Parametric Cost Estimates of Four 20 Ton Payload Mars EDL Vehicle Concepts

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Cost is one of the biggest obstacles to sending humans to Mars. Many studies have examined various vehicles and/or technologies which could be used to send humans to Mars. However, nearly all of these studies have either ignored cost or have used the mass of the vehicles as a surrogate for cost. This work presents parametric cost estimates of four Mars Entry, Descent, and Landing vehicle concepts capable of landing 20 tons of payload on the surface of Mars. The concepts are the Co-Optimization Blunt-body Re-entry Analysis-Mid lift to drag ratio Rigid Vehicle (Cobra–MRV), Adaptive Deployable Entry and Placement Technology (ADEPT), Hypersonic Inflatable Aerodynamic Decelerator (HIAD), and Capsule concepts. Two variants of the HIAD concept are studied, one with a single HIAD heatshield and one with two HIADs. The study presents the development and production cost of each vehicle and the total cost to build five flight units and fifteen flight units. Five flight units is representative of a single mission, or a "Flags and Footprints" style campaign, whereas fifteen flight units is representative of multiple missions going to multiple sites on Mars. All costs presented in this work are normalized to the average of the first unit production cost of the lander concepts. There is a large amount of uncertainty both in the design of the vehicles and the cost of the vehicles and it is impossible to claim with certainty which vehicle will cost the least. However, it is likely that the total cost to produce five flight units will be in the following ranking from least costly to most costly, Single HIAD, Capsule, Dual HIAD, Cobra, ADEPT. The ranking for fifteen units is similar but with the positions of the Dual HIAD and Cobra reversed. This study only considers the cost of the entry vehicle and does not take into account the effect of the entry vehicle's mass on the cost of the in-space propulsion stage or launch vehicle.

I. Introduction

COST is one of the biggest obstacles to sending humans to Mars. There have been several studies comparing different vehicle concepts for landing humans and large payloads on the surface of Mars, but to date, there have been no studies comparing the cost of the vehicle concepts. The Entry, Descent, and Landing Architecture Study (EDLAS) was commissioned as a follow on to the EDL Systems Analysis (EDLSA) study as a part of the Evolvable Mars Campaign (EMC)[1]. The EDLAS study looked in depth at four competing concepts for landing 20 t of payload on the surface of Mars. The Co-Optimization Blunt-body Re-entry Analysis–Rigid Mid lift to drag Ratio Vehicle (Cobra–MRV)[2] and Hypersonic Inflatable Aerodynamic Decelerator (HIAD) concepts from the EDLSA study, as well as two new concepts, the Capsule modeled after the 70° sphere-cone design commonly used to land robots on Mars[3], and a concept making use of a rigid deployable heatshield known as the Adaptive Deployable Entry and Placement Technology (ADEPT)[4, 5]. The initial EDLAS report concluded that HIAD was likely to be the most mass efficient followed by ADEPT and Cobra. The Capsule was not compatible with the EMC assumptions in part because it did not have the required volume to fit the baseline Mars Ascent Vehicle (MAV) inside it. Later the Capsule was redesigned to a shape similar to a Soyuz capsule with a greater usable volume. The MAV was also redesigned to fit within the larger design, but is only able to reach low Mars orbit not the planned 1-sol orbit.

While both the EDLSA study and the EDLAS study have conducted a thorough analysis of the design of many EDL concepts to land humans on Mars, neither study has considered the cost of the concepts. The expected cost of each vehicle concept will be a key driver of which concepts are selected for further development. This work details cost estimation work done as a follow on to (but not funded by) the Mars EDLAS (Entry, Descent, and Landing Architecture Study)[1]. Note that this work only deals with the cost of developing and producing the lander hardware and does not include the cost of the required in-space propulsion stage or launch vehicles. Thus, the effect that the mass of each of the landers will have on the total cost of the mission is not taken into account.

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The following section details the use of the parametric cost modeling tool SEER-H to estimate the cost of future space hardware. Sections III, IV, V, and VI discuss the design and mission concept of operations (ConOps) and provides cost estimates of the Cobra–MRV, ADEPT, HIAD, and Capsule entry vehicle concepts. Section VII compares the HIAD, ADEPT, and Cobra–MRV concepts and presents an uncertainty analysis of their costs. Finally, Sec. VIII provides concluding remarks and discusses future work to be completed.

II. Using SEER-H to Model the Cost of Future Hardware

System Estimation and Evaluation of Resources-Hardware (SEER-H)[*] or simply SEER is a commercial parametric cost estimating tool developed by Galorath Inc. It is one of the standard tools used by NASA to estimate the costs of space missions for early planning and to assess mission proposals. Galorath performed an internal validation study of SEER where it was used to predict the costs of 15 NASA space science missions. Galorath found that SEER's mean error in predicting mission cost was -1% with a standard deviation of 19%[6]. A blind independent validation study by Friz et al. showed that, while SEER's point estimates can be inaccurate, its uncertainty quantification capabilities work as expected[7, 8]. The primary result of the SEER validation study was that when using SEER's uncertainty quantification capabilities 75% of the cases studied fell within SEER's 80% confidence interval,[†] thereby validating the predictions of SEER.

The backbone of SEER is its proprietary Cost Estimating Relationships (CERs) and associated database of cost history. To model the cost of a spacecraft in SEER, the user inputs "Work Elements" which correspond to different components of the spacecraft using SEER's Graphical User Interface (GUI). An annotated screenshot of SEER's GUI can be viewed in Fig. 1. On the left side of the GUI, the user defines a Work Breakdown Structure (WBS) which corresponds

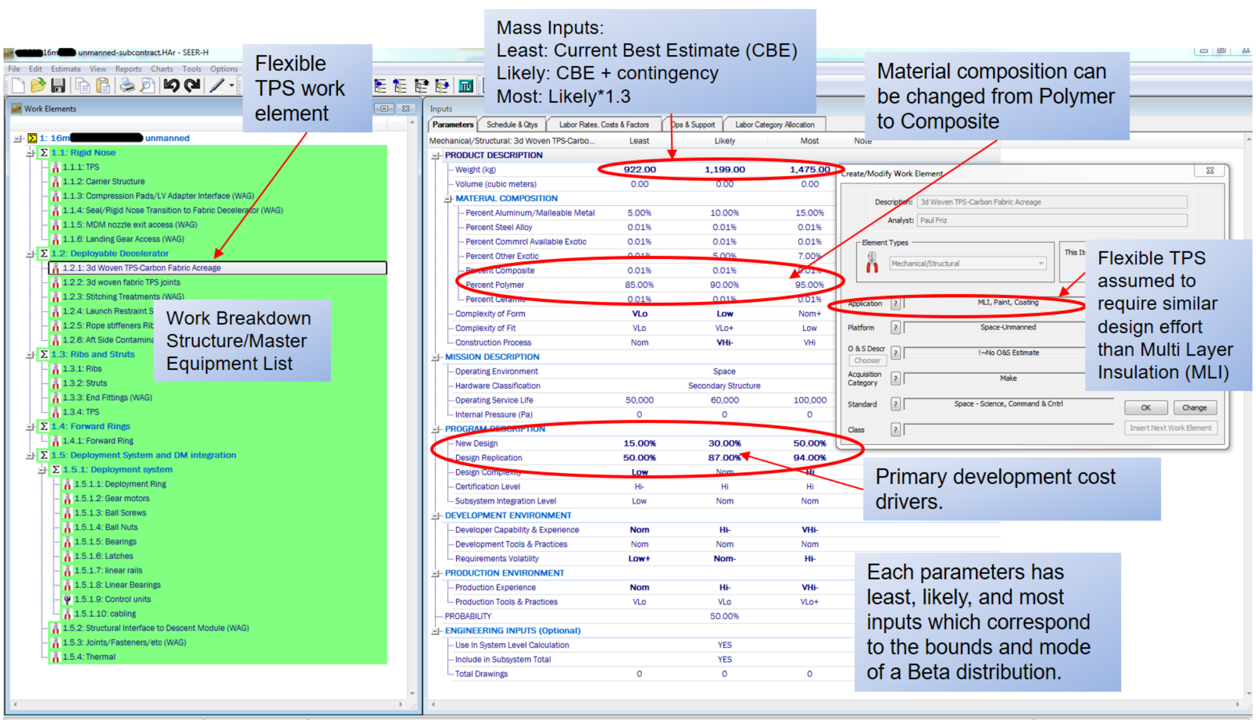


Fig. 1 Annotated screenshot of SEER's GUI. Inputs in this example are for illustration purposes only and are not used in the present work.

roughly to the spacecraft or payload's Master Equipment List (MEL). Each item in the WBS is known as a "Work Element" and can be of the type "Mechanical/Structural," "Electronics," "Electro-Optical System (EOS)," or "Integrated

[*]Reference to or appearance of any specific commercial products, processes, or services by trade name, trademark, manufacturer, or otherwise does not constitute or imply its endorsement, recommendation, or favoring by the U.S. Government or NASA.

[†]SEER only outputs probability levels of an estimate's CDF in intervals of 10%. Thus, 80% confidence intervals must be used as opposed to the more common 95% confidence interval.

Circuit (IC)." After selecting the Work Element type, the user then chooses a "Platform," "Standard," "Acquisition Category," and "Application." The Platform defines the operating environment (i.e., Air-Manned, Air–Unmanned, Sea, Submersible, Space–Manned, Space-Unmanned, etc.). The Standard defines the general reliability requirements (i.e., Commercial, Industrial, Military specifications, Space–Communications and Media, or Space–Science, Command & Control). The Platform and Standard typically have the same setting for all work elements in a SEER model. The Acquisition Category defines the level of development effort required to design the component or integrate an existing design (i.e., build to print, buy and integrate, minor modifications, major modifications, etc.). Finally, Application defines the function of the Work Element. SEER's database contains hundreds of mechanical/structural Applications for hardware components; examples include payload adapter, separation mechanism, space propulsion component, aerodynamic control surface, spacecraft antenna–dish, gimbal mechanism, etc. If the user selects the Platform, Standard, Acquisition Category, and Application of a particular Work Element, SEER will auto fill the majority of the inputs, or "Parameters," of the Work Element. The Parameters are the inputs to SEER's CERs and are different for each Work Element type. The Parameters for Mechanical/Structural Work Elements include mass, material composition, complexity of form, complexity of fit, construction process, amount of new design, design replication, certification level, and a number of other inputs. Electrical Work Elements are modeled using Parameters such as the number of printed circuit boards, number of discrete components per board, number of integrated circuits per board, clock speed, number of pins, percent new design, and a number of other inputs. The EOS and IC Parameters are typically specific to the particular application. For example, an optical bench will include inputs such as number of optical elements, as well as the number and size of the elements, where as a detector will be estimated using the resolution of the sensor, radiation tolerance, pixel size etc. If the user knows all these inputs (or can estimate them) they can potentially improve the accuracy of the estimate or model hardware Applications that are not included in SEER's database. That is, if a spacecraft component does not have a matching Application in SEER, the user can select an analogous technology and alter the Parameter inputs. For example, flexible TPS is not in SEER's database, but it can be accurately modeled as multi-layer insulation with adjustments to material composition, complexity of fit/form, and construction process.

For every Parameter input in SEER the user defines a "least," "likely," and "most" value corresponding to an optimistic, most likely, and pessimistic assumption for the input. Mass modeling in the present work assumed the "least" value was 95% of the Current Best Estimate (CBE), the "likely" value was the CBE plus the Mass Growth Allowance (MGA) and the "most" input was 30% more than the "likely" input. The "likely" and "most" assumptions are recommended in Galorath's SEER Space Guidance document[9]. The NASA Jet Propulsion Laboratory's design principles document recommends that space missions carry a 30% mass margin above the MGA[10] at Preliminary Mission System Review (PMSR). Also, the AIAA and American National Standards Institute state that a mission is at minimal risk of exceeding their mass budget if at ATP (Authority to Proceed) its CBE plus MGA is 30% below the maximum allowable mass for the mission[11]. The "least" assumption is recommended by Friz et. al. to allow for the possibility that the spacecraft decreases in mass over the course of its development[7]. While rare, this does occasionally happen. SEER models uncertainty by assigning each "work element" a distribution of possible costs in addition to a median cost. The least/likely/most inputs for each work element correspond to the lower bound, mode, and upper bound of a beta distribution. By default, SEER uses the median value of each beta distribution as the input to its CERs. To model uncertainty SEER can also output a Cumulative Distribution Function (CDF), however, SEER only outputs the CDF in 10% increments from 10-90%. Thus, an 80% confidence interval instead of the more common 95% confidence interval is used in this study.

III. Cobra–MRV (Co-Optimization Blunt-body Re-entry Analysis–Mid lift to drag ratio Rigid Vehicle)

A. Cobra–MRV Overview

The Cobra–MRV (Co-Optimization Blunt-body Re-entry Analysis–Mid lift to drag ratio Rigid Vehicle) is a high ballistic coefficient, mid lift to drag ratio, rigid vehicle concept. It was developed using the Ames Research Center (ARC) Cobra shape optimization tool[12]. The Cobra–MRV has a higher L/D than typical blunt body entry vehicles but less than a winged vehicle such as the Space Shuttle Orbiter. Thus, the vehicle has greater cross-range capability, entry corridor width, and potentially improved landing accuracy over the HIAD, ADEPT, and Capsule concepts. Another advantage of the Cobra–MRV is that upon landing it sits low to the ground, which allows for easy offloading of crew and cargo. In an effort to reduce the amount of technology development needed, and therefore cost, many of the Cobra–MRV's systems and components are derived from the Space Shuttle. Shuttle derived items include the body flaps, payload doors, and the TPS. The windward TPS is made up of Silicone Impregnated Refractory Ceramic Ablator (SIRCA), while the leeward TPS which is made of Advanced Flexible Reusable Surface Insulation (AFRSI)[2, 13, 14].

B. Cobra–MRV ConOps

Figure 2 shows the ConOps for the Cobra–MRV in various phases of EDL as well as on the surface of Mars as described by Sostaric et al.[2]. After the initial aerocapture of the Cobra–MRV into a nominal 1 Sol orbit around Mars it loiters for a period of time until it is needed on the surface. The vehicle deorbits using its RCS thrusters and at the entry interface has a velocity of 4.7 km/s, a flight path angle of -10.8 degrees, and a 55 deg angle of attack. When at 3.2 km above the surface and traveling at Mach 1.98 the vehicle pitches up to a 90 deg angle of attack and fires its main engines to neutralize its horizontal velocity. The vehicle continues to fire its engines, slowing its descent until it is 12.5 m above the Martian surface at which point it descends at a constant rate of 2.5 m/s until touchdown.

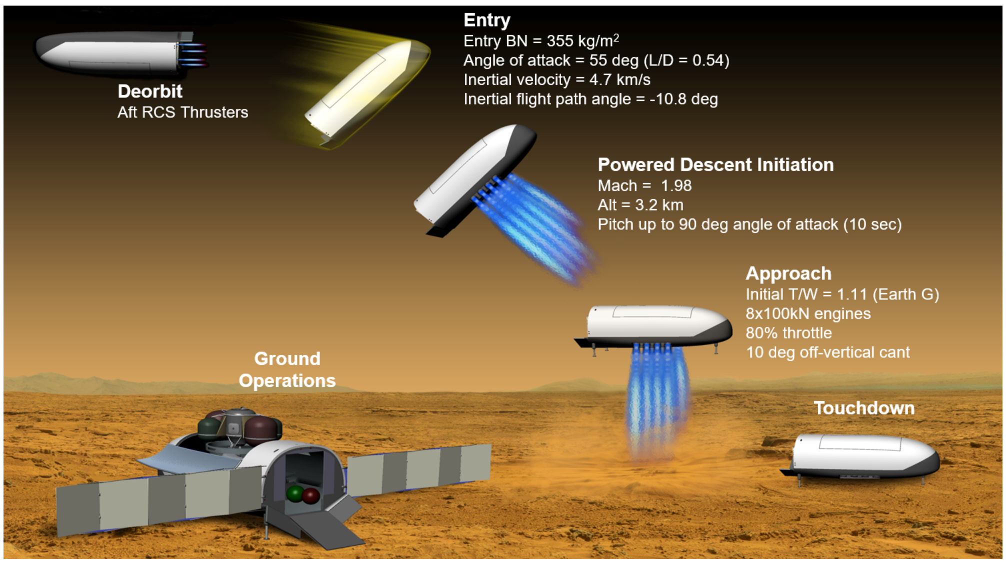


Fig. 2 Cobra EDL ConOps. Aerocapture not shown.

C. Cobra–MRV Cost Estimate and SEER Model

A MEL for the Cobra–MRV was obtained from the EDLAS team and used in conjunction with a document providing technical descriptions of the associated hardware to produce a cost model in SEER. The provided MEL was broken down into 165 different components which translated into 87 work elements in SEER. The MEL and associated documents contain sensitive information which cannot be freely distributed so they are not included in the present work. However, the mass of the subsystems, and a breakdown of the normalized cost from the SEER estimate are provided in Table. 1.

The costs presented in Table 1 are normalized to the average of the first unit production costs of all the lander concepts presented in this work. The data in Table 1 is divided into three sets of two columns. The first two columns show the Predicted Mass of the vehicle and vehicle subsystems, and what fraction of the Vehicle Wet Mass they make up. Since the payload costs are not considered in this study, the payload mass is not included in the vehicle set mass, however, the payload mass is included in the Total Stage Gross Mass which is equivalent to the Trans Mars Injection (TMI) Mass. The TMI Mass is the sum of the payload mass and the vehicle wet mass and represents the mass that the in-space propulsion stage will have to deliver to a TMI orbit. The second set of columns shows a breakdown of the normalized development costs of the vehicle. The development costs are all cost associated with the design, development, testing, and evaluation of a spacecraft; including the costs of prototype hardware, tooling, support equipment, documentation, etc. The third set of columns shows the breakdown of the production costs of the first flight unit produced. These costs include the purchase costs of off the shelf components and raw materials, fabrication, integration and assembly of the final product, production support (planning, scheduling, inventory, shipping etc.), sustaining engineering (resolving manufacturing issues and handling changes in the production process), tool maintenance, etc. The development and production costs do not include science costs, technology development costs, payload costs, mission operations costs (other than flight tests) launch vehicle costs, ground systems not directly related to the development and production of the vehicle (i.e. mission control centers, deep space network antennas, etc.), and education/public outreach. The first two rows of Table 1 show the Mission Systems cost and Vehicle Systems cost for the development and production of the vehicle. Both of these categories of systems costs are made up of Project Management (PM), Systems Engineering (SE), Safety and Mission Assurance (S&MA,) and Integration, Assembly, and Test (IAT). The Vehicle Systems Cost is composed of the PM, SE, and IAT costs of the prime contractor responsible for building the vehicle. The Mission Systems Cost is composed of the NASA level PM, SE, S&MA, and IAT costs of managing and overseeing the contractors, defining and tracking mission requirements, as well as testing and final assembly of the vehicle. Together the systems cost make up 56% of the development cost and 34% of the production cost. The next row down is the structures subsystem which makes up 31% of the vehicle by mass but only 16% of the development and 24% of the production cost. The structures subsystem contains all primary and secondary structures. The next row after structures is propulsion which contains the Main Propulsion System (MPS) and Reaction Control System (RCS). Both the MPS and RCS utilize CH4-LOX (Methane-Liquid Oxygen) as a fuel/oxidizer combination. The MPS utilizes eight engines, each rated at 100 kN, with a specific impulse of 360 s while the RCS is a network of 20 thrusters, each rated at 4.45 kN, with specific impulses of 325 s. The mass and cost breakdown for these subsystems contains all the propellant tanks, lines, fittings, valves, manifolds, cryocoolers, insulation, engines, etc., needed for the MPS and RCS. The power subsystem contains the on-orbit solar arrays, fuel cells, and all harnessing and electronics needed to power the spacecraft. The power subsystem is expected to be modified from the power system of the Orion Multi-Purpose Crew Vehicle. Avionics includes the mass and cost of the Communications, Guidance, Navigation, and Control (GN&C), and Command and Data Handling (C&DH) subsystems. It is assumed that the majority of the components in the avionics subsystems will be modified from existing hardware and no new technology development will need to occur.[‡] Much of the GN&C hardware is modeled based on the hardware developed in the Autonomous Landing and Hazard Avoidance Technology (ALHAT) program[15]. The thermal subsystem is made up of the deployable radiators and heat rejection systems that work in conjunction with the cryocoolers to prevent propellant from boiling off in transit to Mars, as well as cooling the LOX which is produced on the surface of Mars as a part of the ISRU propellant manufacturing included in the MAV payload. Finally, Cobra-MRV subsystems is all the components which are unique to the Cobra-MRV design including the TPS, body flap aerosurfaces, payload bay door, cargo ramp, and landing gear as well as all the associated mechanisms and actuators.

Table 1 Cobra-MRV Mass and Normalized Cost Breakdown

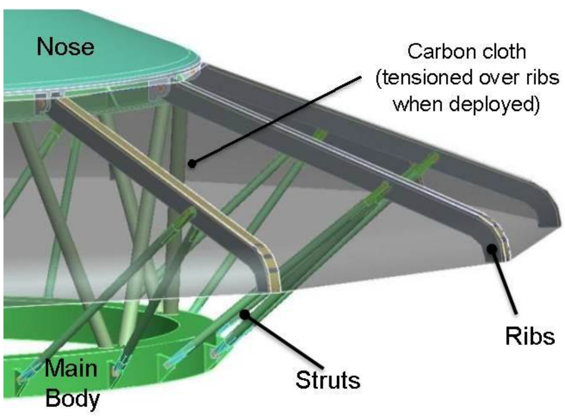
	Predicted Mass (kg)		Development		First Unit	
Mission Systems Cost			2.334	31%	0.126	13%
Vehicle Systems Cost			1.870	25%	0.200	21%
Structures	14,787	31%	1.219	16%	0.237	24%
Propulsion	4,939	10%	0.815	11%	0.170	17%
Power	1,568	3%	0.186	2%	0.053	5%
Avionics	333	1%	0.231	3%	0.047	5%
Thermal	844	2%	0.333	4%	0.017	2%
Cobra-MRV subsystems	5,499	12%	0.616	8%	0.121	12%
Dry Mass	27,970	59%				
Non-Propelled Fluids	1,481	3%				
Used Propellant	18,080	38%				
Vehicle Wet Mass	47,531	100%	7.604	100%	0.971	100%
Payload	20,000					
Total Stage Gross Mass	67,531					

[‡]The assumption that the majority of the components in the avionics subsystems will be modified from existing hardware without requiring a new technology development will likely prove to be incorrect. However, it is likely that much of the hardware will be shared with other elements in the humans to mars campaign such as the deep space habitat, in-space propulsion stage, or Orion, and thus, the cost of developing this hardware is not likely to be carried by the lander. Regardless, the Cobra-MRV, ADEPT, HIAD, and Capsule vehicles are modeled with identical avionics systems so this will not affect cost comparisons between the vehicles.

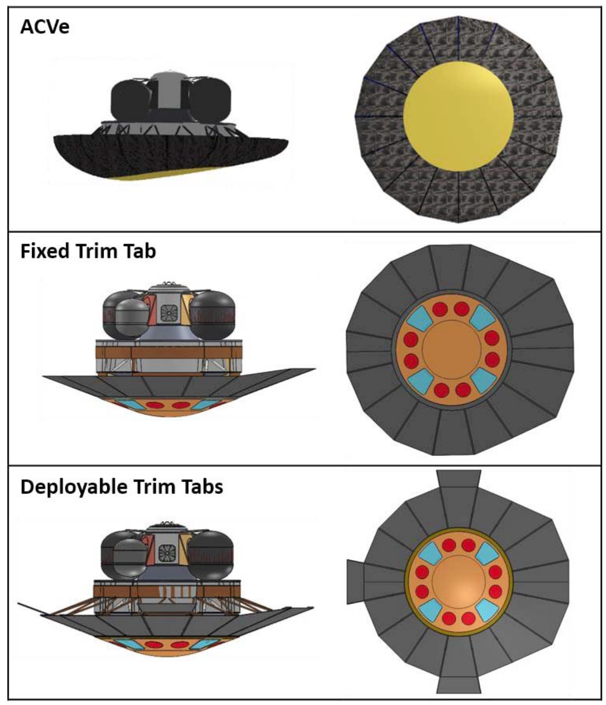
IV. ADEPT (Adaptive Deployable Entry and Placement Technology)

A. ADEPT Overview

Another lander design under consideration to land large payloads on Mars is the Adaptive Deployable Entry and Placement Technology (ADEPT)[4, 5, 16]. The ADEPT concept utilizes a deployable heatshield that is similar mechanically, and in appearance to a large umbrella. When ADEPT is stowed, it allows the lander to fit within the 10 m fairing of the Space Launch System (SLS), but when needed, it deploys to a diameter of 16 m (or greater). The ADEPT heatshield is attached to a Mars Descent Module (MDM) which is made up of the Structures, Propulsion, Power, Avionics, and thermal subsystems. The TPS for the ADEPT heatshield is made of a 3D woven carbon fabric that is held in place by ribs which are attached to movable struts for deployment[17, 18]. The structures of the ADEPT deployable mechanism are shown in Fig. 3a. There are several designs being traded for controlling the vehicle in the hypersonic and supersonic phases of entry which can be viewed in Fig. 3b. The first concept, which is also the examined in the present



(a) ADEPT TPS, Ribs, and Struts structures.



(b) Trades in the ADEPT heatshield design.

Fig. 3 ADEPT structures and design trades.

study, is known as Asymmetric Capsule Vehicle (ACVe) and provides a higher L/D ratio as well as lower convective and radiative heating compared to symmetric forebody shapes[5, 19]. The second is a fixed trim tab which also provides a higher L/D ratio and reduces convective and radiative heating but not to the extent that ACVe does[20]. While the fixed trim tab design lacks the optimized performance of the ACVe it is a simpler design over the ACVe which may save on manufacturing costs. The third design trade is actively controlled deployable trim tabs. The use of deployable trim tabs enables the use of a Direct Force Control algorithm to guide the spacecraft to its targeted landing location on the surface. A study by Cianciolo and Powell showed that Direct Force Control had several advantages to other control algorithms[21]. The advantages of Direct Force Control include saving propellant, reducing landing dispersion, and it does not require the main engines to gimbal or deep throttle. ACVe was selected as the baseline design by the ADEPT team and thus was the only design which had a MEL detailed enough to do a cost estimate at the time of writing[5].

B. ADEPT ConOps

The ConOps for the ADEPT concept is shown visually in Fig. 4, and is very similar to that of the Cobra–MRV discussed in Sec. III.B. The vehicle approaches Mars on a TMI trajectory provided by an in-space propulsion stage not

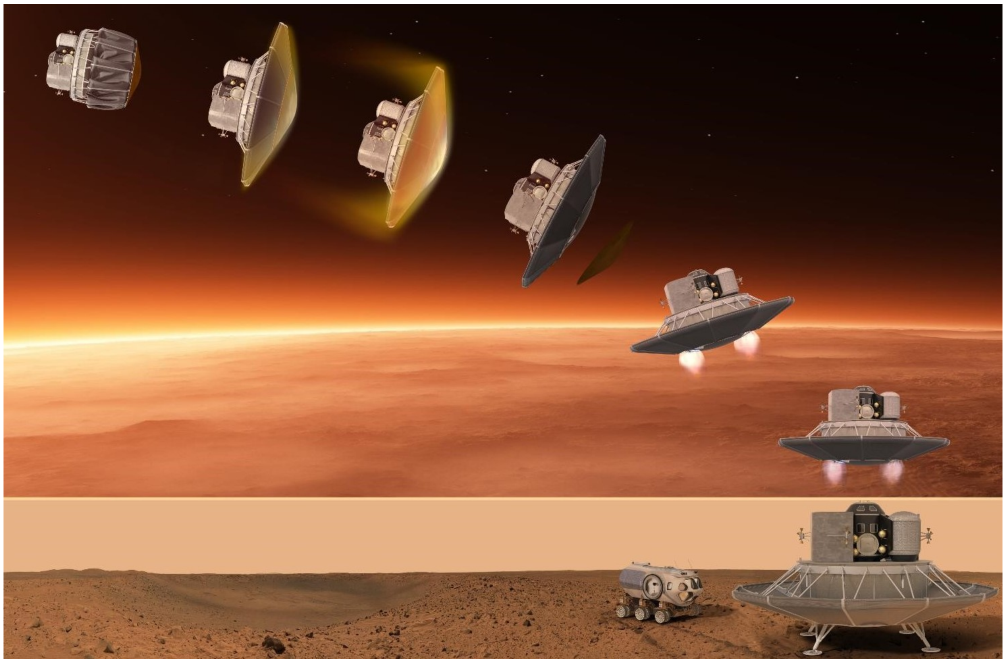


Fig. 4 ADEPT EDL ConOps, rigid heatshield may or may not be detached. Aerocapture not shown.

modeled in this work. The in-space propulsion stage separates from the lander and the ADEPT heatshield is deployed. The ADEPT lander then performs an aerocapture maneuver to a 1 sol orbit around Mars. The ADEPT lander then deploys solar arrays and loiters in the 1 sol Martian orbit for a period of time up to a year until it is needed on the surface. The solar arrays are then jettisoned or retracted and the vehicle performs a ~15 m/s ΔV burn to lower its periapsis and enter the Martian atmosphere. At entry interface, the vehicle has an inertial velocity of 4.7 km/s. Figure 4 shows the heatshield being detached at speeds around Mach 2, however, the heatshield will likely remain attached to the vehicle. Detaching the heatshield is considered a higher risk option as the heatshield may reconnect with the vehicle. The current baseline design which is also the design considered in the present study is to have access doors which open up in the heatshield to expose the main engine nozzles and allow the landing gear to be extended. The engines are then ignited to slow the vehicle and neutralize any horizontal velocity. Nominally the vehicle will continue to slow its descent until it is 12.5 m above the Martian surface at which point it descends at a constant rate of 2.5 m/s until touchdown.

C. ADEPT Cost Estimate and SEER Model

A MEL for the ADEPT lander was obtained from the EDLAS team and used in conjunction with a document providing technical descriptions of the associated hardware to produce a SEER model. The MEL was broken down into 193 different components which translated into 119 work elements in SEER. The MEL and associated documents contain sensitive information which cannot be freely distributed so they are not included in the present work. However, the mass of the subsystems, and a breakdown of the normalized cost from the SEER estimate is provided in Table. 2.

Table 2 shows that the fraction of the costs allocated to Mission Systems and Vehicle Systems are similar to those of the Cobra–MRV. The structures however, are completely different from Cobra–MRV. The primary structure of the ADEPT MDM is a cruciform structure similar in form to that used by the descent module of the Apollo lunar lander[22]. The MDM structures are primarily made of aluminum. The MDM structures make up 14% of the vehicle mass, 8% of the development cost and 23% of the production cost. The ADEPT lander uses the same set of eight CH4-LOX 100 kN engines that the Cobra–MRV uses. This thrust per engine may not be optimal for the ADEPT vehicle but a common 100 kN engine design was decided on to be shared by multiple elements within the Mars campaign (including the MAV) to save on engine development costs. The power and avionics subsystems are all identical to those in the Cobra–MRV model. The major difference is in the addition of the ADEPT deployable heatshield which makes up nearly a quarter of the mass and cost of the entire vehicle. The deployable heatshield contains many large complex structures and mechanisms made out of unique materials. Most of the composite structures will require time intensive hand lay ups and the entire structure is made of moving parts which will require precise tolerances and extensive testing. These factors will increase the cost during both development and production of an ADEPT vehicle.

Table 2 ADEPT Mass and Normalized Cost Breakdown

	Predicted Mass (kg)		Development		First Unit	
Mission Systems Cost			2.223	29%	0.151	13%
Vehicle Systems Cost			1.781	23%	0.239	21%
Structures	5,422	14%	0.615	8%	0.132	12%
Propulsion	4,689	12%	0.840	11%	0.166	14%
Power	1,568	4%	0.179	2%	0.051	4%
Avionics	333	0.8%	0.231	3%	0.047	4%
Thermal	411	1.0%	0.039	0.5%	0.003	0.3%
Aerodecelerator	8,658	22%	1.717	23%	0.361	31%
Dry Mass	21,080	53%				
Non-Propelled Fluids	1,549	4%				
Used Propellant	17,148	43%				
Vehicle Wet Mass	39,778	100%	7.625	100%	1.149	100%
Payload	20,000					
Total Stage Gross Mass	59,778					

V. HIAD (Hypersonic Inflatable Aerodynamic Decelerator)

A. HIAD Overview

The Hypersonic Inflatable Aerodynamic Decelerator (HIAD) lander concept is similar to ADEPT, however, it utilizes a heatshield that deploys via inflation rather than mechanisms. It is currently unknown if a HIAD can withstand a heat pulse from aerocapture followed by another heat pulse for EDL after loitering a year in a 1 sol Martina orbit. Thus, the baseline design for the HIAD lander carries two HIADs one for aerocapture and one for EDL. Carrying two HIADs also helps to mitigate the risk of a micrometeoroid damaging the flexible TPS, inflatable structure of the HIAD, or any of the other lander structures. In addition to the Dual HIAD concept, this work also presents a cost estimate of a lander with a single HIAD. It is unknown whether or not the Single HIAD concept will be able to aerocapture. Thus, it may need to be delivered to an elliptical Mars orbit via an additional burn of the in-space propulsion stage, or be delivered by an additional transfer element. The cost of the additional burn or transfer element are not included in this work.

Flexible TPS wraps around the windward side of the HIAD to absorb and deflect the heat from atmospheric entry. The flexible TPS is made of several layers of thermally insulating fabrics. Figure 5 shows a photo of the first generation of HIAD flexible TPS. The outermost layers are made of a refractory fabric consisting of several layers of woven ceramic materials such as Nextel or Silicon Carbide. The outer layers are sized to handle the heat rate experienced during entry. The inner layers are made of several layers of insulating fabrics such as carbon felt, Pyrogel, or Aerogel. The insulating layers are sized to handle the heat load experienced during entry. The innermost layer is made of a single Kapton layer serves as a gas barrier, since the outer and inner fabric layers are porous, the Kapton layer prevents airflow from reaching the HIAD structures [23–26].

The inflatable structure is made of several stacked inflatable tori held together by Kevlar straps[27, 28]. The configuration of tori and Kevlar straps for a 6 m HIAD test article can be seen in Fig. 6. Each torus is made of a flexible polymer inner bladder surrounded by a composite braid for support.

There have been several previous flight tests of HIAD technology know as the Inflatable Reentry Vehicle Experiments (IRVE)[29, 30]. The IRVE vehicles carried tanks of pressurized gas to inflate the HIAD structures, but future HIADs will likely use gas generators instead[31]. The use of gas generators significantly reduces the mass of the inflation system and will eliminate the risk involved with carrying an additional pressure vessel on a long duration spaceflight.

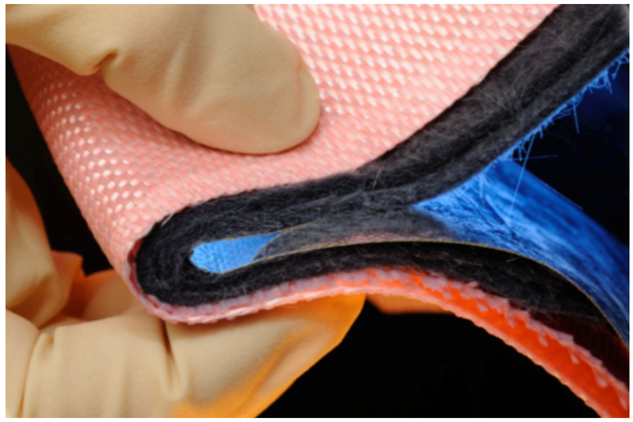


Fig. 5 Photo of first generation HIAD TPS

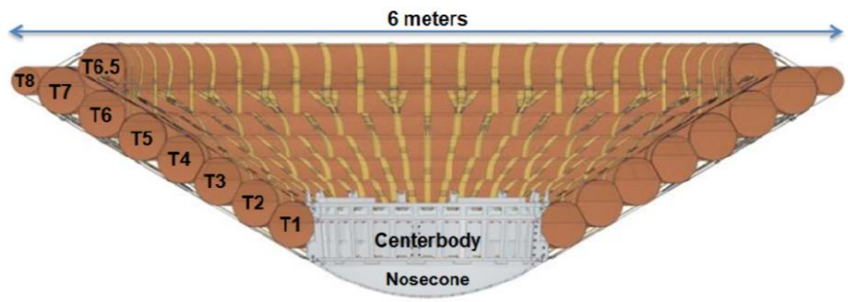


Fig. 6 Cross section of 6 m HIAD test article's structures

B. HIAD ConOps

The ConOps for the Dual HIAD is similar to that of ADEPT. The Dual HIAD lander is delivered to a Trans-Mars Injection (TMI) orbit by an in-space propulsion stage, which is not included in this analysis. After separating from the in-space propulsion stage, the Dual HIAD lander approaches Mars with a hyperbolic velocity of 3,758 m/s and aerocaptures into a Mars one-solar day (1-Sol) parking orbit. When needed on the surface, the Dual HIAD lander jettisons its aerocapture HIAD, deploys solar arrays, and loiters in orbit for up to a year.

There are two options for the ConOps for the Single HIAD lander. The first option is to aerocapture in the same manner as the Dual HIAD lander, however, instead of jettisoning the HIAD used for aerocapture, it will be reused for EDL. This option comes with a number of risks that astronauts and mission planners are unlikely to want to take. It is currently unknown whether or not a HIAD can withstand the heat pulse from two entry events. Even if a HIAD can withstand two heat pulses, the TPS and inflatable structures will present a large target for micrometeoroids and orbital debris while loitering around Mars. Additionally, while loitering the inflatable may lose pressure due to gas leakage and the inflatable structures and TPS may be degraded by UV radiation. Gas leakage could be solved by including additional gas generators, and the UV degradation could be solved by UV resistant coatings or materials. The second option for the Single HIAD lander is to forgo aerocapture and have the in-space propulsion stage or an additional transfer element deliver it to an elliptical orbit around Mars.

The EDL ConOps for the Single and Dual HIAD landers is the same. When the lander is needed on the surface, it deploys the EDL HIAD, makes a deorbit burn, and enters the Martian atmosphere. An artists conception of the entry phase can be seen in Fig. 7. In the hypersonic phase of entry, the lander uses its methane (CH4) and liquid oxygen (LOX) Reaction Control System (RCS) for control. Trim tabs may also be used to provide additional control but they are not included in this iteration of the design. The lander takes advantage of atmospheric drag from its EDL HIAD to slow down to supersonic speeds. The lander then ignites its eight CH4-LOX main engines at the proper time (typically at a speed of Mach 2-3) to reach to a vertical velocity of 2.5 m/s when it is 12.5 m above the surface. The lander then descends at a constant rate for 5 s until it lands safely. Once on Martian soil, the EDL HIAD will be retracted to allow for crew and cargo to be offloaded.

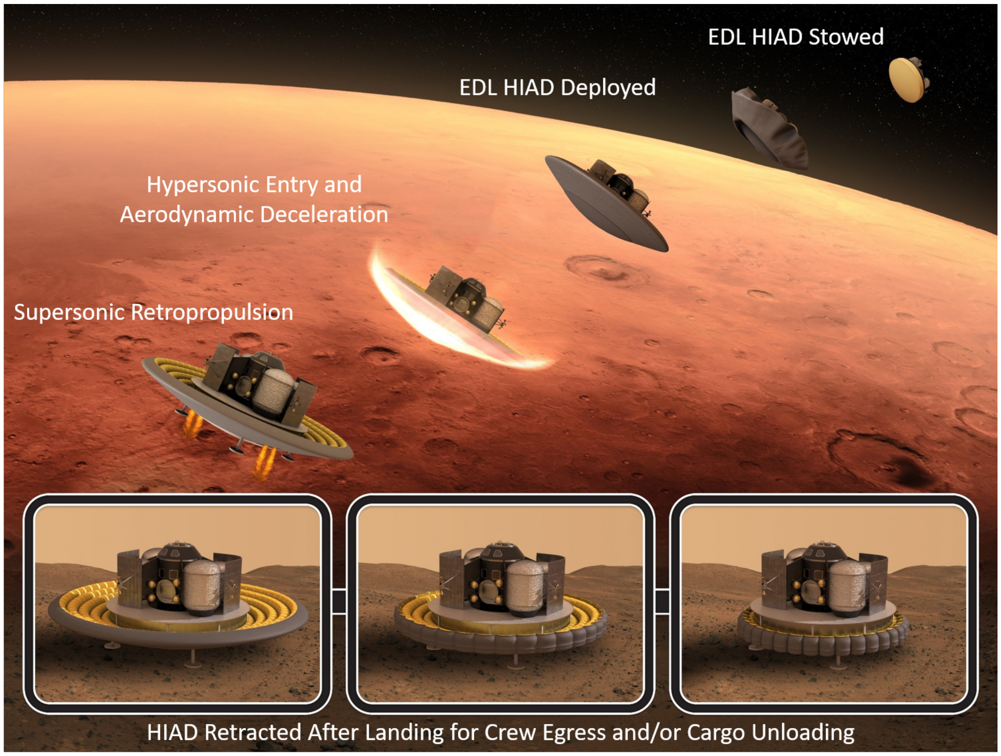


Fig. 7 Baseline HIAD EDL ConOps (Aerocapture not shown).

C. HIAD Cost Estimate and SEER Model

A MEL for both HIAD lander concepts was obtained from the EDLAS team and used in conjunction with a document providing technical descriptions of the associated hardware to produce a SEER model. The Dual HIAD MEL was broken down into 237 different components which translated into 140 work elements in SEER. The Single HIAD MEL was broken down into 203 components which translated into 110 work elements in SEER. The MEL and associated documents contain sensitive information which cannot be freely distributed so they are not included in the present work. However, breakdowns of the subsystems mass and the normalized cost from the SEER estimates of the Dual and Single HIAD landers are provided in Tables. 3 & 4 respectively.

The HIAD and ADEPT MDMs have the same basic design. Many components are identical while others are scaled versions of the HIAD MDM. The Dual HIAD MDM structures are identical to those of the ADEPT MDM; however, the Single HIAD's MDM structures have been scaled down since they do not need to carry the mass of two HIADs and the separation system. As a result, the Single HIAD lander does not require as much propellant and thus its propellant tanks systems are also scaled down. The Power, Avionics, and Thermal subsystems of both HIAD concepts and ADEPT are all identical. The primary cost differences between ADEPT and the Dual HIAD landers are the aerodecelerators and the propulsion systems. The primary reason for the mass/cost savings in the propulsion system is that after aerocapture, the aerocapture HIAD is discarded removing over 3.6 t of mass from the system. This in turn reduces the necessary mass of the propellant tanks, and propellant yielding a total mass savings of yields a total mass savings of 3.1 t. However, the biggest cost savings compared to ADEPT is in the HIADs themselves. The two HIADs are nearly identical, the only differences are the addition of a separation mechanism for the aerocapture HIAD, and doors in the rigid nose section of the EDL HIAD to allow for the main engines to fire and the landing legs to extend. This design replication offers a large savings during development. There are also significantly fewer mechanisms and moving parts involved with the HIADs as compared to ADEPT.

The Single HIAD concept could be significantly less expensive than the Dual HIAD concept depending on the extra costs added to the in-space propulsion stage which are not analyzed in this work. The primary mass and cost savings are from the removal of the aerocapture HIAD which saves on development and production costs. This also allows the structures, and propulsion subsystems to be scaled down saving more cost and mass. In total, the Single HIAD concept saves nearly 6.3 t of mass and development and production costs are reduced by 21% and 24% respectively as

Table 3 Dual HIAD Mass and Normalized Cost Breakdown

	Predicted Mass (kg)		Development		First Unit	
Mission Systems Cost			2.036	29%	0.145	13%
Vehicle Systems Cost			1.631	24%	0.231	21%
Structures	5,422	15%	0.615	9%	0.132	12%
Propulsion	4,665	13%	0.837	12%	0.165	15%
Power	1,568	4.3%	0.186	2.7%	0.053	4.9%
Avionics	333	0.9%	0.231	3.3%	0.047	4.3%
Thermal	411	1.1%	0.039	0.6%	0.003	0.3%
Aerocapture HIAD	3,628	10%	0.673	10%	0.158	14%
EDL HIAD	3,623	10%	0.664	10%	0.159	15%
Dry Mass	19,649	54%				
Non-Propelled Fluids	1,478	4.0%				
Used Propellant	15,535	42%				
Vehicle Wet Mass	36,662	100%	6.912	100%	1.094	100%
Payload	20,000					
Total Stage Gross Mass	56,662					

Table 4 Single HIAD Mass and Normalized Cost Breakdown

	Predicted Mass (kg)		Development		First Unit	
Mission Systems Cost			1.603	29%	0.110	13%
Vehicle Systems Cost			1.284	24%	0.174	21%
Structures	4,694	15%	0.541	10%	0.116	14%
Propulsion	4,637	15%	0.833	15%	0.164	20%
Power	1,568	5.2%	0.186	3.4%	0.053	6.4%
Avionics	333	1.1%	0.231	4.3%	0.047	5.7%
Thermal	411	1.4%	0.039	0.7%	0.003	0.4%
EDL HIAD	3,623	12%	0.725	13%	0.159	19%
Dry Mass	15,266	50%				
Non-Propelled Fluids	1,399	4.6%				
Used Propellant	13,727	45%				
Vehicle Wet Mass	30,392	100%	5.442	100%	0.827	100%
Payload	20,000					
Total Stage Gross Mass	50,392					

compared to the Dual HIAD concept. Again, these figures do not take into account the added mass and cost incurred to the in-space propulsion stage.

VI. Capsule

A. Capsule Overview

The Capsule lander concept was originally developed by a team at the NASA Jet Propulsion Laboratory[3]. Their concept is known as the "Heritage Capsule" because its shape is based on the 70° sphere-cone geometry used by Mars Science Lab, Phoenix, Mars Exploration Rovers, and Viking. The original Heritage Capsule design is 10 m in diameter, has an entry mass of ~75 t, can deliver ~28 t of payload to the surface of Mars, and can support four crew for ~12 days. Its MAV uses a single 250 kN engine using Mono-Methyl Hydrazine and Mixed Oxides of Nitrogen propellants capable delivering the MAV to low Mars orbit. The Mars EDLAS team later took the basic Heritage Capsule design and modified it to fit within their architecture[14]. They changed the design from a 70° sphere-cone to a geometry which more closely resembles a Soyuz capsule in order to increase the usable volume. Even with the increase in volume, the Capsule would not fit the baseline MAV capable of delivering four crew to a 1-sol Martian orbit. Instead, the MAV was redesigned to fit in the Capsule but was limited to taking four crew to low Mars orbit. The propulsion system was also changed to utilize LOX-CH4 engines on both the lander and MAV. These changes resulted in a lander with an entry mass of ~68 t that can deliver ~20 t of payload to the surface of Mars.

B. Capsule ConOps

Figure 8 shows the ConOps for the Capsule in various phases of EDL as well as on the surface of Mars as described by Polsgrove et al.[14]. After the initial aerocapture of the Capsule into a nominal 1 Sol orbit around Mars it loiters for a period of time until it is needed on the surface. It deorbits using its RCS thrusters and at the entry interface has a velocity of 4.7 km/s, flight path angle of -10.6 degrees, and at a -20 deg angle of attack. When at 9.8 km above the surface and traveling at Mach 4.7 the vehicle pitches up to a 0 deg angle of attack and fires its main engines to neutralize its horizontal velocity. The vehicle continues to fire its engines, slowing its descent until it is 12.5 m above the Martian surface, at which point it descends at a constant rate of 2.5 m/s until touchdown.

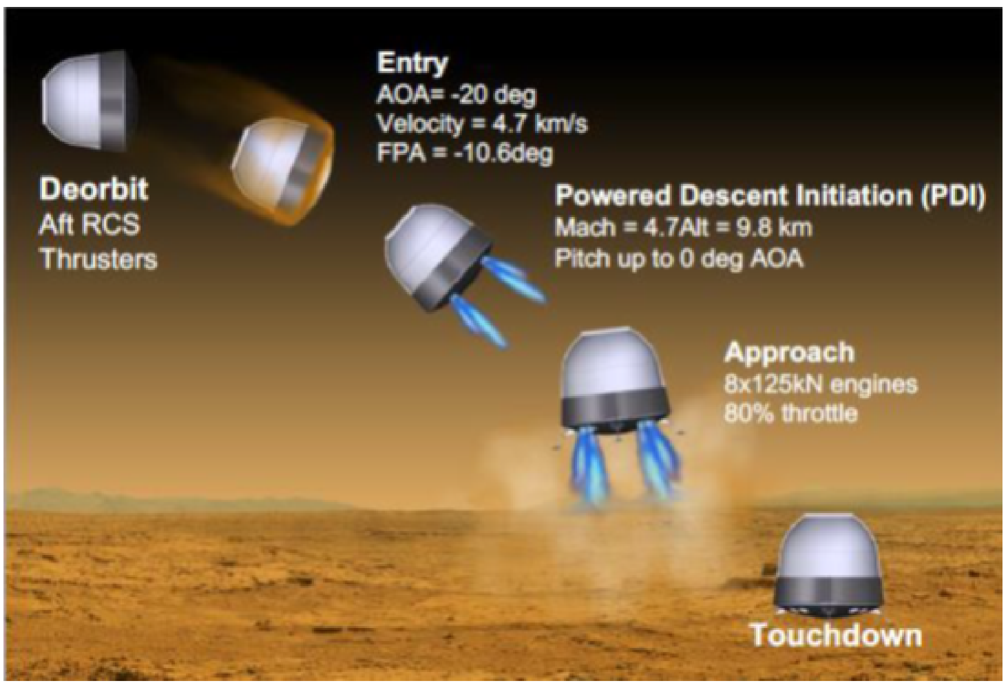


Fig. 8 Capsule EDL ConOps. Aerocapture not shown.

C. Capsule Cost Estimate and SEER Model

A MEL for the Capsule was obtained from the EDLAS team and used in conjunction with a document providing technical descriptions of the associated hardware to produce a cost model in SEER. The provided MEL was broken down into 161 different components which translated into 85 work elements in SEER. The MEL and associated documents contain sensitive information which cannot be freely distributed so they are not included in the present work. However, the mass of the subsystems, and a breakdown of the normalized cost from the SEER estimate are provided in Table. 5.

The costs presented in Table 5 are normalized to the average of the first unit production cost of the five entry vehicles presented in this paper. Table 5 shows that the fraction of the costs allocated to Mission Systems and Vehicle Systems are similar to those of the other concepts. The structures are identical in design and cost to those of the ADEPT and

Table 5 Capsule Mass and Normalized Cost Breakdown

	Predicted Mass (kg)		Development		First Unit	
Mission Systems Cost			1.757	29%	0.124	13%
Vehicle Systems Cost			1.408	23%	0.197	21%
Structures	5,422	11%	0.615	10%	0.132	14%
Propulsion	4,913	10%	0.859	14%	0.169	18%
Power	1,568	3.3%	0.179	3.0%	0.051	5.3%
Avionics	333	0.7%	0.231	3.8%	0.047	4.9%
Thermal	218	0.5%	0.032	0.5%	0.003	0.3%
Capsule Shell	7,025	15%	0.970	16%	0.235	25%
Dry Mass	19,479	41%				
Non-Propelled Fluids	1,944	4.1%				
Used Propellant	26,040	55%				
Vehicle Wet Mass	47,462	100%	6.052	100%	0.958	100%
Payload	20,000					
Total Stage Gross Mass	67,462					

Dual HIAD concepts. The Capsule lander uses a higher thrust set of eight CH4-LOX 125 kN engines than the other concepts use. The only other major difference in the propulsion system of the Capsule compared to HIAD and ADEPT is the scaling of the propellant tanks to allow them to hold the required 26 t of CH4 and LOX. The power and avionics subsystems are all identical to those in the other concepts. The major difference in cost from the other concepts is in the heatshield and backshell that make up the Capsule shell. Since the heatshield and backshell use the same basic design and materials from previous Mars missions, data on the cost of missions such as the Mars Exploration Rovers was used to calibrate the SEER cost estimate. The Capsule shell makes up a quarter of the production cost of the vehicle but only 16% of the development cost.

VII. Comparison of Lander Concepts, Uncertainty Analysis, and Discussion

This section compares the cost estimates of the different lander concepts and discusses the uncertainty in the cost estimates. The point estimates are summarized in Table. 6, where the development and first unit costs are presented along with the total cost to build five units and the total cost to build fifteen units. The development cost is the one time cost associated with designing, testing, and evaluating a given lander design. The first unit cost is the cost of producing the first flight vehicle of a particular design after the design process has been completed. The production cost to build a given number of flight units is given by Wrights Cumulative Average Model given by Eq. 1.

$$P_N = P_i N^b \tag{1}$$

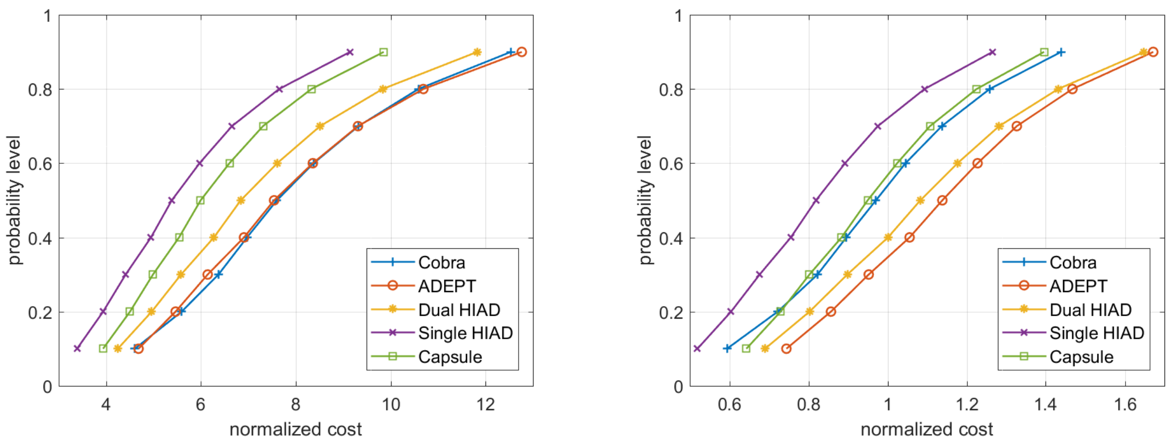
The term P_i is the first unit production cost, N is the total number of units produced, P_N is the total cost to produce the N units, and b is a scaling factor between 0 and 1 that determines how much the cost is reduced for each additional unit produced. In SEER, each component has its own b value determined by SEER's proprietary database and algorithms. Thus, the total cost to build five flight units is the development cost plus P_5 and the total cost to build 15 flight units is the development cost plus P_{15}, where P_5 and P_{15} are determined by Eq 1. Table 6 compares the point estimates of the gross mass, development and first unit costs of each of the lander concepts. Producing five flight units would allow NASA to land a total of 100 t of payload on the surface of Mars and producing fifteen would allow NASA to land 300 t. 100 t of payload would likely only be enough to predeploy assets for a single crewed mission to the surface of Mars. Thus, the five flight units option represents a "Flags and Footprints" style Mars campaign. Whereas the fifteen flight unit option represents a sustained campaign with multiple missions to multiple landing sites on Mars.

Table 6 shows the likely costs of each of the lander concepts, however, just as there is a large uncertainty in the mass of each lander concept there is an even larger uncertainty in the cost of each concept. Validation studies of SEER have shown that the point estimates it produces are frequently inaccurate, however, the costs do fall within the uncertainty

Table 6 Lander Mass and Cost Comparison

	Gross Mass	Development	First Unit	Five Units	Fifteen Units
Cobra	67,531	7.60	0.97	11.25	16.62
ADEPT	59,778	7.62	1.15	11.96	18.35
Dual HIAD	56,662	6.91	1.09	11.03	17.08
Single HIAD	50,392	5.44	0.83	8.58	13.23
Capsule	67,462	6.05	0.96	9.68	15.06

bounds of SEER as expected[7, 8]. In SEER, the user can estimate the uncertainty by doing a Monte Carlo simulation. This is either done with all variables within SEER correlated or uncorrelated. When all variables are correlated this reflects the uncertainty in the actual mission cost from all factors that could affect the cost of a mission, including uncertainty in the design, heritage, mass, capabilities of the contractors, delays in funding, etc. As a result, there is an extremely wide variance in the cost distributions with all variables correlated as seen in Fig. 9. The validation study by



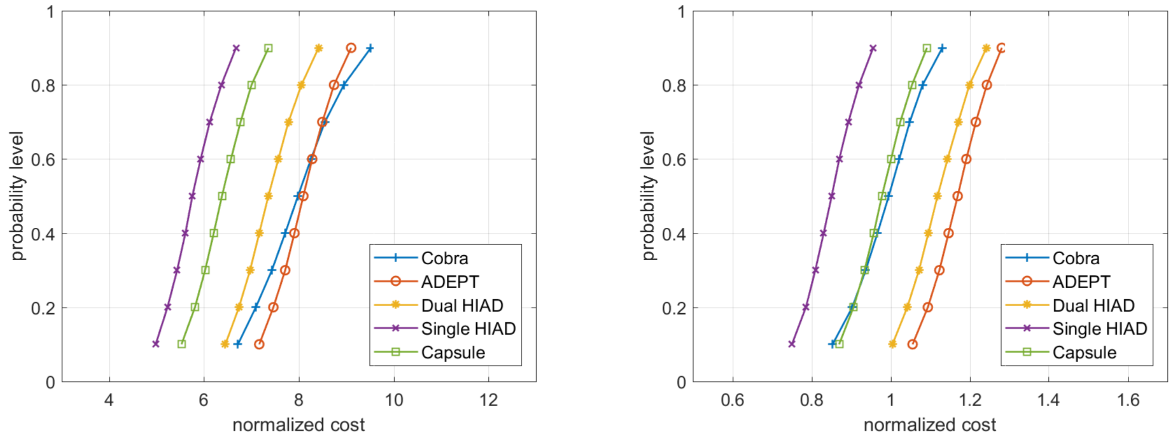
(a) Development Costs Correlated.

(b) First Unit Production Costs Correlated.

Fig. 9 Correlated cost uncertainty representing the uncertainty in the actual cost of developing and producing lander hardware.

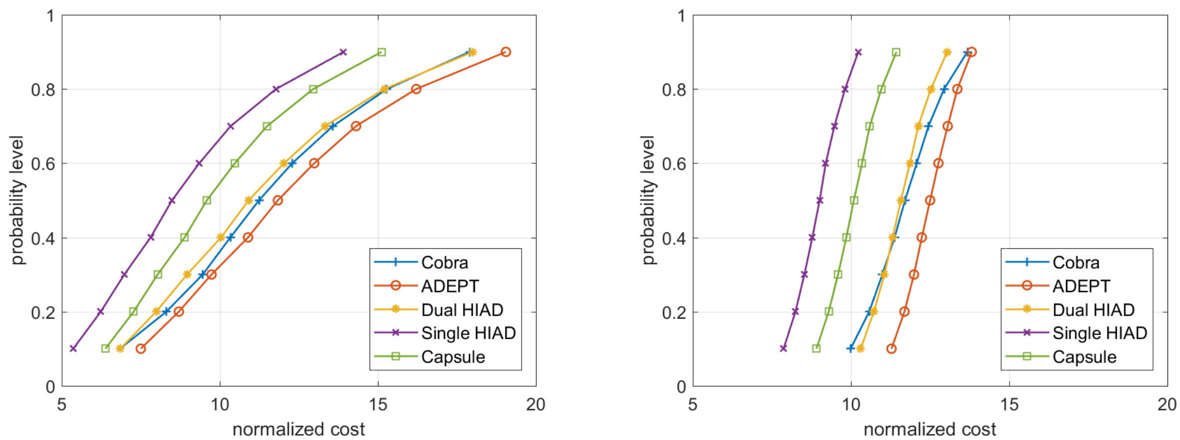
Friz et. al. showed that when using this method the actual costs of missions fall within the 80% confidence interval produced by the distribution as expected[7]. This method is useful from a project manager's perspective when trying to measure the risk of a mission going over budget. However, it is less helpful to a mission planner/decision maker trying to answer the question of whether or not one of the concepts is significantly more likely to be less expensive than the others. For example, each of the concepts uses the same LOX-CH4 engine which needs to be developed. If during testing an engine had an unexpected failure, an investigation would need to take place to determine the cause and ensure that the engine would not fail again in the future. This investigation would add a significant cost to the engine development, and if it delayed the development of other hardware, could affect their cost as well. However, since all concepts use the same engine, this risk will not affect the difference in cost between the different concepts. The ability to define which components will be correlated with the cost of other components within SEER does not exist, the user may only treat all components as fully correlated or uncorrelated. The author is currently working on a methodology to accomplish this outside of SEER. For now though, the CDFs produced from a Monte Carlo simulation with all components uncorrelated is being used as a placeholder. The uncorrelated CDFs of the development and first unit production costs are displayed in Fig. 10. Figure 11 shows the correlated and uncorrelated uncertainty in the total cost of developing and producing five flight units for each lander concept, while Fig. 12 presents the same for fifteen flight units. Note that all Monte Carlo results presented in Figs. 9, 10, 11, and 12 used a sample of 2000 cases. This has been shown in convergence studies to produce errors of less than 1%[32].

Table 6 shows that the Single HIAD lander design is the least massive and least expensive option for developing



(a) Development Costs Uncorrelated **(b) First Unit Production Costs Uncorrelated**

Fig. 10 Uncorrelated cost uncertainty representing the relative uncertainty between the concepts.

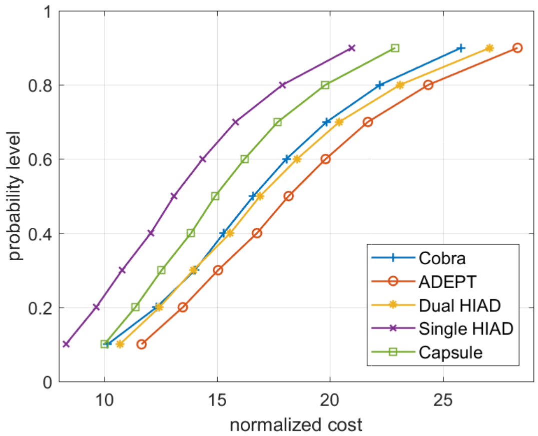


(a) Total cost of five units correlated **(b) Total cost of five units Uncorrelated**

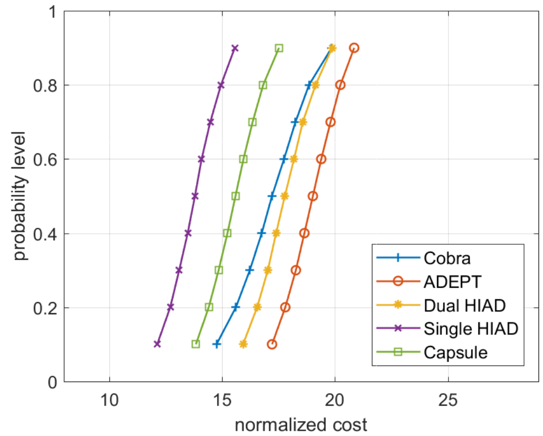
Fig. 11 Correlated and uncorrelated uncertainty in producing five flight units.

and producing both five and fifteen flight units. Figures 9, 10, 11, and 12 all show that there is a large amount of uncertainty in the cost of all the options, but the option which is likely to be the least costly is the Single HIAD lander. Looking at the uncorrelated results in Figs. 11b and 12b the CDFs of the single HIAD and Capsule overlap quite a bit but the distance between the CDFs increases as more units are produced. Meaning that as more and more units are produced it is more likely that the Single HIAD concept will be the least expensive. In both Figs. 11b and 12b there is almost no overlap in the CDF of the Single HIAD concept and the CDFs of Cobra, ADEPT and Dual HIAD, meaning it will almost certainly cost less over a campaign than those three options assuming no additional costs are inured to the in-space propulsion stage. If it is determined that using a Single HIAD for both aerocapture and EDL is too risky, the design can allow the in-space propulsion stage to also perform a large burn to deliver it to an elliptical orbit around Mars. All other lander concepts perform an aerocapture maneuver and thus do not require the in-space propulsion stage to deliver them to an elliptical orbit. The additional cost to the in-space propulsion stage because of the requirement of this additional maneuver will likely outweigh the cost savings of using a Single HIAD lander. However, a detailed cost analysis studying different in-space propulsion technologies and possible trajectories would need to be completed before a definitive answer can be given.

The Capsule concept is likely to have the second least expensive development cost, however, it's mass and production cost are nearly identical to Cobra. Figures 9b and 10b show the CDFs of the production cost of the Capsule and Cobra



(a) Total cost of fifteen units correlated



(b) Total cost of fifteen units Uncorrelated

Fig. 12 Correlated and uncorrelated uncertainty in producing fifteen flight units.

completely overlapping. Figures 11 and 12 show that when building only five units, the Capsule concept is more likely to cost less than Cobra, however, as the number of units produced increases to fifteen the CDFs move closer together making it more uncertain which option will be less costly. If the Single HIAD option proves to be too risky and or incurs too high of a cost on the in-space propulsion stage the Capsule may be the option most likely to cost the least. However, the Capsule has several operational constraints not shared with the other concepts. The capsule has the smallest volume of any of the concepts. Even though the Capsule has the payload capacity to deliver the baseline MAV to the surface, it does not have the required volume to fit the MAV inside. As a result, the MAV would need to be redesigned to only reach low Mars orbit. This would also incur additional costs on the in-space propulsion stage which would need to be redesigned to perform the additional burns to rendezvous with the MAV in low Mars orbit and return the astronauts to Earth. In addition, the Capsule may not have the required volume to carry other necessary elements such as the surface habitat. As a result, the habitat would need to be spread over two landers. The requirement to increase the number of landers on Mars for the same payload would add the unknown costs of additional launch vehicles and in-space propulsion stages which would almost certainly end up making the Capsule more expensive than the other options. A study of the entire Mars campaign would be necessary to fully understand the effect that selecting the Capsule concept would have on the campaigns cost.

Referencing Table 6 and Figs. 9a and 10a the Dual HIAD concept is most likely to have the third most expensive development costs. However, its production costs are likely to be slightly more than that of Cobra. The Dual HIAD lander also has the second lowest mass, and is expected to be nearly 11 t less than the Capsule or Cobra. However, the Dual HIAD's production costs are likely to cost more than Cobra. Figure 11 shows that for a campaign with five landers the cost of Cobra vs. the Dual HIAD are indistinguishable. Increasing the number to fifteen units the CDFs of the Dual HIAD and Cobra in Fig. 12 start to separate due to Cobra's lower production cost but remain nearly indistinguishable. As neither of the HIAD concepts have a backshell they have ample volume to hold the required payloads, but the payload is at an increased risk due to possible flow impingement and radiative heating during entry and aerocapture[33].

ADEPT or Cobra will likely have the highest development cost. Figures 9a and 10a show that the CDFs for the development cost of ADEPT and Cobra are almost entirely overlapping. Figure 10b however, shows that the production cost CDFs of Cobra and ADEPT are overlapping some, but that Cobra is more likely to be less costly.

Turning back to Table 6, the development cost of each concept is six to seven times the cost of the first unit production cost. Also, as the number of units produced increases the production cost per unit decreases. As a result, the cost of total cost of fifteen units is only 53% more expensive than the total cost of five units. In other words three times as many missions to Mars for only a 53% increase in cost.

Landing humans on Mars will be one of the most difficult engineering challenges of the century. It will require many new technologies and vehicles to be developed on a massive scale. With so many unknowns, there will be a large uncertainty in the cost of the mission. Looking at Fig. 11a the cost of developing and producing five Dual HIADs or Cobras ranges from 7-18 in the normalized cost units. This represents an uncertainty range of +63%/-36% from the median cost of just one element of many in a Mars mission. While this uncertainty will shrink as the design matures project managers, mission planners, and those funding the mission should not expect the final cost of a Mars mission to match an initial point estimate.

VIII. Conclusions and Future Work

This work presented cost estimates of four different human rated Mars EDL vehicles capable of landing 20 t of payload on the surface of Mars: Cobra–MRV, ADEPT, HIAD, and Capsule. Currently there is a large amount of uncertainty in both the design of the vehicles and the cost of the vehicles. At this point it cannot be said with certainty which vehicle will be the least costly over a Mars campaign, but it can be said which vehicles are likely to be the least costly. The concept most likely to be the least costly is the version of the HIAD lander with only a single HIAD. However, there are serious doubts that it will be safe to use the same HIAD for both aerocapture and EDL. If it is not possible to use the same HIAD twice it will be necessary to either add a transfer propulsion stage to get the vehicle in orbit around Mars, or increase the propulsive capabilities of the in-space propulsion stage to allow it to deliver the lander to an elliptical Mars orbit. Both of these options are likely to outweigh the cost advantage that the Single HIAD lander has. After the Single HIAD, the Capsule is likely to be the lowest cost option. However, the Capsule concept is severely limited in volume and may require additional flight units to deliver the required assets to the surface of Mars. The Dual HIAD concept is likely to have lower development costs than Cobra, but Cobra is likely to have lower production costs. The Dual HIAD concept is nearly 11 t less massive than Cobra, which could reduce the cost of the in-space propulsion stage, or the HIAD vehicle could be scaled to accommodate a larger payload thereby reducing the cost per ton of payload delivered to Mars as demonstrated by Friz et. al.[34, 35]. ADEPT is likely to have the highest cost. Both HIAD and ADEPT do not have backshells leaving the payload exposed to radiative heating and possible flow impingement. If backshells were added to the ADEPT and HIAD designs it would add both cost and mass to those concepts.

Another finding of this study is that campaigns with more missions are much more cost effective than smaller "Flags and Footprints" style campaigns. It is expected that developing and producing fifteen flight units will only cost 53% more than developing and producing five flight units.

The cost of the in-space propulsion stage must be estimated to determine how each concepts mass will affect the cost of the overall mission. In addition, a more robust uncertainty quantification capability needs to be developed to address the shortcomings of SEER's Monte Carlo simulations. By determining which components of the lander systems have identical cost, independent cost, and dependent cost it will be possible to more accurately quantify the probability that one design will be more or less costly than another.

Even as the designs mature, there will still be a very large uncertainty in the cost of developing vehicles this large and complex. Project managers and those funding the mission should not expect the final cost of the hardware to match the initial point cost estimates.

References

[1] Cianciolo, A. D., and Polsgrove, T. T., "Human Mars Entry, Descent, and Landing Architecture Study Overview," 2016.

[2] Sostaric, R. R., Cerimele, C. J., Robertson, E. A., and Garcia, J. A., "A rigid mid lift-to-drag ratio approach to human Mars entry, descent, and landing," *AIAA Guidance, Navigation, and Control Conference*, 2017, p. 1898.

[3] Price, H. W., Braun, R. D., Manning, R., and Sklyanski, E., "A high-heritage blunt-body entry, descent, and landing concept for human Mars exploration," *54th AIAA Aerospace Sciences Meeting*, 2016, p. 0219.

[4] Venkatapathy, E., Hamm, K., Fernandez, I., Arnold, J., Kinney, D., Laub, B., Makino, A., McGuire, M., Peterson, K., Prabhu, D., et al., "Adaptive deployable entry and placement technology (ADEPT): a feasibility study for human missions to Mars," *21st AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, 2011, p. 2608.

[5] Cassell, A. M., Brivkalns, C. A., Bowles, J. V., Garcia, J. A., Kinney, D. J., Wercinski, P. F., Cianciolo, A. D., and Polsgrove, T. T., "Human Mars mission design study utilizing the adaptive deployable entry and placement technology," *2017 IEEE Aerospace Conference*, IEEE, 2017, pp. 1–16.

[6] Sanchez, S., and Kha, K., "SEER Validation Study Results for NASA Space Science Missions," , 2015.

[7] Friz, P. D., Hosder, S., Leser, B. B., and Towle, B. C., "Blind Validation Study of Parametric Cost Estimation Tool SEER-H for NASA Space Missions," *Acta Astronautica*, 2019.

[8] Friz, P. D., Klovstad, J. J., Leser, B. B., Towle, B. C., and Hosder, S., "Blind Study Validating Parametric Costing Tools Price TruePlanning and SEER-H for NASA Science Missions," *21st AIAA Space Conference*, 2018.

[9] Galorath Inc., “SEER-H Space Guidance: Revision 2.2,” , 2016.

[10] Yarnell, N., “Design, verification/validation and operations principles for flight systems,” *Jet Propulsion Laboratory, D-17868, Rev 1*, Vol. 2, 2003.

[11] “Standard: Mass Properties Control for Space Systems (ANSI/AIAA S-120A-2015) American National Standard,” ????

[12] Brown, J. L., Garcia, J. A., Kinney, D. J., Bowles, J. V., and Mansour, N. N., “Co-optimization of blunt body shapes for moving vehicles,” , May 13 2014. US Patent 8,725,470.

[13] Sepka, S., and Samareh, J. A., “Thermal protection system mass estimating relationships for blunt-body, earth entry spacecraft,” *45th AIAA Thermophysics Conference*, 2015, p. 2507.

[14] Polsgrove, T., Dwyer-Cianciolo, A. M., Robertson, E. A., Percy, T. K., Samareh, J., Garcia, J., Lugo, R., Sostaric, R., Cerimele, C., and Garcia, J. A., “Human Mars Entry, Descent, and Landing Architecture Study: Rigid Decelerators,” *2018 AIAA SPACE and Astronautics Forum and Exposition*, 2018, p. 5192.

[15] Epp, C. D., and Smith, T. B., “Autonomous precision landing and hazard detection and avoidance technology (ALHAT),” *2007 IEEE Aerospace Conference*, IEEE, 2007, pp. 1–7.

[16] Yount, B. C., Kruger, C. E., Cassell, A. M., and Kazemba, C. D., “Deployment Testing of the ADEPT Ground Test Article,” 2014.

[17] Peterson, K. H., Yount, B., Schneider, N. R., Prabhu, D. K., Arnold, J. O., Squire, T. H., Wercinski, P. F., Chavez-Garcia, J. F., and Venkatapathy, E., “Thermal and structural performance of woven carbon cloth for adaptive deployable entry and placement technology,” *AIAA Aerodynamic Decelerator Systems (ADS) Conference*, 2013, p. 1370.

[18] Yount, B., Arnold, J. O., Gage, P., Mockelman, J., and Venkatapathy, E., “Structures and mechanisms design concepts for adaptive deployable entry placement technology,” *AIAA Aerodynamic Decelerator Systems (ADS) Conference*, 2013, p. 1369.

[19] Brown, J., Garcia, J., Kinney, D., and Prabhu, D., “An Asymmetric Capsule Vehicle Geometry Study for CEV,” *45th AIAA Aerospace Sciences Meeting and Exhibit*, 2007, p. 604.

[20] Edquist, K., and Alter, S., “Computational aeroheating predictions for Mars lander configurations,” *36th AIAA Thermophysics Conference*, 2003, p. 3639.

[21] Cianciolo, A. D., and Powell, R., “Entry Descent and Landing Guidance and Control Approaches to Satisfy Mars Human Mission Landing Criteria,” *AAS/AIAA Space Flight Mechanics Conference*, 2017.

[22] Weiss, S. P., “Apollo experience report: Lunar module structural subsystem,” *NASA TN D-7084*, 1973.

[23] Hughes, S., Cheatwood, F., Dillman, R., Calomino, A., Wright, H., DelCorso, J., and Calomino, A., “Hypersonic Inflatable Aerodynamic Decelerator (HIAD) Technology Development Overview,” *21st AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, 2011, p. 2524.

[24] Hughes, S., Ware, J., Del Corso, J., and Lugo, R., “Deployable aeroshell flexible thermal protection system testing,” *20th AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, 2009, p. 2926.

[25] Brune, A. J., Hosder, S., Edquist, K. T., and Tobin, S. A., “Thermal Protection System Response Uncertainty of a Hypersonic Inflatable Aerodynamic Decelerator,” *Journal of Spacecraft and Rockets*, Vol. 54, No. 1, 2016, pp. 141–154.

[26] Calomino, A. M., Dec, J. A., DelCorso, J. A., Sullivan, R. M., Baker, E. H., and Bonacuse, P. J., “Flexible Thermal Protection System Design and Margin Policy,” *9th International Planetary Probe Workshop*, 2012.

[27] Clapp, J. D., Young, A. C., Davids, W. G., and Goupee, A. J., “Bending response of reinforced, inflated, tubular braided fabric structural members,” *Thin-Walled Structures*, Vol. 107, 2016, pp. 415–426.

[28] Swanson, G., Cassell, A., Johnson, R., Hughes, S., Calomino, A., and Cheatwood, F., “Structural strap tension measurements of a 6 meter hypersonic inflatable aerodynamic decelerator under static and dynamic loading,” *AIAA aerodynamic decelerator system (ADS) conference, Daytona Beach, FL*, 2013, pp. 25–28.

[29] Hughes, S., Dillman, R., Starr, B., Stephan, R., Lindell, M., Player, C., and Cheatwood, F., “Inflatable re-entry vehicle experiment (IRVE) design overview,” *18th AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, 2005, p. 1636.

[30] Olds, A., Beck, R., Bose, D. M., White, J., Edquist, K. T., Hollis, B. R., Lindell, M., Cheatwood, F., Gsell, V., and Bowden, E. L., "IRVE-3 Post-Flight Reconstruction," *AIAA Aerodynamic Decelerator Systems (ADS) Conference*, 2013, p. 1390.

[31] Bodkin, R. J., Cheatwood, F., Dillman, R. A., Dinonno, J. M., Hughes, S. J., and Lucy, M. H., "Gas Generators and Their Potential to Support Human-Scale HIADS (Hypersonic Inflatable Aerodynamic Decelerators)," 2016.

[32] Friz, P. D., "A systems and cost analysis of human rated Mars entry, descent, and landing vehicles," 2019.

[33] West, T. K., Theisinger, J., Brune, A. J., and Johnston, C. O., "Backshell Radiative Heating on Human-Scale Mars Entry Vehicles," *47th AIAA Thermophysics Conference*, 2017, p. 4532.

[34] Friz, P. D., Samareh, J. A., and Hosder, S., "An Agile Cost Modeling Approach for Entry Systems Analysis of Human Mars Missions," *21st AIAA Space Conference*, 2018.

[35] Friz, P. D., Samareh, J. A., and Hosder, S., "New Method for Systems and Cost Analysis of Human Mars Entry Vehicles," *Journal of Spacecraft and Rockets*, 2019, pp. 1–15.