A High-Heritage Blunt-Body Entry, Descent, and Landing Concept for Human Mars Exploration

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Human-scale landers require the delivery of much heavier payloads to the surface of Mars than is possible with entry, descent, and landing (EDL) approaches used to date. A conceptual design was developed for a 10 m diameter crewed Mars lander with an entry mass of ~75 t that could deliver ~28 t of useful landed mass (ULM) to a zero Mars areoid, or lower, elevation. The EDL design centers upon use of a high ballistic coefficient blunt-body entry vehicle and throttled supersonic retro-propulsion (SRP). The design concept includes a 26 t Mars Ascent Vehicle (MAV) that could support a crew of 2 for ~24 days, a crew of 3 for ~16 days, or a crew of 4 for ~12 days. The MAV concept is for a fully-fueled single-stage vehicle that utilizes a single pump-fed 250 kN engine using Mono-Methyl Hydrazine (MMH) and Mixed Oxides of Nitrogen (MON-25) propellants that would deliver the crew to a low Mars orbit (LMO) at the end of the surface mission. The MAV concept could potentially provide abort-to-orbit capability during much of the EDL profile in response to fault conditions and could accommodate return to orbit for cases where the MAV had no access to other Mars surface infrastructure. The design concept for the descent stage utilizes six 250 kN MMH/MON-25 engines that would have very high commonality with the MAV engine. Analysis indicates that the MAV would require ~20 t of propellant (including residuals) and the descent stage would require ~21 t of propellant. The addition of a 12 m diameter supersonic inflatable aerodynamic decelerator (SIAD), based on a proven flight design, was studied as an optional method to improve the ULM fraction, reducing the required descent propellant by ~4 t.

Nomenclature

C_3	=	characteristic energy
ΔV	=	delta velocity (change in velocity)
g	=	acceleration, in Earth gravity units
Hz	=	Hertz
I_{SP}	=	specific impulse
kg	=	kilogram
km	=	kilometer
kN	=	kilonewton
kPa	=	kilopascal

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kWe	=	kilowatt, electric
m	=	meter
s	=	second
t	=	metric ton

I. Introduction

Human-scale landers require the delivery of much heavier payloads to the surface of Mars than previously attempted, generally considered to be in the range of 15–40 t. Safely landing a Useful Landed Mass (ULM) of this magnitude is unlikely to be achieved by the entry, descent, and landing (EDL) approaches used to date. Almost all of the human-scale EDL concepts currently being considered utilize some type of heatshield system for entry and rely on aerodynamic forces to shed a large percentage of the entry velocity. Most current concepts for large Mars landers are not able to passively achieve subsonic velocities and must use Supersonic Retro-Propulsion (SRP) to perform the final deceleration and a soft landing on the Martian surface.

The key feature of the concept in this study is the use of a high ballistic coefficient 10 m diameter rigid entry body coupled with SRP. Although blunt body entry vehicles are the standard for Mars, all past flight systems have had ballistic coefficients less than 150 kg/m^2 (most less than 100 kg/m^2). These past EDL approaches utilized supersonic parachutes to achieve the relatively low ballistic coefficients required to achieve subsonic descent conditions in the thin atmosphere of Mars. In the present architecture, the parachute supersonic deployment constraint is removed, allowing the bulk of the deceleration to occur at a lower altitude (5 km or less) where the atmospheric density is thicker and less uncertain. Flying nearly-horizontal, at hypersonic speeds, and at low altitude places additional constraints on the trajectory, landing ellipse, and terminal control strategy. Sensing strategies can be developed, however, and trajectories developed and targeted for most sites of scientific interest at or below 0 km areoid. Removal of the supersonic parachute deployment constraint allows the present architecture to carry a much larger entry mass for a given entry body diameter. The use of high thrust-to-weight SRP (initiating between Mach 3 and 4) couples elegantly with this approach. Despite the low altitude at SRP initiation, the flight path is shallow, the descent rate is low, and SRP can arrest the remaining velocity with relatively little gravity loss.

This study presents an example of a possible lander architecture and is intended as input to the NASA human spaceflight planning process.

II. Background, Past Work, and Past Experience

To date, there have been five successful system designs for soft-landing payloads on the surface of Mars: The Russian Mars 3, Viking, the Mars Pathfinder and Mars Exploration Rover (MER) system, Mars Phoenix, and the Mars Science Laboratory (MSL). They all used heatshields, parachutes, and rockets to land on the surface. Viking and Phoenix had traditional landing legs. Mars 3 used shock absorbing foam and petals to right the lander. The Pathfinder/MER design used airbags and petals. MSL used a rocket propelled descent stage to gently set the rover down on its wheels. The largest of all of these systems was MSL, which is being reused for the Mars 2020 rover and is capable of delivering an approximate 1 t ULM payload to a 0 Mars areoid surface elevation.

Studies by Braun[1] and Steinfeldt[2] have indicated that the traditional EDL approaches used in Mars landers to date have a performance limit of about 2 t ULM. Performance analyses by Steinfeldt also indicate that blunt-body landers should be more mass effective than slender-body landers. Design studies have been performed by Woodcock,[3] Christian,[4] and others for blunt-body landers that could deliver human-scale payloads. They have typically been 10-m diameter or larger, do not require the use of parachutes, and use SRP for final braking. The results of this study are consistent with, and in a similar performance range to Woodcock[3] and Christian.[4]

III. Assumed Lander Capability

The lander concept for this study was designed to have the following key features and capabilities:

- A 10 m diameter blunt-body entry vehicle that would launch in a slight hammer-head configuration on the SLS
- 75 t entry mass, which is based on the performance capability for two SLS launches to inject a payload to Mars on a conjunction-class trajectory
- An ogive-shaped lander backshell that would have a dual use as the launch fairing on the SLS
- The lander would use aerocapture to enter High Mars Orbit (HMO) where it would loiter for an extended time awaiting the arrival of the crew in a separate vehicle
- Able to support a crew of 2 for 24 days, a crew of 3 for 16 days, or a crew of 4 for 12 days in the MAV

- A non-cryogenic biprop system with multiple pump-fed engines for descent stage propulsion
- A simple single-stage non-cryogenic high-heritage biprop system with a single pump-fed engine for Mars Ascent Vehicle (MAV) propulsion
- A fully-fueled MAV that would allow for abort-to-orbit capability both during EDL and after landing, without requiring interaction with any Mars surface infrastructure
- Ascent is to Low Mars Orbit (LMO) where the MAV docks with a boost stage in LMO to return to HMO
- An aerodynamic MAV moldline with a hinged nosecone to protect the docking system from dust and debris
- The MAV has no airlock, and egress and ingress is performed in a similar manner to the Apollo LEM
- The lander could also be configured as an uncrewed cargo delivery system that could provide a habitat, pre-deployed supplies, and ground transportation for longer duration stays

IV. Assumptions for Mars Ascent Vehicle

For this study, a MAV concept was developed for a fully-fueled single-stage vehicle that could deliver the crew from the Martian surface to a circular LMO with an altitude of about 300 km where it would rendezvous with a pre-deployed boost stage that would lift the MAV and its crew to HMO. The lower orbit for the MAV requires about 4.2 km/s of ΔV. Mono-Methyl Hydrazine (MMH) and Mixed Oxides of Nitrogen (MON-25) were selected as the propellants with the advantage of high density liquid storage in an easy-to-maintain temperature range. A single pump-fed 250 kN gimbaled engine, based on a scaled-up RS-72 engine, was chosen as a design point. This would be a new engine development. The rationale for this approach was to provide a simple, low-risk vehicle that could enable the crew to abort to orbit during much of EDL and after landing, without requiring the support of other surface assets. The MAV concept is similar to an approach previously described by Geels[5].

In order to provide a realistic basis for mass estimates, an existing propulsion system—the Russian Briz-M stage—was chosen as a reference, but with its 19.6 kN S5.98M main engine replaced with the conceptual 250 kN engine described above. The propellant loading of 19.8 t used for the MAV concept is the same as the Briz-M capacity.

The crew cabin for the MAV assumed a 3.2 m diameter by 3.0 m tall pressure vessel with equipment and provisions that could support a crew of 2 for 24 days, a crew of 3 for 16 days, or a crew of 4 for 12 days. The seating would be reclined and on a single level. The total mass of the crew cabin was estimated to be ~3.3 t. An additional 2 t of equipment and supplies for the crew was included in the design of the descent stage, but this would be unavailable after the end of the surface phase of the mission. The MAV concept, depicted in Fig. 1, is aerodynamic in shape, with a hinged nosecone that protects the forward docking ring and hatch during all mission phases. The nosecone would be swiveled ~150° out to the side to enable docking and crew transfer operations.

The functioning of the MAV would be similar to the Apollo Lunar Module Ascent Stage. There would be no airlock, so the crew would egress and ingress on the surface in a manner similar to Apollo. For a standalone short-stay surface mission, the crew would utilize the MAV cabin as a small surface habitat, probably with a crew of 2 or 3. For long-stay surface missions, the crew would transfer to a pre-placed large habitat (delivered in advance by a separate and identical EDL system) and would return to the MAV only for periodic checkouts or maintenance and for the final ascent from the surface.

The total wet mass of the MAV was estimated to be ~26.3 t. The mass breakdown is provided in Table 1. A specific impulse (I_{SP}) of 338 s was used in the ascent analysis. Although this is a high-performance value, the I_{SP} for both the RS-72 engine and the Russian Dnepr third stage engine is 340 s. Both are pump-fed hydrazine/NTO class engines.

Figure 1. Mars Ascent Vehicle

Table 1. Mars Ascent Vehicle Mass List

Component	Mass (kg)	# of Units	CBE (kg) Dry	Growth Contin-gency	CBE+Contin-gency (kg) Wet	Description/Comments
Mars Ascent Vehicle (MAV)			5,239		26,292	
Crew Cabin			1,661		3,308	
Structure (pressure vessel)	680	1	680	50%	1,020	Final Altair ascent cabin structure mass w/o prop. tank sppt.
Consumables (by crew member days)	7	48	336	30%	437	H_2O, O_2, food, etc. for 2 for 24 days. Waste jettisoned for ascent.
Crew accommodations	125	2	250	30%	325	Incl. seats, EVA suits, etc.
Crew	85	2	170	0%	170	
Science equipment (add'l. science on descent stage)	50	1	50	0%	50	Incl. cameras. Mass is traded for sample mass on ascent.
ECLSS	175	1	175	30%	228	Additional ECLSS is on descent stage
Other subsystems			830	30%	1,079	
Propulsion			2,748		22,809	
Structure, tanks, and propulsion components	2,275	1	2,275	5%	2,389	Briz-M actual
Main rocket engine	438	1	438	30%	569	Scaled-up RS-72 engine; assumed Isp=338
Briz-M mods for larger main engine	35	1	35	50%	53	Incl. gimbal mass increase and higher flow rates
Unusable propellant					1,121	Assuming improvements in Briz-M to achieve 6% residuals
Usable propellant					18,678	Provides 4.2 km/s ΔV for ascent
System Margin					174	Margin on dry mass

V. Description of Lander Concept

The lander concept is a 10 m diameter traditional blunt-body entry vehicle with a heatshield similar to the MSL design (see Fig. 2). The vehicle incorporates an ogive-shaped backshell that would provide aerodynamic and thermal protection during Earth launch, aerocapture, and entry. Although the backshell's planform is different, its construction would be similar to that of the MSL composite backshell. In this concept, the backshell would also serve an additional function as the payload fairing for launch on an SLS block 2 launch vehicle. Fig. 3 presents a size comparison between the 10 m lander concept and several blunt-body entry vehicles that have previously flown.

The lander concept considered in this study did not utilize a separate cruise stage to support the lander in its transit to Mars. A restowable solar array would be deployed through a hatch in the backshell to provide power during cruise. It would be restowed for aerocapture and then redeployed after aerocapture. An array size of ~5 kWe was sized for this study. Low-gain omnidirectional antennas protruding through the backshell would provide telecommunication coverage. Reaction Control System (RCS) thrusters firing through holes in the backshell would provide 3-axis attitude control.

The top portion of the backshell would serve a tertiary function as the nose cone of the MAV during its ascent to Mars orbit. The MAV nose cone would require mechanisms and structural load management features to enable that tertiary function. The nose cone would also be hinged to enable rotating it ~150° out to the side in Mars orbit to provide access to the docking port at the forward end of the MAV for crew transfer. After crew transfer, the nose cone would be rotated back to the closed position to protect the docking port from dust and debris.

In addition to the heatshield and backshell, the major systems of the lander are the descent stage and the landed payload. This system would be capable of delivering a wide range of payloads to the Mars surface. For this study, the landed payload is the fully-fueled MAV, as shown in Fig. 2. The descent stage is a conical structure that supports the payload, descent stage propulsion system, landing gear, avionics (including EDL sensors and instrumentation), deployable solar arrays, and batteries. As conceived, the descent stage would also carry additional equipment to support the crewed surface mission that would not be required for the ascent phase. This includes crew consumables, power generation and storage, science equipment, and possibly a technology demonstration ISRU unit to generate

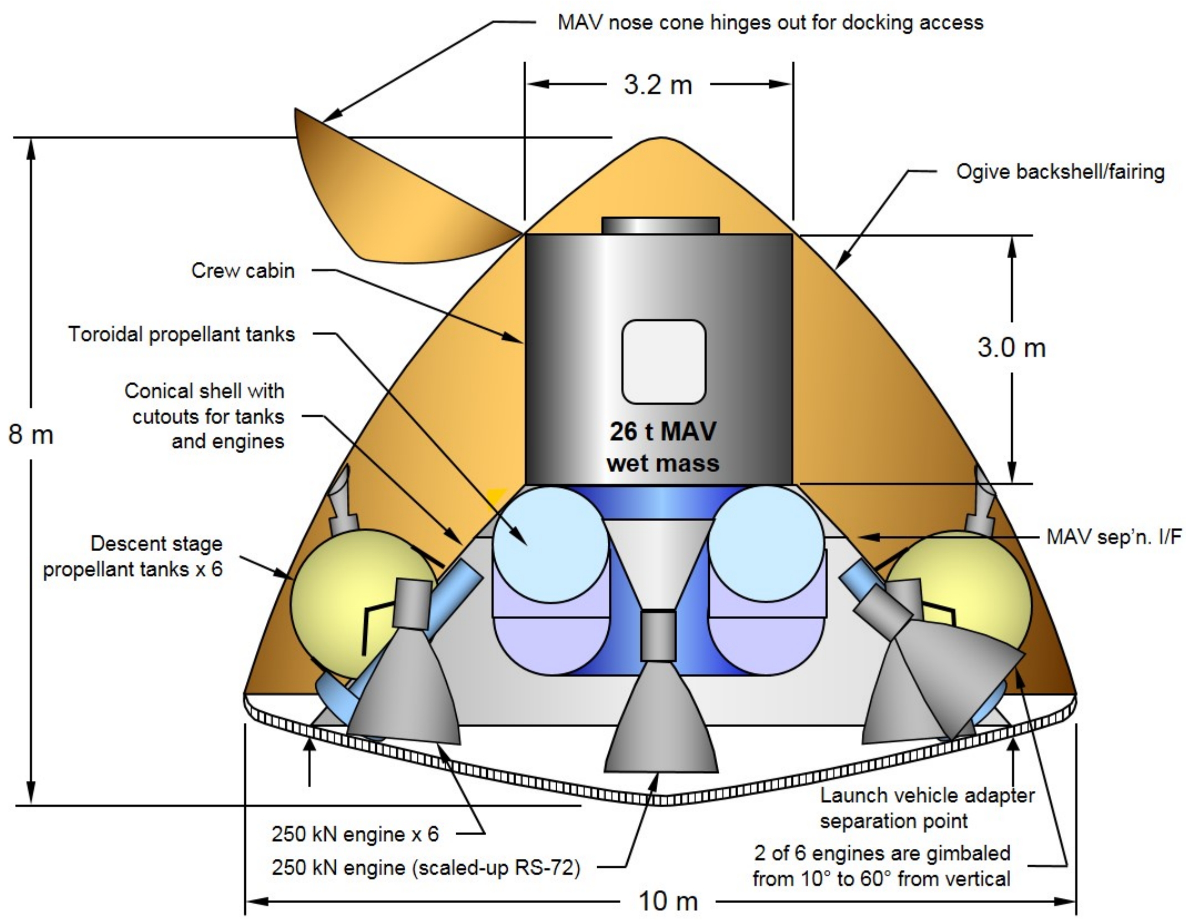

Figure 2. Mars lander concept with MAV.

supplemental breathing oxygen from the Martian atmosphere. An unpressurized rover could also be included, which might be equivalent to a modernized version of the Apollo rover.

The EDL sequence adopted for this study is depicted in Fig. 4. The vehicle would enter the Martian atmosphere using its heatshield in a manner similar to previous Mars landings. As the lander was slowed by aerodynamic forces, it would go through a phase of peak heating with a deceleration of about 6 g. The time span for pure aerodynamic breaking would be about 3 minutes.

At about Mach 3.5, the backshell would be jettisoned, and six 250-kN rocket engines would be ignited to begin the SRP phase of descent. The heatshield would have mechanisms to open six areas in the heatshield for the engines to fire through. These could be hatches that slide out of the way or plugs that are blown out by the engines.

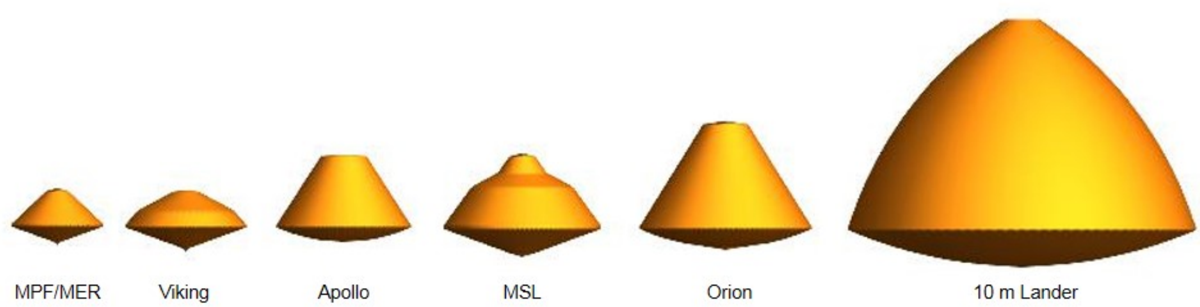

Figure 3. Size comparison of blunt-body entry vehicles.

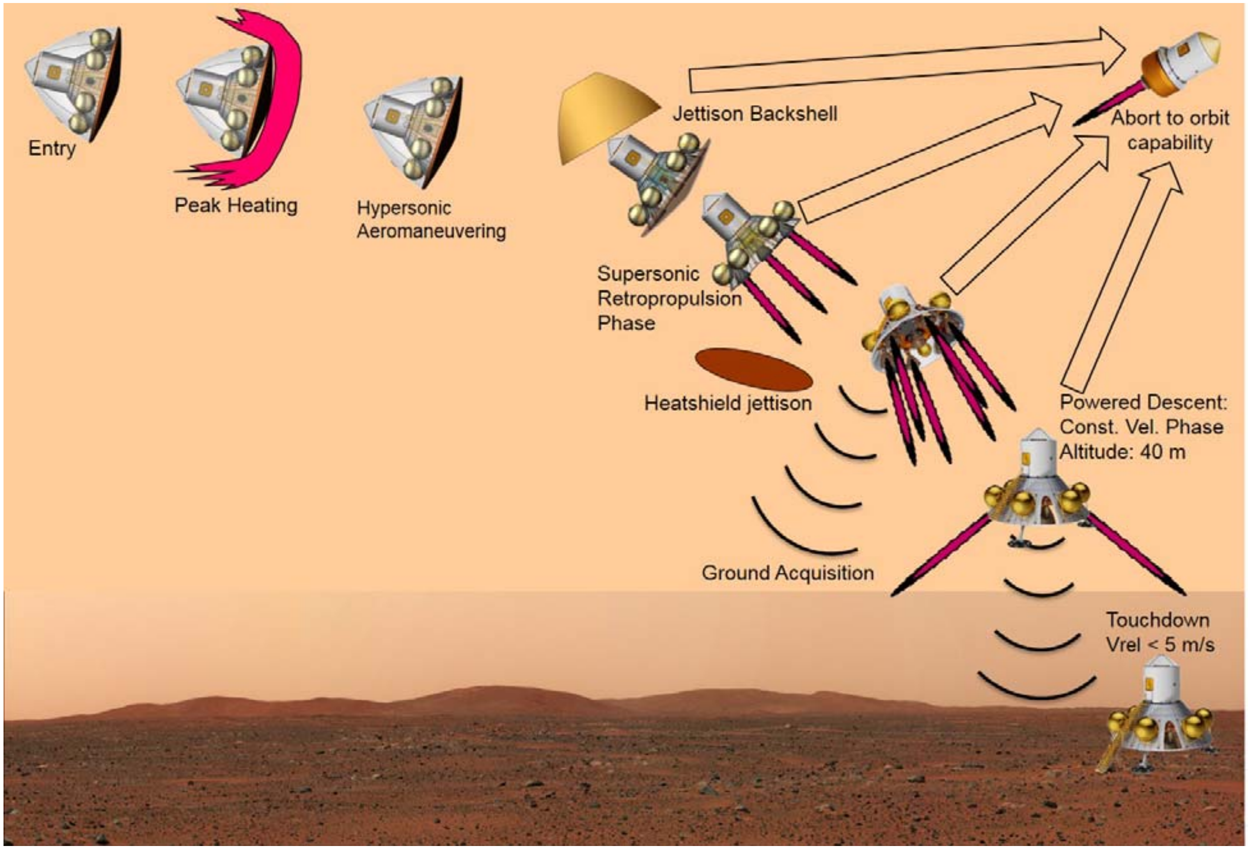

Figure 4. Entry, descent, and landing sequence.

The vehicle would be decelerated and steered toward the designated landing site using Terrain Relative Navigation (TRN). At Mach 1.8 or lower, the dynamic pressure would be low enough to jettison the heatshield. When the vehicle reaches a target altitude of 40 m, four of the engines would be shut down, and the two remaining engines would be throttled down and gimbaled outward to an angle of about 50° from vertical. This is done to reduce the thrust-to-weight ratio to slightly less than 1 and allow for a constant velocity phase final descent to the surface, with terminal guidance to meet a precise target.

Gimbaling the engines out to a large angle for terminal descent provides other benefits. It should significantly reduce, if not eliminate, soil and surface erosion directly underneath the vehicle and blow much of the debris out and away rather than up toward the lander. The lander can be clocked to a preferred orientation in order to blow debris in directions away from nearby surface assets. Gimbaling the engines outward can also provide a clear downward field of view for terminal guidance sensors, so as not to be obscured by the rocket plumes.

Thrust would be terminated at or just before touchdown, ensuring a final touchdown velocity under 5 m/s. The landing gear reference design for this concept has telescoping tubular legs that deploy the footpads to a distance of about 2 m below the bottom of the main structural cone, providing large ground clearance and a long stroke for the shock absorbing system. The powered phase of the descent would be less than 1 minute long.

Since the MAV is fully fueled and can provide ~4.2 km/s of ΔV, it is conceivable that it could provide abort-to-orbit for most of the failure modes that might be encountered after the lander is past the peak heating portion of EDL. For example, in the event of a descent engine failure, the lander should be able to reorient to a MAV-forward flight. At that point, descent stage propulsion would be shut down, and the MAV would be ignited, separated from the lander, and used to carry the crew back up to LMO for eventual rendezvous and docking with the pre-placed orbital transfer stage.

On the Martian surface, the descent stage would provide power resources for the delivered payload with ~25 kWe (at Mars) solar arrays based on those being developed for the Asteroid Redirect Mission (ARM) and batteries to maintain ~5 kWe through the Martian night.

The total wet mass of the descent stage is estimated to be 47.4 t, and the mass breakdown is listed in Table 2. It should be noted that ~5 t of Reaction Control System (RCS) propellant is planned to be depleted prior to EDL for performing en route maneuvers. The packing density inside the aeroshell is in family with previous blunt-body entry vehicles, as shown in Fig. 5.

Table 2. Mars Lander Descent Stage Mass List

Component	Mass (kg)	# of Units	CBE (kg) Dry	Growth Contin-gency	CBE + Contin-gency (kg) Wet	Description/Comments
Descent Stage			19,557		53,820	
Structure			10,304		13,276	
Primary structure	3,931	1	3,931	30%	5,110	Based on FEM analysis. Incl. prop. tank sppt.
Backshell	3,157	1	3,157	30%	4,105	
Entry heatshield (incl. TPS)	2,816	1	2,816	30%	3,661	FEM using scaled-up MSL model (HEEET might replace PICA)
Heatshield ballast	400	1	400	0%	400	
Propulsion			4,666		33,670	
Propellant Tanks	46	6	277	30%	360	Low pressure spherical Ti tanks
Main rocket engines	438	6	2,628	30%	3,416	Scaled-up RS-72; Isp=325 (derated for SRP)
Main engine assy. lines and fittings	55	1	55	30%	72	
Engine gimbal assy.	15	2	30	30%	39	
HP gas propulsion components	49	1	49	30%	64	
LP gas propulsion components	92	1	92	30%	120	
LP liquid flow components	895	1	895	30%	1,164	
Pneum. purge, back-flush components	84	1	84	30%	109	
RCS tanks and components	444	1	444	30%	577	
RCS thrusters	7	16	112	30%	146	
RCS propellant (1,000 kg left at EDL)					6,250	200 m/s TCMs, aerocapture raise, & de-orbit + 1,000 kg for EDL
He and gas generator propellant					55	
Usable main engine propellant					20,000	
Unusable main engine propellant					1,300	6.5% incl. tank & line residuals and mixture ratio reserves
Mechanisms			2,282		3,027	
Landing gear			1,682	30%	2,187	CBE scaled from LM (222 kg) for landed mass and gravity
Upper cone MAV support release	100	1	100	30%	130	to remove conical shell structure required for separation clearance
L/D control flap mechanisms	100	1	100	30%	130	Mass of flaps is included in backshell structure
Rocket engine port mechanisms	50	6	300	50%	450	To enable firing engines through the heatshield
Hatches for redeployable HGA & array	10	2	20	30%	26	Mass of hatches is included in backshell structure
Release mechanisms	5	16	80	30%	104	Heat shield, backshell, MAV staging, etc.
Other Components			2,305		2,847	
Batteries	42	15	630	30%	819	15 x 5 kWh advanced Li ion batteries
Surface solar arrays & booms	220	2	440	30%	572	Based on ARRM; 50 kWe @Earth (25 kWe @Mars)
Restowable solar array & boom	125	1	125	30%	163	For use during cruise and Mars orbit
ECLSS to support MAV prior to ascent	200	1	200	30%	260	
Cabling	150	1	150	30%	195	
Thermal control	200	1	200	30%	260	
Landing sensors	10	2	20	30%	26	LIDAR, camera, etc.
Science equipment	400	1	400	0%	400	Allocation (could incl. advanced Apollo-type rover)
Super Moxie	100	1	100	0%	100	For generating supplemental crew O_2 (allocation)
Electronics	20	2	40	30%	52	REUs, power conversion & switching, etc.
System Margin					1,000	Margin on dry mass

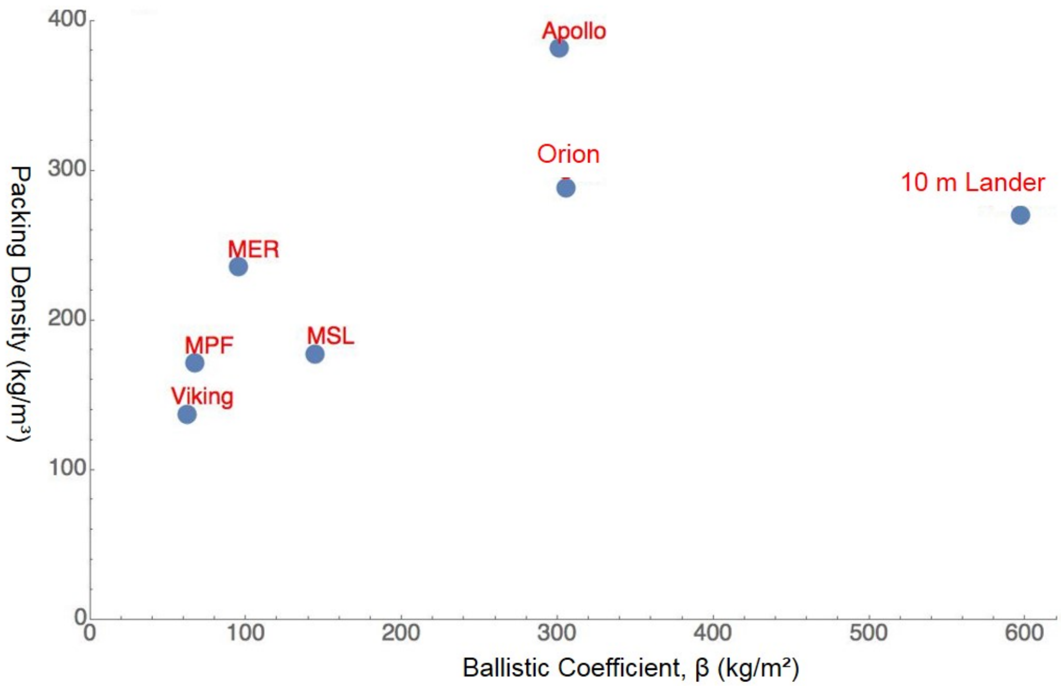

Figure 5. Packing density for blunt-body entry vehicles.

VI. Propulsion Concept

The propulsion system concept for the descent stage is a traditional pump-fed biprop design with six 1.8 m diameter spherical titanium tanks and six 250 kN engines based on similar technology to the RS-72 engine or the second stage engine of the Dnepr launch vehicle. The lander requires a deep throttling capability to be able to perform high-thrust SRP and also be able to perform terminal descent and landing with a thrust-to-weight capability of less than 1. Throttling is achieved through a combination of three methods: (1) The new-design 250-kN engine would have dual injectors to allow 50% throttling; (2) terminal descent would be on only two of the engines, providing another 67% reduction in thrust; and (3) the two terminal engines would gimbal out to an angle of up to 60°, providing additional throttling through the cosine loss of up to 50%.

The lander propulsion would use MMH and MON-25 propellants, because they have lower freezing points than other formulations of hydrazine and nitrogen tetroxide (NTO) and would provide more manageable thermal control on the surface of Mars, requiring less heater power during the night. The conceptual propulsion design includes a helium pneumatic purge and backflush subsystem to clear the pumps and lines of hypergolic propellants after the engines are shut down. The subsystem would push the holdup propellants back into the propellant tanks. Any residual propellants left in the engines downstream of the valves should rapidly evaporate, in a similar manner to the Apollo LM, and not present a hazard to the crew during Extra Vehicular Activity (EVA).

The lander Reaction Control System (RCS) would be a separate biprop system with its own tankage. Although it would use the same propellants as the descent propulsion, the heritage RCS thrusters are pressure fed and require a higher regulated pressure than the pump-fed descent propulsion system can support. The RSC thrusters would fire out of holes in the backshell and provide 3-axis attitude control for the vehicle. The RCS thrusters would also be used for in-space Trajectory Correction Maneuvers (TCMs), periapsis raise after aerocapture, and the deorbit burn for landing on Mars.

As described in Section IV, the MAV propulsion system concept used for this study was based on the existing Briz-M stage with very few modifications. The 19.6-kN S5.98M main engine would be replaced by a single 250 kN engine that would be a slightly modified version of the engine design used on the lander descent stage. The modified engine would not require the throttling capability of the descent engines.

VII. Structural Design Concept

The core structure concept for the lander descent stage is an aluminum conical shell with reinforcing longerons that carry the launch loads from the top of the 4.1 m diameter MAV propulsion system to an 8.4 m diameter cylindrical adapter that carries the loads to the 8.4 m diameter Exploration Upper Stage (EUS) on the SLS. As shown in Fig. 6, the descent structure cone attaches to the cylindrical adapter through 12 release bolts that are mounted to 12 hard points in the heatshield. The heatshield is supported by the 12 interface points and carries the loads of the backshell, which is mounted to its perimeter. The heatshield is also supported by a matching contoured pad in the boat tail portion of the adapter that provides an aerodynamic fairing between the 8.4-m diameter base of the adapter and the 10-m diameter perimeter of the heatshield and backshell.

The six spherical propellant tanks for the descent stage are supported by bipods that are attached to the longerons on the conical core structure. In order to fit within the space limitations inside the backshell, the tanks protrude into the conical shell through holes in the structure that are provided to accommodate the tanks. The six rocket engines are mounted to reinforced holes in the conical structure. Two of the engines, on opposite sides, are gimballed, and slots are cut into the conical shell to enable the engines to gimbal out to 60° from vertical. In order to stiffen the cone, the slots are closed at the bottom by beams, and the beams are separated and jettisoned just prior to the engines gimballing outward.

The descent stage structure was analyzed and sized with a Finite Element Model (FEM) as shown in Fig. 7. Load cases were assessed for launch, maximum deceleration on entry, and peak descent engine thrust. The lowest frequency mode was the vehicle lateral load at launch at 8.5 Hz.

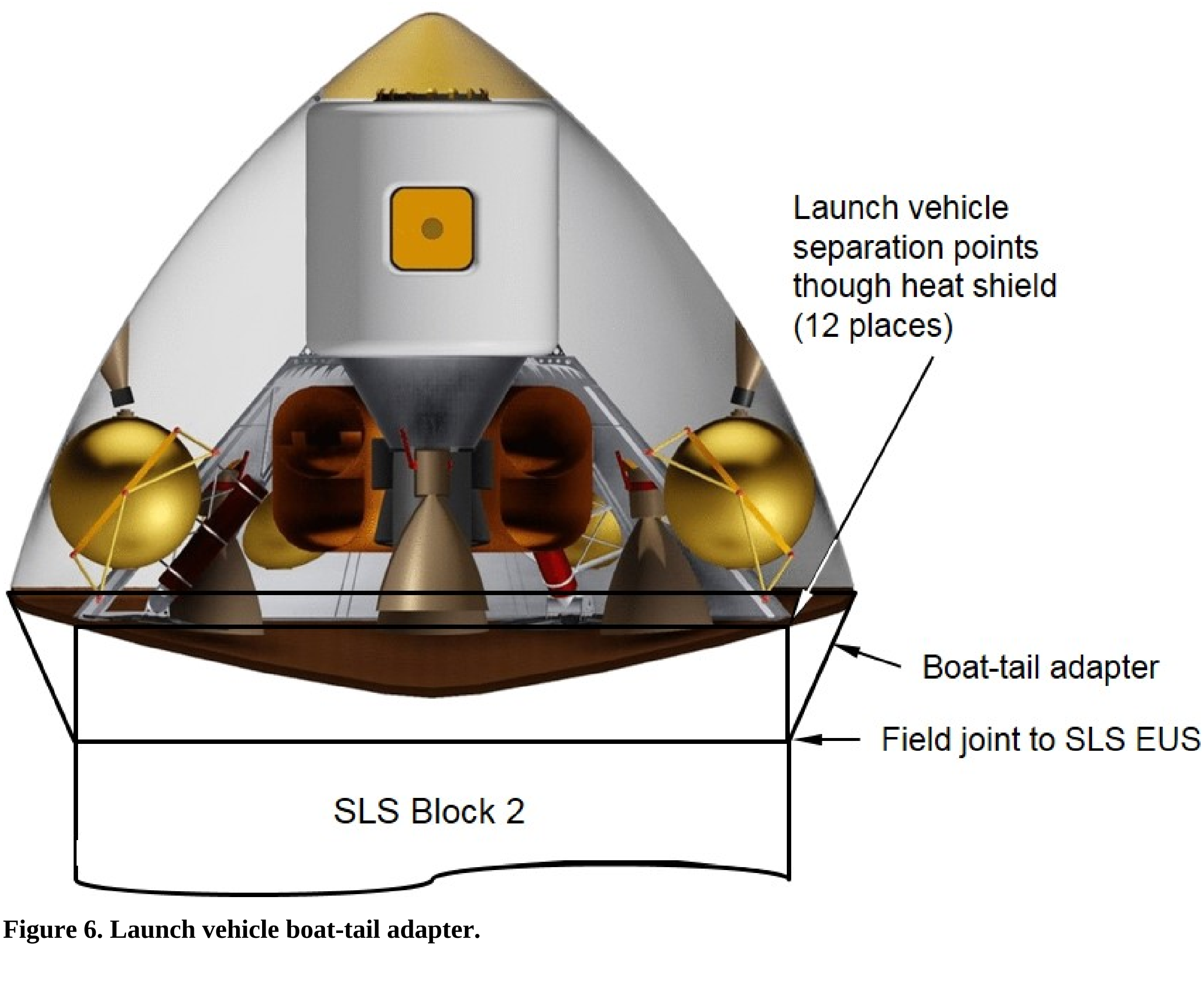

Figure 6. Launch vehicle boat-tail adapter.

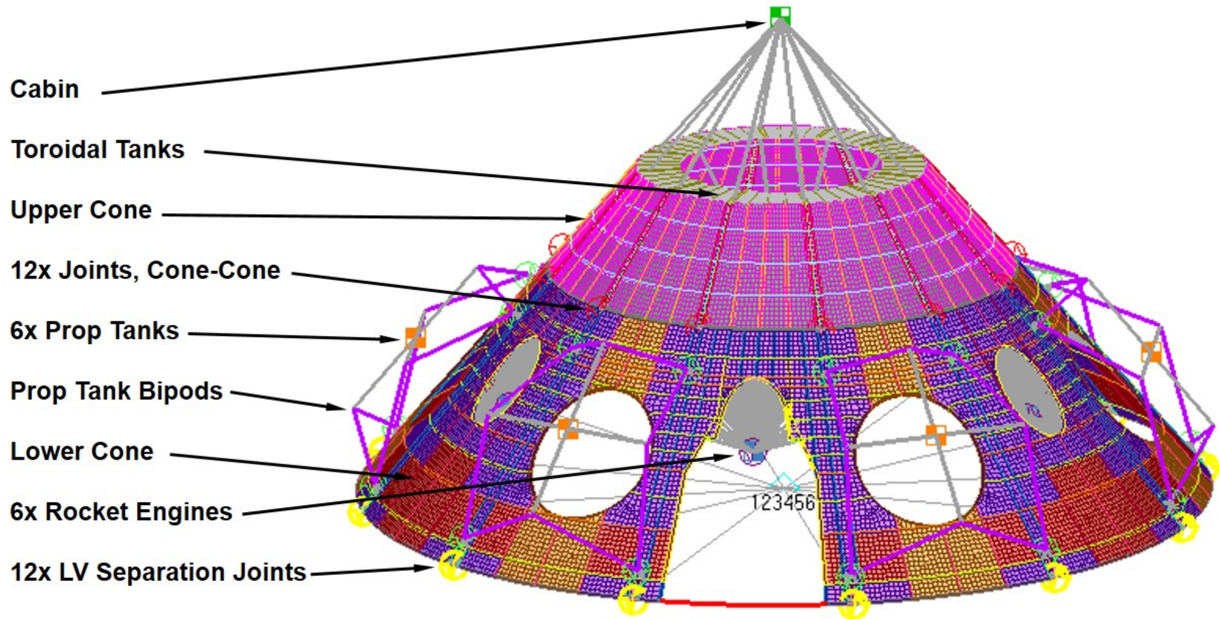

Figure 7. Descent stage structure Finite Element Model.

VIII. Heatshield and Backshell Concepts

The heatshield concept is architecturally similar to the MSL composite heatshield, except that passthroughs for transfer of the launch loads will be required (similar to terrestrial capsules like Apollo, Orion, and Dragon). The MSL FEM was scaled up and used to analyze the heatshield in this study for a design case loading of 65 kPa. It would be a composite structure with PICA thermal protection system (TPS) with high heritage to MSL. There are more advanced TPS materials under development, and a more advanced TPS could be on-boarded into the design as it becomes qualified and available. The heatshield would be a dual pulse system capable of performing aerocapture into HMO and then, a number of months later, performing EDL with crew. If analysis and test shows that the dual pulse approach presents a risk to the crew, then a feasible fallback would be to include a separate aerocapture heatshield contoured to fit in front of the EDL heatshield. The aerocapture heatshield, in that case, would be jettisoned after the orbit insertion pulse, exposing a pristine heatshield that would be used for EDL.

The backshell concept is based on the MSL backshell design, but it is scaled up in size and is an ogive shape rather than a dual conical shape. For MSL, the backshell carried launch loads for the lander system, but in this case it would only be carrying its own weight. It would, however, be subjected to aerodynamic loading during launch on the SLS. It would also require acoustic blankets to enable it to serve a dual function as the launch payload fairing. The backshell would require access doors for launch pad access and also incorporate motor-driven reclosable hatches to deploy and retract a solar array and communication antennas during both cruise to Mars and orbital operations at Mars.

IX. Supersonic Retro-Propulsion

In this investigation, the propulsive capability currently utilized during subsonic descent is extended to supersonic initiation velocities (i.e. supersonic retro-propulsion). SRP descent architectures offer the ability to land larger payload masses while providing additional control authority (thrust-vectoring and throttling) throughout the descent. SRP scales well across large-scale robotic and human exploration missions and affords both cost and technology feed-forward benefits for large-scale missions. As entry vehicle and landed mass requirements increase, the benefits of SRP become more significant, while the use of alternative decelerator technologies become more challenging.

SRP was identified as a technology investment area in NASA's *Space Technology Entry, Descent and Landing Roadmap* and was recently cited as a high priority in the National Research Council (NRC) Life and Physical Sciences Survey, *Recapturing a Future for Space Exploration: Life and Physical Sciences Research for a New Era*.

In addition, SRP was identified as a "critical path technology" and baselined in a large number of the NASA Mars EDL systems analysis concepts.

Initially studied in the 1960s, interest in SRP technology has been recently renewed.[6] Technology efforts from 2005 to 2012 focused on gaining a fundamental understanding of aerodynamic-propulsive fluid dynamic interactions with cold gas plumes at supersonic freestream conditions. Systems analysis, computational fluid dynamics (CFD) simulations, and small-scale air-in-air wind tunnel testing were also performed in this timeframe.[7,8] Blunt body aeroshell configurations, similar to the concept baselined in this investigation, have been the focus of these efforts. Flight dynamics simulations have demonstrated that SRP initiation generally occurs at a minimum altitude boundary subject to subsequent timeline constraints, with resulting high values of thrust. SRP can also be utilized as additional control authority for precision landing.[9,10] CFD tools have been shown to be capable of capturing major flowfield features, including unsteadiness, albeit at considerable computational expense.

From 2013 to 2015, through a partnership with Space X, NASA received its first insight into the performance of a flight-qualified propulsion system operated into an opposing supersonic freestream. These efforts focused on analysis of Space X first stage recovery flight data. To return this launch stage safely to Earth, operation of its propulsion system in the supersonic regime at the right altitudes on Earth to yield Mars-relevant conditions is required. To date, Space X has performed SRP maneuvers during recovery operations of seven Falcon 9 first-stage systems. NASA personnel have independently reviewed these data sets. Multiple flights are in the specific Mach and dynamic pressure regime required by the present Mars EDL system. While the Space X first stage is not Mars-like in configuration, no showstoppers have been identified for this technology.

SRP computational, ground-based, and flight data have demonstrated that aerodynamic force and moment modeling uncertainty in the SRP phase is low for steady state, Mars-relevant conditions. Some uncertainty remains for SRP operation during startup and transition to steady-state operation, but performance uncertainty during this small time period may be mitigated by robust control system design. Combined with ground-based test data, the Space X flight data bounds the range of SRP thrust coefficients needed for human Mars EDL. Taken in total, these computational, ground-based, and flight test efforts significantly reduce the SRP flight system development risks for Mars EDL.

X. Entry, Descent, and Landing Analysis and Results

The trajectory design for the EDL concept was analyzed with the Dynamics Simulator for Entry, Descent and Surface landing (DSENDS) program developed at JPL for the MSL program. The entry altitude and velocity used for the 75 t vehicle was at a Mars radius of 3,522 km and a velocity of 4.9 km/s from a highly elliptical orbit. The 10 m diameter heatshield geometry used for the analysis was a 70° sphere-cone shape similar to MSL, Phoenix, MER, and Viking. This representative geometry is not optimized and was chosen from a heritage standpoint, which allowed for the usage of previously developed aero-databases that were available.

The EDL scenario can be split into three phases: 1) entry hypersonic aeromaneuvering, 2) SRP braking, and 3) a constant velocity powered descent phase. During the entry phase, the vehicle utilizes lift with a nominal lift-to-drag ratio (L/D) of 0.24 to perform a bank-controlled guided flight to a designated target. The entry guidance portion of the design is based on the Apollo implementation that was used for the MSL mission. After completing the heading alignment portion of the trajectory, the vehicle transitions to an angle of attack of 0° in preparation for the SRP phase. The SRP rocket firing is initiated when the optimum solution for the propulsive braking is found using a descent guidance logic. Nominally, SRP starts at about Mach 3.6 and an altitude of 3 km above the ground level (AGL). During the SRP burn, once the dynamic pressure has been reduced to 1 kPa, the heatshield is jettisoned to reduce mass and enable the landing gear to later deploy. After a burn duration of about 43 s from the start of SRP, the vehicle transitions to a constant velocity powered descent at ~5 m/s and an altitude of 40 m AGL.

The EDL performance was evaluated using a Monte Carlo trajectory simulation method with a sample size of 4,001. Error sources were included in the atmospheric model and for winds. Uncertainties in the aero-database coefficient and for mass properties were also included. For conservatism, the analysis only used 80% of the thrust capability of the engines in order to reserve the rest of the vehicle control authority for attitude control and differential throttling. The nominal trajectory, plotted as an altitude versus velocity curve, is presented in Fig.8. To read the chart, follow the curve from the right to the left. The timeline is indicated by tick marks at 10 s intervals along the curve from the entry point of 4.9 km/s. The maximum dynamic pressure, shown by the contours on the plot, is ~40 kPa. The lander dives deep into the atmosphere to maximize aerodynamic breaking. At this low entry altitude, where the bulk of the deceleration occurs, the atmosphere density uncertainty is less than half that at higher more traditional deceleration altitudes (10- 25 km). This results in lower dispersions, better targeting performance, and lower risk. As described above, the lander decelerates aerodynamically until SRP is initiated.

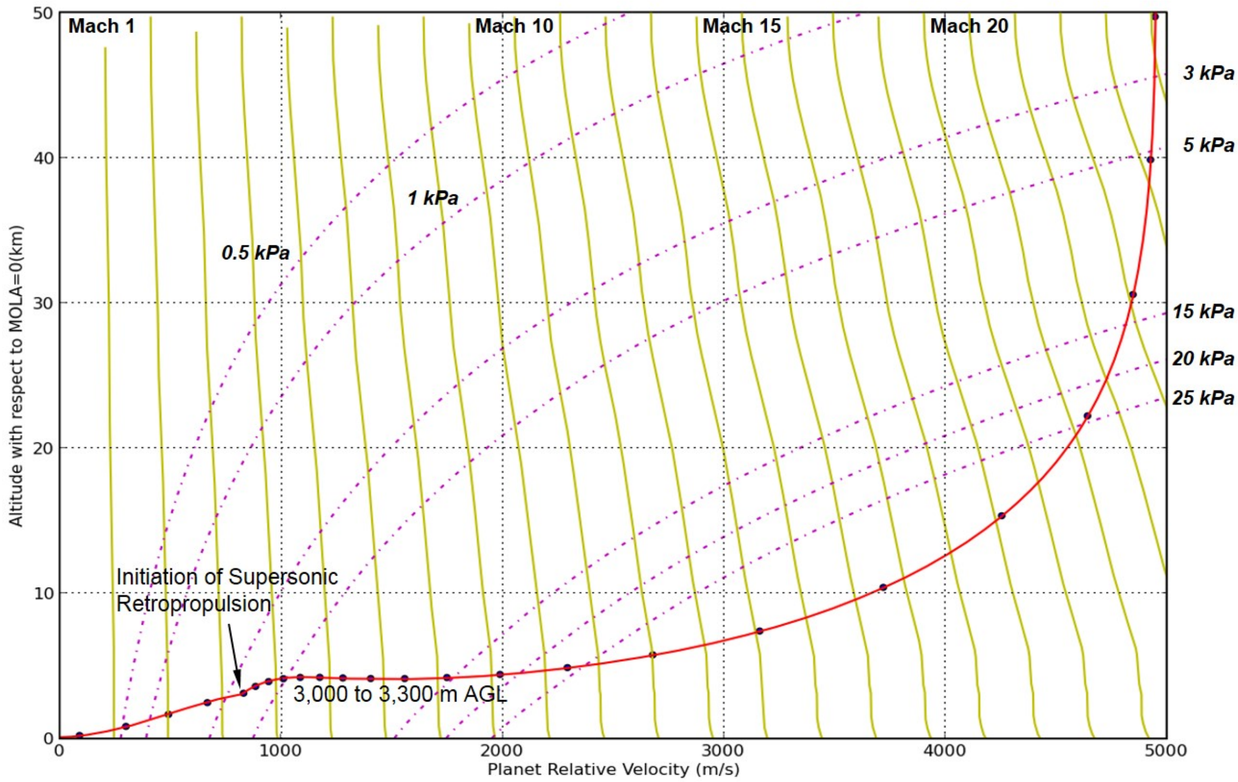

Figure 8. EDL altitude vs. velocity plot for non-SIAD case with 10 second time marks.

Fig. 9 shows a histogram plot for propellant usage. The propellant capacity in the descent stage design was chosen to be able to capture the 95th percentile propellant usage which, based on the Monte Carlo analysis, is about 19 t. Fig. 10 shows a plot of acceleration versus time during EDL. The ULM of the system is the sum of the 26 t MAV plus other equipment on the descent stage that provides useful functions for the crew, such as power, consumables, and science equipment. For the design example studied in this paper, the total ULM is 28.4 t. This results in a gear ratio of entry mass to ULM of 2.6.

XI. Optional Addition of Supersonic Inflatable Aerodynamic Decelerator

If a higher ULM fraction is desired, this architecture may be augmented with either a hypersonic or supersonic deployable aerodynamic decelerator. For this study, a 12 m deployed diameter supersonic inflatable aerodynamic decelerator (SIAD) was sized, based on the configuration and construction of the 6 m diameter SIAD that was twice successfully demonstrated in flight relevant conditions by the Low Density Supersonic Decelerator (LDSD) Project. The SIAD to aeroshell diameter ratio for this system is 1.2, whereas the SIAD to aeroshell diameter ratio demonstrated by LDSD was 1.33. The 12 m SIAD, shown in Fig. 11, would deploy under conditions at Mars which are very similar to those experienced by the LDSD flights. The mass of the system, including the inflation system, was estimated to be 895 kg (including contingency) with high heritage to LDSD. EDL scenarios were analyzed for the 10 m lander concept, both with and without a SIAD.

The SIAD, when deployed, increases the breaking area by a factor of 1.44. Analysis indicates this would place the lander at a lower velocity at the time of SRP initiation and would reduce propellant consumption by about 4 t. Taking into account the mass of the SIAD and reduction in propulsion dry mass, this results in an increase in ULM of about 3.5 t.

If SIAD technology can be advanced beyond that demonstrated by the LDSD flights, a larger SIAD could reduce propellant consumption even more. This is a case where more advanced technology could be on-ramped when it became available to increase performance and reduce costs.

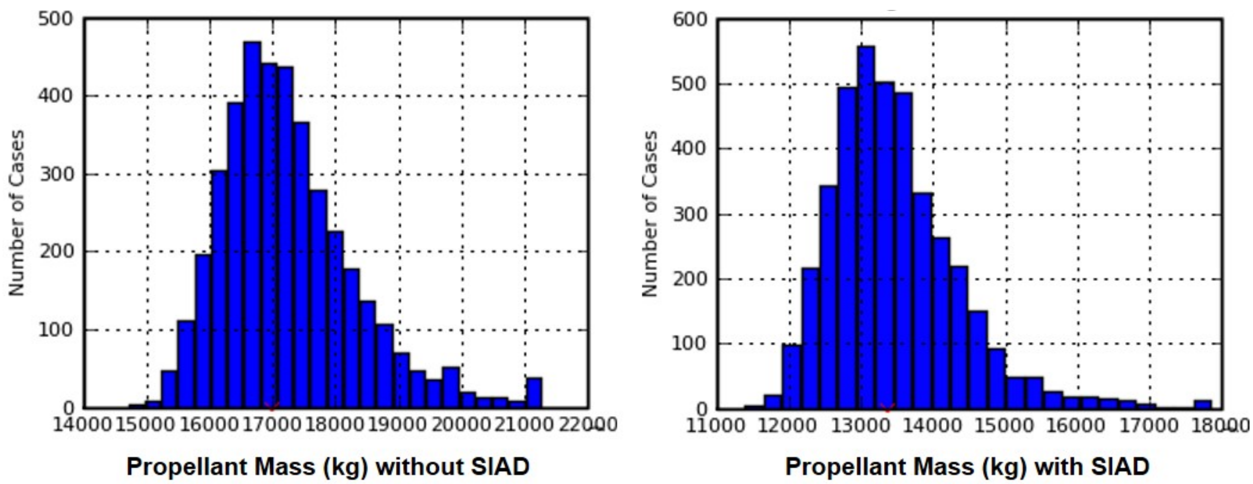

Figure 9. EDL propellant usage histogram with and without SIAD.

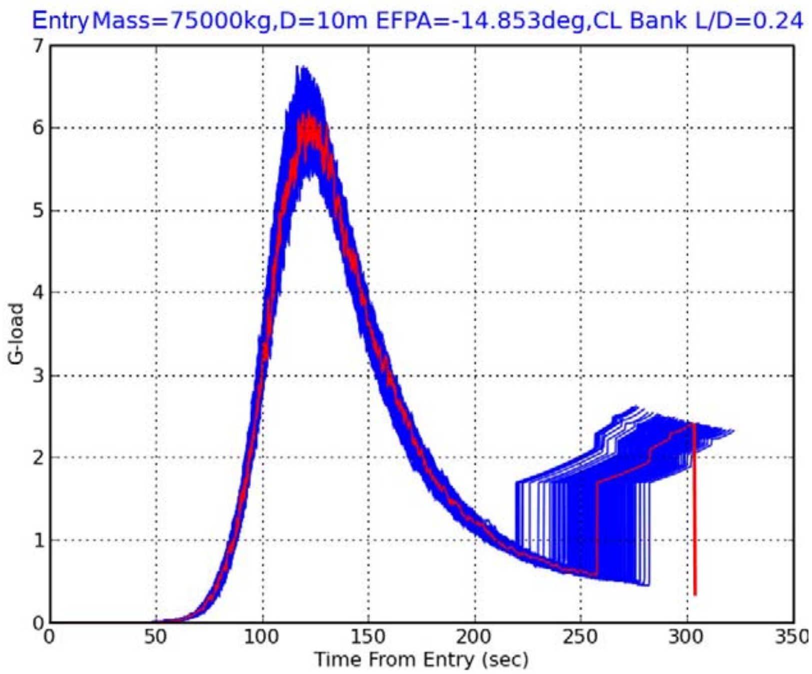

Figure 10. EDL g loading vs. time (4,001 Monte Carlo cases).

XII. Concept for Delivering Lander to Mars Orbit

Delivering a 75 t lander to Mars is a challenging undertaking. A single SLS Block 2 launch vehicle can inject about 40 t onto a conjunction class trajectory to Mars. For this study, a two-launch scenario with the Block 2 SLS was chosen as the reference approach for Trans-Mars Injection (TMI). The first SLS launch would deliver the lander to an elliptical HEO with a perigee of about 2,500 km (based on orbital debris avoidance) and an apogee of about 50,000 km. This launch could take place many months prior to the departure window to Mars, and the lander would be left in this parking orbit to await the second launch.

The second SLS launch would be timed for the departure window for the conjunction class Mars transfer trajectory. The EUS, which is the standard upper stage for the SLS Block 2 vehicle, would be launched with only a docking kit for a payload. The docking kit would have an active docking ring, an RCS system to enable rendezvous and docking, and extra insulation for the EUS cryogenic tanks to minimize boil-off and increase the loiter time on

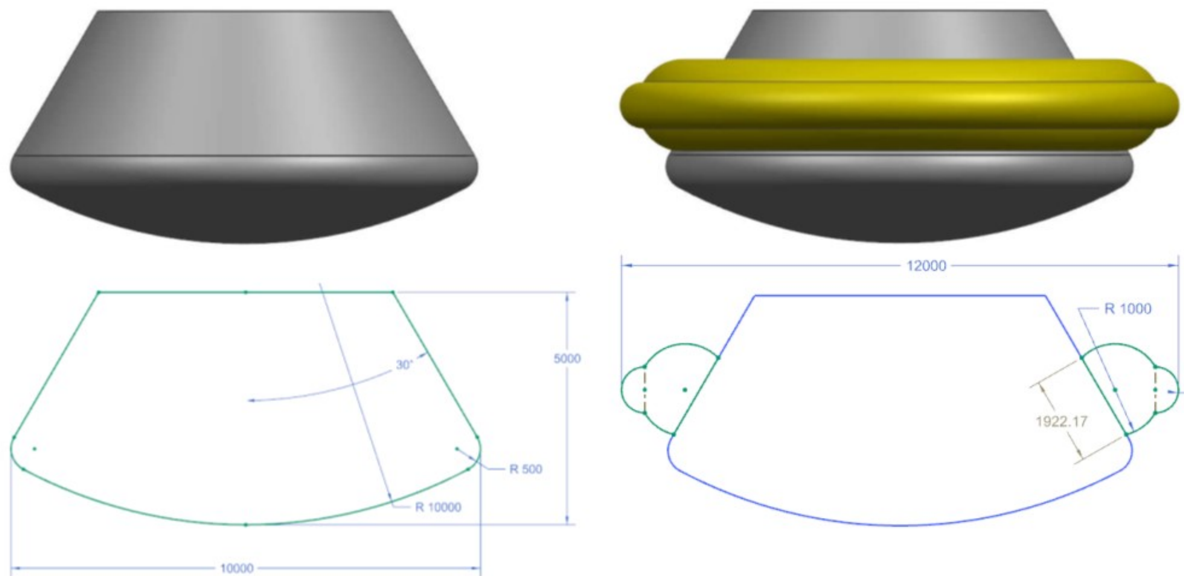

Figure 11. 12-m diameter SIAD based on LDSD.

orbit. The docking kit would also include solar arrays for power, or alternately, a fuel cell system to tap the LOX and LH_2 in the EUS for power.

The EUS from the second launch would rendezvous and dock with the lander in HEO, and the EUS would be reignited at perigee to perform TMI. Prior to the second SLS launch, the lander orbit would need to be properly phased for departure, and perigee would be lowered to maximize the effectiveness of the TMI burn. The performance that can be obtained by the two-launch scenario has been estimated to be about 85 t for a C_3 of 15 km^2/s^2.

XIII. Concepts for Cargo Landers

As described by Price,[11] the first crewed landing on the Martian surface might be a short-stay mission with a crew of only 2 or possibly 3. The advantage would be that it could take place at an earlier date, since it would require no other surface infrastructure, and it could be a pathfinder for follow-on missions that would have long surface stays. For the follow-on missions, infrastructure would need to be pre-placed on the surface to support the crew outside of the MAV.

For cargo delivery to the Martian surface, a lander identical to that designed for the MAV could be used to minimize additional development cost, reduce risk through common design, and to minimize production cost through a common production line. Fig. 12 shows two concepts for cargo carriers. One is a lander where the MAV has been replaced by a 27-t surface habitat that could support a crew of 4 on the surface for one Earth year. Solar arrays and batteries on the common descent stage, similar to the ones used on the MAV lander, would provide the power for the habitat.

Another possible cargo lander variant could deliver logistical supplies such as crew consumables, a pressurized rover, additional power systems, and science equipment such as deep drilling units. With a continuing series of surface missions, infrastructure could be built up over time at a Mars base location that could evolve toward a permanent presence. In-Situ Resource Utilization (ISRU) systems could be brought on line, providing for the generation of air and water for the crew and for propellant production and transfer. With increased experience and confidence in MAV reliability and performance, it could be evolved to a vehicle that could have all of its propellants manufactured from indigenous resources, improving the performance of the MAV landers and potentially reducing mission costs.

XIV. Conclusions

A conceptual design has been developed for a 10 m diameter 75 t crewed Mars lander with a fully-fueled MAV that could offer abort capability for much of EDL after the peak heating phase. The lander and MAV functioning

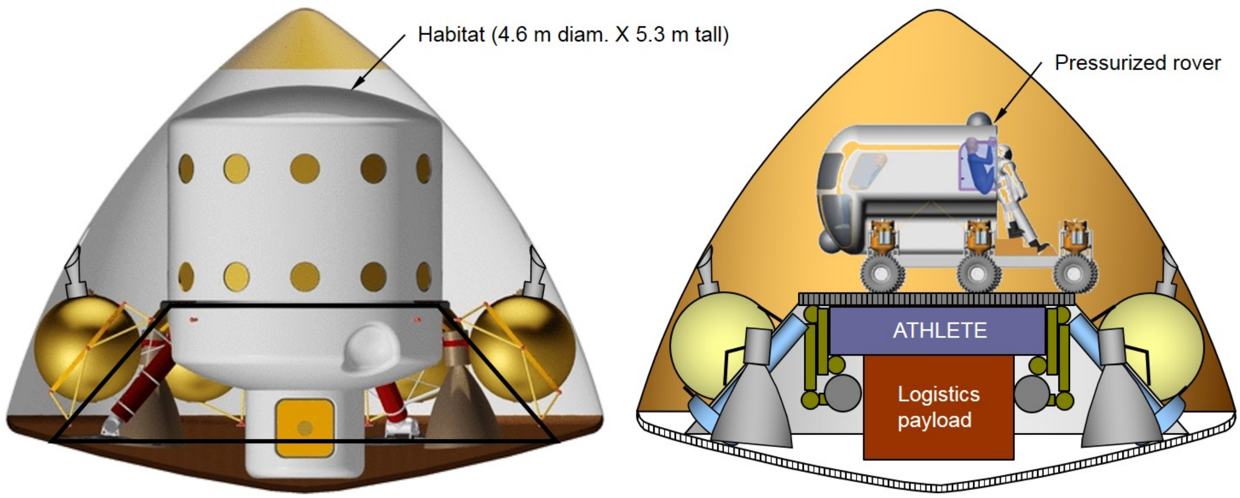

Figure 12. Cargo version concepts for landers with surface habitat and logistics carrier.

would not depend on interacting with any other surface infrastructure on Mars. Most of the technology is high TRL, and development could proceed in the near-term with current aerospace industry capabilities. The lander could be delivered to Mars with two Block 2 SLS launches, and the time between the two launches could be as long as one year. Aerocapture would be used for MOI into HMO to await the later arrival of crew in another vehicle.

The EDL design would use a traditional blunt-body heat shield, with high heritage from past systems, coupled with SRP. The propulsion system is sized to have adequate throttling capability to enable a constant-velocity controlled landing and safely place a 28-t payload at or below 0 km areoid. Improvements in the performance of this system are possible to deliver payloads in the range of 31-32 t ULM with the inclusion of SIAD technology, proven twice in flight relevant conditions by the LDSD project. Other options for improving performance could be increasing the rigid blunt-body diameter and increasing the thrust-to-weight ratio for SRP.

The lander concept could provide a several week surface stay for a crew of two for an early Mars landing, and could provide a several day stay for a crew of four on later missions where a habitat and other infrastructure was pre-placed on the surface. The lander design, with few changes, could be used as an uncrewed cargo carrier to deliver a habitat and other supplies to the surface to support crews for long stays and to build up infrastructure for an evolving Mars base.

The early versions of the lander could support oxygen-generating ISRU equipment to provide supplemental breathing supplies from the Martian atmosphere to enable additional EVAs for the crew. Later versions of cargo landers could provide more advanced ISRU capabilities to generate LOX or other propellants from the atmosphere. The MMH/MON-25 MAV, for example, could be upgraded to a MMH/LOX propulsion system for later missions to utilize ISRU propellants and enable more capable transportation for future crews.

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