He has worked a number of flight projects including flight operations for Mars Odyssey, stage separation analysis for Hyper-X, and booster separation and recovery analysis for Ares I-X. Mr. Tartabini holds engineering degrees from Purdue University (B.S. Aeronautical/Astronautical Engineering, 1992) and Virginia Tech (M.S. Aerospace Engineering, 1996) as well as a degree in Engineering Management from George Washington University (M.S., 2000).

James R. Clark is pursuing his doctorate in Aeronautics and Astronautics at the Massachusetts Institute of Technology. Jim received a B.S. in 2014 in Aeronautics and Astronautics and M.S. in 2016 in Aeronautics and Astronautics from MIT. Jim's research areas include laser communication, mission and trajectory design, and advanced concept development. Jim interned with NASA Langley in 2014, Jet Propulsion Labaratory in 2015, and NASA Langley in 2017.

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14. ABSTRACT

The Hercules single-stage reusable vehicle is designed to support crewed missions to both the lunar surface and Mars surface. The design maximizes crew safety by providing full coverage crew abort capability during ascent and entry/descent, either through abort-to-surface or abort-to-orbit. This paper outlines each of the abort modes and discuss the Hercules vehicle design along with the base and orbital infrastructure required to enable the full coverage abort capability. For each abort mode, trajectory simulations are flown that illustrate the requisite design capabilities and highlight the sensitivity to key design variables.

15. SUBJECT TERMS

Aborts; Ascent vehicle; Crew safety; Lander; Operational flexibility; Reusable; Single-stage; Space transportation

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