

Atmospheric Entry Performance of the Hercules Single-Stage Reusable Vehicle at Earth

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Braxton Brakefield
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ABSTRACT

The Hercules Single-Stage Reusable Vehicle (HSRV) is an innovative rocket that has been designed for use at Mars. However, with the renewed focus on returning to the moon first, it has been proposed to use the Hercules design for missions in the Earth-Moon system. To test its reentry performance, 9 types of entries were simulated, with multiple sensitivity cases for each. Overall, the current design of the vehicle is able to return to Earth safely for a wide range of different entry conditions, but some improvements could be made to optimize the vehicle for Earth operations.

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CHAPTER ONE

INTRODUCTION

The Hercules Single-Stage Reusable Vehicle (HSRV) is a novel rocket that has been designed for use at Mars [1]. It was envisioned as a reusable ferry to go from a space station in Mars orbit, down to the surface, refuel, and then return to orbit, carrying crew or cargo in either direction. This would support the development and sustenance of a Mars outpost or colony [2]. However, with the renewed focus on returning to the Moon first [3], it has been pondered if the HSRV can be used in the Earth-Moon system. For this, the Hercules concept needs to be evaluated for use at the Moon, in a similar capacity to its proposed role at Mars, but also needs its performance evaluated for Earth conditions. The vehicle needs to somehow go from the surface of the Earth into orbit, but it will also have to return to Earth's surface for various missions. Therefore, its reentry, descent, and landing performance at Earth needs to be evaluated. It needs to be determined if the current design is feasible for use as-is, or if the design can be adapted to better fit this new operation space.

Using the Hercules vehicle in the Earth-Moon system initially also allows for testing the capabilities of the design before it is used for a mission to Mars. Testing at or close to Earth allows for much more data to be gathered, as various ground tracking and sensing stations can monitor the tests, as well as allowing for much easier recovery of the vehicle during nominal or abort scenarios. Progressive testing can be utilized as well, such as ground testing, then suborbital flights, then flights to low earth orbit (LEO), and then continuing to further destinations. Each test can improve the confidence level of the vehicle, allowing the design to progress forwards.

This study builds off of previous studies developing the Hercules' design [1] and its abort capabilities at Mars [4]. This study also builds off of previous reentry studies conducted at the University of Tennessee [5] [6] [7]. Prior studies on the reentry of similar mid lift-over-drag vehicles have primarily been focused on such vehicles at Mars [8] [9], while some have been conducted for vehicles at Earth [10].

Vehicle Design

The general Hercules design is similar to most rocket designs, having a cylindrical main body longer than it is wide, with rocket engines protruding from its aft side, and the fore side being aerodynamically rounded. The current design of the vehicle can be seen in Figure 1. All tables and figures referenced in this paper are located in the appendix. The two main distinctions of the Hercules over other rocket designs are its single-stage and reusable properties. The single stage design was chosen as it is a viable approach for launching from Mars, and it keeps

the overall vehicle and mission design much simpler compared to a multistage rocket. The reusable property of the HSRV allow for a cheaper overall mission cost, and also allow for greater viability of a Mars mission as one HSRV can perform many functions.

The Hercules vehicle is further differentiated from other similar rocket designs by having a focus on commonality, crew safety, and operational flexibility. Commonality in the design leads to lowered initial and continuing costs for a space program utilizing multiple Hercules vehicles, even in different configurations. Prioritizing safety leads to slightly more complexity and cost for the design, but greatly adds to the chances of the design being selected for any manned missions. Operational flexibility allows for a single Hercules vehicle to perform many varied missions, and so can reduce the total mission cost. This study primarily helps to support the operational flexibility focus, by examining the Hercules Vehicle's performance at Earth. If the vehicle is able to perform as intended at Earth, that opens up a multitude of mission options for the vehicle. Mission at or around Earth can also serve to advance the readiness level of the vehicle and its necessary technologies for use at Mars, and will allow the vehicle to be used for Martian mission sooner.

Size

The nominal vehicle design is 6 meters in diameter, with a length-to-diameter ratio of 3.2, giving a length of 19.2 meters. The nominal weight is 20 tons of dry mass and 160 tons fully fueled. A full breakdown of the mass of the vehicle is shown in Table 1.

Propulsion

The propulsion system consists of two different engine groups. The first is the main Ascent/Descent System (ADS) engines, located on the bottom of the vehicle, and the second group is the Abort/Terminal Landing System (ATLS) engines arranged around the top of the vehicle. The ADS engines are fairly typical, being five identical liquid oxygen (LOX) and liquid methane (LCH₄) burning engines estimated to produce 246,700 N each, totaling 1,233,500 N. The ADS engines have an ISP of 360 s and are designed to be used for the main descent stage, as well as the ascent from Mars' surface [1]. The ATLS engine group consists of eight smaller engines located towards the top of the vehicle, canted downwards at 30° from vertical. These also use LOX and LCH₄, producing 59,700 N each, 477,600 N total, and have an ISP of 351.55 s. These engines were originally designed for the final descent phase of landing on Mars, as their high location should create less disturbance on any surface particles, producing less dangerous debris. This type of operation is shown in Figure 2. The ATLS engines were also designed as an abort system for the vehicle, being powerful enough to lift the upper portion of

the rocket away from the main body in the event of an emergency during ascent or entry.

The fuel tanks for the two engine groups are planned to be separate but connecting, with the tanks for the ADS engines towards the bottom of the rocket, while the ATLS tanks are located in the nose of the vehicle. Fuel would be able to be pumped in either direction and would also be connected to the reaction control system (RCS), as it too would use liquid oxygen and methane. The entire propulsion system would be pressurized using the fuels themselves, with the waste heat from part of the electrical energy generation system used to vaporize enough liquid propellant to provide sufficient pressure.

Aerodynamics

The Hercules vehicle is a mid lift-over-drag (L/D) vehicle, with a design L/D of 0.5. Hercules is not designed to glide during reentry, but to have a small ability to target a landing site and correct for perturbations in its descent.

A flap is mounted to the aft of the rocket, on the windward side as defined during reentry. This flap is similar to the one found on the Space Shuttle [11] and serves two purposes. The first is it is the primary aerodynamic surface for trim and down-range control during the vehicle's atmospheric entry. The small lift capability of the vehicle's shape allows for a modest amount of control, and the addition of this flap allows for much more precise control. The second function is to protect the engines from the heating during reentry, by shielding them from the flow.

Thermal Protection System (TPS)

The thin atmosphere of Mars allows for low reentry heating and thus a fully-reusable thermal protection system (TPS). The current design of the Hercules is planned to use an advanced carbon-carbon (ACC6) hot structure TPS. However, the thicker atmosphere and higher gravity on Earth create a much harsher entry environment and generally necessitates an ablative heat shield. Reusable heat shields have been used successfully previously at Earth, such as the ceramic tiles used by the Space Shuttle [12], and newer designs are currently being developed [13]. This study will help to characterize the heating on Hercules during reentry and will inform its TPS design for operation at Earth.

CHAPTER TWO

METHODS

Simulations

All of the various cases discussed later in this report were performed using the Program to Optimize Simulated Trajectories II (POST2). POST2 is a generalized point mass, discrete parameter targeting and optimization program developed by NASA [14]. It was selected for this study for its proven record, and its strength in simulating spacecraft trajectories through an atmosphere. The simulations were all performed using 3 degrees of freedom. Reference trajectories for the multiple types of entries tested were created assuming “nominal” values for the vehicle and trajectory parameters. A data file was created to contain all of the values that are changed between cases, and then the reference POST input files were made to use the data file values. This allowed for only needing to make multiple data files, and then using the same main input file for each type of entry.

Each case was setup to use 4th order Runge-Kutta integration, as it is fast and accurate. The projected gradient method was selected for the parameter targeting along with using central finite differences for partial derivatives. Generally, two or three variables are targeted during each run, those will be outlined specifically later on. In addition, each case has the final weight targeted for optimization, in order to minimize the fuel used during the entire trajectory.

The 1976 Standard Atmosphere model was used, as it is built into POST and is well proven. Wind models were turned off, as they were deemed unnecessary for this study. This should be examined later on in the design process of the vehicle. The J8 model of a planet’s gravity, using the values specific to Earth, was used, as it better accounts for the oblateness and other irregularities of Earth’s gravity over a simpler model.

Aerodynamics and Heating

The full aerodynamics of the vehicle were calculated at NASA Langley by a team not including this author. The coefficients of lift and drag, as well as the coefficient of the moment about the nose of the vehicle were all calculated for Mach numbers from 0.001 to 50 and various angles of attack and altitudes. Sample plots of some of the generated data are shown in Figure 3. The data for Mach 0.001 to Mach 3 was generated using Missile DATCOM [15] with a second order shock expansion, and was calculated for altitudes of 0, 30,000, 60,000, and 98,400 ft, as well as angles of attack from -5° to 60°. The data for Mach 3 to 8 was produced using Hypersonic Arbitrary Body Program (HABP), also at altitudes of 0, 30,000, 60,000, and 98,400 ft, and angles of attack from -5° to 60°. The aerodynamics of

the vehicle were only calculated using continuum assumptions, free molecular flow was neglected.

For Mach 10 to Mach 50, the aero data was generated with CBAERO [16], without the viscous approximation, as this was deemed negligible. The angle of attack range for these was only 40° to 60° , as the vehicle should only be at these high Mach numbers during the beginning of the reentry and should be positioned for optimal trim.

The aerodynamic effect of the aft flap was also calculated, using HABP [17] with viscous forces included. The data was generated for Mach 5 to 12, at altitudes of 0, 30,000, and 60,000 ft, angles of attack of -5° to 60° , and flap deflection angles of -20° to 50° . It was found that the vehicle trims optimally at a 55° angle of attack.

As well as the aerodynamics of the vehicle, the aerothermodynamics of the vehicle were calculated too. The convective and radiative heating rates were calculated in CBAERO for Mach numbers from 10 to 50, and angles of attack from 40° to 60° . It was determined that both heating rates became negligible at lower Mach numbers, and so full aerothermal data was not calculated below Mach 10. At each time step in the reentry simulation, the worst-case heating point as determined by the CBAERO calculations was used as the heating rate for the vehicle. This point varies throughout the trajectory, as the conditions experienced over the vehicle change. Both the convective and radiative heating rates are used, and are added together to get the total heating rate. The total heat load was also found by integrating the heat rates by the time step over the entire trajectory.

MATLAB Script

A MATLAB script was made in order to automate the process of running multiple different entry cases. The type of entry to run, and the values for the desired variables are input into the MATLAB script, and then the script creates every combination of those input variables. Each combination of variables is then written to a data file, and then each data file is fed into POST along with the correct input file that uses the data file. The output files, along with copies of the input and data files, are then saved into an organized folder structure. This script allowed for running multiple sets of cases at once and sped up the whole process over setting up and running each case by hand.

Types of Entries

In order to test the atmospheric entry capabilities of the HSRV at Earth, four main types of entries, each with a few variations, need to be considered. The four types are a descent from LEO, a direct entry from a hyperbolic return trajectory, an aerocapture from a hyperbolic return trajectory, and an aero-assisted orbit transfer.

Several entry subtypes were examined for each of the aforementioned types, differing in starting and ending locations. The starting locations include LEO, a Lunar-distance high Earth orbit (LDHEO), and hyperbolic return trajectories from a fast-transit or minimum-energy return from Mars. For this study, LEO is defined as a 400x400 km Earth orbit, LDHEO is defined as a 400x380,000 km Earth orbit, the fast-transit Mars return results in a surface relative entry velocity of 12.0 km/s, and the minimum-energy Mars return results in a surface relative entry velocity of 11.5 km/s. The ending locations include the Earth's surface (a generalized landing zone of Cape Canaveral was selected), LEO, or LDHEO. The specific combinations of a starting and ending location give multiple different entry subtypes.

Furthermore, for all of the entry types simulated, a trade space of cases was created consisting of different ballistic coefficients and maximum decelerations. Ballistic coefficient was used as the ballistic coefficient of an object correlates to the reentry conditions experienced by that object. Also, while the ballistic coefficient incorporates the size and mass of the object, vehicles with similar ballistic coefficients but different sizes or masses still perform comparably. The maximum deceleration experienced by the vehicle relates to the loads on the vehicle and any crew carried. This helps to estimate if the structure of the vehicle is sufficient, and if the deceleration can be withstood by any crew. The deceleration also relates to the heating of the vehicle, with higher deceleration signifying higher heating. Targeting specific decelerations allows for evaluating the vehicle's performance at those specific levels, as well as giving incremental increases in the severity of the entry.

Specifically, initial masses of 30, 35, 40, 45, and 50 metric tons, giving ballistic coefficients of 303, 354, 404, 455, and 505 kg/m² were used, along with max decelerations of 3, 3.5, 4, 4.5, and 5 Earth Gs. The combinations of these give a total of 25 cases per entry subtype. A "nominal case" for each entry subtype was defined as a ballistic coefficient of 404 kg/m², and a max deceleration of 4 Gs. The effects of the ballistic coefficient and max deceleration were compared separately, with the nominal values used for the other variable, as well as the effects of those two variables being changed in conjunction with each other.

Multiple variables were used to quantify the performance of each case, such as the velocity over altitude, the heat rate and deceleration over time, and the total ΔV , heat load, and max heat rate over the whole simulation. Any ΔV references and values given in this paper are of effective ΔV , calculated by POST based on the ISP of the engine and the length of the burns. The velocity, heat rate, and deceleration over time give a view into the conditions of the vehicle throughout its entire trajectory, and can show local maximums and minimums, and how those relate to the entry events. The total ΔV , total heat load, and max heat rate over the entire trajectory give quantifiable values for the performance of the vehicle over the entire entry, descent, and landing. These allow for comparing between cases quickly, and for ensuring design changes can adequately account for the max values experienced.

The effect of the initial thrust to weight ratio on the required ΔV was also examined. As each case has the same thrust due to using the same engine configuration and performance, the thrust to weight ratio is inversely related to the ballistic coefficient, both scaling with the mass of the vehicle. However, it was decided that showing the total ΔV required based on the initial thrust to weight ratio was the better comparison, as it is more common. All of these comparisons were done to see the effect of scaling the mass and size of the vehicle on its trajectory and performance, and also to see how well the vehicle can handle a more or less intense entry environment. As the design of the vehicle is still being iterated upon, this data allows for potential selection of better design parameters or can inform about the potential performance of a proposed design change.

CHAPTER THREE

RESULTS AND DISCUSSION

Descent from LEO Case

A descent from LEO would be a common situation, as there are many applications for having a vehicle in LEO towards the beginning or end of a mission. For the Hercules specifically, it is assumed in-orbit refueling will be used, and so tanker flights will need to return from their rendezvous in LEO. It is also possible that a return mission from Mars or the Moon would aerocapture into LEO before descending to the surface. For any of these cases, the HSRV first would perform a deorbit burn, in order to lower part of its trajectory into Earth's atmosphere. The vehicle would enter belly-down in order to produce as much drag as possible to dissipate as much energy as possible. Roll reversals, such as used for the Space Shuttle reentry, are used to further slow down, and also to coarsely target the landing site. Next, during the heading alignment phase, the vehicle more precisely targets its landing site through changing its bank angle. The bank angle was allowed to change instantaneously as needed, as rate-limiting the bank angle caused problems with the simulation. After reaching low subsonic speeds, the vehicle realigns for its retropropulsion burn by transitioning from belly-down to engines-down. Once the engines are opposing the velocity vector, the main descent burn phase starts with all of the main engines firing at full power. Finally, when a vertical velocity of 2.5 m/s is reached, the main engines are throttled to maintain that velocity until touchdown.

Using just the ATLS engines for the descent and landing would be preferred, as was originally proposed for Mars, but the full thrust available from those is not sufficient to counteract the weight of the Hercules vehicle under Earth's stronger gravity. Sample cases of using both the main engines as well as the ATLS engines were simulated to compare the performance of both sets of engines to the performance of just the main engines. While the increased thrust provided by using all of the engines gives a lower needed ΔV , it was not a significant improvement. It was also decided it would be better to reserve the ATLS engines for abort scenarios instead of using them for the small performance increase.

Another brief study was conducted on the transition from belly-down to engines-down. This has been a complicated regime to model for some time [18], with multiple proposed options. After considering a few options, the simplest method of assuming a constant pitch rate over a set time was decided upon. This was the easiest method to implement, and it was determined the results would not be significantly affected by using this simplification. Further study of this portion of the trajectory should be conducted in order to obtain a more accurate simulation though.

Only a single descent from LEO subtype was examined, as there is only one possible starting and ending location for this type of entry: starting in LEO and ending at the landing site. Plots of the performance of the nominal case for this trajectory subtype are displayed in Figure 4. Figure 5 shows how the angle of attack, bank angle, and entry angle change over time.

The POST simulations for this subtype were started with the vehicle in its 400x400 km orbit, with an instantaneous burn used for deorbit. The ΔV used for the burn is one of the targeted variables controlled by POST, with the targeted condition being the requested max G load during the reentry. A higher ΔV burn gives a steeper entry angle and thus a higher max G load. The other targeted variable for this subtype is the initially altitude for the start of the retropropulsion burn. Starting the burn too low leads to insufficient deceleration before reaching the ground, while too long of a burn can lead to altitude gains if sufficiently long and powerful, or simply using more propellant than is necessary to land safely.

The various phases of the reentry can be clearly seen, as well as when the peak deceleration and heating occur. This nominal case started in its 400x400 km LEO orbit with an initial mass of 40,000 kg. The deorbit burn required 200 m/s of ΔV , resulting in an entry mass of 37611 kg. The initial thrust to weight ratio at the start of the descent burn was 2.68, and the whole descent burn used 2270 kg of fuel representing 220 m/s of ΔV . The final landed mass was 35341 kg. The effect of changing the ballistic coefficient for this entry type was examined next and can be seen in Figure 6: Effect of ballistic coefficient on the trajectory of the descent from LEO subtype.

Varying the ballistic coefficient has a minimal effect on this entry type, with a slight rise in the heat rate with a higher ballistic coefficient being the only somewhat significant effect. Even the highest ballistic coefficient still has a relatively mild heating rate of a little under 400 W/cm². Next, the effect of changing the max deceleration was examined, as seen in Figure 7.

Again, there is minimal variation from the nominal case found from varying the max deceleration. The max deceleration value itself varies, as each case had a value that was specifically targeted. The heat rate does have a small increase as the max deceleration increases, similar to the effect of increasing the ballistic coefficient, but it is still mild. Finally, the effect of changing both the ballistic coefficient and the max deceleration simultaneously was examined, and the results are shown in Figure 8.

Even varying both parameters to either extreme had only a small effect on the overall conditions experienced by the Hercules vehicle during its reentry. A higher initial thrust to weight did decrease the ΔV modestly, but not significantly. One property measured exclusively for this entry type was the ΔV needed for the deorbit burn. The ΔV needed mostly correlated linearly with regards to the initial thrust to weight, as the burn in orbit did not have to fight against gravity. However, the needed ΔV increased with an increased max deceleration, but this is because of the greater ΔV needed to target a steeper entry angle in order to produce the higher deceleration. As discussed previously, the max heat rate increases with an

increase in either ballistic coefficient or max deceleration, but the total heat load has not been considered yet. The total heat load does increase with an increase in ballistic coefficient but decreases with an increase in max acceleration. This is due to the longer heating pulse for the lower deceleration, instead of the quicker pulse for the higher decelerations.

Direct Entry Cases

The second type of entry is a direct entry from a hyperbolic return trajectory. This type of entry would be used for a vehicle returning from a mission to Mars or similar. The vehicle's trajectory would be aligned on the return journey to directly encounter the Earth's atmosphere. An entry angle sufficient to not allow the vehicle to skip back out would also be targeted prior to reaching the atmosphere. Upon entering the atmosphere, the bank angle is modulated to sustain a targeted constant G load before performing the roll reversals as in the descent from LEO entry type. After the roll reversals, the aforementioned heading alignment phase, transition to engines-down, main descent burn, and final constant velocity descent are all performed as well.

Three different direct entry subtypes were simulated, and all three ended with the HSRV landing successfully on Earth's surface. Three different starting locations were used though, a fast-transfer return from Mars, a minimum-energy return from Mars, and a return from the Moon. The return from the Moon subtype does not actually start in a hyperbolic orbit, and potentially could be better modeled similar to the descent from LEO, but with a higher starting orbit. However, as the atmospheric entry itself is the focus for this study, the return from the Moon subtype is better grouped with the Mars return trajectories, as its entry velocity is almost as fast as the Mars return entries. This lunar return subtype has an entry velocity of 11.1 km/s compared to the 12.0 or 11.5 km/s for the Mars return subtypes. Furthermore, the deorbit portion of the lunar return trajectory is tiny, as changing the periapsis from 400km to inside of Earth's atmosphere while at the 380,000km apoapsis requires very minimal ΔV .

Fast-Transfer Mars Return

The fast-transfer Mars return case is examined first. Plots of the performance of the nominal case for this trajectory subtype are displayed in Figure 9, and its angle of attack, bank angle, and flight path angle over time are all shown in Figure 10.

The slightly modified trajectory for this type of entry can be seen, as well as how the peak deceleration and heating line up with the new phases. Most notable is the plateau in the sensed acceleration, holding the max deceleration relatively constant for some time. This nominal case started in its hyperbolic return trajectory with an initial mass of 40,000 kg. An entry angle of -6.51° was targeted. The initial

thrust to weight ratio at the start of the descent burn was 2.52, and the whole descent burn used 2530 kg of fuel representing 231 m/s of ΔV . The final landed mass was 37469 kg.

By targeting a constant G load, the effect of that specific deceleration on the vehicle can be more easily seen. However, sustaining the targeted deceleration leads to a long time spent at that deceleration. For the current entry subtype being examined, and for most of the cases examined after this one, the max G load is sustained for around 200 seconds. This is a long time for human occupants to be under this level of deceleration. Previous studies have found this G load for this duration is tolerable though, especially if no performance is needed from the passengers [19] [20] [21].

The effect of changing the ballistic coefficient for this entry type was examined next and can be seen in Figure 11. As with the descent from LEO entry type, varying the ballistic coefficient has a minimal effect, with the same slight rise in the heat rate as the ballistic coefficient increases. However, it should be noted that the heating rate is much higher overall for this type of entry, around 1200-1400 W/cm² compared to the 300-400 W/cm² for the descent from LEO case.

Two previous papers have studied similar entries to this paper but found fairly disparate results. Both were studying capsules instead of rockets, but both were examining vehicles entering Earth's atmosphere from a Mars return trajectory, and included results for similar velocities, ballistic coefficients, and L/D ratios. The first paper found a capsule with an L/D of 0.5 reentering with a velocity of 12 km/s to have a peak stagnation point heating rate of 650 W/cm² for a ballistic coefficient of 300 kg/m², and 1000 W/cm² for a ballistic coefficient of 500 kg/m² [22]. These values are quite a bit lower than the values found from the present study. The Hercules heating rates are the worst rate across the entire vehicle, and so should be worst-case and somewhat higher than the general heating across the vehicle. The second paper found that a 310 kg/m² capsule with an L/D of 0.5 and coming in at 12.5 km/s experiences a max heating rate of 627.5 W/cm² [23]. This is significantly lower than the values found in this current paper.

Next, the effect of changing the max deceleration was examined, as seen in Figure 12. Once again, there is minimal variation from the nominal case when varying the max deceleration. For these cases though, the max deceleration is sustained for a shorter amount of time with a higher max deceleration. This is due to the trigger for moving to the roll reversal phase being the vehicle reaching a velocity of 5,000 m/s. A higher deceleration reaches that target velocity faster, and so does not need to be sustained as long. The heat rate does have a small increase as the max deceleration increases, similar to the effect of increasing the ballistic coefficient, but it is still mild. Another thing to note is the lack of a case for the 3 G max deceleration. This is too low of a condition for the vehicle to successfully target with such a high entry speed. An entry angle shallow enough to meet the 3 G deceleration limit results in the vehicle skipping out of the atmosphere creating an aerocapture trajectory. For this reason, this specific case was excluded from this study. Finally, the effect of changing both the ballistic

coefficient and the max deceleration simultaneously was examined, and the results are shown in Figure 13.

Similar results to the descent from LEO case are seen here as well. A higher initial thrust here does see a slightly better decrease in the needed ΔV . It might be curious that the max deceleration does not have an effect on the needed ΔV , but there is a good explanation for that. Because the Earth's atmosphere is relatively thick, all of the cases are slowed to roughly the same velocity by roughly the same altitude. Specifically, when the heading alignment phase begins at 3,000 m/s, all of the cases are close to 42,000 m in altitude. Due to this, when the mass of the vehicle is held constant, the same amount of ΔV is needed for the descent burn that starts at the same velocity and altitude independent of the prior conditions. A higher massed vehicle will need extra ΔV as it has to oppose gravity during the powered descent. This is what causes the change based on initial thrust to weight, as the thrust is the same for every case tested, and the weight is what is changing this variable. As seen previously, the max heat rate increases with an increase in either ballistic coefficient or max deceleration, and the total heat load increases with an increase in ballistic coefficient but decreases with an increase in max acceleration.

Minimum-Energy Mars Return

The second subtype of the direct entry cases examined was the minimum-energy Mars return trajectory. This is very similar to the fast-transit subtype, but with a slightly lower arrival velocity. The plots of the trajectory for the nominal case of this subtype are shown in figures Figure 14 and Figure 15.

As noted previously, this trajectory is very similar to the fast-transit Mars return seen above. This nominal case started in its hyperbolic return trajectory with an initial mass of 40,000 kg. An entry angle of -6.37° was targeted. The initial thrust to weight ratio at the start of the descent burn was 2.52, and the whole descent burn used 2535 kg of fuel representing 232 m/s of ΔV . The final landed mass was 37465 kg. The effect of changing the ballistic coefficient for this entry type was examined next and can be seen in Figure 16.

As with the fast-transit Mars return, varying the ballistic coefficient has a minimal effect, with the same slight rise in the heat rate as the ballistic coefficient increases. It can be seen however that the heating rate is slightly lower when compared to the fast-transit return subtype, peaking around 900-1300 W/cm² versus 1200-1400 W/cm². These results are still quite different from the two papers studying similar entry conditions. The study by Lyne, Tauber, and Braun found that a capsule entering at 11.5 km/s with a L/D of 0.5 experienced a peak stagnation point heating rate of 580 W/cm² for a ballistic coefficient of 300 kg/m² and 900 W/cm² for a ballistic coefficient of 500 kg/m². The study by Braun, Powell, and Lyne found an even lower heating rate of 333.9 W/cm² for a ballistic coefficient of 310 kg/m².

Next, the effect of changing the max deceleration was examined, as seen in Figure 17. There is minimal change in the trajectory when the max G load is changed, as previously seen. The heating rate is affected, but less so across the different targeted G loads than across the different ballistic coefficients. As expected, the max G load is different depending on the targeted load, and as noted previously, the length of time spent at the max load changes based on the load. Finally, for this case, the effect of varying both the max G load and the ballistic coefficient is examined, with the results shown in Figure 18.

The results seen here are very similar to the results found for the fast-transit Mars return subtype. The same phenomenon of the max deceleration having no effect on the ΔV required for landing is present here as well. Again this is due to the vehicle being slowed to the same speed at the same altitude, thus requiring the same ΔV for the final descent phase. If compared directly to the fast-transit return data, the necessary ΔV is almost identical between the two cases for the same initial thrust to weight ratio. This shows that even for an increased entry velocity, the vehicle ends up being slowed to its terminal velocity at a sufficiently high altitude to require the same ΔV for landing. The difference between the two entry velocities does result in a greater amount of energy requiring dissipation for the fast-transit cases, and so their max heating rates and total heat loads are both higher for the same ballistic coefficient and max G load.

Lunar Return

The third and final direct entry subtype simulated was a return from lunar orbit. This is to model a Hercules returning from a lunar mission but coming from a temporary lunar orbit instead of directly from the lunar surface. As this type of trajectory is not actually a hyperbolic return such as the two Mars return subtypes, this case could potentially be better modeled similar to the descent from LEO subtype, but with a much higher initial orbit. However, as the altitude of the starting orbit is significantly higher than a LEO orbit, the entry speed of the vehicle is much higher than a descent from LEO, and so it was determined it would be better modeled similar to a hyperbolic return entry. To this end, this entry subtype has been modeled and grouped with the two Mars return subtypes. The nominal case trajectory plots for this subtype are shown in Figure 19 and Figure 20.

This case results in similar performance to the two Mars return subtypes, but with a lower entry velocity. The flight and final conditions for this subtype are comparable with those two entries, but with a slightly milder environment overall. This nominal case started with an initial mass of 40,000 kg, targeting an entry angle of -6.03° . The initial thrust to weight ratio was 2.52, with the descent and landing burns using 2541 kg of propellant, resulting in 232 m/s of ΔV . The final landed mass was 37458 kg. The effect of changing the ballistic coefficient for this entry type was examined next and can be seen in Figure 21.

This entry subtype also shows a minimal effect from varying the ballistic coefficient. There is a slight rise in the heat rate as the ballistic coefficient

increases, as was seen with the previous two subtypes. The heating rate is lower still from the minimum-energy Mars return though, as the entry velocity is lower. However, it is comparable, at 700-1000 W/cm² versus 900-1300 W/cm² for the minimum energy return. This lunar return heating rate is approximately double the descent from LEO heating rate but is about halfway between the descent from LEO heating rates and the fast-transit Mars return heating rates. Next, the effect of changing the max deceleration was examined, as seen in Figure 22.

Changing the max deceleration has a minimal change on the conditions experienced by the vehicle, as also seen in the previous cases. The heating rate is affected, but less so across the different targeted G loads than across the different ballistic coefficients. As expected, the max G load is different depending on the targeted load, and as noted previously, the length of time spent at the max load changes based on the targeted load. Finally, for this case, the effect of varying both the max G load and the ballistic coefficient is examined, with the results shown in Figure 23.

The results of comparing both parameters together are almost identical to the two previous entry subtypes, but with a lower max heat rate and lower total heat load when compared with the same ballistic coefficient and G load. This is due to the lower energy location the vehicle starts in, thus requiring less energy dissipation. The ΔV required exhibits the same behavior of no effect due to a change in the max deceleration, and when all three direct entry subtypes are compared together, it can be seen that the entry velocity of these three has no discernable effect either. All three cases eventually reach the same velocity at the same altitude, leading to the same ΔV needed for the descent and landing propulsive phases.

Aerocapture and Aerotransfer Cases

The third and fourth entry types have the same phases once they enter Earth's atmosphere, but they originate from different locations. The aerocapture entry type starts in a hyperbolic return trajectory, as would be the case for a return from Mars. The aero-assisted orbit transfer, or aerotransfer, type starts in a high Earth orbit, such as a Lunar distance orbit after a mission to the Moon. Both trajectory types target the correct entry angle before entry in order to achieve a desired max G load. The first phase of the entry involves orienting the vehicle belly-down so that the lift vector mainly points upwards to reduce the G load and heating somewhat. Shortly after the flight path angle reaches 0°, the vehicle banks 180° to be belly-up, thus pointing its lift vector down. This allows it to stay in the atmosphere somewhat longer and allows for a shallower initial entry, reducing the heating and deceleration. The HSRV maintains its lift down orientation until it exits

the atmosphere, and then coasts to its apoapsis. It then performs a short burn to raise its periapsis above the atmosphere and to the desired altitude.

Four different aerocapture subtypes were examined, with two starting and two ending locations. The starting locations were either a fast-transit or minimum-energy Mars return hyperbolic trajectory, which give a relative entry velocity of 12.0 or 11.5 km/s respectively. The ending locations were either LEO or LDHEO. The different combinations of these give the four subtypes.

A single aerotransfer subtype was simulated, starting in LDHEO and ending in LEO, representing a mission to the Moon transferring to LEO. As with the direct entry from lunar orbit cases, this entry subtype is simulated the same as the hyperbolic trajectory aerocaptures, with the simulation beginning shortly before the atmospheric entry interface, instead of simulating the deorbit and coast phases.

Fast-Transfer Mars Return to LEO

The first aerocapture subtype examined is the fast-transfer Mars return aerocapturing into LEO. The plots showing the nominal case trajectory are shown in Figure 24 and Figure 25.

This case is clearly different from the previous cases examined, as it should be as an aerocapture is very different from a descent and landing. The altitude versus velocity plot shows the vehicle descending into the upper atmosphere before climbing back out and shows how it is decelerated while in the atmosphere. It also shows how the main events occur closer to the start of the entry, with a fairly long coast after flipping. This nominal case targeted an entry angle of -7.08° and had a final periapsis of 86.1 km, requiring 91.4 m/s of ΔV after the aerocapture to circularize. Next, the effect of the ballistic coefficient on the trajectory is shown in Figure 26.

The ballistic coefficient has a minimal effect on the trajectory for this subtype. The actual trajectory itself varies by a few kilometers depending on the value of the ballistic coefficient, but this is minor compared to the overall altitude. The heat rate also varies a little, but as with the previous subtypes this variation is also minor. The heat rate variation is even less with this subtype though, with a low of 350 and a high of 450 W/cm². This range is also much lower than the direct entry subtypes, and comparable to the descent from LEO subtype. This showcases why aerocaptures can be a good option, as the heating can be reduced from 1200-1400 to 350-450 W/cm² for the same vehicle with the same entry speed. This still leaves the vehicle in LEO instead of on the Earth's surface though, and doesn't take into account the effects of reusing a heatshield or dealing with the heat absorbed from the aerocapture heating. The next parameter examined was the max G load of the vehicle, and these results can be seen in Figure 27.

Changing the max deceleration experienced by the vehicle also has a minimal effect on the trajectory and conditions. The heat rate is affected around the same amount as changing the ballistic coefficient, and still with the same small variation. As the max G load is not targeted to stay constant, and instead the value

for the max is the only thing targeted, the G load over time for the different cases is mostly the same, except for the peak value. Finally, the effect of both the ballistic coefficient and max deceleration changed simultaneously is shown in Figure 28.

This data is fairly typical, being similar to the previously seen cases. For the aerocapture subtypes, there is no ΔV required for the entry, and so any effects on it do not need to be examined. As the ballistic coefficient increases, the max heat rate as well as the heat load both increase. As the max deceleration increases, the max heat rate increases, but the total heat load decreases, which is consistent with the prior subtypes. However, while the variation in the max heat rate is consistent with the variation of the direct entry subtypes, the variation for the heat load is much smaller as the max deceleration is changed. It is more similar to the descent from LEO subtype. This is most likely as there is no constant G load phase, as is the same with the descent from LEO. This phase is fairly variable in length between the different max decelerations, and so leads to a larger spread in the total heat load.

Fast-Transfer Mars Return to LDHEO

The second aerocapture subtype simulated was a fast-transfer Mars return aerocapturing into a LDHEO orbit. Because of the great altitude of the final orbit, higher max G loads cannot be targeted, as they will remove too much energy and not allow the vehicle to end up in the desired final orbit. This effect has been previously studied [24]. Due to this, please note the change in case parameters examined for this subtype. This subtype was still able to attain its desired final conditions for a ballistic coefficient of 404 kg/m^2 and a max G load of 4 Earth G's, and so these parameters are still defined as the nominal case for this aerocapture subtype. The trajectory plots for this nominal case can be seen in Figure 29.

This trajectory is quite similar to the previous aerocapture subtype, using the same entry and guidance structure. The final velocity targeted by this subtype is much higher than the aerocapture to LEO, as the final orbit is much higher than LEO, and so requires a faster velocity at periapsis. This aerocapture is also much shorter in the time domain, primarily as less energy needs to be dissipated for the reasons just explained. The G load and heating rates are also comparable to the aerocapture into LEO, but peak and then decrease over the shorter timeframe for this subtype. This trajectory targeted an entry angle of -6.74° . It resulted in a periapsis altitude of 63.18 km, which requires 4.7 m/s of ΔV after the aerocapture to obtain the final $400 \times 380,000 \text{ km}$ LDHEO. Next, the effect of the ballistic coefficient on the trajectory was examined and is shown in Figure 31. Varying the ballistic coefficient over the usual range did not cause the vehicle to miss the final trajectory, so these values were used as normal.

As previously found, the ballistic coefficient has only a small effect on the trajectory. The biggest effect is on the heating rate, but this is still minimal, varying from $350\text{-}450 \text{ W/cm}^2$, almost identical to the fast-transit return aerocapture to LEO. The effect of the max deceleration is shown next in Figure 32. For this, the

range of max G loads was changed from 3, 3.5, 4, 4.5, and 5 to 2.5, 3, 3.5, and 4 Earth G's. This is due to any higher G load resulting in the vehicle in a lower orbit than the desired LDHEO, and any lower G load not dissipating enough to capture into the desired orbit.

The targeted max G load also has minimal influence on the vehicle's trajectory. The max heating rate does somewhat decrease with decreasing peak G load, as seen previously. The heating rate also is higher for longer with a lower max deceleration, primarily as the lower deceleration requires the vehicle to stay in the atmosphere longer in order to attain the same total ΔV from the atmosphere. Next, how changing both the ballistic coefficient and max G load simultaneously is shown in Figure 33.

The results shown here are expected given the results of the previous cases. As the ballistic coefficient increases, both the max heat rate and the heat load increase as well. The max heat rate also increases with increasing G load, and the heat load increases with decreasing G load also as previously seen. As with the previous aerocapture subtype, the spread of the heat load when varying the max G load is smaller than the spread of the direct entry subtypes, most likely due to less spread in time in the atmosphere between cases. The total heat load is lower when compared to the previous aerocapture subtype. This is due to the shorter duration in the atmosphere and the lower deceleration needed to end up in the higher energy LDHEO over LEO. The max heating rate is somewhat lower as well, but it is relatively close to the values found for the previous aerocapture subtype.

Minimum-Energy Mars Return to LEO

The third aerocapture subtype examined was a minimum-energy Mars return aerocapturing into LEO. This subtype is very similar to the first aerocapture subtype examined, except for a slightly slower entry velocity due to the difference in the return from Mars trajectory. The nominal case trajectory is shown in Figure 34 and Figure 35. This subtype was able to use all of the standard nominal case parameters, as it is much easier to target LEO over LDHEO.

This subtype is very similar to either of the two previous aerocapture subtypes examined, being most similar to the first due to having the same destination, LEO. This nominal case targeted an entry angle of -6.93° , ended with a periapsis altitude of 86.2 km, and needs a ΔV of 91.4 m/s after the aerocapture to circularize into LEO. This periapsis and ΔV required are the same for this subtype as for the fast-return Mars trajectory aerocapture, as Earth's atmosphere is sufficient to bring the vehicle into the same orbit after exiting the atmosphere for either entry speed. Since they end up in the same orbit, still with the periapsis inside the atmosphere, it requires the same amount of ΔV to lift the periapsis to LEO. The effect of ballistic coefficient on the vehicle's trajectory is shown next in Figure 36.

The ballistic coefficient still has a minimal impact on the trajectory of this aerocapture subtype. The heating rate does increase slightly with an increase in ballistic coefficient, from around 320-420 W/cm². This is comparable to the other aerocapture cases, although slightly lower due to the lower entry speed of the vehicle for this subtype. Next the effect of the max G load on the trajectory is examined in Figure 37.

The max G load also still has a minimal effect on the trajectory. There is a small increase in the heating rate with an increase in the G load, and this is comparable to the increase in heating rate from an increase in ballistic coefficient for this subtype. Finally, the effect of both ballistic coefficient and max G load on the trajectory of the vehicle is shown in Figure 38.

The trends seen here are very similar to the trends seen previously. Both the max heat rate and total heat load increase with an increase in ballistic coefficient, while the max heat rate increases, and the total heat load decreases, with an increase in max G load. The max heating rate is somewhat lower than the max heating rate of the Mars fast-transit aerocapture to LEO, but still a little higher than the same return but to LDHEO. The total heat load is also noticeably lower than the first aerocapture subtype but is still not as low as the second. This can all be attributed to the amount of energy needed to be dissipated to get to the destination orbit from the original trajectory.

Minimum-Energy Mars Return to LDHEO

The next aerocapture subtype simulated was a minimum-energy Mars return aerocapturing into LDHEO. This case is most similar to the second aerocapture subtype, the fast-transit Mars return to LDHEO. As with that entry subtype, there is a narrower entry corridor than compared to capturing into LEO, and so the normal parameters for max G load could not be used. For this case, only a max G load of 2 Earth G's was able to attain the desired orbit. Any G load higher or lower did not result in a successful capture into the desired orbit. The normal sensitivities for ballistic coefficient all allowed for a successful aerocapture, and so they were still used. The plots for the trajectory of the nominal case, 404 kg/m² and 2 Earth G's, are shown in Figure 39 and Figure 40.

This case is mostly the same as previously seen, being most similar to the second aerocapture subtype, the fast-transit Mars return to LDHEO. For this trajectory, an entry angle of -6.088° was targeted. This resulted in a periapsis altitude of 71.77 km and a needed ΔV of 4.5 m/s after the aerocapture in order to raise the periapsis out of the atmosphere and to 400 km. Next the effect of the ballistic coefficient was examined and is shown in Figure 41.

The ballistic coefficient has the expected effect on the conditions experienced by the vehicle, with the heating rate slightly increasing as the ballistic coefficient increases. It should be noted that the range for this subtype's cases are the lowest heating rates of any of the cases simulated, from about 240 to 310 W/cm². This is primarily due to the lower peak G load targeted, leading to the

vehicle staying higher in the atmosphere and experiencing lower peak heating. Because this subtype is targeting LDHEO and starts in the slower Mars return trajectory, it also requires less energy dissipation than the other cases, which contributes to the lower heating. As only one max G load could successfully be targeted, a study on the effects of changing the max G load on the trajectory was not done, and so the next results presented are the effect of the ballistic coefficient on the max heating rate and total heat load in order to compare this subtype to the other aerocapture subtypes. The plots for this are shown in Figure 42.

As stated previously, the max heat rate is lower than any of the other subtypes examined. It is relatively close to the other aerocapture subtypes, but still somewhat lower. The total heat load is significantly lower than the heat load for the aerocaptures targeting LEO, but is comparable, while still slightly lower than, the Mars fast-transit return to LDHEO.

Aerotransfer from LDHEO to LEO

The singular aero-assisted transfer examined was a transfer from LDHEO to LEO, representing a vehicle returning from a lunar mission. As with the direct entry from LDHEO being grouped with the direct entries from Mars return, this aerotransfer is grouped with the aerocaptures from Mars return because it results in similar conditions. This aerocapture was able to complete successfully for the full range of normal max G and ballistic coefficient sensitivities, and so those were all used for this subtype. The nominal case trajectory is shown in Figure 43 and Figure 44.

These results are very comparable to the previous aerocapture cases, especially the two other subtypes targeting LEO as their final destination. This case targeted an entry angle of -6.43° , had a final periapsis of 81.8 km, needing a ΔV of 92.7 m/s to circularize into LEO. This is very similar to, but not identical to, the final conditions of the other aerocaptures to LEO. Next the effect of the ballistic coefficient on the trajectory is shown in Figure 45.

There is minimal effect on the trajectory when the ballistic coefficient is changed, as has been seen previously. There is a slight increase in the heating rate with an increase in the ballistic coefficient, from around 280 to 350 W/cm². Next, the effect of the max G load on the trajectory is examined and shown in Figure 46.

There is once again a minor effect on the trajectory when the max deceleration is changed. Increasing the max deceleration does slightly increase the max heating rate, but this increase is even less than the already small increase when the ballistic coefficient is increased. Finally, the effect of both ballistic coefficient and max G load on the trajectory is plotted in Figure 47.

The max heat rate and total heat load both increase with increasing ballistic coefficient, as seen previously. Additionally, the max heat rate increases, and the total heat load decreases with increasing max G load, also seen previously. However, the actual values for these two measures are somewhat lower than the

similar aerocapture subtypes. The total heat load has fallen even further from the Mars minimum-energy return to LEO, and is about as far from that subtype as that subtype is from the fast-transit Mars return to LEO. The minimum-energy Mars return to LDHEO is still lower on both accounts than this subtype is, but that is primarily due to the lower max G load needed to successfully target the final orbit for that subtype.

CHAPTER FOUR

CONCLUSION

Through examining the performance of the HSRV through different entry regimes at Earth, it has been shown that the current vehicle design can successfully complete all of these entry types over a range of conditions if an ablative heat shield is used. PICA has been shown to work with heat rates at or below 1200 W/cm^2 , and all of the entry subtypes have cases that fall under that maximum [25]. However, one of the main goals of the Hercules vehicle is reusability, and requiring an ablative, non-reusable TPS greatly hinders this goal.

It is possible that the HSRV can use a reusable heat shield for certain entries at Earth, as the descent from LEO subtype, along with the five aerocapture subtypes, all have max heat rates of $300\text{-}400 \text{ W/cm}^2$. This is still higher than current reusable TPSs can handle, but this is a very high estimate for the heating on the vehicle. As explained previously, the heating calculations use the worst point on the vehicle at the current point in time as the heating for the entire vehicle at that point in time. Once more confidence in the vehicle design and simulation methods is reached, better heating calculations can be conducted that give a much more accurate result for the heating.

One option to make use of a reusable TPS more likely would be to use propellant to slow the vehicle before its entry. This would require a fairly large ΔV to slow down sufficiently, and so is probably not the best idea for most missions, but could allow for more mission profiles. This option could also enable a heavier version of the Hercules to return to Earth safely with an ablative heat shield even from the most energetic trajectories studied.

Another possibility to incorporate reusability would be to use an ablative TPS on the portions of the vehicle that experience the most severe heating, and then use a reusable TPS on the sections that are mild enough to allow for that. This option requires much further study in order to effectively find the heating environment across the entire vehicle. Further study is also needed in order to determine the exact entry corridor allowed for each of the entry subtypes studied in this paper.

Outside of the TPS for the vehicle, another possible modification would be to decrease the number of engines. The ATLS engines are not used at all in these entry cases, as their thrust alone is insufficient at Earth. They could potentially be used as abort engines in case of an emergency during the reentry, as they were partially designed for at Mars, but they have not been evaluated for this option at Earth. Additionally, it is thought that many of the missions involving this vehicle will not carry crew, and so those missions have no need for an abort system unless the cargo it is carrying is deemed to be extremely valuable.

The number of main engines could potentially be lowered as well, as the 5 engines currently in place provide much more thrust than is needed for the descent. These were originally sized for launching the whole vehicle from Mars'

surface to orbit, and much less thrust is needed for a descent and landing at Earth. Having less thrust available will increase the needed ΔV somewhat, but it is fairly low currently, so a small increase should be acceptable. Another reason to keep all of the main engines is to give redundancy in case of failure of one or more engines. Only 2 or 3 main engines are needed to counteract the weight of the vehicle at Earth, depending on the exact mass of the vehicle, and so theoretically 2 engines could malfunction, and a successful landing could still be attained. Finally, a major goal of conducting missions with the Hercules vehicle at Earth is to prove the concept for eventual use on Mars, and so leaving it designed for Mars as much as possible, even if that is not necessarily optimal at Earth, helps with the final goal of the vehicle.

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APPENDIX

Table 1: Mass breakdown for the Mars variant of the HSRV

Subsystem	Ascent Mass (kg)	Entry Mass (kg)
Predicted Mass	18,898	18,898
Structures	5,701	5,701
Thermal Protection	2,080	2,080
Landing Legs & Actuation	1,036	1,036
Ascent Propellant Tank	1,936	1,936
Descent Propellant Tanks	318	318
ADS Propellant Feed	474	474
ADS Engines	1,383	1,383
ATLS Propellant Tanks	725	725
ATLS Propellant Feed	543	543
ATLS Engines	511	511
RCS Thrusters	114	114
Power & Avionics	850	850
Growth/Margin	3,216	3,216
Propellant Mass	138,171	12,959
Ascent Tank	121,714	0
Descent Tanks	8,286	6,949
ATLS Tanks	8,171	6,010
Payload Mass	5,750	5,500
Cargo	5,000	5,000
Crew & Suits	500	500
Samples	250	0
TOTALS	162,819	37,358



Figure 1: Current design of the Hercules Single-Stage Reusable Vehicle.



Figure 2: ATLS engines being used during the final descent stage of a Mars landing.

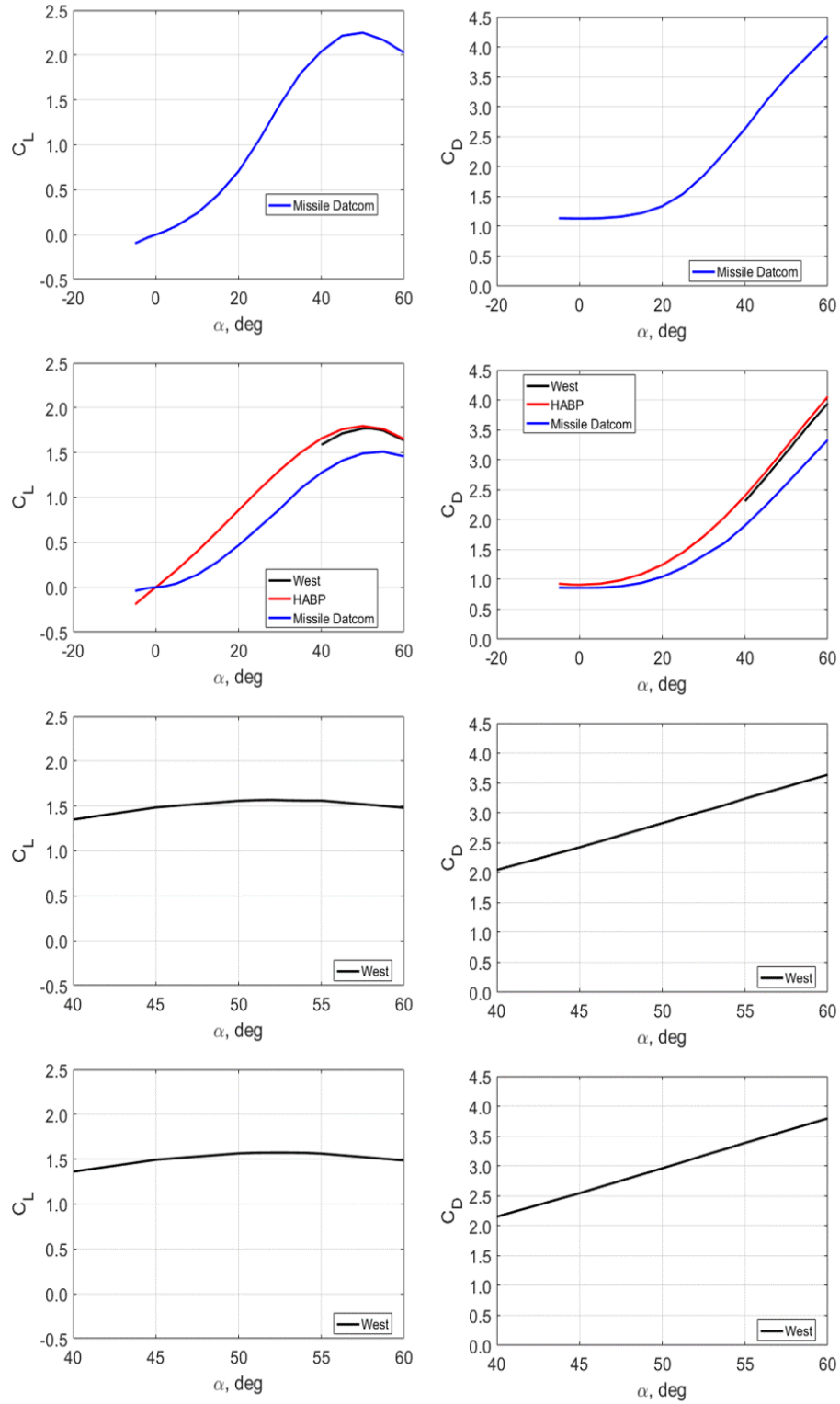


Figure 3: Selected plots of the coefficients of lift and drag calculated for the Hercules Vehicle at Mach number of 1, 3, 10, and 50.

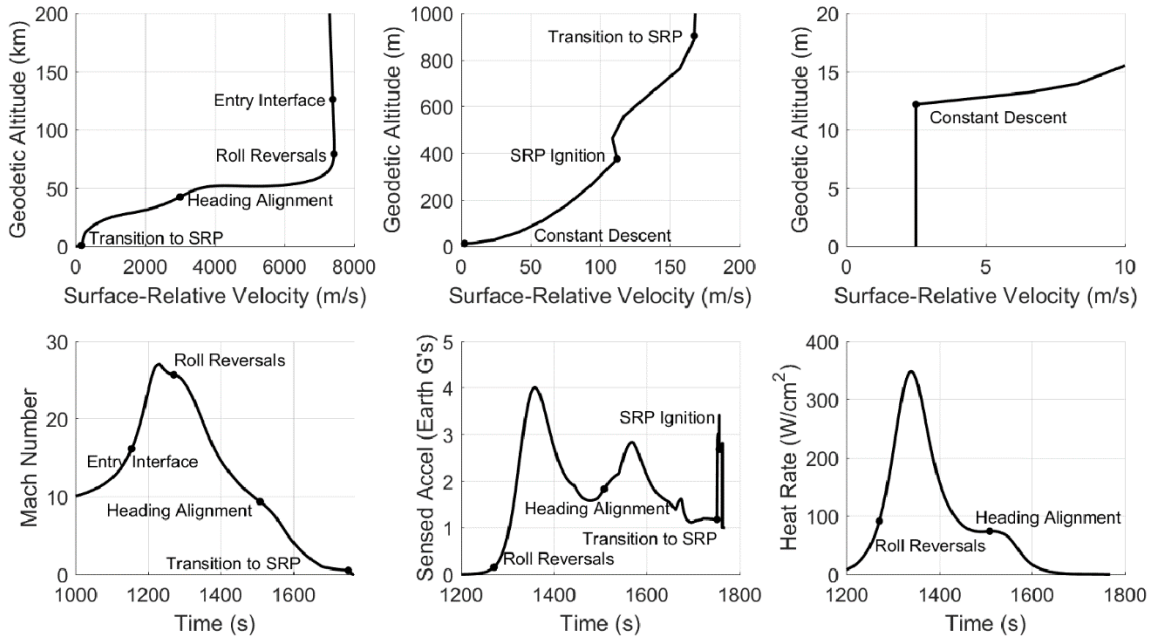


Figure 4: Plots of the trajectory of the nominal descent from LEO subtype.

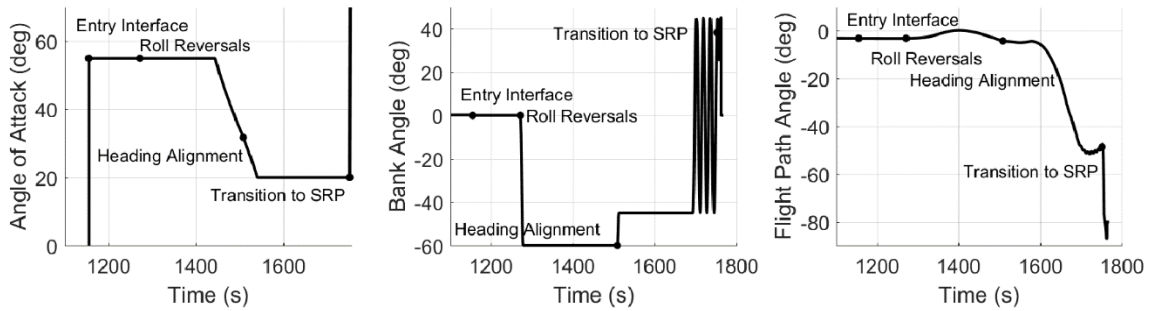


Figure 5: Angle of attack, bank angle, and flight path angle for the nominal descent from LEO subtype.

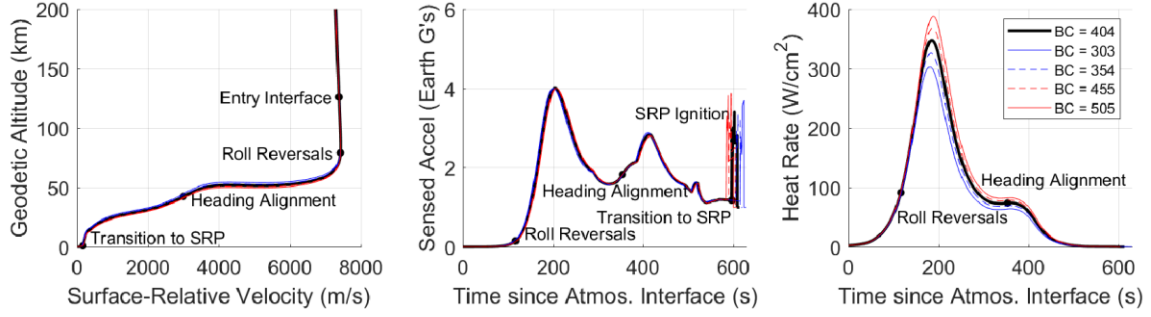


Figure 6: Effect of ballistic coefficient on the trajectory of the descent from LEO subtype.

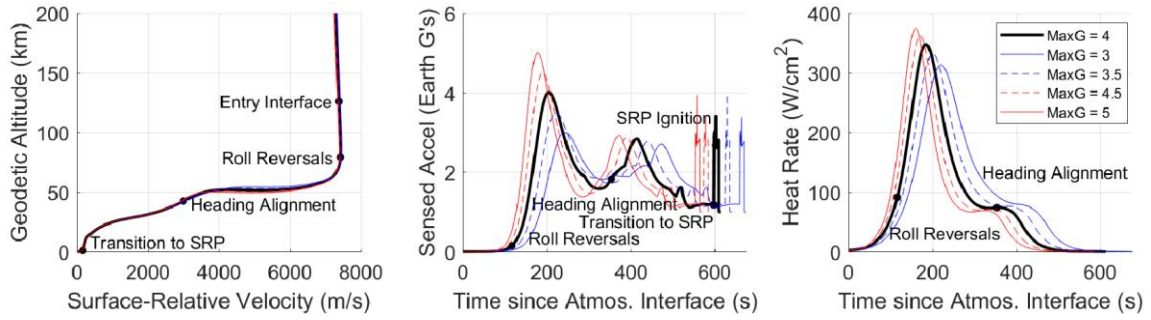


Figure 7: Effect of max G load on the trajectory of the descent from LEO subtype.

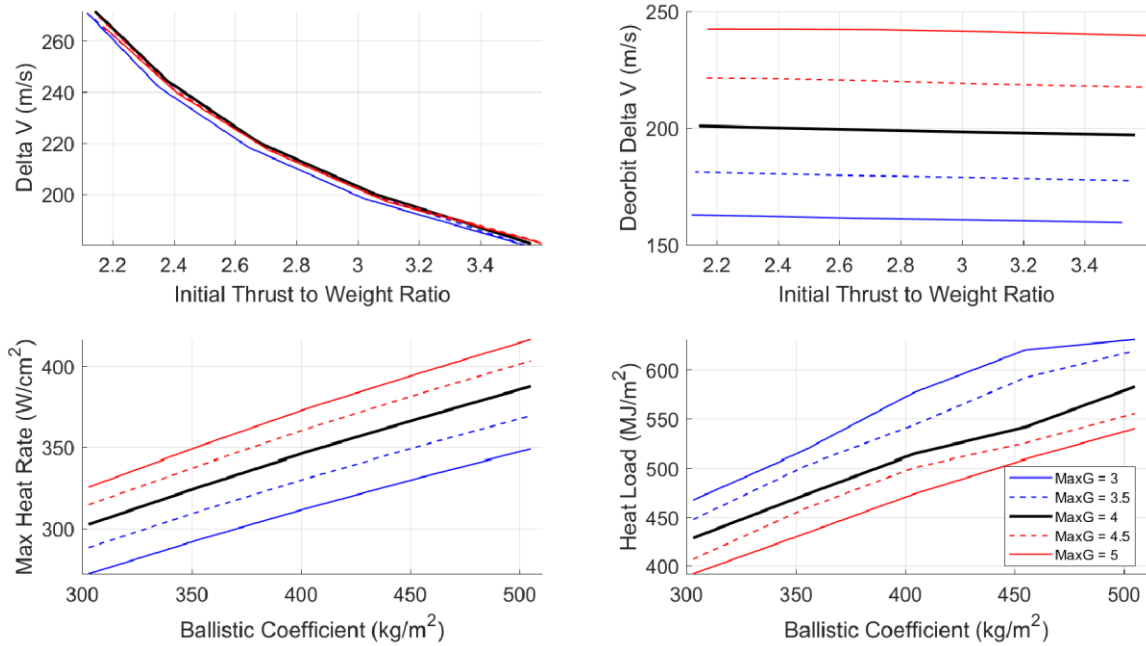


Figure 8: Effect of both the ballistic coefficient and max G load on the experienced conditions of the descent from LEO subtype.

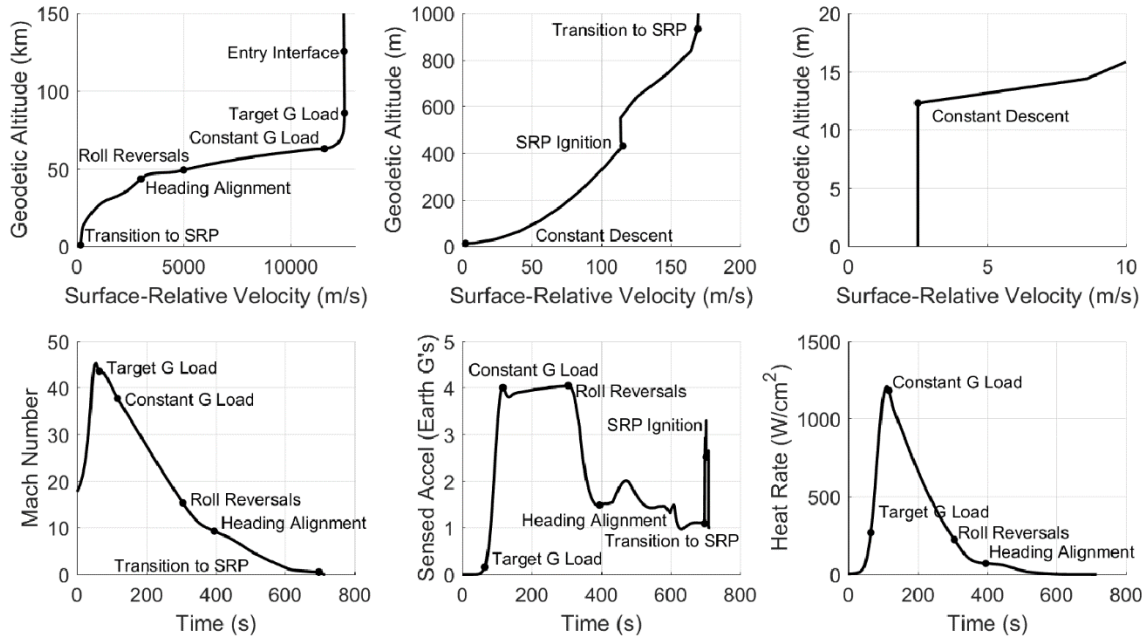


Figure 9: Plots of the trajectory of the nominal fast-transfer Mars return subtype.

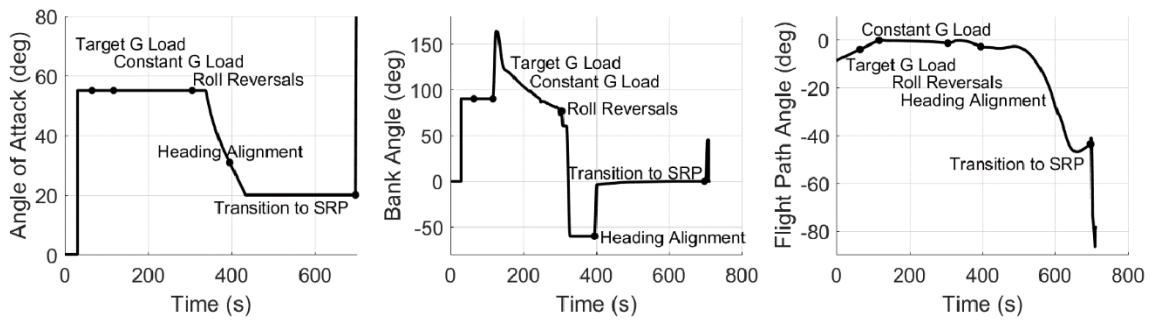


Figure 10: Angle of attack, bank angle, and flight path angle for the nominal fast-transfer Mars return subtype.

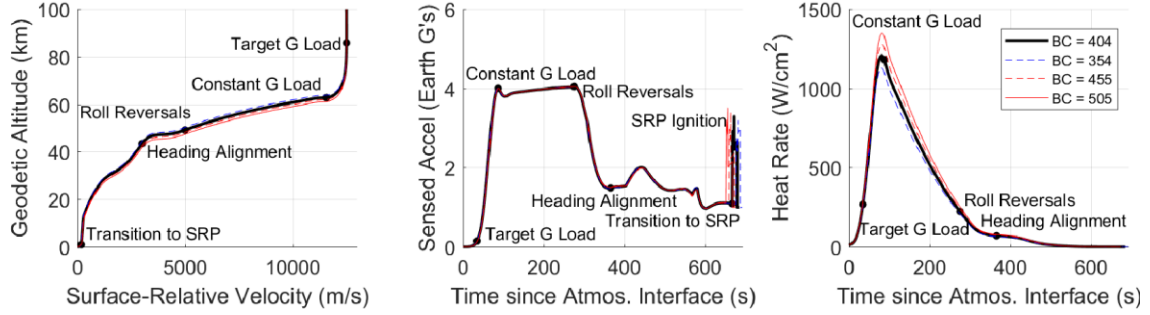


Figure 11: Effect of ballistic coefficient on the trajectory of the fast-transfer Mars return subtype.

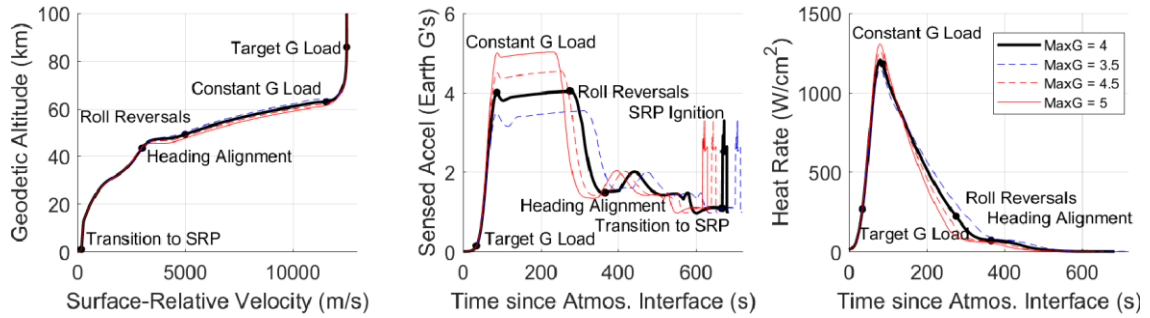


Figure 12: Effect of max G load on the trajectory of the fast-transfer Mars return subtype.

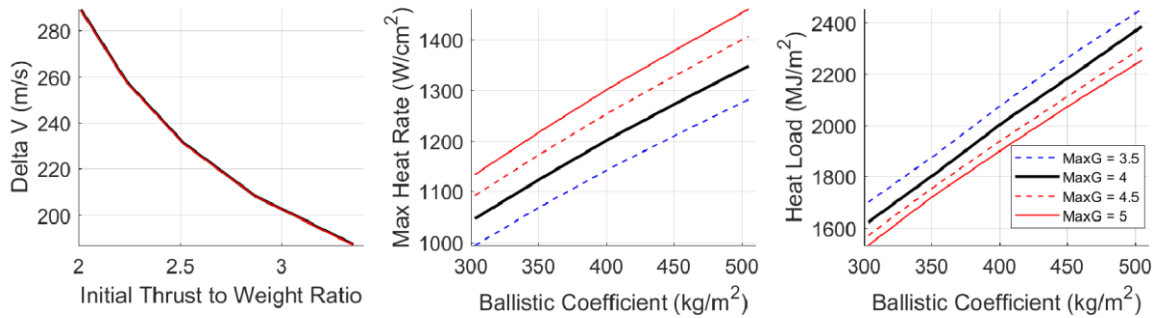


Figure 13: Effect of both the ballistic coefficient and max G load on the experienced conditions of the fast-transfer Mars return subtype.

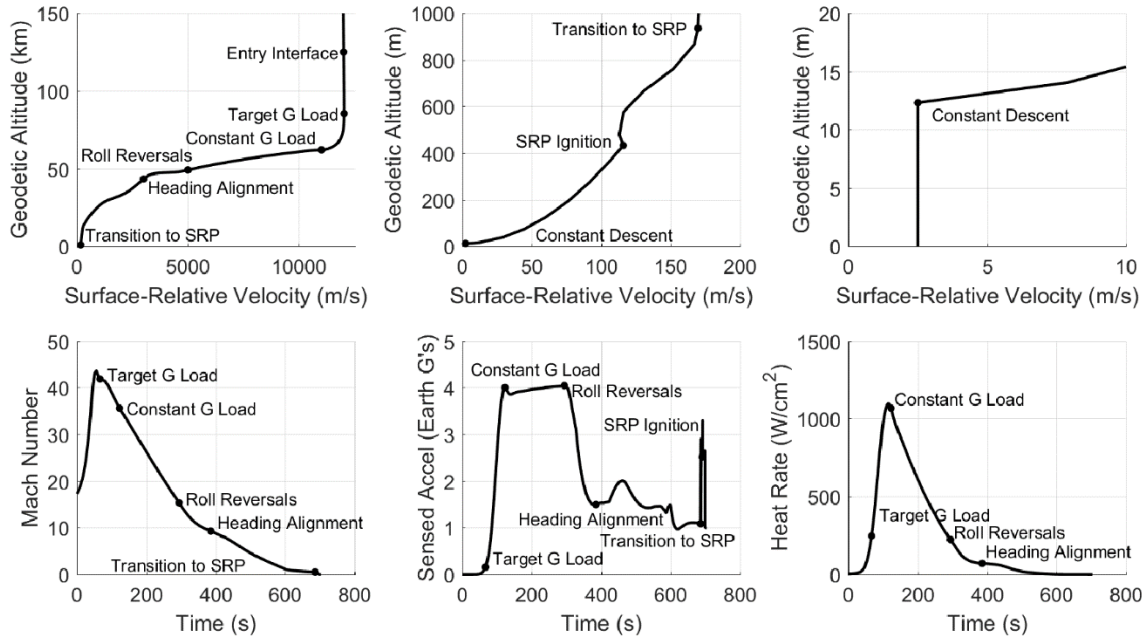


Figure 14: Plots of the trajectory of the nominal minimum-energy Mars return subtype.

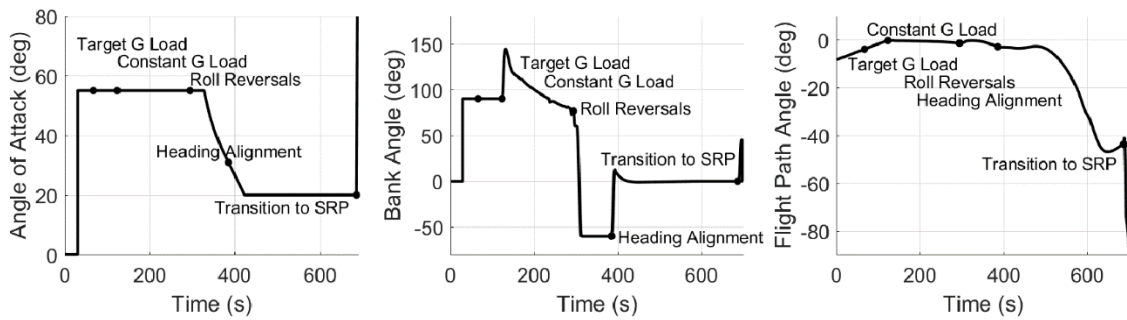


Figure 15: Angle of attack, bank angle, and flight path angle for the nominal minimum-energy Mars return subtype.

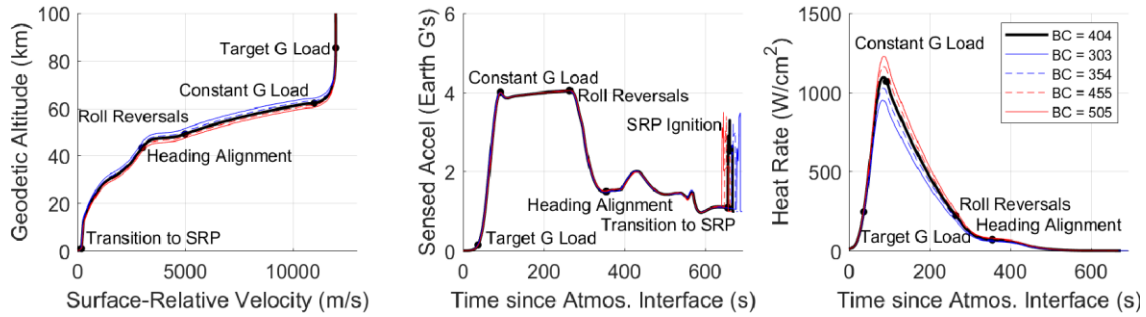


Figure 16: Effect of ballistic coefficient on the trajectory of the minimum-energy Mars return subtype.

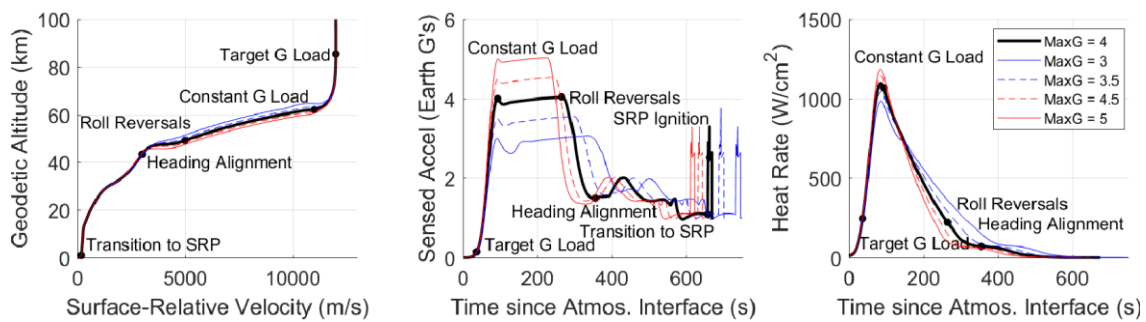


Figure 17: Effect of max G load on the trajectory of the minimum-energy Mars return subtype.

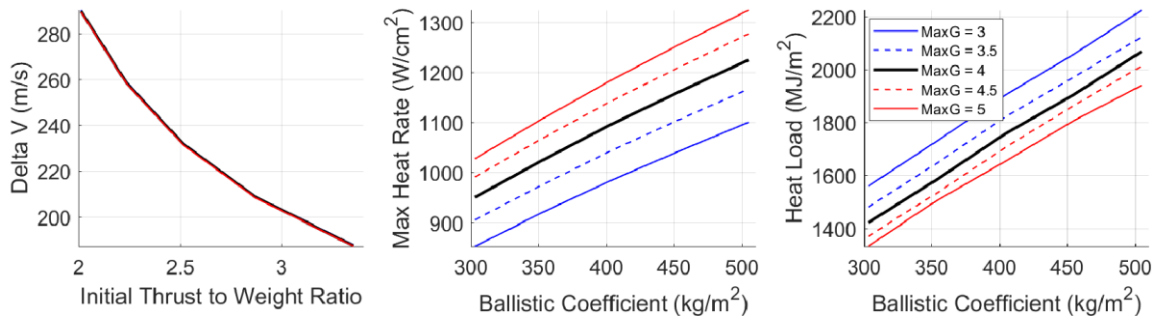


Figure 18: Effect of both the ballistic coefficient and max G load on the experienced conditions of the minimum-energy Mars return subtype.

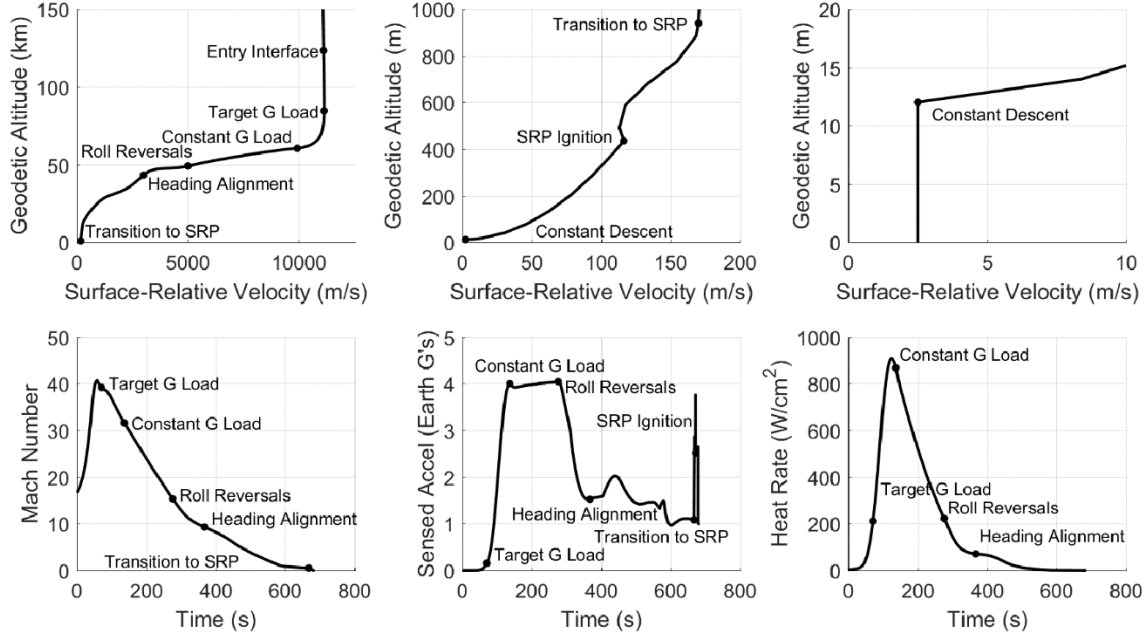


Figure 19: Plots of the trajectory of the nominal Lunar return subtype.

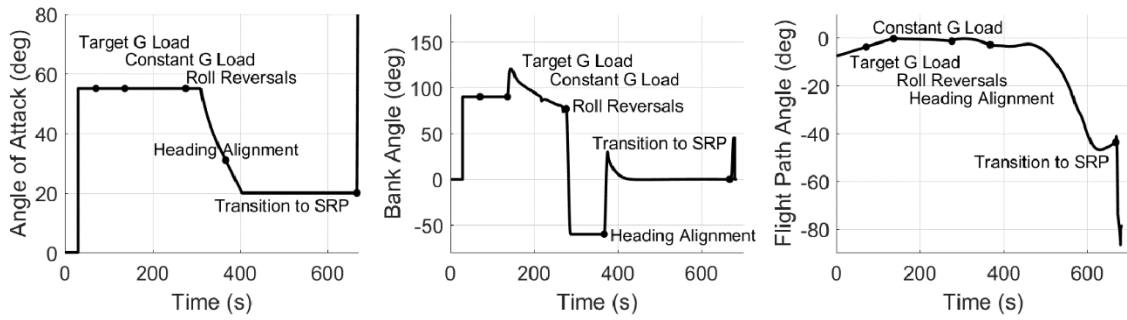


Figure 20: Angle of attack, bank angle, and flight path angle for the nominal Lunar return subtype.

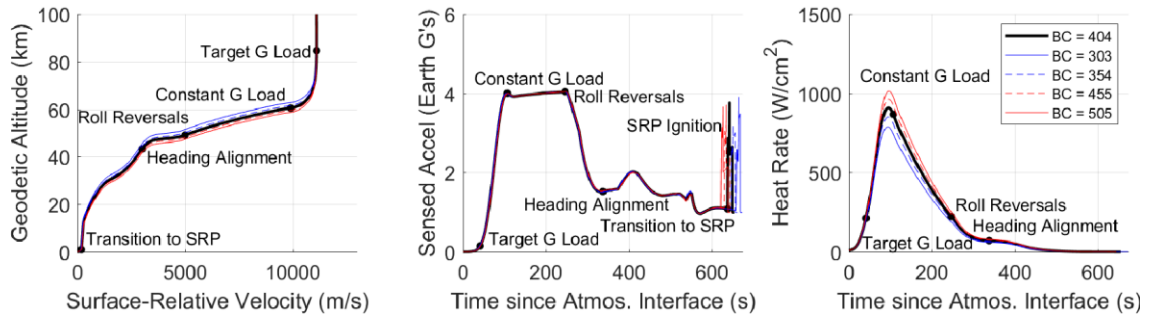


Figure 21: Effect of ballistic coefficient on the trajectory of the Lunar return subtype.

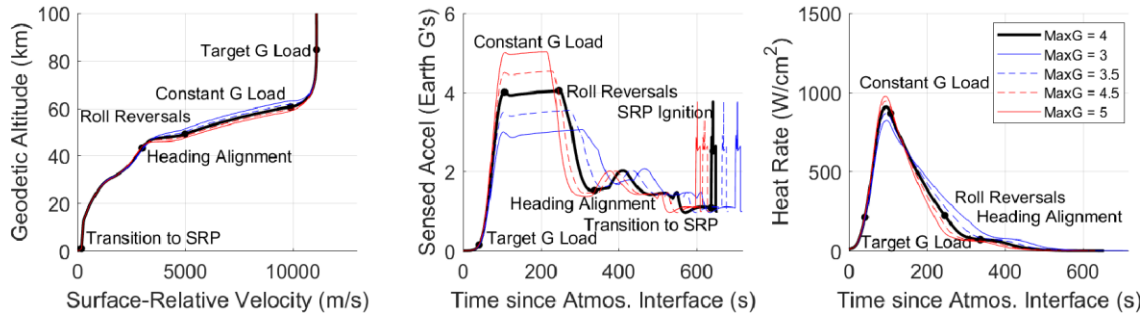


Figure 22: Effect of max G load on the trajectory of the Lunar return subtype.

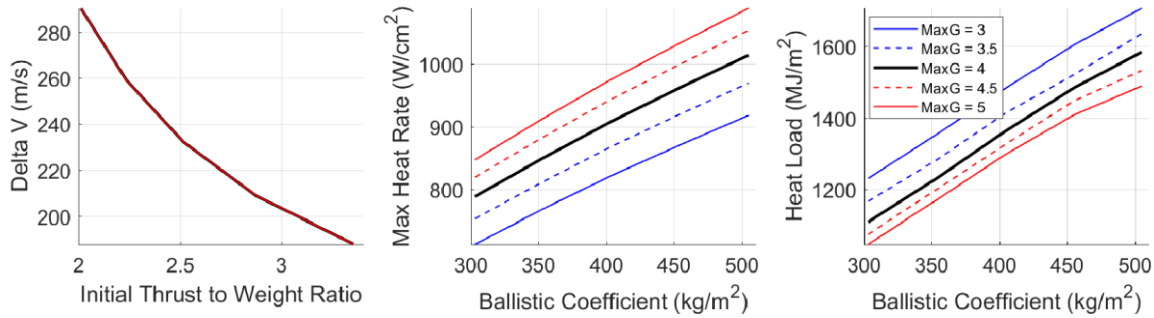


Figure 23: Effect of both the ballistic coefficient and max G load on the experienced conditions of the Lunar return subtype.

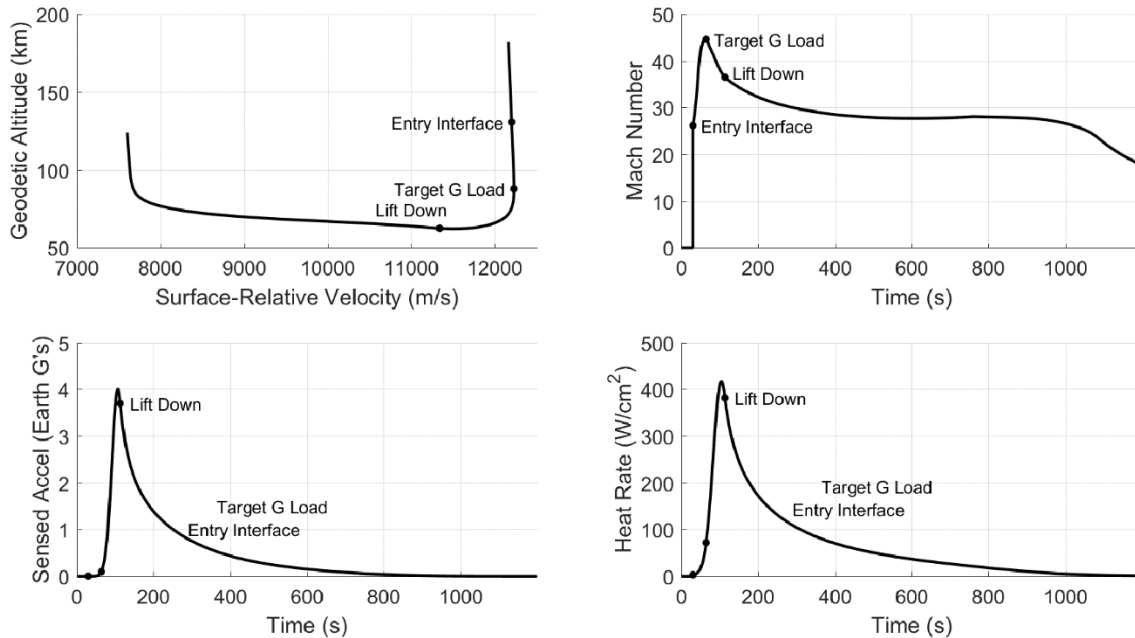


Figure 24: Plots of the trajectory of the nominal fast-transfer Mars return to LEO subtype.

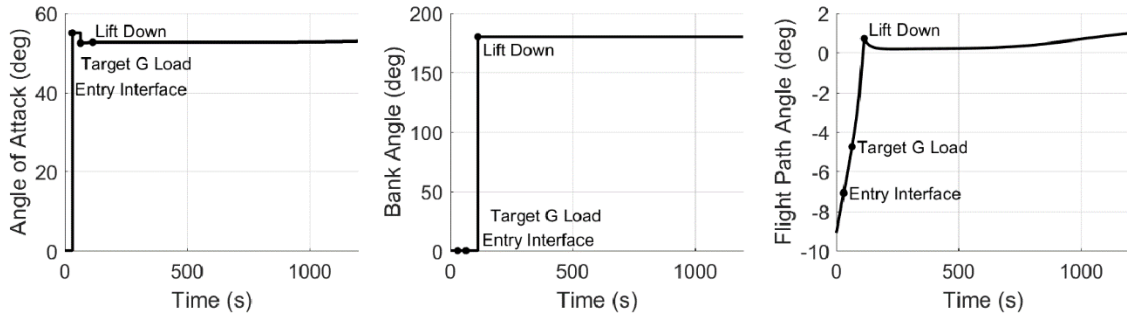


Figure 25: Angle of attack, bank angle, and flight path angle for the nominal minimum-energy Mars return subtype.

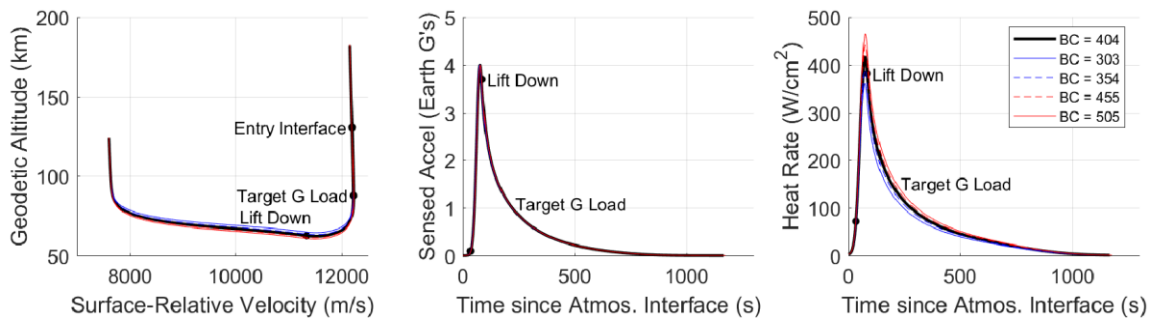


Figure 26: Effect of ballistic coefficient on the trajectory of the fast-transfer Mars return to LEO subtype.

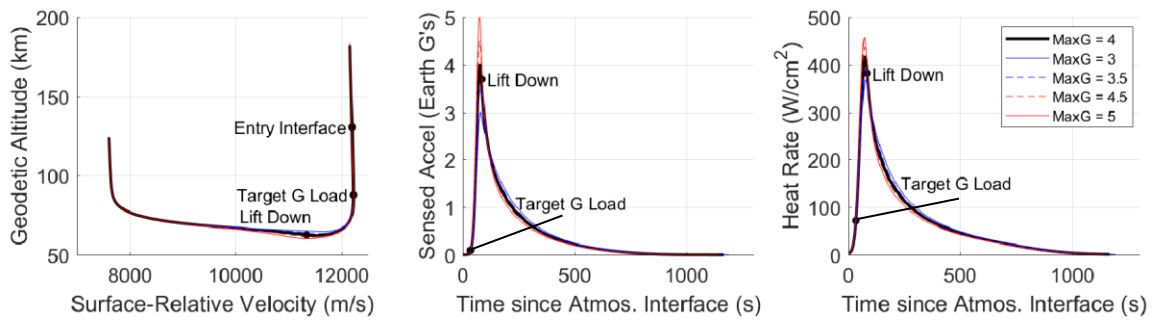


Figure 27: Effect of max G load on the trajectory of the fast-transfer Mars return to LEO subtype.

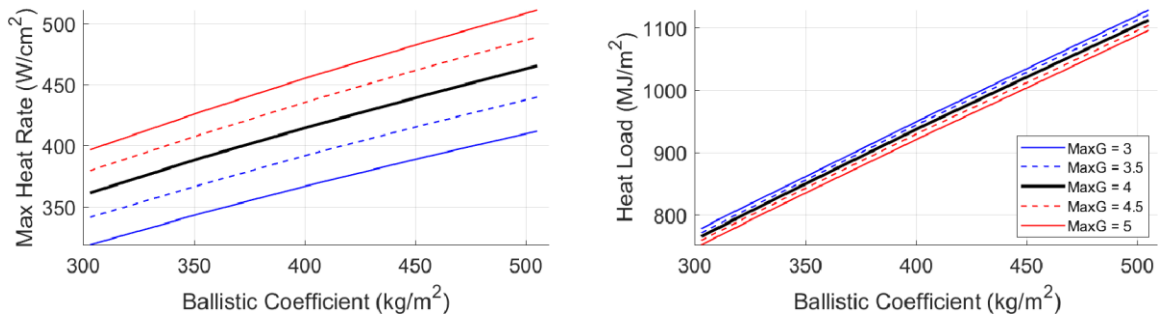


Figure 28: Effect of both the ballistic coefficient and max G load on the experienced conditions of the fast-transfer Mars return to LEO subtype.

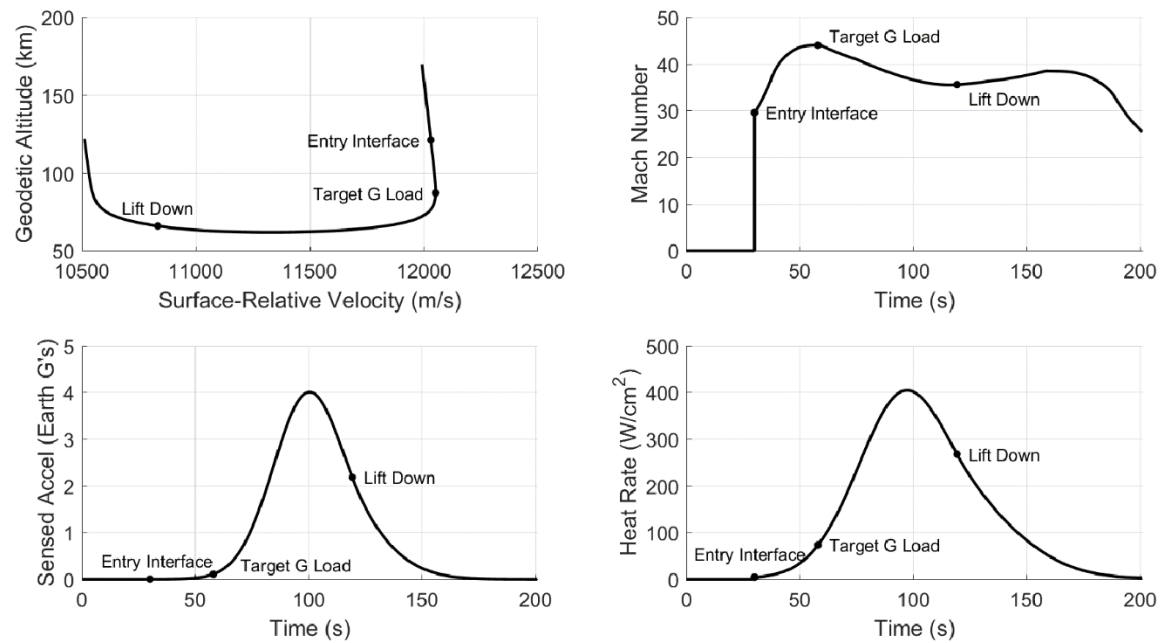


Figure 29: Plots of the trajectory of the nominal fast-transfer Mars return to LDHEO subtype.

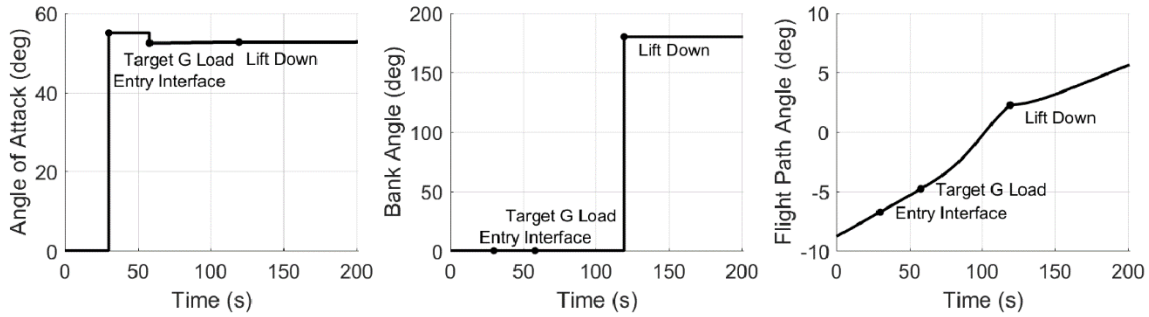


Figure 30: Angle of attack, bank angle, and flight path angle for the nominal fast-transfer Mars return to LDHEO subtype.

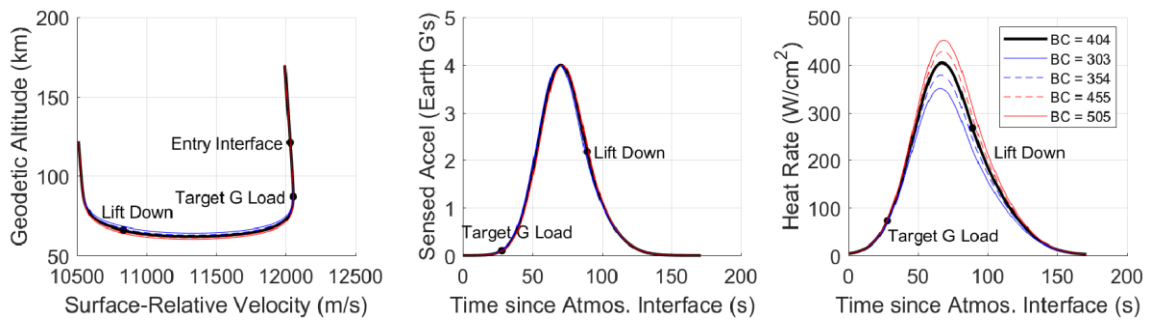


Figure 31: Effect of ballistic coefficient on the trajectory of the fast-transfer Mars return to LDHEO subtype.

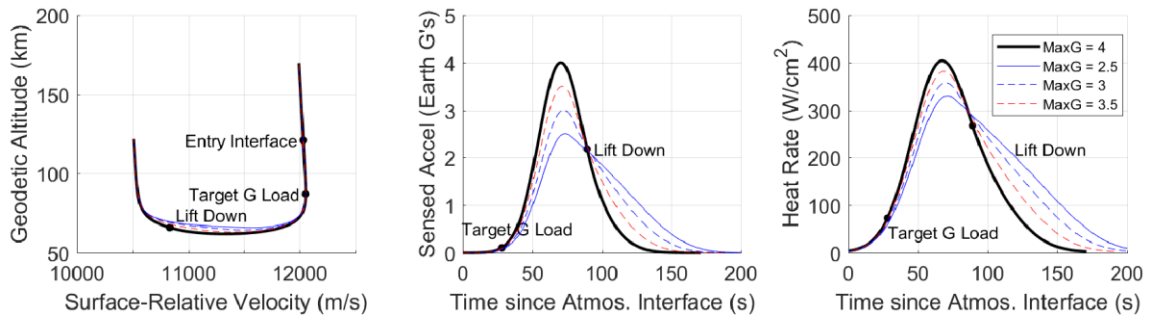


Figure 32: Effect of max G load on the trajectory of the fast-transfer Mars return to LDHEO subtype.

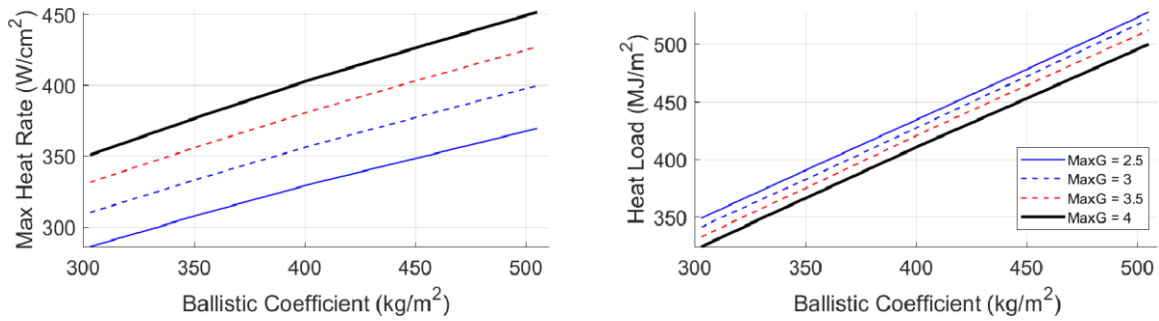


Figure 33: Effect of both the ballistic coefficient and max G load on the experienced conditions of the fast-transfer Mars return to LDHEO subtype.

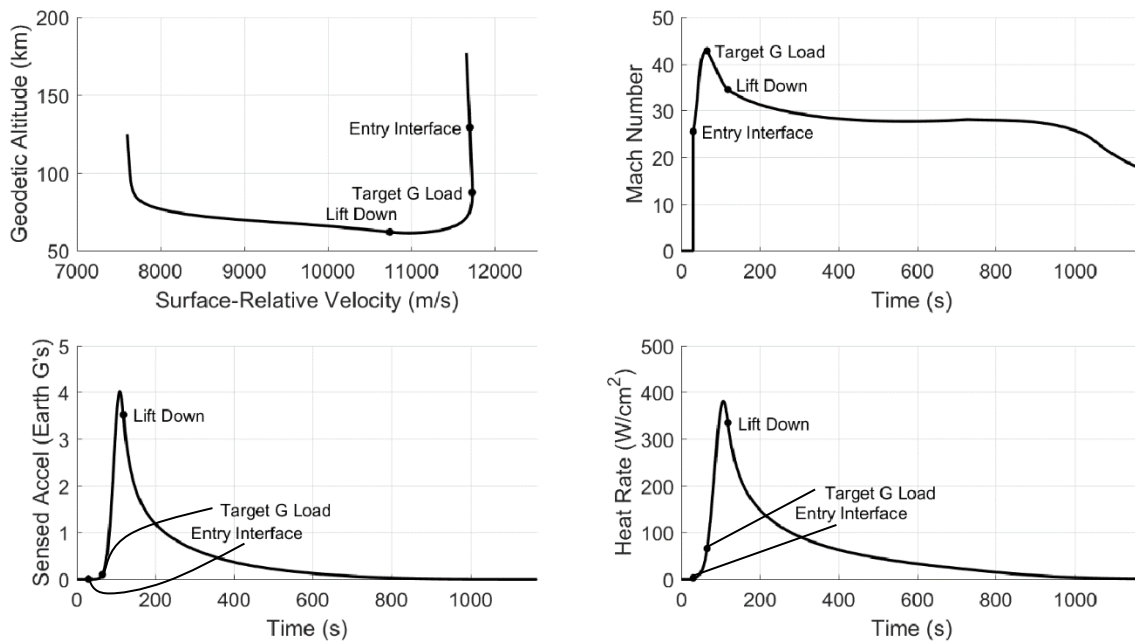


Figure 34: Plots of the trajectory of the nominal minimum-energy Mars return to LEO subtype.

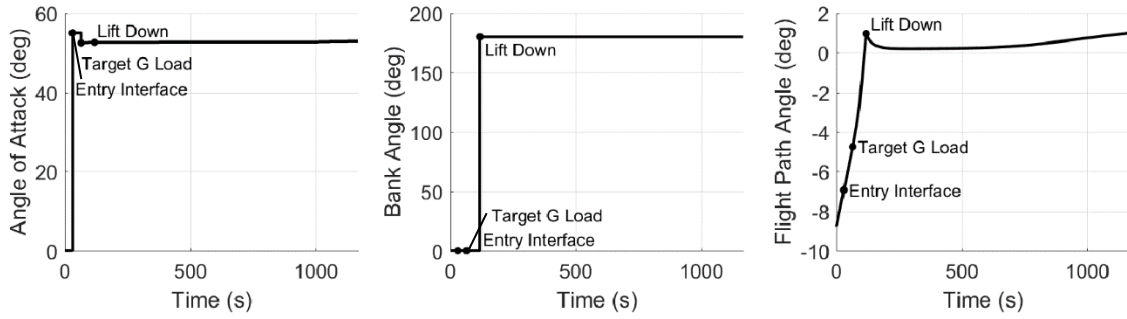


Figure 35: Angle of attack, bank angle, and flight path angle for the nominal minimum-energy Mars return to LEO subtype.

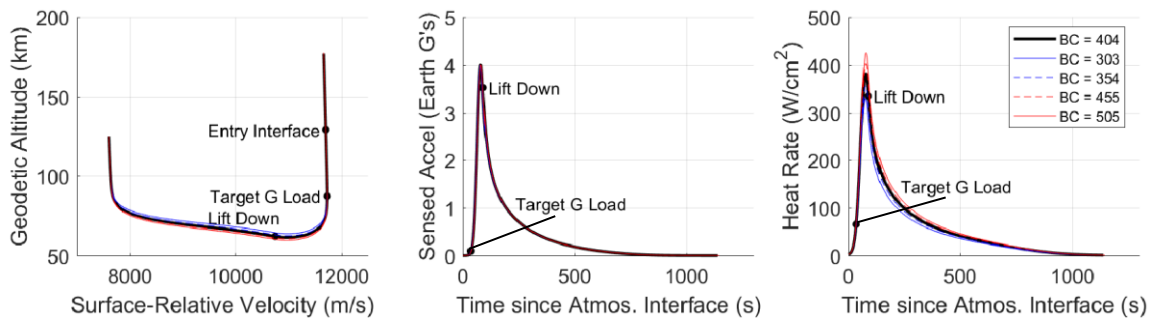


Figure 36: Effect of ballistic coefficient on the trajectory of the minimum-energy Mars return to LEO subtype.

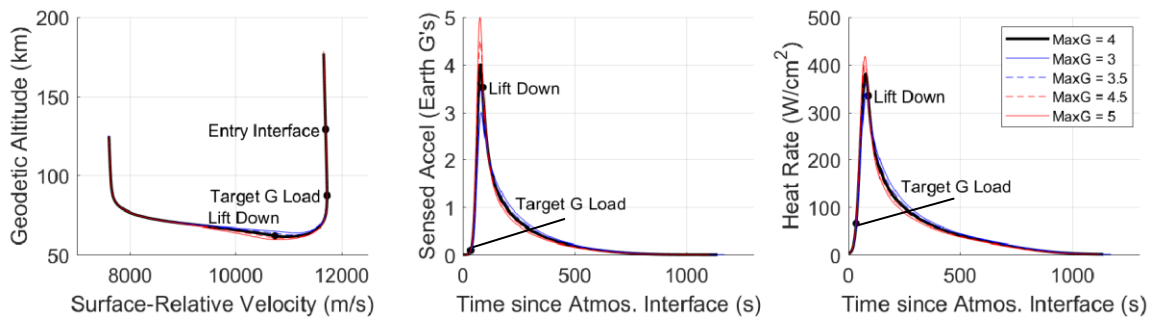


Figure 37: Effect of max G load on the trajectory of the minimum-energy Mars return to LEO subtype.

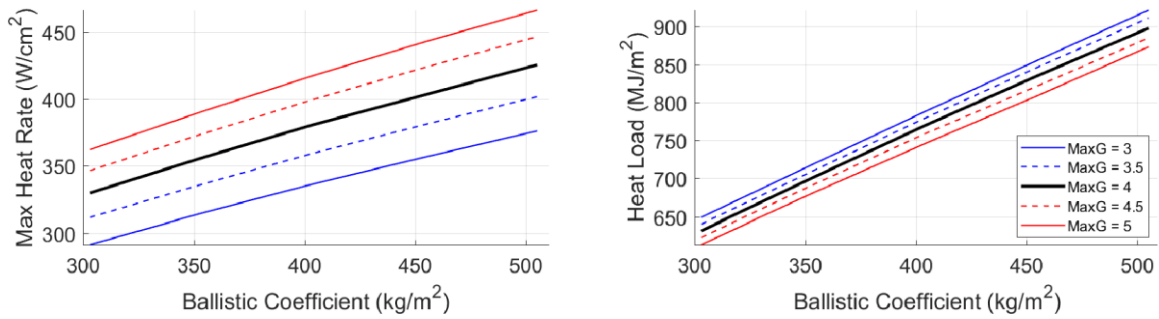


Figure 38: Effect of both the ballistic coefficient and max G load on the experienced conditions of the minimum-energy Mars return to LEO subtype.

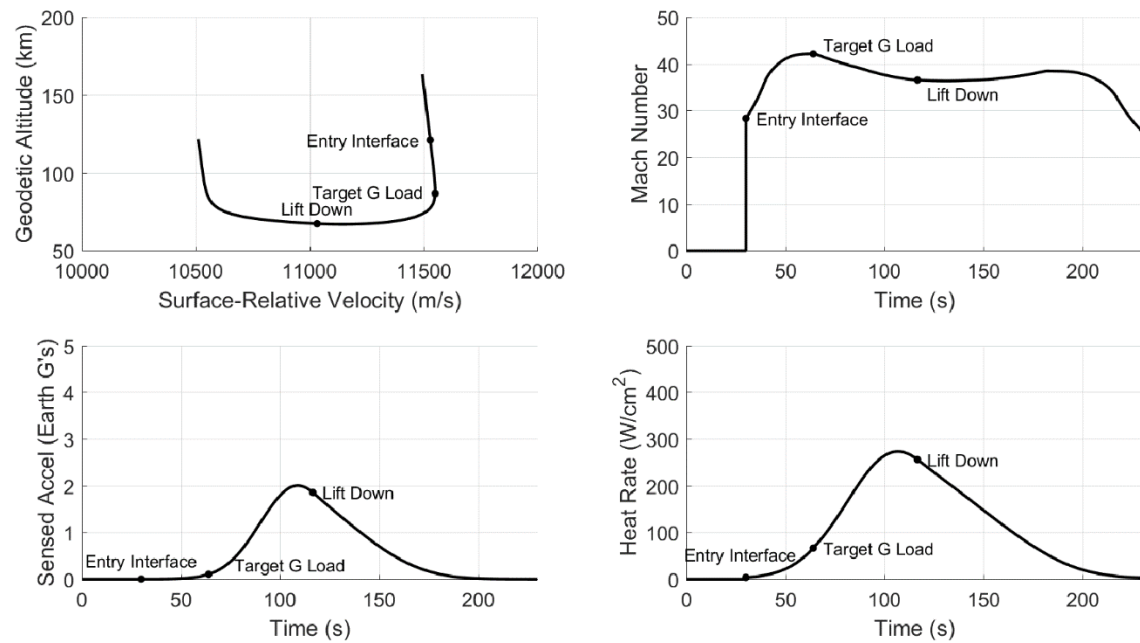


Figure 39: Plots of the trajectory of the nominal minimum-energy Mars return to LDHEO subtype.

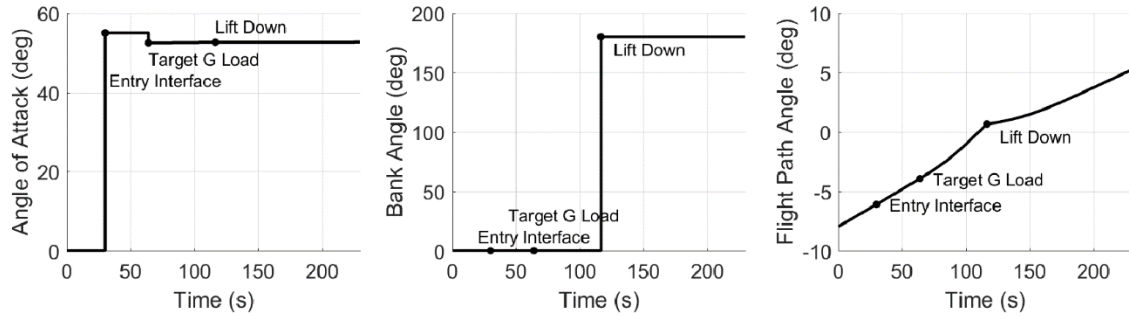


Figure 40: Angle of attack, bank angle, and flight path angle for the nominal minimum-energy Mars return to LDHEO subtype.

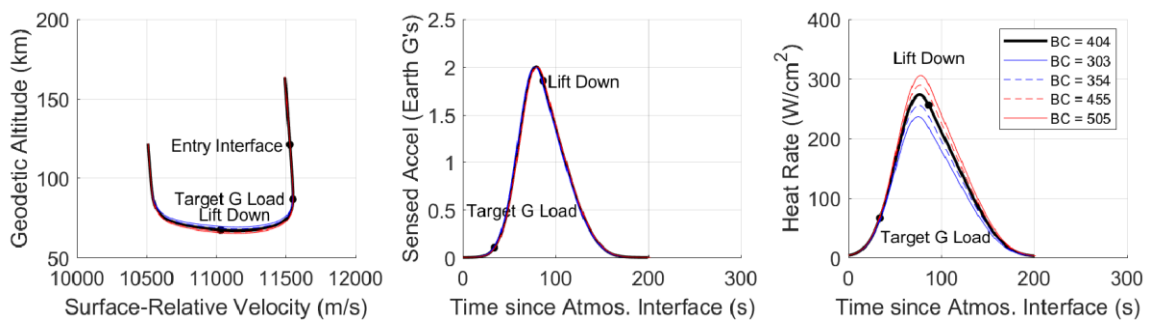


Figure 41: Effect of ballistic coefficient on the trajectory of the minimum-energy Mars return to LDHEO subtype.

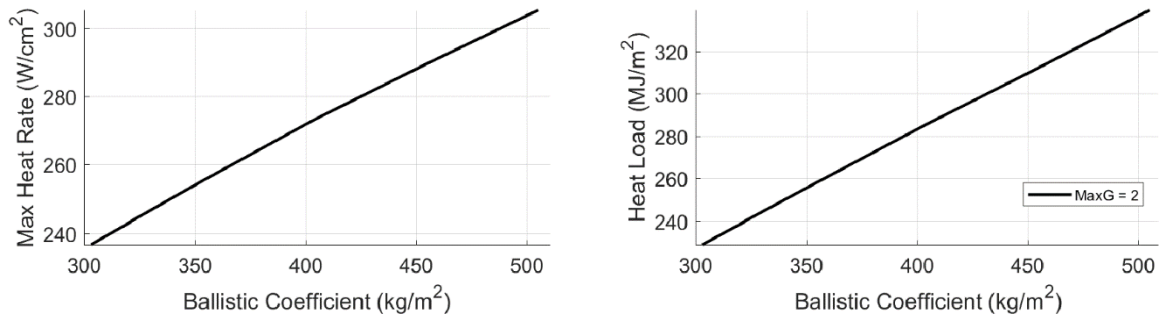


Figure 42: Effect of both the ballistic coefficient and max G load on the experienced conditions of the minimum-energy Mars return to LDHEO subtype.

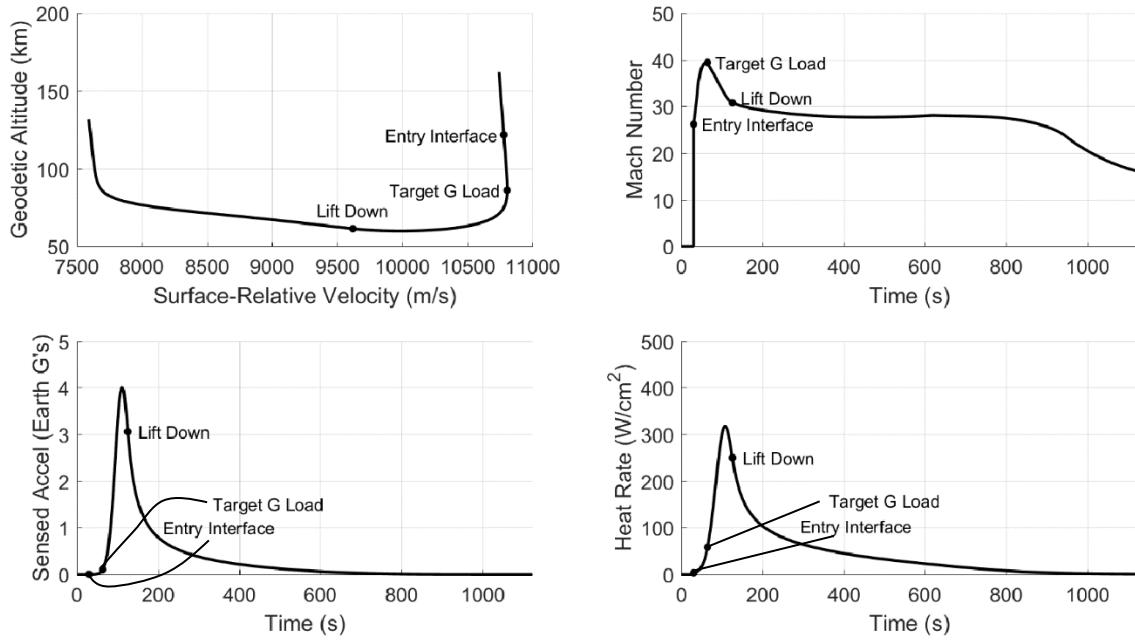


Figure 43: Plots of the trajectory of the nominal aerotransfer from LDHEO to LEO subtype.

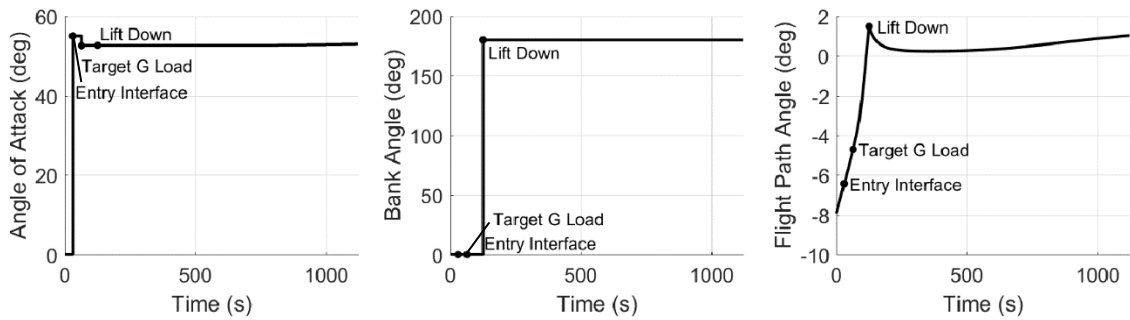


Figure 44: Angle of attack, bank angle, and flight path angle for the nominal aerotransfer from LDHEO to LEO subtype.

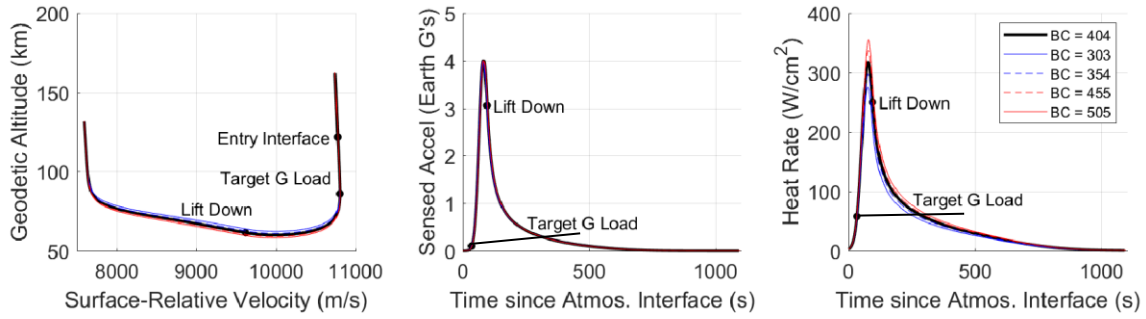


Figure 45: Effect of ballistic coefficient on the trajectory of the aerotransfer from LDHEO to LEO subtype.

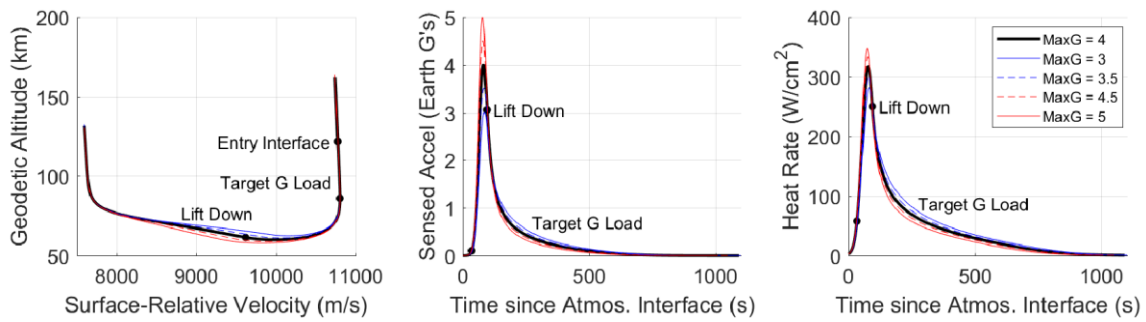


Figure 46: Effect of max G load on the trajectory of the aerotransfer from LDHEO to LEO subtype.

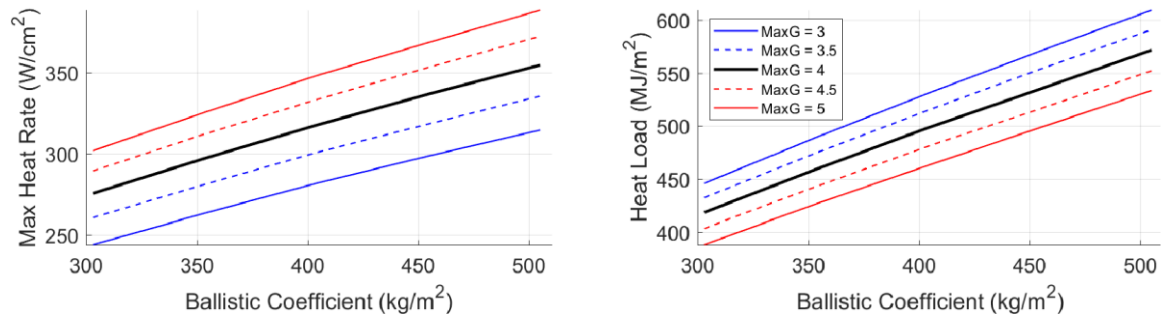


Figure 47: Effect of both the ballistic coefficient and max G load on the experienced conditions of the aerotransfer from LDHEO to LEO subtype.

VITA

Braxton Brakefield was born and raised in Nashville, Tennessee. He went to Hillsboro High School, as well as attended the School for Science and Math at Vanderbilt. While there, he was exposed to engineering directly, and decided that was a path he wanted to continue following. He enrolled in the University of Tennessee, Knoxville for undergrad, initially as an undecided engineering major. After watching a few livestreamed rocket launches his freshman year, he realized he could be a part of that, and picked aerospace engineering as his focus. During his junior year, he had Dr. Evans Lyne as a professor, and got involved with research under him when the opportunity arose. The research was into hybrid rocket motors, primarily through experimental testing, and while very interesting, Braxton decided he would rather be in air conditioning over the hot, humid, and buggy test stand facility. Therefore, when it was time to pick senior design projects, Braxton took a chance on an interplanetary mission design over continuing to work on hybrid motors. That ended up being an excellent choice, as the project, designing a sample return mission to Saturn's moon Enceladus, was awesome and taught him a lot. That project also cemented his desire to work on the computational side of aerospace, and specifically introduced him to trajectory simulation, something he became quite interested in. He decided he wanted to pursue that interest further, and so he chose to stay at UT to pursue a master's in aerospace under Dr. Lyne. During this time, he also came into contact with D.R. Komar at NASA Langley, who was looking for an intern to do some trajectory simulations of the Hercules Vehicle. That culminated in Braxton performing that trajectory work and using the methods and data for his master's thesis, this document. Braxton is now about to graduate with his master's from UT, and is moving to Seattle to join Blue Origin as a Trajectory Analyst.