

Flight Envelope Assessment of SmallSat Aerocapture Trajectories at Venus and Mars

Jack A. Joshi
University of Illinois at Urbana-Champaign
104 S. Wright St., Urbana, IL 61801; 312-282-5648
jjoshi8@illinois.edu

Rohan G. Deshmukh
Analytical Mechanics Associates
21 Enterprise Pkwy., Hampton, VA 23666; 757-864-4552
rohan.g.deshmukh@nasa.gov

Soumyo Dutta, Angela Bowes
NASA Langley Research Center
1 Nasa Dr., Hampton, VA 23666
soumyo.dutta@nasa.gov, angela.bowes@nasa.gov

ABSTRACT

Aerocapture is an increasingly studied orbit insertion concept for small satellite (SmallSat) missions beyond low Earth orbit (LEO). Compared to fully propulsive methods, aerocapture reduces the orbit-insertion propellant mass by approaching on a hyperbolic path and using the planetary atmosphere to reduce the vehicle's velocity such that the final target orbit is achieved. This allows for an increase in payload mass delivered to orbit and a reduction in launch-to-orbit time. To analyze the feasibility at Venus and Mars, aerocapture flight envelope analysis is conducted by assessing the guidable trajectory space during atmospheric flight given entry conditions, vehicle properties, target parameters, and planet-dependent trajectory dispersions. The Program to Optimize Simulated Trajectories II (POST2) is used to simulate both ballistic and lifting aerocapture trajectories with SmallSat-compatible aeroshell designs. The entry flight path angle is optimized to achieve a final target orbit for lift up/down and max/min control configurations. When plotted, the resulting area between the steep and shallow trajectories forms a flight envelope with planet-dependent $\pm 3\sigma$ atmospheric, aerodynamic, and delivery state dispersion profiles applied. The results presented in this paper show that SmallSat aerocapture is feasible for lifting aeroshell designs at Mars and Venus as well as ballistic vehicle designs at Mars.

NOMENCLATURE

C_D	=	drag coefficient
C_L	=	lift coefficient
L/D	=	lift-to-drag ratio
m	=	mass of SmallSat [kg]
S	=	aerodynamic reference area [m ²]
β	=	ballistic coefficient [kg/m ²]
ΔV	=	change in velocity [m/s]
σ	=	standard deviation

INTRODUCTION

Currently, only one small satellite (SmallSat) mission (Mars Cube One) has successfully traveled to an interplanetary destination. The Mars Cube One (MarCO) mission was designed by NASA's Jet Propulsion Laboratory with the purpose of relaying signals to the InSight lander on Mars.¹ While the pair of CubeSats for the MarCO mission only weighed 13.7 kg each, they were able to successfully communicate with the InSight

lander during its entry, descent, and landing phase, demonstrating the potential of SmallSats in future missions. Aside from the MarCO mission, there have been few SmallSat missions targeting interplanetary destinations beyond low-Earth orbit, specifically Mars and Venus. In contrast, there have been numerous past, current, and proposed SmallSat missions targeting low-Earth orbits, as seen in Figure 1. Aerocapture provides a potential avenue for increasing the feasibility of SmallSat missions at planetary destinations beyond low-Earth orbit.

SmallSats are typically anywhere from 10 to 180 kg in mass and are more compact than a standard satellite. While they have reduced space for onboard science instruments, their smaller size proves to be advantageous. Specifically, there is the unique capability of having multiple SmallSats onboard the same launch vehicle as part of rideshare missions. Not only does this save time and launch costs, but it also allows for

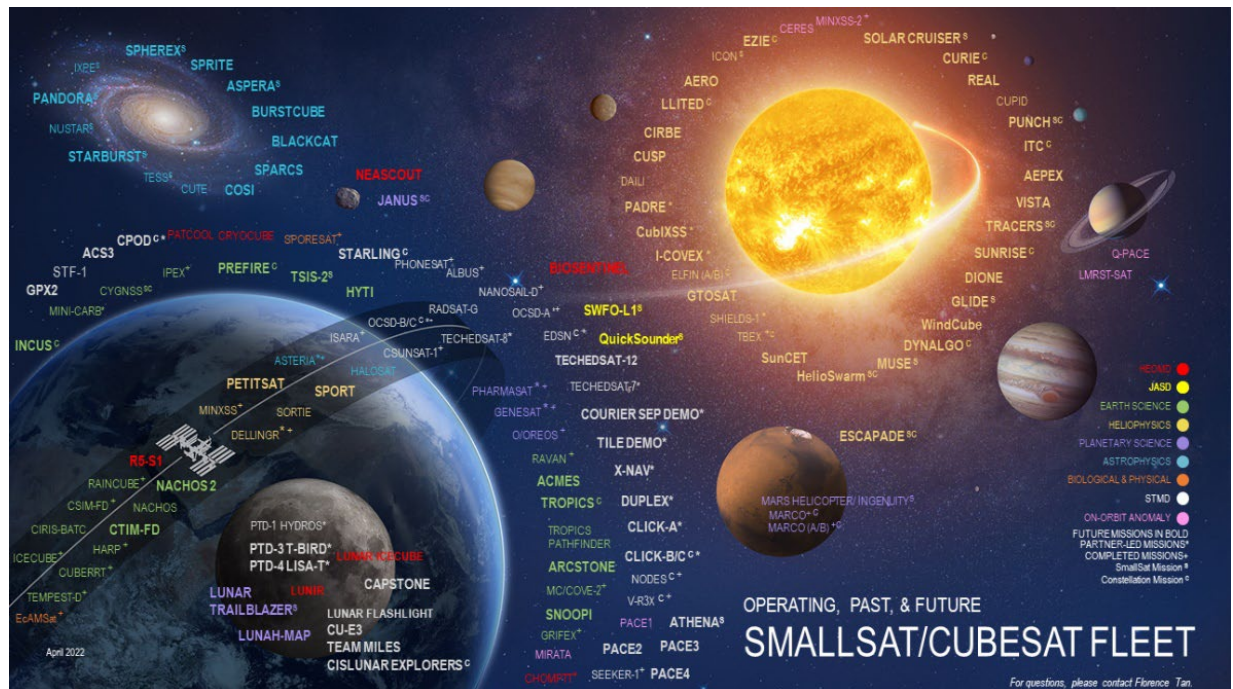


Figure 1: Past, current, and proposed SmallSat missions to Venus and Mars, among other destinations.²

increased science returns since they can reach a greater number of destinations.

One consequence of their size is that SmallSats have reduced onboard fuel capacity that otherwise would be used for propulsive orbit insertion. However, aerocapture provides a powerful alternative compared to traditional fully propulsive methods, allowing for more efficient propellant usage. By leveraging aerocapture techniques, a SmallSat can use a planet's atmosphere to decrease its interplanetary velocity and capture into the final target orbit. The corresponding reduction in propellant mass could potentially allow for additional SmallSat science instrument mass. Hence, SmallSat missions employing aerocapture principles can potentially provide an increased science return. Additionally, since aerocapture performance is less sensitive to entry velocities, the SmallSats can approach at higher hyperbolic velocities, which can reduce the launch-to-orbit time and allow them to begin their mission earlier.

Previous studies have primarily focused on quantitative results when modeling SmallSat aerocapture missions.^{3,4} However, qualitative results can provide a broader perspective of the feasibility of various missions and their corresponding parameters. Additionally, they can give insight into performance capability without computationally expensive Monte Carlo simulations. More specifically, flight envelope assessments produce a visually bounded control space that can determine the

remaining margin of error and highlight any limiting factors. Furthermore, flight envelope analysis can provide estimates for aeroshell design requirements given a specific set of entry, planetary, and target parameters. By assessing the guidable trajectory space for SmallSat aerocapture, the feasibility of using aerocapture as a mission-enabling orbital insertion technique can be explored. This paper presents an overview of aerocapture techniques, applications of aerocapture to SmallSat missions, and flight envelope assessment of candidate missions at Mars and Venus.

AEROCAPTURE OVERVIEW

For an aerocapture maneuver, the vehicle begins from a hyperbolic trajectory on an approach towards the atmospheric interface. Upon entering the sensed atmosphere, guidance commands are activated such that the vehicle can target a final apoapsis upon exiting the atmosphere. These guidance commands allow the vehicle to remain on course while correcting for any unforeseeable trajectory dispersions and are discontinued just before atmospheric exit. During this atmospheric pass, the SmallSat employs an aeroshell and thermal protection system. While this adds additional mass to the SmallSat, it allows the vehicle to handle the planet-dependent aeroheating environment.

After passing through the atmosphere on a single drag pass, the vehicle moves along an elliptical orbit with an apoapsis close to the final target apoapsis and a periapsis inside the planet's atmosphere. When the SmallSat

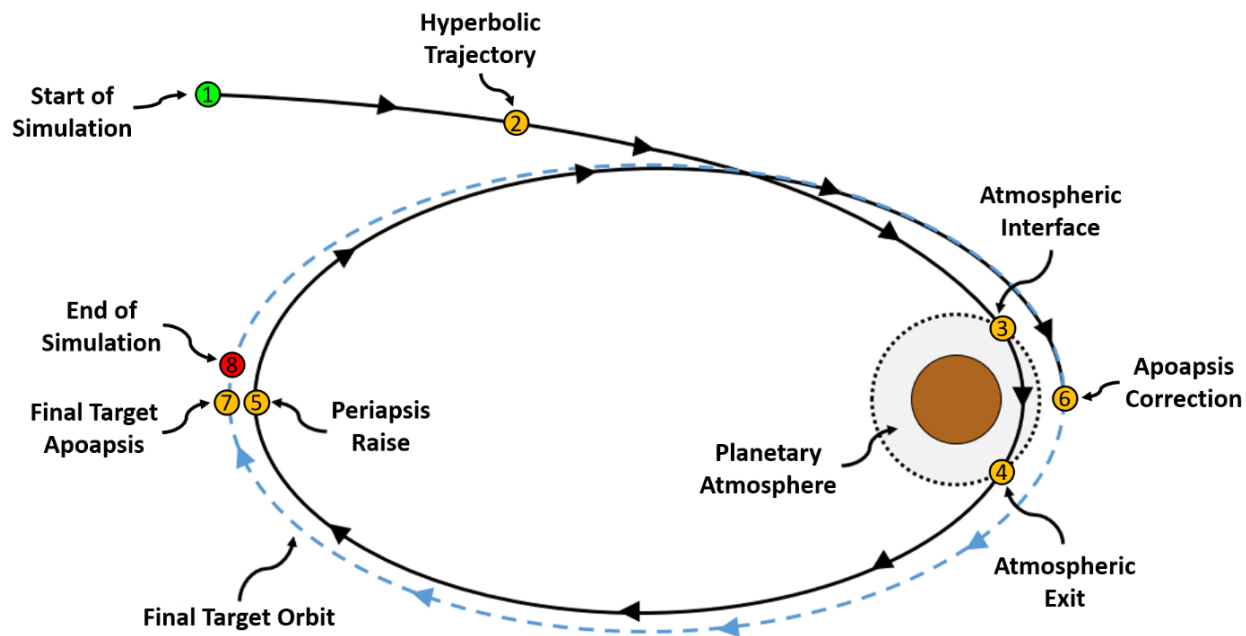


Figure 2: Nominal aerocapture trajectory for SmallSat vehicle.

reaches the apoapsis of its new orbit, a periapsis raise maneuver is employed in order to increase the periapsis of the orbit to the final target value. Similarly, once the vehicle reaches the final periapsis, an apoapsis correction maneuver is performed such that the vehicle's final orbit aligns with the target orbital parameters. For both the periapsis and apoapsis maneuvers, the combined ΔV required is a fraction of the hyperbolic orbit insertion cost associated with fully propulsive methods. These various phases of aerocapture for a SmallSat vehicle are illustrated in Figure 2.

SMALLSAT MISSION DESIGN

In order to identify appropriate entry states, target values, and vehicle parameters for a SmallSat aerocapture mission, an in-depth study of past, current, and proposed interplanetary missions is performed.

Mars Mission Parameters

For past and current Mars missions, the 2001 Mars Odyssey, Mars Global Surveyor (MGS), and Mars Reconnaissance Orbiter (MRO) all had circular or nearly circular polar target orbits with orbital altitudes ranging from 250 km to 450 km.^{5,6,7} Additionally, all three missions employed hundreds of high-altitude atmospheric drag passes, known as aerobraking, as a means to reduce the propulsive ΔV -cost associated with orbit insertion.⁸ These orbital parameters align closely with the values seen in the proposed Next Mars Orbiter

(NeMO) and Aeolus missions, the latter of which has a SmallSat vehicle design.⁹ More specifically, the Aeolus mission envisions a 40.8 kg launch mass and a 383 km circular target orbit with the goal of measuring the polar regions of Mars.

Additionally, previous Mars aerocapture studies have targeted circular orbits within a similar orbital altitude range. For example, one study targeted a 400 km circular Martian orbit with an inclination of 30° .³ Another study conducted by NASA explored an aerocapture mission with a 500 km circular target orbit with an inclination of 45° at Mars.¹⁰ Combining the findings from the aerocapture studies along with the past, current, and proposed Martian missions, the orbital parameters seen in Table 1 are selected.

Table 1: Mars Target Orbit

Apoapsis Altitude	400 km
Periapsis Altitude	400 km
Inclination	30°

While these values serve as a sample target orbit, a range of orbital altitude values from 200 km to 600 km, based on the aforementioned missions and studies, are used in the SmallSat aerocapture analysis presented in this paper. The corresponding atmospheric entry values are seen in Table 2 and the inertial velocity reflects the value seen in a previous SmallSat aerocapture study at Mars.⁴

Table 2: Mars Atmospheric Entry Values

Inertial Velocity	6.0 km/s
Planetodetic Altitude	150 km
Latitude	0°
Longitude	0°
Azimuth Angle	90°

These entry values again provide an example set of parameters for a Mars SmallSat aerocapture mission with a range of entry velocities from 5 km/s to 8 km/s are used for additional analysis. This range of values is consistent with realistic entry velocities based on the Mars aerocapture studies.

Venus Mission Parameters

As compared to Mars, there have been fewer missions that have targeted a Venus orbit. As a result, the primary mission considered for the SmallSat mission parameters is the upcoming VERITAS mission. The VERITAS mission expects to target a 220 km circular orbit around Venus with a nearly polar inclination of 88.5°. ¹¹ This intended target orbit aligns closely with the theoretical circular target orbits of 300 km and an inclination of 90° seen in two Venus aerocapture studies. ^{3,12} Using the parameters from the VERITAS mission and these studies, the target orbit parameters in Table 3 are formed.

Table 3: Venus Target Orbit

Apoapsis Altitude	300 km
Periapsis Altitude	300 km
Inclination	90°

Similar to the Mars target orbit, the orbital altitude values are varied from 200 km to 600 km to provide a range of low altitude, circular target orbits at Venus. Circular target orbits can be especially advantageous for aerocapture missions when compared to fully propulsive methods that would typically require large ΔV insertion burns. Additionally, the entry values, including the inertial velocity, are also based on the previous Venus aerocapture studies, with an emphasis on the parameters for a SmallSat vehicle design.

Table 4: Venus Atmospheric Entry Values

Inertial Velocity	11.5 km/s
Planetodetic Altitude	150 km
Latitude	0°
Longitude	0°
Azimuth Angle	90°

The inertial velocity value seen in Table 4 reflects a middle value for an example Venus entry state but is varied from 10 km/s to 14 km/s to provide a range of entry velocities for analysis.

For this SmallSat aerocapture scenario, the inertial entry velocity at Venus has a greater magnitude than for Mars. Ideally, to reduce interplanetary transfer time, a SmallSat vehicle wants to enter the atmosphere at the highest velocity possible, since faster arriving interplanetary trajectories correlate with larger entry velocities. To dissipate more orbital energy, the aerocapture trajectory must fly lower into the atmosphere where density is higher thereby increasing aerodynamic drag.

However, for thinner atmospheres like Mars, larger entry velocities may necessitate lower altitudes resulting in the vehicle crashing instead of exiting the atmosphere as intended. On the flip side, the escape velocity of the planet will influence how low the entry velocity can be since aerocapture is associated with hyperbolic arrival trajectories. As a result, the feasible range of entry velocities is planet dependent.

Vehicle Parameters

For both planets of interest, two different SmallSat vehicle designs are used. The first vehicle design is a lifting SmallSat design, with a varying bank angle. The lifting vehicle has a L/D ratio of 0.25 and a ballistic coefficient of 50.13 kg/m² as shown in Table 5. These vehicle parameters are for a lift up/down control configuration and are based on a 70° sphere-cone geometry for a SmallSat vehicle with a Hypersonic Inflatable Aerodynamic Decelerator (HIAD) design. ¹³

Table 5: Lifting Vehicle Parameters

Mass	150 kg
Nose Radius	1 m
Diameter	1.5 m
Reference Area	1.76 m ²
Drag Coefficient	1.7
Lift Coefficient	0.425
L/D Ratio	0.25
Ballistic Coefficient	50.13 kg/m ²

The second vehicle design is a ballistic SmallSat configuration with a varying ballistic coefficient. The vehicle parameters are seen in Table 6 and the vehicle has a fixed bank angle of 0°. The vehicle has a mass of 75 kg instead of 150 kg, both of which are within the range of typical SmallSat values.

Table 6: Ballistic Vehicle Parameters

Mass	75 kg
Nose Radius	1 m
Drag Coefficient	1.7
Lift Coefficient	0
L/D Ratio	0
Min. Ballistic Coefficient	25 kg/m ²
Max. Ballistic Coefficient	500 kg/m ²

In order to control the trajectory of the ballistic vehicle, the ballistic coefficient is varied from a minimum to a maximum ballistic coefficient. For this analysis, a change in ballistic coefficient by a factor of 20 from a base value of 25 kg/m² is assumed.

OPTIMIZATION METHODS

In order to simulate both lifting and ballistic aerocapture trajectories, the Program to Optimize Simulated Trajectories II (POST2) is used given a SmallSat-compatible aeroshell design parameterized by ballistic coefficient (β) and lift-to-drag ratio (L/D).¹⁴ The vehicle parameters, entry values, and target orbits align with the information seen in Tables 1-6. For both vehicle configurations, flight path angle is chosen as the independent parameter and POST2 is used to optimize this parameter given a set of target values. In order to improve convergence, the range of flight path angles are constrained within a set range and varied such that the final target orbit error is minimized.

Lifting Configuration

For the lifting vehicle design, the lift up and lift down configurations are simulated by fixing the vehicle's bank angle accordingly. Specifically, the lift up design corresponds to a bank angle of 0° while the lift down design corresponds to a bank angle of 180°. These lift up and lift down control configurations result in steep and shallow trajectories providing a bounded region of maneuverability for the SmallSat vehicle.

Ballistic Configuration

For the ballistic vehicle design, maximum and minimum ballistic coefficients, calculated according to Equation 1, are simulated to bound the corresponding control region.

$$\beta = \frac{m}{C_d S} \quad (1)$$

To vary the ballistic coefficient, the diameter of the vehicle ranges from 0.3352 meters and 1 meter, which subsequently results in bounded reference area values.

As can be seen in Equation 1, the largest reference area value results in a minimum ballistic coefficient (25 kg/m²) whereas the smallest reference area results in a maximum ballistic coefficient (500 kg/m²). Like the lifting vehicle design, the corresponding minimum and maximum control configurations result in a set of steep and shallow trajectories forming a bounded region.

TRAJECTORY ANALYSIS

For both lifting and ballistic vehicle configurations, a flight envelope assessment is conducted to determine the guidable trajectory space during atmospheric flight. The simulated trajectories use the entry values, vehicle parameters, and target orbits specified in Tables 1-6. Initially, the guidable trajectory space consists of a flight envelope with no trajectory uncertainties factored in. However, to provide a more accurate assessment of the control region, atmospheric, aerodynamic, and delivery state uncertainty values are added.

The atmospheric uncertainty used for this analysis is based on the variability of a given point in the atmosphere as modeled in the planet-specific Global Reference Atmospheric Models (GRAM).¹⁵ The respective GRAM models, Mars-GRAM 2010 and Venus-GRAM 2005, factor in variability in density, temperature, pressure, and speed of sound and use these values to create a 3 σ atmospheric uncertainty profile. These $\pm 3\sigma$ atmospheric uncertainty profiles are then applied to both the steep and shallow trajectories, with the two most constraining trajectories forming a new maximum bound for the flight envelope.

For the aerodynamic uncertainty, the lift-to-drag ratio is varied by ± 0.05 , corresponding to an expected $\pm 3\sigma$ dispersion for both a lifting and ballistic vehicle design. More specifically, this results from a $\pm 20\%$ uncertainty for a lifting configuration with a mean lift-to-drag ratio of 0.25.¹⁶ Similar to the atmospheric uncertainty, this aerodynamic uncertainty is applied to the initial steep and shallow trajectories and subsequently reduces the remaining flight envelope.

Lastly, the delivery state uncertainty creates a flight envelope bounded between steep and shallow trajectories that are formed by $\pm 3\sigma$ dispersion profiles. The delivery state uncertainty varies depending on mission-based values and corresponds to a flight path angle dispersion of $\pm 0.2^\circ$ and $\pm 0.28^\circ$ for Mars and Venus, respectively.^{10,12} However, instead of applying the flight path angle dispersions to the steep and shallow trajectories, the dispersions are applied to the middle entry flight path angle value, essentially creating a new flight envelope. This new flight envelope represents the region of maneuverability for the SmallSat vehicle and

ideally falls within the constraining bounds formed by the aerodynamic and atmospheric uncertainties.

Lifting Configuration at Venus

As aforementioned, the initial flight envelope does not factor in any uncertainties and is simply bounded by the optimized steep and shallow trajectories given an initial set of parameters. This initial flight envelope is shown in Figure 3 represented by the blue lines and corresponds to the outermost bounds. However, once the aerodynamic and atmospheric uncertainties are factored in, represented by the green and red lines respectively, the remaining space between the steep and shallow trajectories is greatly reduced. Finally, the delivery state uncertainty is added which creates a final bounded region based on the delivery entry flight path angle dispersions, which is the true flight envelope. For a SmallSat vehicle with a lifting configuration at Venus, the flight envelope, as shown in Figure 3, clearly lies within the bounds formed by the most restrictive aerodynamic and atmospheric uncertainty constraints.

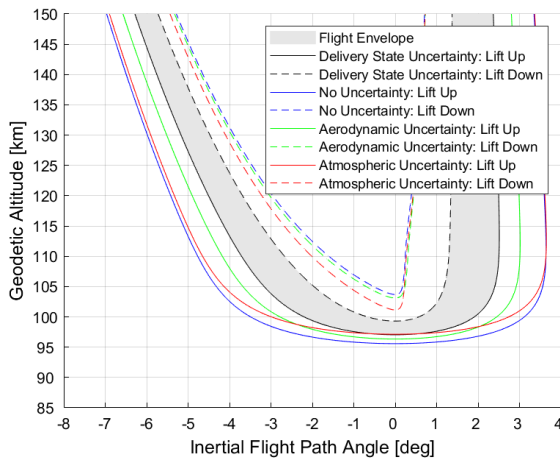


Figure 3: Flight envelope for lifting vehicle design with 300 km circular target orbit at Venus.

While an adequate remaining flight envelope exists for a 300 km circular target orbit at Venus, Figure 4 explores how the corridor width and subsequent flight envelope varies as a function of orbital altitude for a lifting configuration. The initial corridor width seen consists of the difference between initial entry flight path angles for the steep and shallow trajectories. Additionally, the combined uncertainty, represented by the magenta line in Figure 4, consists of the combined effect of the most constraining steep and shallow trajectories affected by the atmospheric and aerodynamic uncertainties. For a lifting vehicle design at Venus, the most constraining shallow trajectory corresponds to the result of the -3σ atmospheric dispersions whereas the steep trajectory

corresponds to the effects of the $+3\sigma$ aerodynamic dispersions.

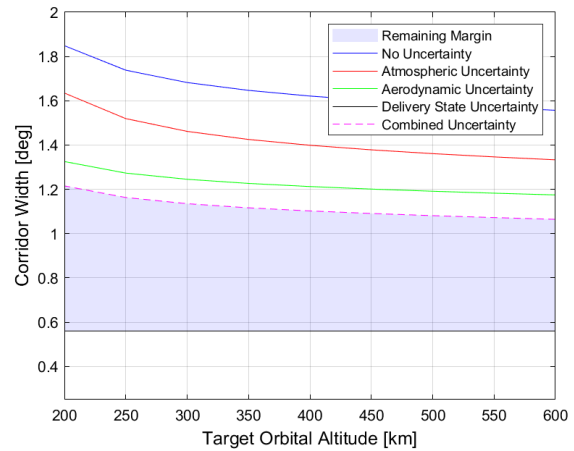


Figure 4: Corridor widths as a function of varying Venus target orbits for a lifting configuration.

As shown in Figure 4, for all target orbital altitudes within the 200 km to 600 km range, there is a remaining margin for maneuverability indicated by the buffer region between the combined uncertainty and delivery state uncertainty lines. This is further highlighted in Figure 5, which shows that there is a substantial remaining margin during the atmospheric pass portion of the aerocapture maneuver for an example 300 km circular target orbit at Venus. Additionally, the delivery state uncertainty and combined uncertainty equally contribute to the reduction of the remaining margin for the SmallSat vehicle.

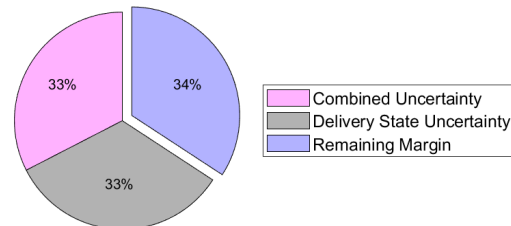


Figure 5: Remaining margin for a 300 km circular Venus target orbit for a lifting configuration.

To gain a better sense of how the remaining margin varies as the initial parameters change, the initial corridor width can provide insight. Similar to Figure 4, Figure 6 displays the corridor width as a function of orbital altitude while also considering the effects of varying the entry velocity. As the orbital altitude at Venus decreases, the corridor width increases showing a slight inverse relationship between the two parameters. In contrast, there is a direct relationship between entry

velocity and the corridor width, which indicates that a higher entry velocity leads to a larger corridor width.

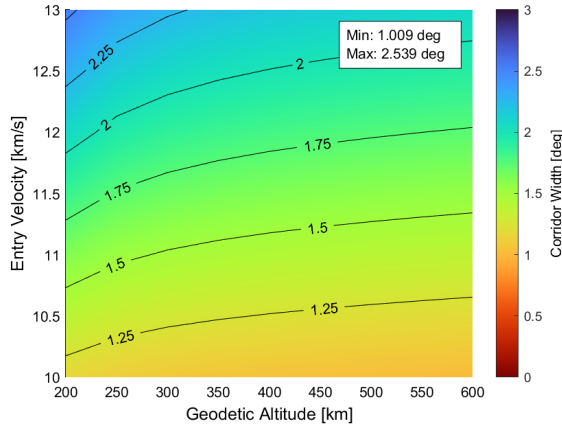


Figure 6: Corridor width as a function of entry velocity and target orbital altitude at Venus for a lifting configuration.

Figure 6 also indicates that the entry velocity has a much greater effect on the corridor width compared to the orbital altitude of the circular target orbit. However, both an increase in entry velocity and a reduction in orbital altitude can feasibly result in a greater remaining margin for a lifting SmallSat design at Venus. Additionally, the results indicate that the same corridor width could be achieved by reducing the vehicle's required control capability but increasing its entry velocity.

Ballistic Configuration at Venus

In order to assess the feasibility of aerocapture for a ballistic configuration at Venus, similar techniques are used to apply atmospheric, aerodynamic, and delivery state uncertainties to an initial flight envelope.

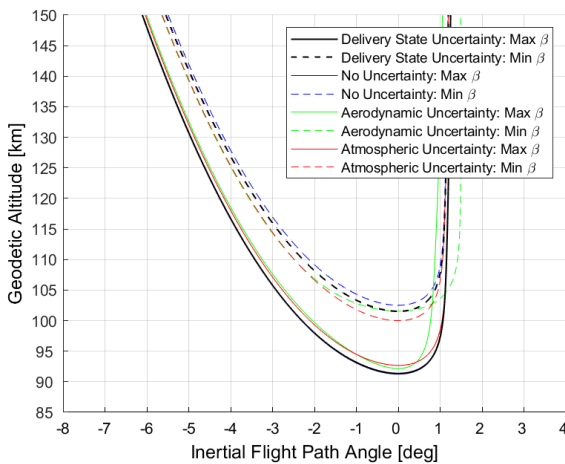


Figure 7: Flight envelope for ballistic vehicle design with 300 km circular target orbit at Venus.

However, after applying the uncertainties, no remaining flight envelope exists as shown in Figure 7. This lack of flight envelope results from the aerodynamic and atmospheric uncertainties being too constraining for a ballistic vehicle at Venus, along with the variation in the delivery entry flight path angle being too large. One potential method to improve the performance of a ballistic vehicle at Venus is to reduce the delivery state uncertainty. As shown in Figure 8, the corridor width associated with the delivery state uncertainty lies above the constraining bounds formed by the combined uncertainty.

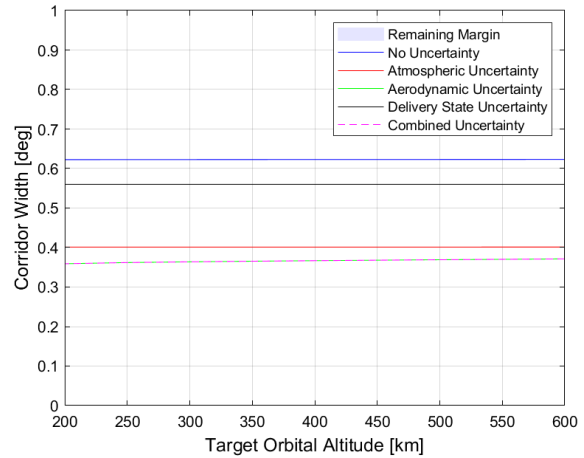


Figure 8: Corridor widths as a function of varying Venus target orbits for a ballistic configuration.

By reducing the corridor width by 0.2° though, the delivery state uncertainty would no longer exceed the combined uncertainty bounds. This corresponds to reducing the entry flight path angle uncertainty from $\pm 0.28^\circ$ to $\pm 0.18^\circ$ which is a substantial reduction but indicates that a ballistic SmallSat mission design at Venus can be possible with improved delivery accuracy. Figure 9 highlights the lack of a remaining margin given the current parameters and visually demonstrates that the delivery state uncertainty is the limiting factor.

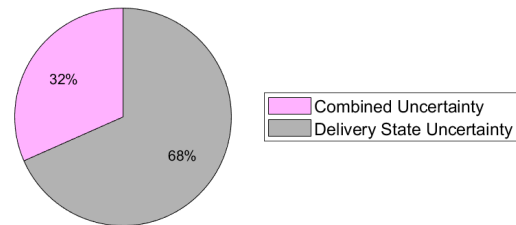


Figure 9: Remaining margin for a 300 km circular Venus target orbit for a ballistic configuration.

The results from the 300 km circular Venus target orbit can be applied to a range of orbital altitudes from 200 km

to 600 km and further indicate that reducing the delivery state uncertainty could provide a remaining margin. Additionally, the final target orbital altitude has minimal impact on the corridor width.

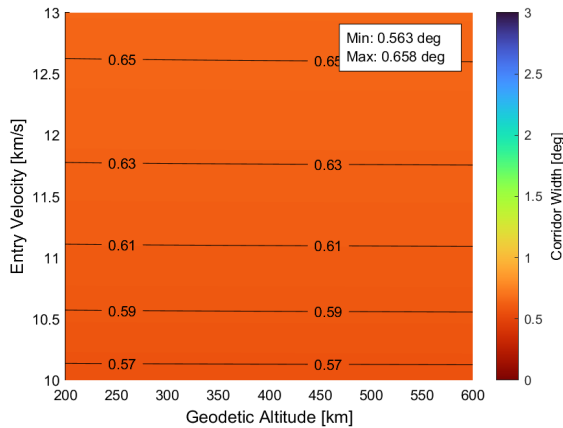


Figure 10: Corridor width as a function of entry velocity and target orbital altitude at Venus for a ballistic configuration.

While increasing the entry velocity slightly increases the corridor width, the change is not substantial. As indicated by the minimum and maximum corridor width values in Figure 10, the greatest possible improvement in corridor width is 0.093° which is well below the 0.2° improvement necessary. Thus, for aerocapture with a ballistic SmallSat vehicle design to be feasible at Venus, reducing the initial delivery state uncertainty would be most effective.

Lifting Configuration at Mars

For the flight envelope assessment at Mars, a 400 km circular target orbit is used to represent the expected performance of SmallSat aerocapture.

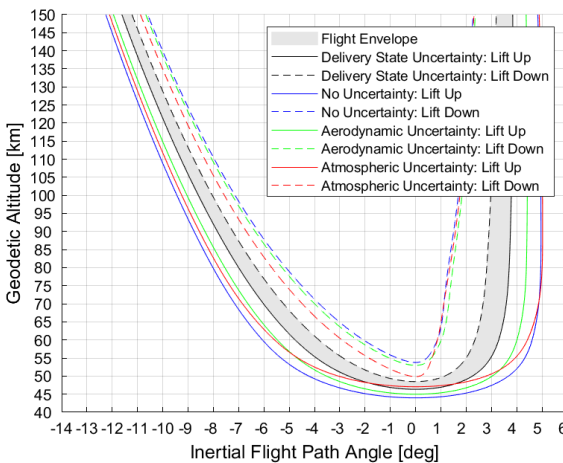


Figure 11: Flight envelope for lifting vehicle design with 400 km circular target orbit at Mars.

Figure 11 indicates that the flight envelope formed by the delivery state uncertainty bounds exists between the most restrictive constraints created by the aerodynamic and atmospheric uncertainties.

In contrast to SmallSat aerocapture at Venus, the trajectory reaches lower altitudes, about 50 km lower, in the atmosphere. This is to effectively achieve higher density values in an already thin Martian atmosphere such that sufficient velocity is dissipated. To achieve this, the optimal entry flight path angles for the lift up/down configurations are much steeper for SmallSat aerocapture at Mars. Similarly, to avoid crashing since the vehicle must approach at a steep entry flight path angle, the feasible range of entry velocities is lower than at Venus. Like the Venus lifting configuration though, the corridor width can be analyzed as a function of orbital altitude, shown in Figure 12.

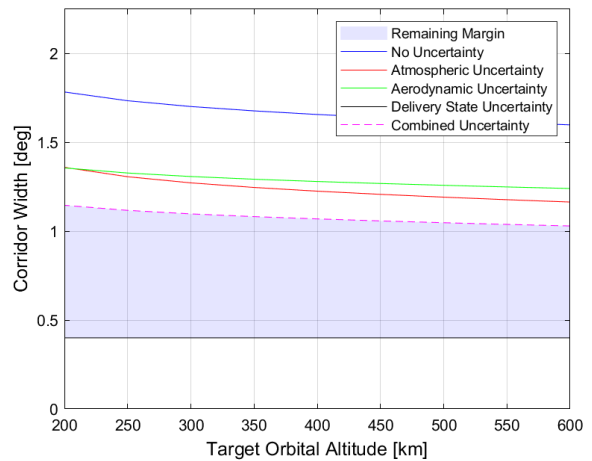


Figure 12: Corridor widths as a function of varying Mars target orbits for a lifting configuration.

Figure 12 shows that the delivery state uncertainty lies well below the reduced corridor width associated with the combined uncertainty. For the combined uncertainty, the -3σ atmospheric dispersions applied to the shallow trajectory and the $+3\sigma$ aerodynamic dispersions applied to the steep trajectory form the most restrictive bounds for the SmallSat vehicle. The results seen in Figure 12 are further reflected in Figure 13 where there exists a substantial remaining margin after factoring in all the uncertainties. The remaining margin for a 400 km circular target orbit at Mars is indicative of the results for the 200 km to 600 km range of target orbital altitudes as shown by the significant area between delivery state and combined uncertainty corridor widths, seen in Figure 12. Figure 13 also indicates that the combined atmospheric and aerodynamic trajectory dispersions account for a greater reduction in the remaining margin than the delivery state uncertainty.

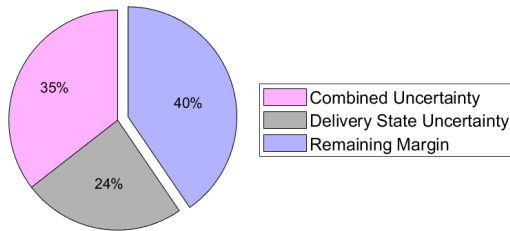


Figure 13: Remaining margin for a 400 km circular Mars target orbit for a lifting configuration.

More specifically, Figure 14 demonstrates that the corridor width increases slightly as the target orbital altitude decreases, similar to the results for a lifting vehicle at Venus. Additionally, the results also show that the entry velocity has a significant impact on the corridor width and that increasing the entry velocity provides more room for maneuverability when entering the atmosphere for aerocapture.

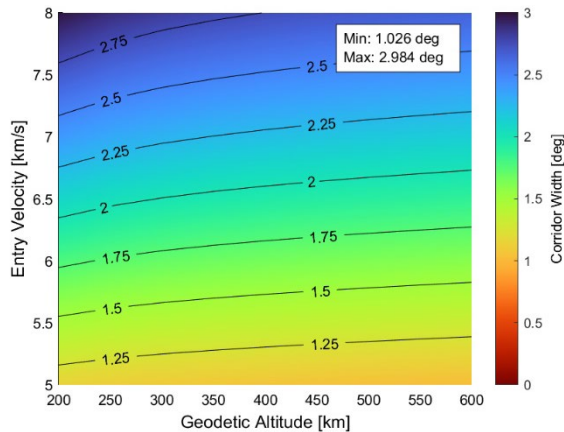


Figure 14: Corridor width as a function of entry velocity and target orbital altitude at Mars for a lifting configuration.

While the current entry and target parameters listed in Tables 1-6 provide a substantial remaining margin for a lifting SmallSat design at Mars, an even greater buffer is possible by adjusting the entry velocity and target orbital altitude parameters appropriately. With the current parameter suite analyzed, these results indicate that aerocapture for a lifting configuration is potentially feasible even after accounting for the various 3σ trajectory dispersions.

Ballistic Configuration at Mars

Using the same 400 km circular target orbit at Mars, the flight envelope visualization for a ballistic SmallSat vehicle is seen in Figure 15. For a ballistic configuration, the steep and shallow trajectories are formed by using maximum and minimum ballistic coefficients (β). The

visualization shows that there is a feasible flight envelope after factoring in the delivery state, aerodynamic, and atmospheric uncertainties.

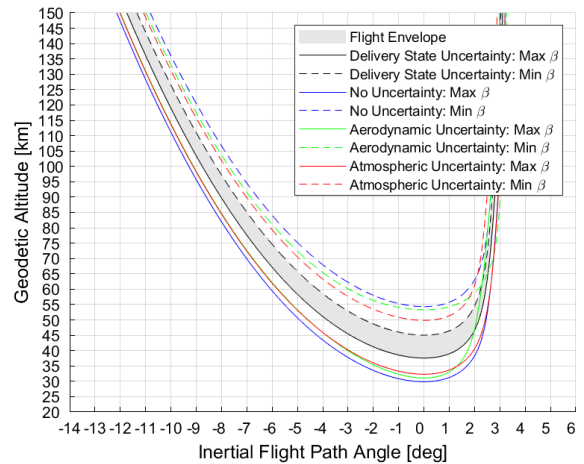


Figure 15: Flight envelope for ballistic vehicle design with 400 km circular target orbit at Mars.

As the SmallSat begins its ascent out of the atmosphere, the width of the flight envelope becomes greatly reduced. Compared to a lifting configuration at Mars, a ballistic SmallSat design requires greater controllability after passing through the initial periapsis inside of the atmosphere. However, the ballistic design is still effective as shown by the corridor width values for a range of target orbital altitudes including 400 km, seen in Figure 16. Like the lifting configuration analysis at Mars, the corridor width values corresponding to the combined uncertainty for a ballistic configuration are a result of the -3σ atmospheric dispersions and the $+3\sigma$ aerodynamic dispersions applied to the shallow and steep trajectories, respectively.

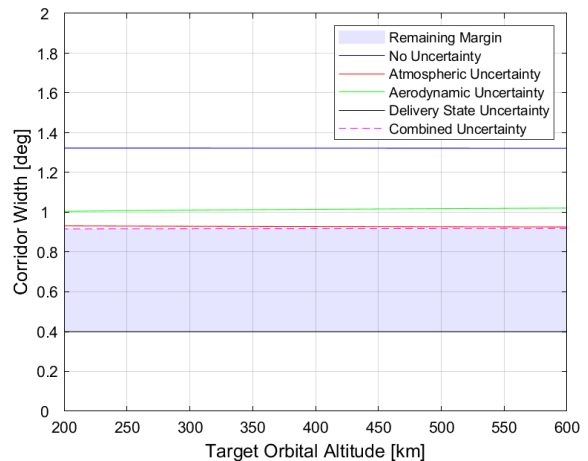


Figure 16: Corridor widths as a function of varying Mars target orbits for a ballistic configuration.

In contrast to the lifting configuration though, the target orbital altitude has almost no impact on the corridor width for the range of values shown. Regardless, the remaining margin that exists between delivery state and combined uncertainty corridor widths is emphasized in Figure 17 for a 400 km target orbit.

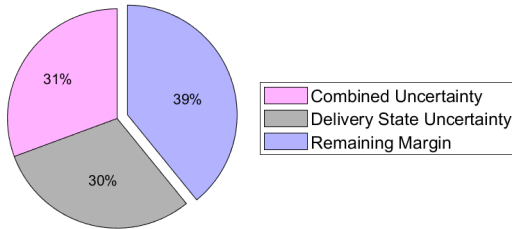


Figure 17: Remaining margin for a 400 km circular Mars target orbit for a ballistic configuration.

Out of the total initial corridor width before any uncertainties are factored in, the remaining margin accounts for 39% and the delivery state and combined uncertainties reduce the corridor width by an equal amount. Additionally, the geodetic altitude of the target orbit along with the entry velocity have minimal effects on the initial corridor width. Since the initial corridor width is less sensitive to entry velocity for a ballistic SmallSat design, this can provide more flexibility from a mission design perspective. More specifically, since SmallSats are typically secondary payloads that may rely on launch vehicles to initiate their interplanetary trajectory, unforeseen changes in launch time or remaining onboard propellant could lead to higher or lower entry velocities than intended. However, Figure 18 indicates that similar vehicle controllability is possible despite potential variation in entry velocity.

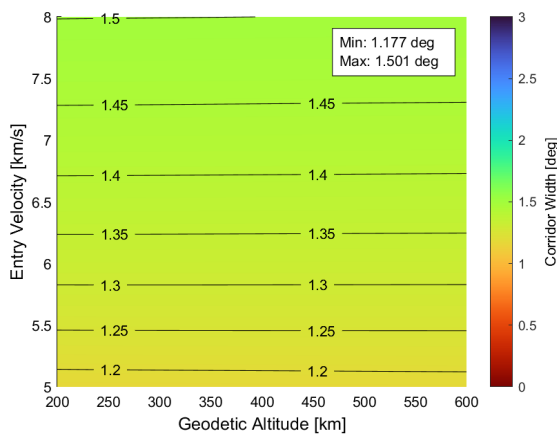


Figure 18: Corridor width as a function of entry velocity and target orbital altitude at Mars for a ballistic configuration.

The trends seen as a function of orbital altitude and entry velocity are like those seen for the ballistic configuration at Venus. However, the ballistic SmallSat design provides an adequate remaining margin at Mars in addition to the lifting configuration.

DISCUSSION

The results demonstrate that SmallSat aerocapture is feasible for lifting aeroshell designs at Mars and Venus as well as for ballistic vehicle designs at Mars. However, the ballistic design at Venus does not result in enough of a remaining margin of error for a SmallSat vehicle. In terms of flight control strategies, this indicates that drag modulation may not be an effective method for vehicle control at Venus. However, the results demonstrate that drag modulation is effective at Mars in addition to lift modulation for both planetary destinations. Additionally, variation in the target orbital altitude at both planetary destinations has minimal effects on the corridor width and remaining margin for both lifting and ballistic vehicle configurations. In contrast, varying the inertial entry velocity has a substantial impact on the corridor width, especially for lifting configurations at both planets. This indicates that the remaining margin of controllability for a lifting SmallSat design can be improved by altering the initial entry parameters. In contrast, a ballistic SmallSat design can provide consistent control capability despite potential unforeseen variations in the entry and target values. Overall, the flight envelope assessments and corridor width analysis demonstrate that aerocapture is a mission-enabling orbit insertion technique for SmallSats targeting both Mars and Venus orbits.

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