

Small Satellite Aerocapture Concepts for Future Interplanetary Missions

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The recent developments in small satellite technology has opened the door to a new paradigm of space missions. Traditional large-scale planetary orbiters and atmospheric probes can be potentially augmented with smaller and cheaper small satellite architectures. The recent developments in entry, descent, and landing (EDL) technology associated with deployable aeroshells may allow for small satellite orbit insertion via aerocapture. This paper serves to understand the design trade space for small satellite mission concepts while identifying potential candidate designs for more detailed analysis.

I. Introduction

THE recent developments in small satellite (SmallSat) technology has opened the door to a new paradigm of space missions. A recent NASA Technical Paper thoroughly documents the current state-of-the-art in small spacecraft technology [1]. SmallSats are a smaller-sized form factor of a traditional satellite. SmallSats are compelling to space mission designers due to their lower-cost associated with the utilization of commercial-off-the-shelf components and rideshare opportunities as secondary payloads. Secondary payload adapters impose restrictive mass and volume requirements for SmallSats, which must stow prior to launch and deploy once released from the adapter, such as the EELV Secondary Payload Adapter (ESPA) [2]. Currently there are many variants of the ESPA platform, including some that are configured to serve as orbital transfer vehicles. Figure 1 depicts the standard ESPA variant. As of 2018, NASA Science Mission Directorate (SMD) has an active policy of integrating ESPA rings on SMD missions that have excess ascent performance for secondary payload rideshare opportunities [3].

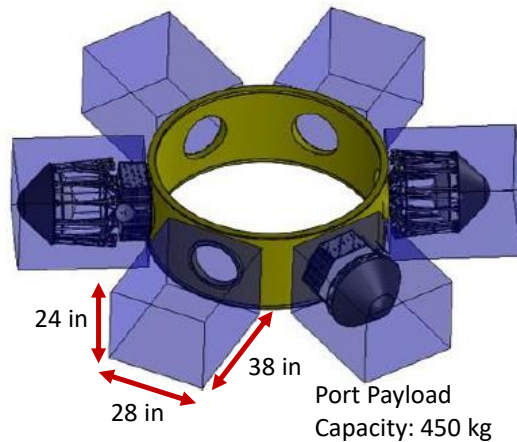


Fig. 1 Standard ESPA configuration with example secondary payloads and port mass/volume requirements. Image adapted from Reference [4]

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Figure 2 shows a visualization of past, current, and future NASA SmallSat missions. The success of the Mars Cube One SmallSats in 2018 signaled the potential applicability of interplanetary SmallSats. Consequently, future missions may require orbit insertion about the destination of interest to achieve the desired science orbit. Unlike traditional satellites, the lower mass and smaller dimensions of SmallSats may make achieving insertion costs on the order of a few km/s quite challenging. Rideshare integration with primary payloads reaching the same destination may alleviate some orbit insertion concerns; however, the desired orbit is not guaranteed. Electric propulsion is a potential alternative for high ΔV , low propellant mass orbit insertion, but at the cost of long insertion time. For destinations with an atmosphere, aerodynamic drag can be used to quickly achieve the high ΔV maneuver through a method known as aerocapture, as shown in Figure 3.

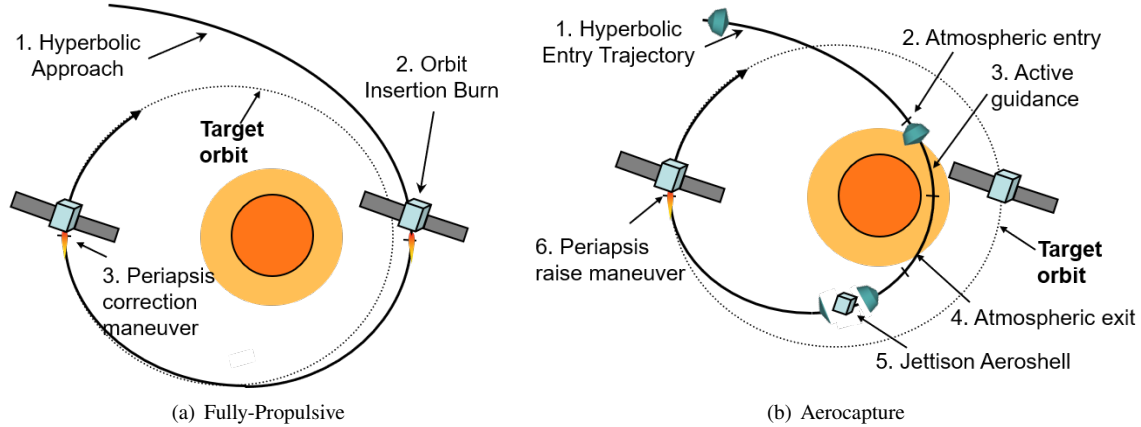


Fig. 3 Comparison of orbit insertion techniques about planetary destinations with an atmosphere. Image adapted from Reference [7]

Aerocapture is an aeroassist technique that aims to significantly reduce orbit insertion costs. Despite not having been demonstrated in space, aerocapture leverages many flight-proven technologies from EDL. These include a protective aeroshell, active guidance, and thermal protection system, amongst others. In 2016, a panel of aerocapture technologists from NASA concluded that, given the shared technology with EDL, aerocapture is ready for flight implantation at Mars and Venus [8, 9]. Nevertheless, a fundamental engineering challenge to the applications of existing NASA EDL technology is the consideration of SmallSats as secondary payloads.

Rigid aeroshells are the most used configuration for atmospheric flight missions. The secondary payload fairing specifications will constrain the maximum rigid aeroshell diameter conceivable. Existing atmospheric entry probes, such as Pioneer Venus, have diameters suitable for fairing specifications. Rigid aeroshells have advantages in certain areas, such as ability to withstand higher aeroheating environments and mechanical simplicity. On the other hand, deployable aeroshells can stow compactly during launch and deploy to much larger sizes than rigid aeroshells. The investigation of deployable aeroshells is an on-going technological investment by NASA, including mechanically or inflatable deployable hypersonic decelerators. The Adaptable Deployable Entry Placement Technology (ADEPT) is an aeroshell platform that deploys mechanically like an umbrella. The platform has been analyzed for SmallSat payload configurations and mission concepts to Venus and Mars [4]. Earth sounding rocket tests [10] of the CubeSat-sized platform variant, known as Nano-ADEPT [11], demonstrated the potential viability of SmallSat deployable decelerators. The Hypersonic Inflatable Aerodynamic Decelerator (HIAD) is an aeroshell platform that deploys through toroid inflation. The recent work on the platform included a November 2022 Earth-based flight test of a 6m diameter variant [12]. Although HIAD has been proposed for larger Mars human-scale missions, it has seen limited analysis for downscaling to small satellites variants [13]. Understanding the potential capabilities of such a variant is imperative for producing alternative deployable aeroshell designs. As a result, this paper focuses on the investigation of small satellite aerocapture concepts using the HIAD platform. Design analysis for SmallSat-sized HIAD aeroshells and aerocapture orbit insertion performance at Earth, Mars, and Venus are presented.

II. SmallSat HIAD Aeroshell Design

This section provides highlights of the downscaling the HIAD aeroshell for small satellite applications. Detailed analysis of the process, including design trades, can be found in Reference [14]. The purpose of this section is to highlight the key findings that came out of the analysis.

The HIAD aeroshell is downscaled to a variety of potential configurations compatible with secondary payload adapter requirements when stowed. The downscaling analysis is conducted using engineering models developed for the LOFTID aeroshell. Three SmallSat-scaled HIAD designs were analyzed as parameterized by the deployed diameter: 1.5m, 2m, and 2.5m. A 70° sphere-cone with a nose radius equal to one third the diameter is assumed. Figure 4 shows a visualization of the stowed and deployed HIAD design as well as a depiction of the design's compatibility with a standard ESPA payload adapter.

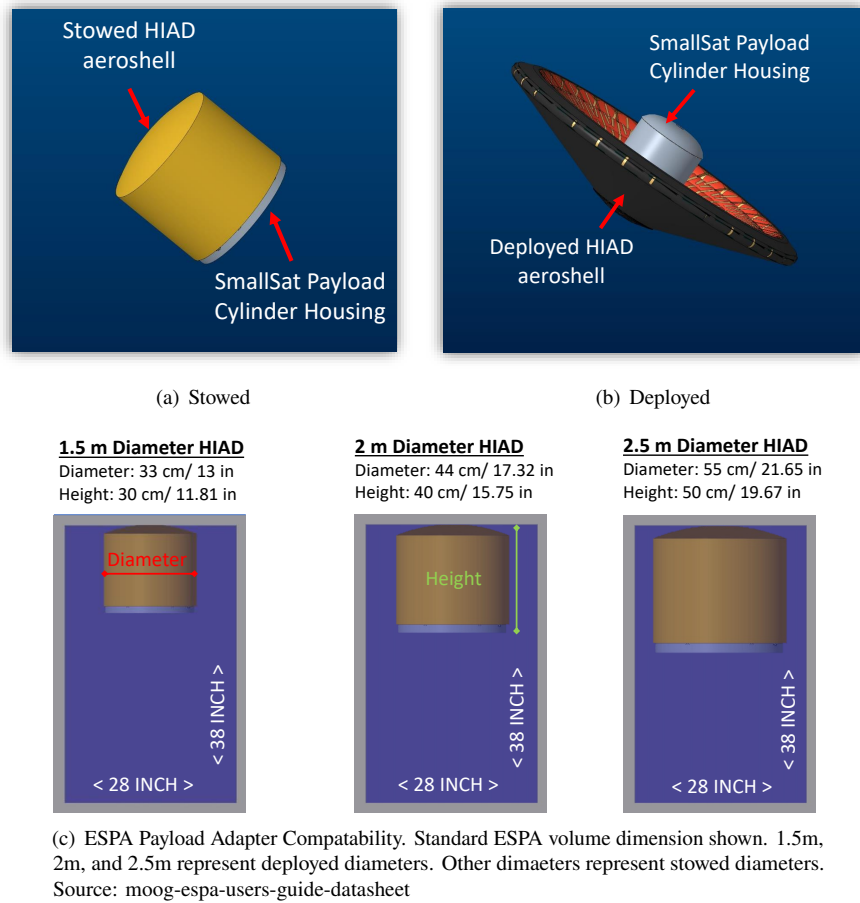


Fig. 4 SmallSat HIAD Entry Vehicle Visualization

Table 1 shows the configuration sizing estimates including the HIAD aeroshell mass, deployed reference area, and ballistic coefficient (BC). As the HIAD scaling increases with deployed diameter, the aeroshell mass increases but the ballistic coefficient decreases. Note that the ballistic coefficient calculation does not include SmallSat payload mass. However, all the configuration masses fall well below the 450kg maximum mass requirement for a standard ESPA ring port indicating positive SmallSat payload mass margin.

Table 1 Configuration Sizing Estimates for SmallSat HIAD Aeroshell

Configuration ^a	Mass (kg) ^b	Area (m ²)	BC (kg/m ²) ^c
1.5m	61	1.767	19
2m	85	3.142	15
2.5m	111	4.909	13

^a deployed diameter

^b HIAD aeroshell mass based on parametric scaling of 6m LOFTID design

^c calculated with drag coefficient of 1.8 and aeroshell mass only.

An engineering challenge associated with aerocapture is ensuring the aeroshell is designed to survive the aeroheating environment. Understanding how downscaling HIAD aeroshell effects the environment is important for the selection of potential configurations. A Sutton-Graves peak stagnation point convective heat rate model is utilized to provide an estimate for the aeroheating environment experienced for each HIAD configuration at Earth, Mars, and Venus. Radiative heat calculations are not modeled in this analysis. Figure 5 provides a comparsion of the peak heat rate for

each configuration at each destination targeting the same 600km circular orbit. For the plot, the associated aerocapture trajectories for Earth and Venus have identical entry velocities of 10.3 km/s while the associated trajectory for Mars has an entry velocity of 6 km/s. The results show that peak convective heat rates below 100 W/cm² are achievable with 1.5m, 2m, and 2.5m diameter designs at each destination with the heat rate decreasing with increasing diameter. For the same HIAD configuration and entry velocity, the heat rate experienced at Venus is roughly 1.5 times larger than values at Earth. Despite the differing entry velocity, the heat rate experienced at Mars is roughly 33% and 22% the value at Earth and Venus respectively.

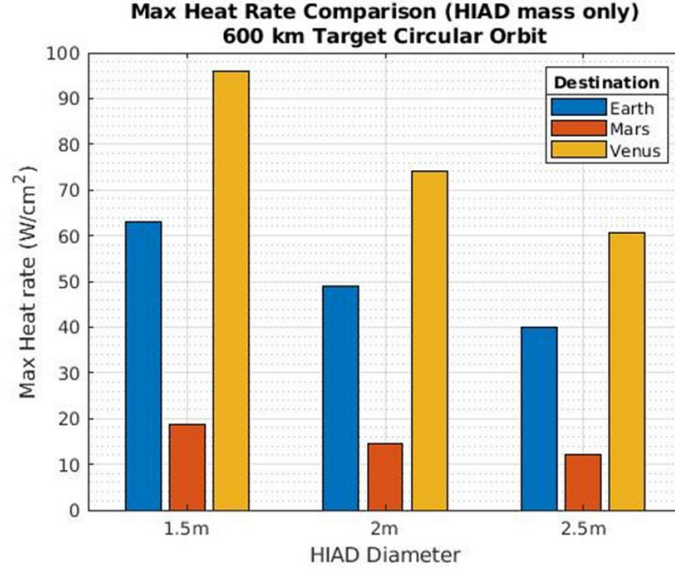


Fig. 5 Peak convective heat rate vs HIAD diameter for Earth, Mars, and Venus aerocapture trajectories

The results presented in Figure 5 neglect payload mass. Consequently, adding payload mass will increase the ballistic coefficient and thereby the resulting peak convective heat rates. For this analysis, a heat rate limit of 75 W/cm² is established based off HIAD flexible thermal protection system (TPS) capabilities. Given the trends, Mars aerocapture with a significantly large SmallSat payload mass (e.g. 150 kg) might be conceivable given the current TPS capabilities. The large heat rates at Venus may require high-performance TPS materials along with large diameter HIAD aeroshells to maximize SmallSat payload mass. Earth-return aerocapture may also follow similar trends as Venus (though lower heat rates allows for larger SmallSat payload masses for a given TPS material limit).

Moreover, Earth-based SmallSat missions may not directly employ using aerocapture in the traditional sense of hyperbolic-to-elliptical orbit insertion but rather as an elliptical Geostationary Transfer Orbit-to-Low Earth Orbit orbital transfer maneuver. Figure 6 shows peak convective heat rate values for such a scenario using two entry mass configuration for each HIAD aeroshell diameter. In the figure, β denotes ballistic coefficient. The min β_E pertains to an entry mass equal to the HIAD mass only, which leads to a minimum value for the ballistic coefficient. The payload β_E pertains to a SmallSat payload mass, with values shown on each respective bar chart, added to the HIAD mass. The payload mass values selected are representative of typical Small Satellite-class payloads, including CubeSats, microsatellites, and minisatellites. Adding the payload masses increases the ballistic coefficient of the 1.5m, 2m, and 2.5m diameter configurations to 24 kg/m², 25 kg/m², and 30 kg/m² respectively. The resulting peak heat rates remain below the 70 W/cm² limit.

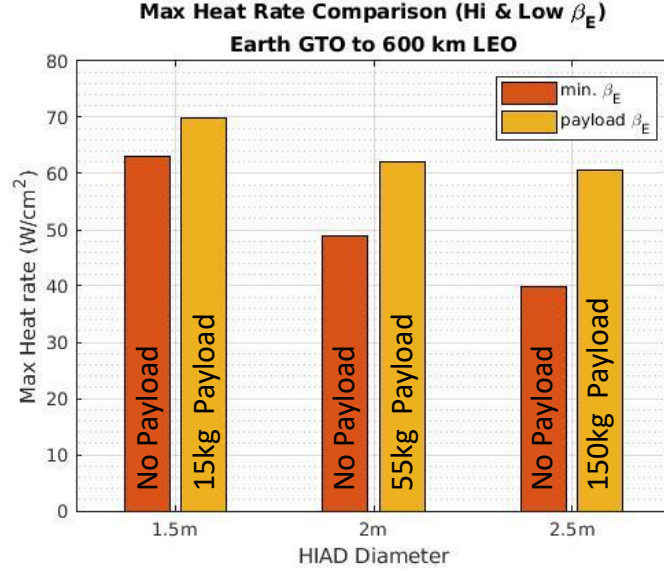


Fig. 6 Peak convective heat rate vs HIAD diameter for Earth GTO to LEO aerocapture trajectories

Overall, this section highlights downsized HIAD aeroshell configurations for SmallSat payloads. Stowed configuration in secondary payload adapters can potentially achieve a deployed diameter up to 2.5m. Analysis of aeroheating environment from SmallSat HIAD trajectories and current flexible thermal protection system limits supports the potential implementation of such configurations for Mars and Earth aerocapture scenarios. The high aeroheating environment experienced at Venus may necessitate larger diameter aeroshell configurations and/or new thermal protection system materials that can increase maximum heat rate limits.

III. Aerocapture Analysis

This section provides aerocapture orbit insertion performance analysis for SmallSat-sized HIAD vehicles. First, the design space for candidate aerocapture mission concepts at Earth, Mars, and Venus is presented to provide qualitative insight into the design space. Second, Monte Carlo simulation orbit insertion results are presented to provide quantitative insight to the qualitative findings. Analysis in this section is conducted using the Program to Optimize Simulated Trajectories II (POST2).

A. Aerocapture Flight Envelope

The uniqueness of SmallSat-focused aerocapture vehicles necessitates an understanding of the trade-space. The potential science missions drives the selection of orbits targeted during aerocapture. Such targeted orbits can vary from low=altitude circular to highly elliptical orbits. The low-mass and small form-factor designs of the SmallSat vehicles may lead to a different range of ballistic coefficients and/or lift-to-drag ratios as compared to traditional rigid aeroshells used to deliver larger satellites and robotic payloads. Each planetary destination has a unique flight environment that influences the orbit insertion performance during aerocapture. It is imperative to analyze the design-space to identify feasible SmallSat mission concepts. One method to understand the trade-space is via the aerocapture flight envelope.

The flight envelope is a metric to understand whether different parameterized vehicle designs have sufficient control authority in the presence of trajectory dispersions. These unforeseeable dispersions, such as atmospheric, aerodynamic, and delivery state, can potentially affect the risk posture of aerocapture. Most aerocapture literature present a metric known as the corridor width. This width, defined by the inertial entry flight path angle, is typically determined by solving for the trajectories associated with the minimum and maximum control limit * that achieve the target apogee altitude. The steep-side is defined by the most negative flight path angle that achieves the target orbit using the maximum control limit while the shallow-side is defined by the least negative flight path angle that achieved the target orbit using the minimum control limit. Individually applying the $\pm 3\sigma$ dispersions and solving for the corresponding width

*lifting: lift down/lift up ; ballistic: min/max ballistic coefficient

allows for a visualization of their effect on diminishing the guidable trajectory-space. The most restrictive bounds that define the flight envelope are determined through aggregation of each dispersion source. Figure 7 provides an example representation of a flight envelope in altitude vs flight path angle space. The solid and dashed blue lines bound the corridor width associated with the flight control limits. The green, red, and black lines show in the individual application of each potential trajectory dispersion type, which visually serve to reduce the bounds of the corridor width. The greyed region, the flight envelope, denotes the resulting restrictive bounds. As a first-order method, having a positive corridor width indicates that the vehicle design has sufficient control authority to overcome the most restrictive of the 3σ trajectory uncertainties. Although Monte Carlo simulations are best used to quantitatively demonstrate control robustness, the qualitative observations and results obtained from the flight envelope analysis generally agree with the overall trends from the Monte Carlo results.

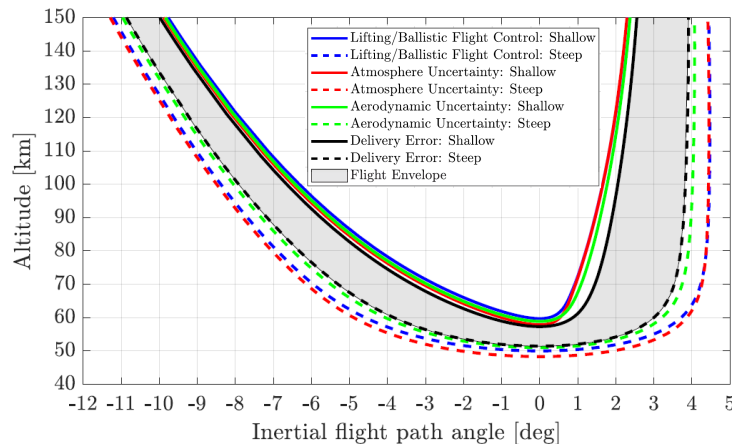


Fig. 7 Sample visualization of an aerocapture flight envelope

The trends from the flight envelope assessment as applied to SmallSat aerocapture trajectories are presented in the paper. Detailed analysis pertaining to the flight envelope assessment can be found in Reference [15]. The purpose of this section is to provide highlights of the key findings that came out of the analysis and how they tie to the overall study conducted.

The aerocapture flight envelope is computed to understand whether the resulting corridor widths are sufficient control-wise. The lifting and ballistic aerocapture trajectories are parametrized by lift-to-drag ratio and ballistic coefficient. For lifting vehicles, the L/D and ballistic coefficient are assumed to be 0.25 and 25 kg/m², respectively. The latter value is selected to be representative of values achieved with a SmallSat HIAD aeroshell design explored in Section II. For ballistic vehicles (L/D = 0), the ballistic coefficient is assumed to be able to increase to a value 20 times the base value of 25 kg/m² (ballistic coefficient ratio (BCR) equal to 20). The built-in optimizer in POST2 is utilized to determine the required inertial entry flight path angle that achieves the target circular orbit altitude between 200 and 600 km at Venus, Earth, and Mars. To normalize the comparison between lifting and ballistic, the same entry velocity is selected and corresponding $\pm 3\sigma$ dispersions for the atmospheric profile and delivery state are assumed. The resulting atmospheric and delivery state dispersions at each planet are different (e.g different GRAM models, different flight path angle dispersion amount). Due to the differences in flight control, the resulting aerodynamic dispersions differ ($\pm 20\%$ in L/D for lifting, ± 0.05 in L/D for ballistic). For each planet, a breakdown of the resulting corridor widths as a function of target orbit with each uncertainty source individually applied is shown as well as a visualization of a sample target orbit's flight envelope.

Figure 8 shows the results for Venus aerocapture with an entry velocity of 11.5 km/s. The atmospheric uncertainties produced by VenusGRAM lead to a small reduction in the corridor width for lifting vehicles but a substantial reduction for ballistic vehicles (both numerically and visually). Aerodynamic uncertainty, as applied to Venus, is shown to produce a pronounced decrease in the corridor width to both lifting and ballistic vehicles. The combined uncertainty, shown by the magenta dashed lines in subplots (a) and (b), correspond to the most restrictive bounds produced by the aerodynamic and atmospheric uncertainties. This can be visualized by the example result shown in subplot (c) where the shallow-side atmospheric (red dashed) and steep-side aerodynamic (green solid) uncertainties form the most restrictive reduction

to the original corridor width bound by the blue solid and dashed lines. The resulting combined uncertainty width should be larger than the delivery uncertainty to indicate that the vehicle has sufficient controllability. At Venus, the lifting vehicle has sufficiently large margin to overcome delivery state dispersions; however, ballistic vehicles have no margin. For the applied delivery state error of $\pm 0.28^\circ$, the resulting width of 0.56° constitutes 68% of the original (no uncertainty) corridor width (see subplot (b)). This indicates that delivery state uncertainty can potentially influence the robustness of the assessed ballistic vehicle at Venus. Subplot (d) indicates that the applied delivery dispersion leads to a negative flight envelope (e.g. grey shaded region does not exist). To achieve a zero-width flight envelope, the delivery state dispersion would need to be decreased to $\pm 0.18^\circ$. This reduction is potentially achievable with improvements in deep space navigation but may not be practical for a secondary payload SmallSat mission. As a result, lifting vehicle configurations for SmallSat payloads might be more desirable for Venusian mission concepts.

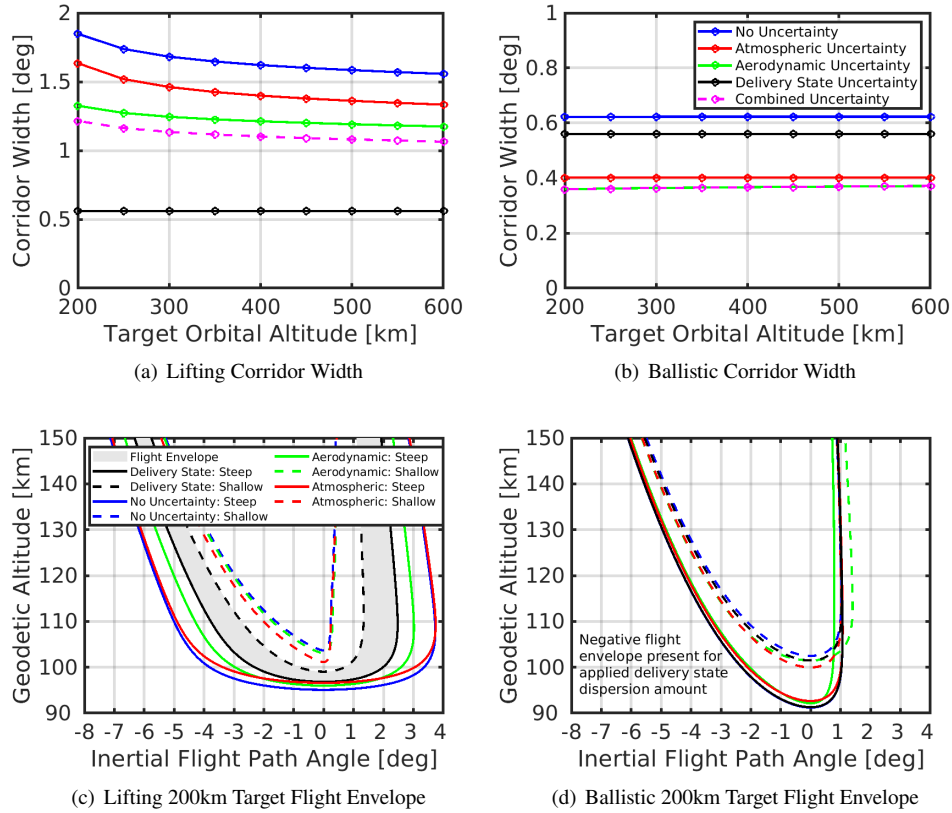


Fig. 8 Comparison of lifting and ballistic flight envelopes at Venus for entry velocity of 11.5 km/s

Figure 9 shows the results for Earth aerocapture with an entry velocity of 11.5 km/s. The atmospheric uncertainties produced by EarthGRAM and simulated aerodynamic uncertainties lead to similar-sized reductions in the corridor width for both lifting and ballistic vehicles. Like Venus, the combined uncertainty is bounded by the shallow-side atmospheric and steep-side aerodynamic uncertainties. Unlike Venus, the ballistic vehicle produces a non-zero margin to allow for potential mitigation of a $\pm 0.25^\circ$ delivery state dispersion. Reducing the BCR down from 20 may lead to a significant reduction or elimination of the margin. As a result, vehicles with smaller BCR capability may require further reductions in the delivery state uncertainty. Overall, both lifting and ballistic vehicle designs housing SmallSat payloads provide sufficient controllability.

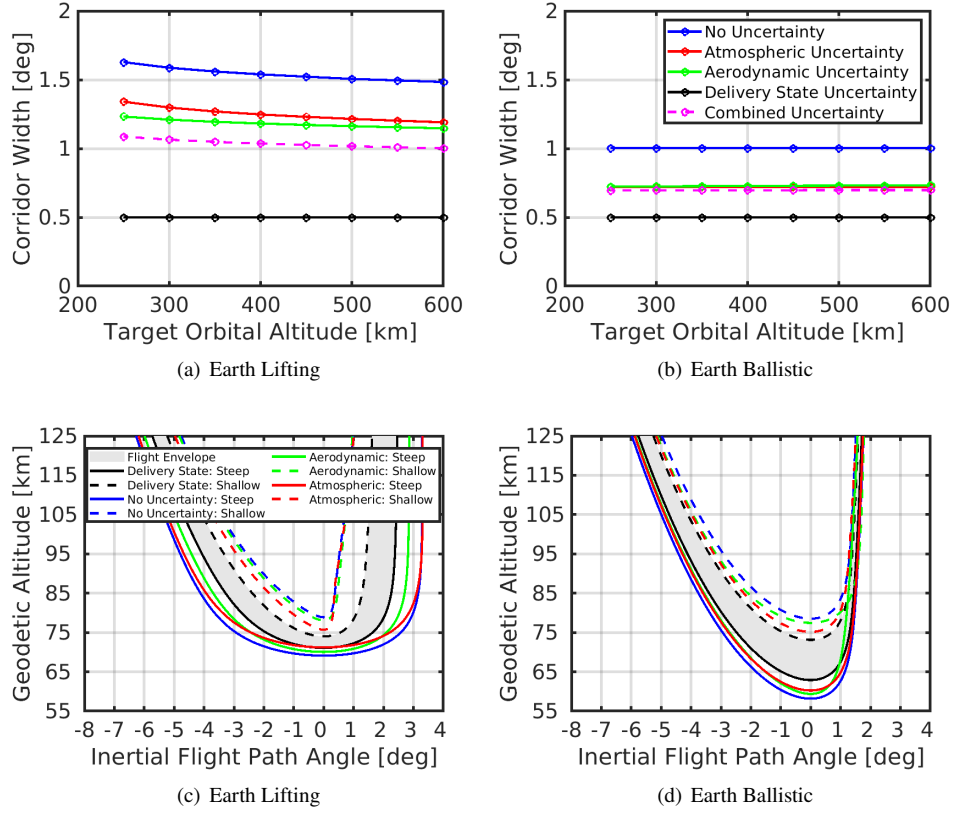


Fig. 9 Comparison of lifting and ballistic flight envelopes at Earth for entry velocity of 11.5 km/s

Figure 10 shows the results for Mars aerocapture with an entry velocity of 6 km/s. The atmospheric uncertainties produced by MarsGRAM and simulated aerodynamic uncertainties lead to similar reductions in the corridor width for both lifting and ballistic vehicles. Like Venus and Earth, the combined uncertainty is bounded by the shallow-side atmospheric and steep-side aerodynamic uncertainties. Like Earth, the ballistic vehicle produces a non-zero margin to allow for potential mitigation of a $\pm 0.2^\circ$ delivery state dispersion. Reducing the BCR capability from 20 may lead to a significant reduction or elimination of the margin. As a result, vehicles with smaller BCR capability may require further reductions in the delivery state uncertainty. Overall, both lifting and ballistic vehicle designs housing SmallSat payloads provide sufficient controllability.

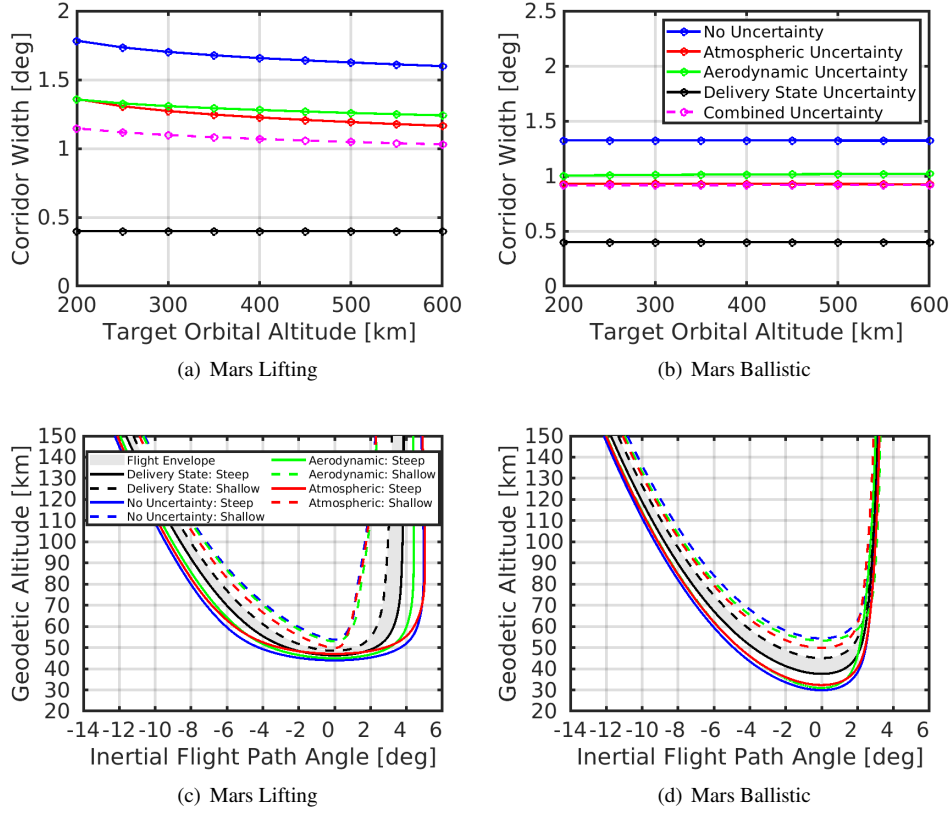


Fig. 10 Comparison of lifting and ballistic flight envelopes at Mars for entry velocity of 6 km/s

B. Performance Under Uncertainty

The aerocapture flight envelope assessment in the previous section provided qualitative insight into how trajectory dispersions may affect the controllability of SmallSat aerocapture trajectories. To provide quantitative insight, Monte Carlo simulations are conducted at Venus, Earth, and Mars.

Orbit insertion performance analysis via Monte Carlo simulations is conducted at each destination using POST2. 8001 Monte Carlo simulations are conducted for each planetary destination using an outer-inner loop POST2 implementation as shown in Figure 11. The outer-loop serves as the truth trajectory model with the same 3σ dispersions sources of atmospheric, aerodynamic, and delivery state as used in the flight envelope assessment. The inner-loop consists of a recursive instance of POST2 running as a generalized numerical predictor-corrector aerocapture (GENNPC) guidance algorithm. The inner-loop contains a simplified nominal knowledge of the vehicle aerodynamics as well as atmosphere. Every guidance cycle, the inner-loop POST2 GENNPC determines the bank angle that minimizes the apoapsis error by numerically integrating the trajectory from the current vehicle state to atmospheric exit. Perfect state knowledge is assumed. A first-order fading memory filter is utilized for calculating correction gains to the inner-loop trajectory model based on comparison of sensed aerodynamic acceleration values from the outer-loop to corresponding values computed using inner-loop trajectory model. A HIAD SmallSat-sized lifting vehicle is assumed in this analysis with a L/D of 0.25, ballistic coefficient of 25 kg/m^2 , and bank angle flight control. For this simplified analysis, no out-of-plane performance is considered. For Earth the guidance targets a 500km circular orbit, for Mars the guidance targets a 400km circular orbit, and for Venus the guidance targets a 300km circular orbit. All performance metrics, including propulsive ΔV costs are computed using POST2.

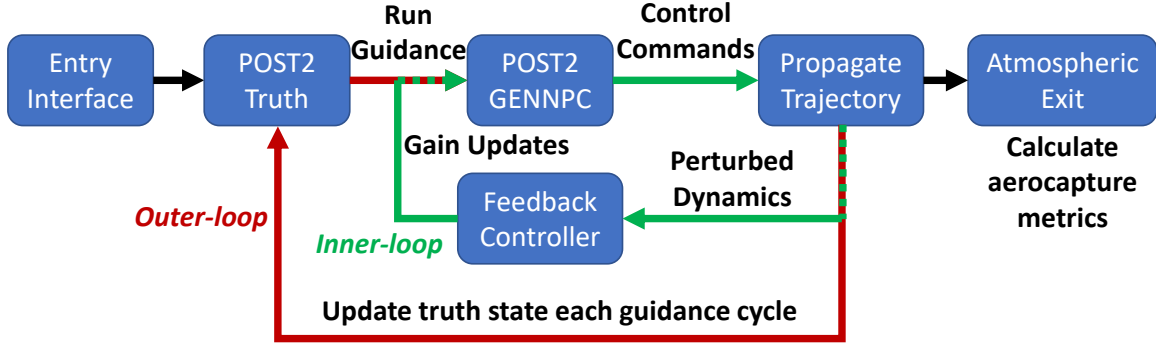


Fig. 11 Flowchart of Aerocapture Monte Carlo Simulations

Figure 12, Figure 13, and Figure 14 show the orbit insertion histograms for Venus, Earth, and Mars respectively. In each figure, the cumulative distribution function (CDF) corresponds to the right vertical axis while the mean, 1σ , 3σ , and 99th percentile statistical values are indicated by vertical lines. At each destination, the bank angle lifting guidance algorithm is able to sufficiently target the desired apoapsis with zero hyperbolic or ground-impact cases. Despite having some noticeable apoapsis error, the corresponding correction ΔV is generally small. The total ΔV contains both the periapsis raise and apoapsis correction costs. Though the exact values vary for each destination, the resulting values are within the 100-150 m/s range. The orbit insertion performance could be improved through further refinements to the guidance algorithm. Nevertheless, the qualitative results from the Monte Carlo simulations corroborate the qualitative findings from the flight envelope indicating that the analyzed lifting SmallSat HIAD design with 0.25 L/D provides sufficient controllability to overcome trajectory dispersions at each planetary destination.

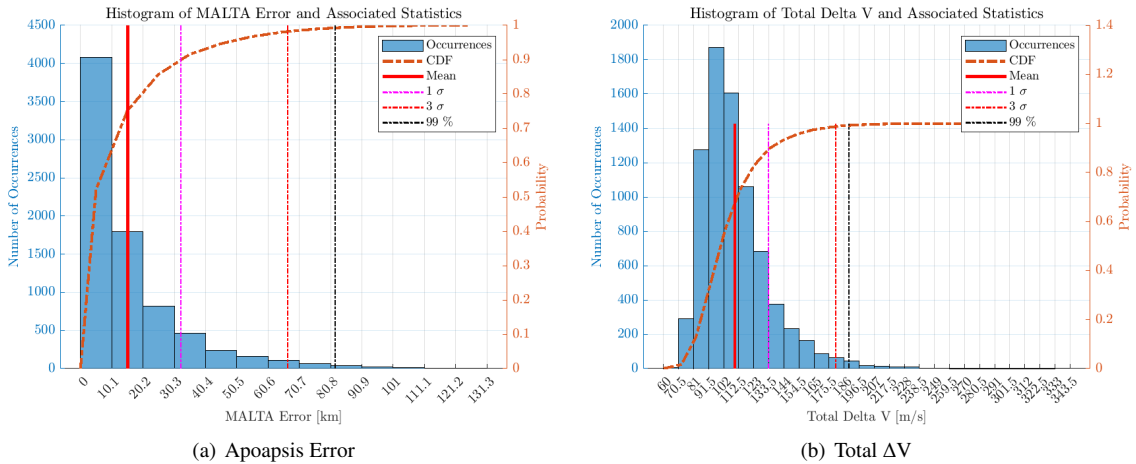


Fig. 12 Venus Aerocapture Orbit Insertion Histograms

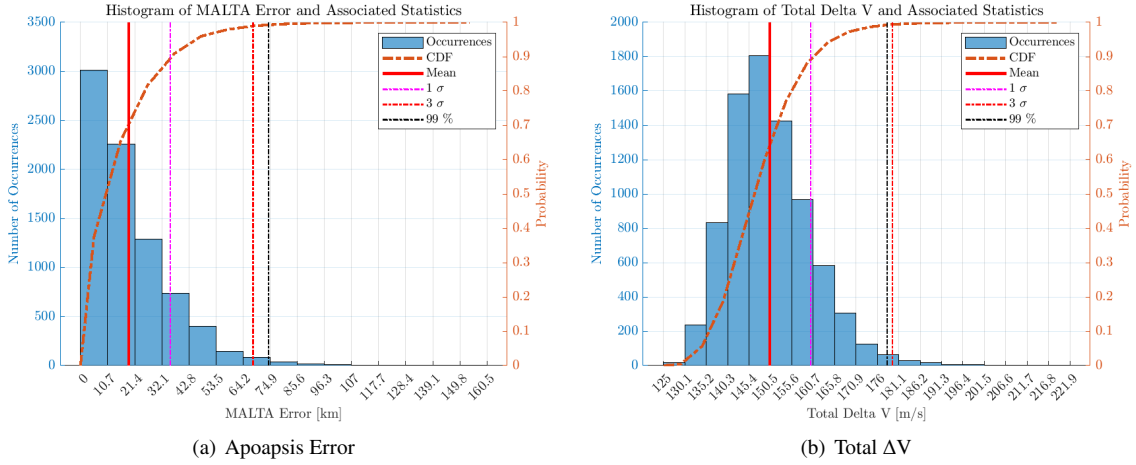


Fig. 13 Earth Aerocapture Orbit Insertion Histograms

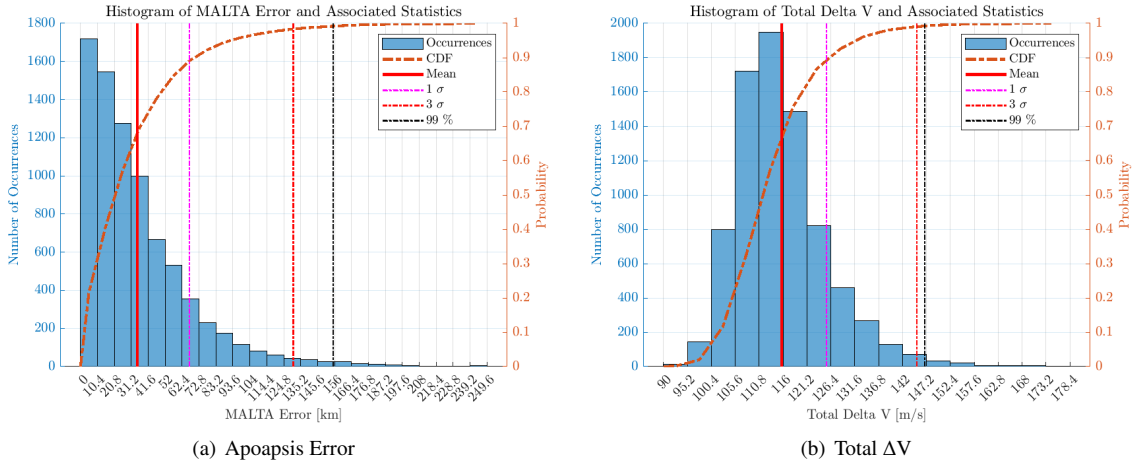


Fig. 14 Mars Aerocapture Orbit Insertion Histograms

To obtain propellant mass estimates, the 99th percentile total ΔV values are utilized assuming the ideal rocket equation. A hydrazine thruster with a specific impulse of 200 sec is assumed along with a entry wet mass of 150kg (25 kg/m² ballistic coefficient using 2m SmallSat HIAD aeroshell + 55kg payload). The resulting propellant mass estimates are shown in Table 2. Despite seeming large, the resulting 99th percentile ΔV values lead to small propellant mass requirements, generally around 10-15 kg. This correlates to less than 10% of the entry mass being allocated to propellant. With a higher specific impulse system, this mass allocation can be potentially reduced. For SmallSat missions, the entry wet mass can be potentially increased up to 450kg based on secondary port payload max capacity. This has the effect of increasing the vehicle's ballistic coefficient and the corresponding aerothermodynamic heating. Generally, increasing the ballistic coefficient generally doesnt affect the orbit insertion performance significantly due to the fact that numerical predictor-corrector guidance algorithms are not dependent on ballistic coefficient in their respective formulations. The resulting 99th percentile total ΔV values can be used as first-order approximations in determining how the corresponding propellant mass changes with changes to entry wet mass. Figure 15 depicts these trends and shows a nearly linear relation of increasing propellant mass with increasing wet mass. But, the corresponding propellant mass allocation percentage is nearly constant generally staying below 10%.

Table 2 SmallSat Aerocapture Orbit Insertion Performance Results

Planet	Capture Rate	99th %-tile Apoapsis Error (km)	99th %-tile Burn Cost (m/s)	99th %-tile Propellant Mass (kg)
Earth	100%	74.189	178	13.045
Mars	100%	155.072	147	10.823
Venus	100%	82.231	189	13.809

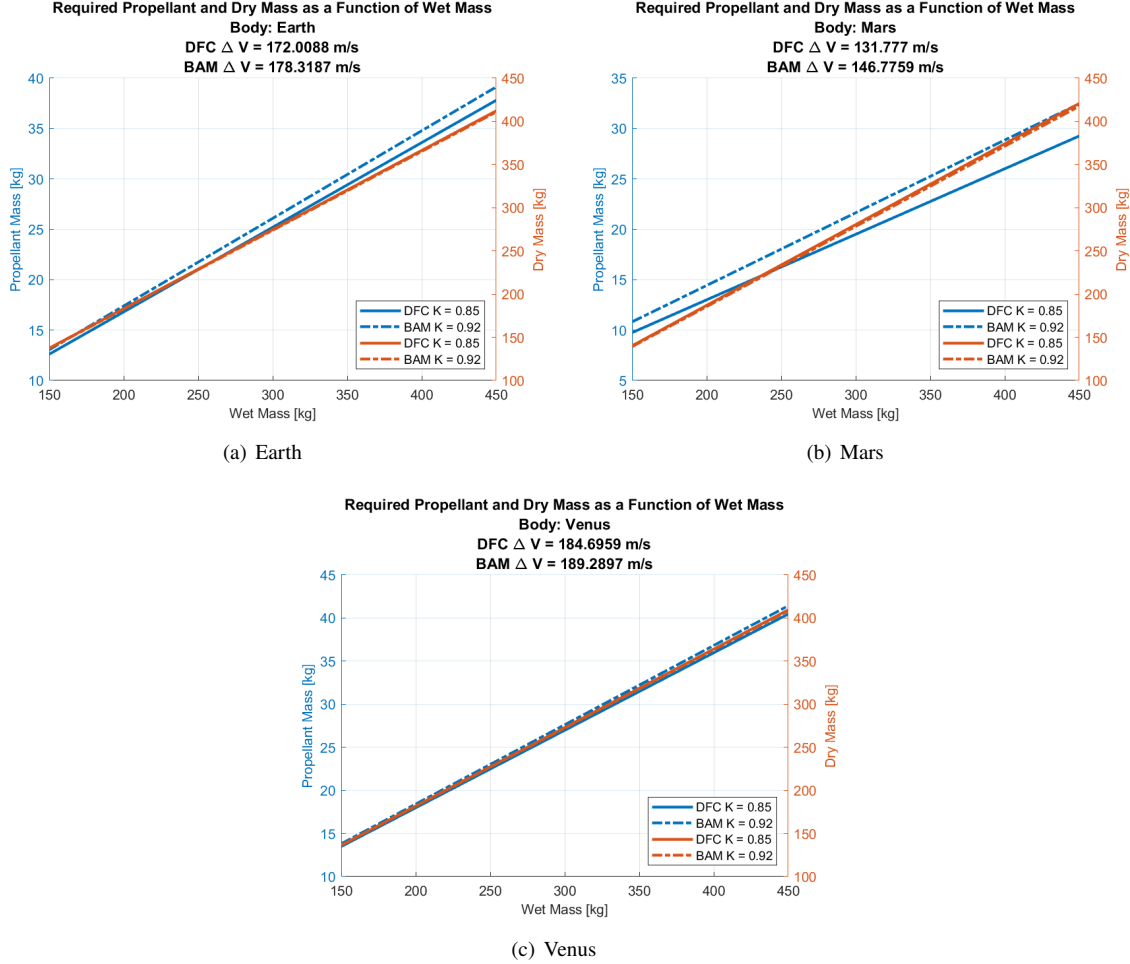


Fig. 15 Propellant mass estimates as function of vehicle wet mass

IV. Conclusion

This paper presents the investigation of SmallSat HIAD aerocapture concepts. Candidate HIAD aeroshell designs that conform to secondary payload requirements are presented along with trends associated with aeroheating environments and aeroshell mass. The design-space for SmallSat aerocapture trajectories is assessed using the aerocapture flight envelope. Trends from the flight envelope are used to inform design considerations in quantitative aerocapture orbit insertion performance simulations. The key findings are discussed below.

Downscaling of aeroshell geometries to accommodate secondary payload constraints is shown to provide a unique challenge in potentially high aeroheating environments, such as Venus and Earth. Smaller aeroshell geometries (e.g. smaller nose radius) along with lower max heat rate value associated with flexible TPS, limits the design-space for SmallSat-sized aeroshell designs. Potential design considerations to mitigate this challenge include lowering the ballistic coefficient and increasing the nose radius. The 1.5m, 2m, and 2.5m HIAD designs analyzed in this paper are shown to

not only potentially achieve this design consideration, but also deliver a significant SmallSat payload mass to Venus, Earth, and Mars. Aerocapture flight envelope assessment indicate lifting flight control with modest L/D provides sufficient control margin for SmallSat HIAD aerocapture trajectories at Venus, Earth, and Mars. Although ballistic flight control is shown to provide sufficient control margin at Earth and Mars, the large required increase in ballistic coefficient may prove to be difficult to achieve mechanically. At each destination, Monte Carlo aerocapture simulations of a candidate lifting guidance algorithm indicate 100% capture success rate with orbit insertion ΔV propellant mass allocations below 10% of the total entry mass.

While the results in the paper are not fully extensive, they provide insight into the applicability of interplanetary SmallSat missions using aerocapture and HIAD technology. Further analysis can be done in TPS analysis with higher-fidelity aeroheating models to refine TPS mass fractions and SmallSat payload mass estimates. Furthermore, trade studies between ADEPT and HIAD SmallSat aeroshell design can provide further insight into design considerations.

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