Coupled Aero-Structural Modelling of Deployable Aero-Decelerators for Mars Entry

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Analysis of deployable aero-decelerators has been performed using a developed 6 degree-of-freedom entry trajectory simulator coupled with a structural model of the deployable structural members, or ribs, to investigate the effect of aero-decelerator flexibility on the trajectory and configuration design. The Modified Newtonian method is used in the entry trajectory simulator, and the deployable ribs are modelled as Euler-Bernoulli beams. It is shown that, although flexibility is beneficial in reducing mass and volume of the deployed ribs, an increase in peak heat flux will result. However, if mass savings from flexible ribs can be reallocated towards increasing the diameter of the entry vehicle, significant benefits can be gained.

Nomenclature

\[ a = \text{aerodynamic acceleration (m/s}^2) \text{ or distance between rib end and support strut (m)} \]

\[ c_p = \text{pressure coefficient} \]

\[ e_0, e_1, e_2, e_3 = \text{attitude quaternions of the entry vehicle} \]

\[ E = \text{Young’s Modulus (Pa)} \]

\[ F = \text{aerodynamic force (N)} \]

\[ g = \text{gravitational acceleration (m/s}^2) \]

\[ G_{GC2B} = \text{rotation matrix to transform between the geocentric frame and the body frame} \]

\[ I = \text{inertia matrix (kgm}^2) \text{ or second moment of area (m}^4) \]

\[ L = \text{length of deployed rib (m)} \]

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\( M \) = moment (Nm)
\( M_\infty \) = upstream Mach number
\( r \) = radial distance from the entry vehicle to the centre of Mars (m)
\( R \) = reaction force (N)
\( u, v, w \) = North, East and Down components of the velocity vector (m/s)
\( x \) = distance along rib from the hinged end (m)

\( \gamma \) = ratio of specific heat capacities
\( \delta \) = deflection (m)
\( \theta \) = angle between mesh element normal and velocity vector, used in Modified Newtonian equation (rad)
\( \vartheta \) = longitude (°E)
\( \varphi \) = latitude (°N)
\( \Omega_{Mars} \) = rotational velocity of Mars (rad/s)
\( \omega \) = roll, pitch and yaw rotation rates (rad/s)

Subscripts

\( \text{end} \) = at the hinged end of the deployable rib
\( i \) = mesh node under consideration
\( \text{ss} \) = at the support strut location
\( u, v, w \) = North, East and Down components
\( x, y, z \) = body-frame components

I. Introduction

FUTURE Mars lander missions will require very different atmospheric entry vehicles to past and present designs. The existing rigid aeroshells of diameters up to 4.5 m have reached their mass delivery limit with NASA’s recent Mars Science Laboratory mission, which delivered approximately 1000 kg to the Martian surface [1]. The next phase
of Mars exploration will require surface payloads of up to 20,000 kg, to enable missions such as Southern Highlands rovers and human landing on Mars. Delivery of such large payloads requires much larger diameter entry vehicles, or radically different designs. As the diameter of available launcher fairings is currently fixed at around 4.5 m, larger diameters are only achievable with deployable aero-decelerators that are stowed and folded inside the launch vehicle fairing during launch. These then extend either via mechanical deployment or inflation once in space and after the fairing has been jettisoned, at any point along the transfer. As well as enabling higher mass payloads to be delivered to the surface of Mars, deployable and inflatable aero-decelerators would also enable landing at higher surface elevations, and the extended time available for entry would allow for guidance maneuvers to be used to achieve more precise landings.

Mechanically deployable aero-decelerators consist of structural elements such as ribs and struts that are covered in flexible thermal protection system (TPS) material and deployed via mechanisms. Inflatable aero-decelerators comprise layers of bladder fabrics that inflate via gas generators, also covered in flexible TPS material. Both inflatable and mechanically deployable aero-decelerators have been investigated in recent years, but the major focus has been on inflatable designs such as NASA’s Hypersonic Inflatable Aerodynamic Decelerator (HIAD) [2], which has seen extensive technology development and testing. Although mechanically deployable concepts are now being developed in NASA’s ADEPT project and the Italian Space Agency/European Space Agency IRENE project, the technology readiness level remains low, and the technology is considered underexplored.

Mechanically deployable aero-decelerators offer a number of potential advantages over an inflatable concept. They are resilient to micrometeoroid impact [3]. They are capable of deployment testing and re-stowage at any point, including in Earth orbit to demonstrate safety. They are also capable of withstanding a dual heat pulse to perform both aerocapture and entry, descent and landing with a single aero-decelerator, if the ballistic coefficient is sufficiently low that no ablation occurs, whereas inflatables cannot retain inflation for extended periods of time due to leakage. Guidance of the vehicle could be enabled by individual control of the deployed elements and hence lift vector. In addition, there is potential to use the deployed elements as landing gear [4], thus reducing the overall mission mass, and eventually there could be full re-use in later missions.

In this paper, we focus on mechanically deployable aero-decelerators for the above reasons. The primary objectives are to advance the technology readiness level of the deployable aero-decelerator configuration, and to propose and assess a low-mass, robust and high performance design for such a decelerator. This paper proceeds with a brief review
of past deployable concepts, followed by presentation of the 6DOF entry trajectory simulator with aero-structural coupling developed specifically for assessing deployable aero-decelerator concepts, which inherently have some flexibility associated to their design. Finally, the results from an optimization study are presented in the context of trade-offs and the unique potential offered by flexible deployable designs.

II. Configuration Review

Concepts of mechanically deployable aero-decelerators have existed for decades, with numerous designs proposed, including those detailed in Table 1 and illustrated in Fig. 1. However, the majority of these concepts were early paper designs only, with little hardware development and no further progression. Two designs – ADEPT and IRENE – have moved on from the concept phase to perform hardware development and testing in recent years, with further tests and technology development planned.

Table 1 Past mechanical deployable aero-decelerator configurations. See Fig. 1 for illustrations

<table>
<thead>
<tr>
<th>Concept</th>
<th>Key Parameters</th>
<th>Design</th>
</tr>
</thead>
</table>
Various assessments of the mass and performance of mechanically deployable aero-decelerators have also taken place in recent years, with widely differing results as seen in Fig. 2. Wercinski [13] estimated a human-class ADEPT decelerator would require 46% of the entry vehicle mass based on Yount [11], whereas Cassell [3] estimated a mass fraction of 15.4% would be required. Braun [14] investigated a 22 m diameter deployable decelerator requiring a mass fraction of 8%, whereas Underwood [12] estimated a 5.7 m diameter IRENE-type decelerator would need 36%.

Fig. 1 Illustrations of past mechanically deployable aero-decelerator concepts.
Fig. 2 Past mass estimates for mechanically deployable aero-decelerators.

Fig. 2 plots the decelerator mass fractions against the initial entry vehicle mass for the concepts described in Table 1, where mass fraction is defined as the mass of the deployable aero-decelerator divided by the total entry vehicle mass. For the lower mass entry vehicles, below 10,000 kg, the mass fractions vary significantly, by tens of percent. As the entry vehicle masses (and hence diameters) increase, the deployable aero-decelerator mass fraction estimates decrease to much lower levels than for the smaller vehicles, suggesting the technology is best suited to larger vehicles. Mass comparisons with other entry vehicle designs such as inflatables yield conflicting results. Cianciolo [15] contains a recent mass comparison of the three Mars entry vehicle concepts currently under consideration by NASA for human exploration: ADEPT, HIAD and the rigid Mid Lift-to-Drag body. The margin philosophy applied to these three concepts differs significantly, with the Mid Lift-to-Drag vehicle incorporating maturity margin of approximately 21.5% [16] compared to the ADEPT vehicle [3], which includes 35.5% maturity margin. The maturity margin applied to HIAD is unclear. The margin philosophy has a very significant effect on the overall mass estimates.

It is clear that developing robust mass estimates for next generation Mars entry vehicles is challenging, and no consistent approach has been applied across the many concepts. Good quality mass estimates are, however, critical for determining the decelerator performance in terms of ballistic coefficient, entry trajectory capability, and payload mass fraction. To enable deployable aero-decelerator mass estimates to be made with confidence, the entry trajectory
and aero-structural simulator described in the following sections has been developed and correlated. This tool will improve the deployable rib mass estimation process and facilitate the exploration and optimization of potential deployable aero-decelerator designs. This is particularly important for the unique challenge of modelling very large diameters, where deformations can become very significant.

III. 6DOF Entry Trajectory Simulator

A 6 degree-of-freedom (6DOF) entry trajectory simulator has been developed to assess the performance of the potential entry vehicle configurations under consideration. This has been implemented in the Matlab suite [17]. This reduced-order model allows fast assessment during early phase design, and uses the Modified Newtonian local surface inclination method to determine the aerodynamic pressures and forces. The resulting aerodynamic coefficients are updated at every timestep, so no aerodynamic database is required for the trajectory analysis, in contrast to many industrially-used tools, and a wide range of shapes can therefore be assessed.

A. Simulator Description

The simulator requires the initial trajectory conditions at the atmospheric entry interface point, and the dimensions and parameters of the entry vehicle as inputs. Note that relative entry velocity is required in the initial conditions. The Mars atmosphere is modelled using the European Mars Climate Database [18], which provides the major atmospheric data of interest for specific entry dates and locations. Additional freestream properties are calculated using ideal gas assumptions.

A geometry mesh is created based on the entry vehicle input parameters, and the surface normal for each mesh element determined. This allows $\theta$, the angle between the mesh element normal and the velocity vector, to be calculated – the major variable required to determine the pressure coefficient at each mesh element using the Modified Newtonian method [19], which is applicable for hypersonic speeds only and states that

$$c_p = c_{p,\text{max}} \sin^2 \theta$$  \hspace{1cm} (1)

where $c_p$ is the pressure coefficient, and $c_{p,\text{max}}$ is the maximum pressure coefficient, defined by Anderson [19] such that
\[ c_{p,max} = \frac{2}{M_{\infty}^2 \gamma} \left\{ \frac{(y+1)^2 M_{\infty}^2}{4 y M_{\infty}^2 - 2 (y-1)} \frac{y}{y-1} \left[ \frac{1 - y + 2 y M_{\infty}^2}{y+1} \right] - 1 \right\} \]  

in which \( M_{\infty} \) is the upstream Mach number and \( \gamma \) is the ratio of specific heats. Various entry vehicle shapes can be modelled, and the mesh resolution can be tailored according to the requirements and convergence analysis results. The mesh co-ordinate system comprises the \( x \)-axis pointing forwards from the nosetip, and the \( y \) and \( z \)-axes making up a right-handed set.

The pressure applied to each mesh element is calculated using the pressure coefficient, the forces at each mesh element are then determined, and the \( x, y, z \) component of the forces are summed across all mesh elements. The aerodynamic coefficients are calculated in terms of the axial, normal and side forces, and the roll, pitch and yaw moments. In order to determine the moments, the mass properties of the entry vehicle are incorporated via an inertia matrix, \( I \), either directly input or calculated based on input mass properties. The drag and lift coefficients are also determined based on the angle of attack, followed by the lift-to-drag ratio and drag, lift and side forces. These are the major contributors to the aerodynamic deceleration that the entry vehicle experiences.

The moments of force are calculated about the centre of gravity of the entry vehicle for each mesh element and then summed across the entire entry vehicle to give the moments \( M_x, M_y, M_z \). These moments act on the entry vehicle to produce angular accelerations, which are determined using Euler’s rotation equation for the rotation of a 3D rigid body in space

\[
\begin{bmatrix}
M_x \\
M_y \\
M_z
\end{bmatrix} = M = I \cdot \dot{\omega} + \omega \times (I \cdot \omega) \tag{3}
\]

where \( \omega \) is the angular velocity and \( \dot{\omega} \) the angular acceleration. This equation is rearranged to get the angular accelerations as a first order differential equation, as in Eq. 11.

The overall acceleration on the entry vehicle consists of a gravitational component, an aerodynamic component, a centrifugal component, and a Coriolis component. Each of these is applied via a set of equations of motion, defined in Titterton and Weston [20], with additional components as required for the specific situation, as

\[
\dot{r} = -w \tag{4}
\]

\[
\dot{\phi} = \frac{u}{r} \tag{5}
\]

\[
\dot{\theta} = -\frac{v}{r \cos \phi} \tag{6}
\]
\[
\dot{u} = a_u + g_u + \frac{1}{r}(uw - v^2 \tan \varphi) - 2\Omega_{Mars} v \sin \varphi \tag{7}
\]

\[
\dot{v} = a_v + g_v + \frac{1}{r}(uv \tan \varphi + vw) + 2\Omega_{Mars}(w \cos \varphi + u \sin \varphi) \tag{8}
\]

\[
\dot{w} = a_w + g_w - \frac{1}{r}(u^2 + v^2) - 2\Omega_{Mars}v \cos \varphi \tag{9}
\]

\[
\begin{bmatrix}
\dot{e}_0 \\
\dot{e}_1 \\
\dot{e}_2 \\
\dot{e}_3
\end{bmatrix} = \frac{1}{2}
\begin{bmatrix}
-e_1 & -e_2 & -e_3 \\
 e_0 & -e_3 & e_2 \\
- e_2 & e_1 & e_0 \\
- e_3 & e_0 & e_1
\end{bmatrix}
\begin{bmatrix}
p \\
q \\
r
\end{bmatrix} - \frac{1}{r} G_{GC2B}
\begin{bmatrix}
v \\
-u \\
-v \tan \varphi
\end{bmatrix} - G_{GC2B}
\begin{bmatrix}
\Omega_{Mars} \cos \varphi \\
0 \\
- \Omega_{Mars} \sin \varphi
\end{bmatrix}
\tag{10}
\]

\[
\begin{bmatrix}
p \\
q \\
r
\end{bmatrix} = I^{-1}
\left(M - \begin{bmatrix} p \\ q \\ r \end{bmatrix} \times \begin{bmatrix} p \\ q \\ r \end{bmatrix}\right)
\tag{11}
\]

where \(r\) is the distance between the entry vehicle and the centre of Mars, \(u, v, w\) are the entry vehicle velocity components in the North, East and Down (NED) co-ordinate frame directions, \(\varphi\) is the latitude in degrees North, \(\vartheta\) is the longitude in degrees East, \(\Omega_{Mars}\) is the rotational velocity of Mars, \(a_u, a_v, a_w\) are the components of aerodynamic deceleration in the NED directions, \(g_u, g_v, g_w\) are the components of gravitational acceleration in the NED directions, \(e_0, e_1, e_2, e_3\) are the attitude quaternions determined from the pitch, yaw and roll angles, \(p, q, r\) are the rotation rates in roll, pitch and yaw directions, and \(G_{GC2B}\) is the rotation matrix required to transform between the geocentric NED frame to the body frame as defined in Karlgaard [21].

The internal Matlab numerical solver ‘ode45’ is then used to integrate these equations of motion over the length of the trajectory to a specified altitude, using a 4th-order Runge-Kutta method. The solver records the state variables for each timestep, and further output parameters can be derived for each timestep and stored in data tables as required. The mesh sizing is optimized for each case to ensure convergence, and temporal convergence is also assured through checks of the timestep size so that the trajectory does not differ by more than 1%.

### B. Correlation

The simulator has been used to recreate trajectories of past NASA Mars missions as a validation exercise, and the results compared to actual flight data in the literature for the Mars Exploration Rover Spirit [22] and Phoenix [23]. These are shown in Fig. 3.
Fig. 3 Trajectory simulation and comparison with actual data for Spirit (left) and Phoenix (right).

The results for the Mars Exploration Rover Spirit align within 5% of the actual flight data, and the results for Phoenix align closely, within 1-2% of the flight data. It should be noted that the exact starting conditions and assumptions for the actual missions are generally not stated in the literature, which likely causes some of the difference between the simulator results and the flight data. However, further investigation was warranted to ensure accurate results for all variables.

The simulator has therefore been correlated against results from the ArianeGroup in-house trajectory tool BL43, using a simple correlation case. The industry tool, like most 6DOF trajectory tools, utilises an aerodynamic database of aerodynamic coefficients built up using analysis initially, and refined during test campaigns. A version of this aerodynamic database was replicated within the Matlab tool, and took the place of the Modified Newtonian aerodynamics calculations. Note that it is not possible to create one of these aerodynamic databases for a flexible deployable aero-decelerator due to the high complexity in analyzing and testing such a large number of variables.
Fig. 4 Velocity-Altitude and Flight path angle-Altitude correlation plots for a rigid body of 2.4 m diameter, 600 kg mass and 5.83 km/s entry velocity.

The position and velocity results shown in Fig. 4 align with the industry tool to within 0.01%. The correlation of the Matlab simulator demonstrates the position and velocity results match those of an aerospace industry tool used for launch and re-entry vehicle trajectory calculations, with the attitude results also matching closely. This gives confidence that the tool can be used for performance assessment of the deployable aero-decelerator concepts.

C. Reference Mission Selection

Two reference missions have been selected as design starting points, to demonstrate the expected mission boundaries within the next twenty years. Ballistic entry trajectories are assumed for both of these missions, along with direct entry from transfer orbit at 6 km/s, and landing at an altitude of 0 km MOLA. The reference missions are:

1. Robotic Mission: with 2000 kg surface payload (e.g. large rover or Mars ascent vehicle). Limited to existing launch vehicles with 4.5 m diameter fairing, and assumed to use a supersonic parachute for the descent phase.

2. Human Cargo Mission: with 20,000 kg surface payload (the minimum payload assumed in recent human mission architectures [15]). Assumed to also fit within existing 4.5 m diameter fairings, with the use of supersonic retropropulsion engines for the descent phase. It is noted that the proposed SLS Block 2 launch vehicle may allow for 10 m diameter fairings in future.

The 6DOF simulator has been used to determine the required entry vehicle diameter that will give the necessary conditions for either supersonic parachute deployment or supersonic retropropulsion ignition. Supersonic parachute
deployment should occur at a Mach number between 1.1 and 2.2 [24], with lower Mach numbers preferred to limit oscillatory motion. In addition, the Viking parachute qualification allows for dynamic pressures at deployment of 239-850 Pa [25]. In order to determine a preferred deployment altitude within this box, an extrapolation from Mars Science Laboratory (MSL) has been assumed. MSL landed at -4.44 km MOLA elevation and a parachute deployment altitude higher than 6 km was required, with 8 km or higher being preferred. Extrapolating from this, in order to land at 0 km MOLA elevation, a parachute deployment altitude of 10 km or higher is likely to be desirable and this is taken as a starting point. The required conditions for supersonic retropropulsion initiation are not yet clear due to the low maturity of SRP for Mars, but it is assumed Mach numbers below 3.5 will be required, potentially at altitudes of at least 3 km [26].

In order to identify trends, we have made some starting point assumptions for the overall entry vehicle mass for these two reference missions. Assuming an entry vehicle mass of 6000 kg based on extrapolation from Mars Science Laboratory, and an entry velocity of 6 km/s, the Robotic Mission requires a 7 m diameter entry vehicle to ensure the required parachute deployment conditions are met, as shown in Fig. 5. Assuming an entry vehicle mass of 50,000 kg based on recent NASA mass estimates [15], and an entry velocity of 6 km/s, the Human Mission requires a minimum of 16 m diameter entry vehicle to enable SRP initiation at Mach 3.5 above 3 km altitude, as shown in Fig. 5. Note that these trajectories assume a fully rigid entry vehicle.

![Fig. 5 Robotic Mission (left) and Human Cargo Mission (right) example entry trajectories.](image)
IV. Aero-Structural Modelling

With the simulator now aerodynamically validated, it is necessary to consider the impact of deployable structural elements on the shape of the entry vehicle. No deployable elements will ever be fully rigid, and it is therefore important to quantify the amount of structural deformation and flexibility that will occur during atmospheric entry, and the subsequent effect on the overall trajectory. Note however that other effects will also occur due to deployable elements, such as flexible TPS deformation and warping. In this model, the structural effect of the flexible TPS is neglected, and the TPS is also assumed to remain flat between adjacent ribs. The configuration assumed in the following description includes deployable ribs that are hinged at one end at the attachment to the entry vehicle, with a single support strut at some variable distance from the hinge. In this study, we assume a support strut distance of 1.1 m for all cases.

Modelling of the structural deformation of the deployable ribs has been included in the simulator as an additional module. This models the deployable ribs as simply-supported beams undergoing deflection, and is coupled into the trajectory simulator so that the structural deformation responds to the changing aerodynamic pressures experienced by the entry vehicle. The geometry mesh is updated at every timestep using the rib deformation model, to give updated aerodynamic coefficients all along the entry trajectory. The deployable rib material, size, cross-section and the number of ribs can be varied as required, to optimize for a particular mission.

A. Aerodynamic Forces

The entry trajectory simulator provides the aerodynamic forces being applied to each mesh element. These forces are assumed to be transferred from the heatshield surface to the ribs by a method that sums the forces, \( F \), in rows mid-rib to mid-rib, and applies these forces to the outer rib nodes for each row as illustrated in Fig. 6.
The exact load path from the flexible TPS to the deployable ribs is dependent on the type of attachment of the TPS to the rib. The above approach assumes the TPS is attached at a number of fixed points along the rib using, for example, rivets or studs. As mesh resolution increases, the above force summing assumption tends towards a continuous TPS attachment along the rib. The TPS material is assumed to maintain tension regardless of rib deformation. The assumption that the flexible TPS between ribs remains flat significantly simplifies the problem and enables a first estimate for aerodynamic forces and heating. It is noted that the amount of TPS flexure is dependent on both the TPS material and the amount of tension in the material when introduced by the deployment mechanisms. However, we acknowledge that some effects due to the likely flexure of the TPS that would occur – particularly increases to local heating – are not captured.

B. Rib Bending

The deployable ribs are each modelled as individual Euler-Bernoulli beams bending under the applied loads from the aerodynamic forces. The bending moment must be calculated for each node of the deployable ribs based on the point load at that location resulting from the summing technique described earlier. First, the reacting forces at the hinged end and at the support strut location, $R_{\text{end}}$ and $R_{ss}$ respectively as shown in Fig. 7, must be determined.
Fig. 7 Free body diagram of a simple deployed rib and applied aerodynamic loads.

The total length of the deployed rib is $L$, and the distance of the support strut location from the hinged end is $a$. The reacting forces are then found from simple equilibrium, and the bending moment, $M$, at each node, $x_i$, is then determined using the equation

$$M(x_i) = R_{\text{end}} x_i + R_{\text{ss}} (x_i - a) - \sum_{n=1}^{n=x_i} F_n (x_i - x_n)$$

in which $x$ is the distance along the beam from the hinged end, and where the angled brackets denote Macauley functions. The bending moment is inserted into the Euler-Bernoulli beam equation so that

$$M(x_i) = -EI(x_i) \frac{d^2 \delta}{dx^2}$$

where $\delta$ is the deflection, $E$ is Young’s Modulus, and $I$ is the second moment of area (which can vary with $x$ e.g. in the case of a tapered beam).

This final equation must then be integrated twice to get the deflection at each node point. An analytical solution for the deflection is not practicable due to the varying second moment of area and resulting complexity, so the trapezoidal numerical method has been applied to perform a stepwise integration along the length of the rib, assuming zero deflection at the hinged end and at the support strut. The resulting rib deflections are applied to the mesh which then updates all co-ordinates. The new deformed mesh is then passed back to the 6DOF code, and the aerodynamic coefficients recalculated for the new conditions and geometry, repeating the process for each timestep.

It should be noted that although the initial geometry of the decelerator is axisymmetric, the deformation of each rib is calculated and passed back to the 6DOF code separately so the subsequent deformations are not necessarily
axisymmetric. The overall centre of gravity is updated at each timestep based on the input heatshield mass properties and the current deformed shape.

C. Deflection Correlation

A correlation has been performed of the deflections obtained with the aero-structural code against linear deflection results from the Abaqus v6.14 commercial finite element solver [27], to ensure validity. The Human Cargo reference mission has been selected as the correlation case, with a 20 tonne surface payload requiring a 16 m diameter entry vehicle as in Fig. 5. With a 4.5 m fixed nose-cone (to ensure fit within existing launch vehicles) and a 70° sphere-cone shape, the length of the deployable rib under consideration is 6.12 m. A beam representing the deployable rib was set up in Abaqus with matching dimensions, cross-section shape and material properties, and was pinned at one end and at the location of the support strut. The beam is modeled using three-dimensional two-node Euler-Bernoulli beam elements.

Cases were run in the Matlab entry trajectory simulator assuming the deployable rib had 5, 15, 30, 60 and 120 mesh elements. The summed aerodynamic loads at an example fixed trajectory point were extracted from the simulator for a single rib for each of these cases, and these loads were applied at the matching load points to the Abaqus beam element. The mesh resolution within Abaqus was set to 0.001 of the rib length. Note that the number of mesh elements in Matlab corresponds to the number of load points in Abaqus.

As seen in Fig. 8, both the Matlab and Abaqus results show greater deflection for a lower number of load points/mesh elements. The maximum deflection decreases as the number of mesh elements/load points increases, and the solution converges to 0.0636 m. This was obtained by plotting the maximum deflection against the inverse of the stiffness, and locating the point where the resulting straight line meets the y-axis.
Fig. 8 Maximum deflection at the rib tip for the Matlab simulator and the Abaqus case against the converged solution.

The Matlab simulator results can then be compared against the converged solution, as shown in Table 2.

<table>
<thead>
<tr>
<th>Mesh Density</th>
<th>Matlab Deflection [m]</th>
<th>Difference from Converged Solution</th>
</tr>
</thead>
<tbody>
<tr>
<td>5</td>
<td>0.06998</td>
<td>10.02%</td>
</tr>
<tr>
<td>15</td>
<td>0.06682</td>
<td>5.06%</td>
</tr>
<tr>
<td>30</td>
<td>0.06556</td>
<td>3.08%</td>
</tr>
<tr>
<td>60</td>
<td>0.06458</td>
<td>1.54%</td>
</tr>
<tr>
<td>120</td>
<td>0.06406</td>
<td>0.72%</td>
</tr>
</tbody>
</table>

The aero-structural module of the Matlab simulator is therefore shown to give deflections with an accuracy of the order of 5% when a mesh density of 15 or greater is used along the length of the deployable rib. An accuracy of 5% is considered acceptable for this application, as it offers a compromise between accuracy and reasonable runtime. The deflection results in Table 3 confirm that the majority of deflections are within the appropriate range for linear analysis,
but it is important to recognize that at some higher deflections there may be nonlinear effects that are not accounted for in this analysis.

V. Simulator Results and Discussion

The 6DOF entry trajectory simulator with coupled aero-structural modeling enables the investigation of the effect of deployable rib flexibility on the deployable aero-decelerator and entry trajectory, with a focus on any potential beneficial effects to important quantities such as peak heat flux and maximum acceleration.

A. Effect of Stiffness Variation of Non-Tapered Deployable Ribs

Initially, non-tapered deployable ribs were investigated by varying the bending stiffness, $E I$, of the ribs, in order to explore the effect of stiffness in isolation. The same reference Human Cargo mission was assumed as for the aero-structural correlation, assuming a constant mass 50 tonne entry vehicle. The maximum deflection at the tip encountered during the complete entry trajectory for the range of bending stiffnesses explored are listed in Table 3.

<table>
<thead>
<tr>
<th>Case</th>
<th>Rib Stiffness, $EI$ [Nm²]</th>
<th>Maximum Rib Deflection [m]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Rigid</td>
<td>0</td>
</tr>
<tr>
<td>2</td>
<td>$84 \times 10^6$</td>
<td>0.08</td>
</tr>
<tr>
<td>3</td>
<td>$50 \times 10^6$</td>
<td>0.14</td>
</tr>
<tr>
<td>4</td>
<td>$42 \times 10^6$</td>
<td>0.17</td>
</tr>
<tr>
<td>5</td>
<td>$34 \times 10^6$</td>
<td>0.21</td>
</tr>
<tr>
<td>6</td>
<td>$16 \times 10^6$</td>
<td>0.41</td>
</tr>
<tr>
<td>7</td>
<td>$7 \times 10^6$</td>
<td>0.92</td>
</tr>
<tr>
<td>8</td>
<td>$4 \times 10^6$</td>
<td>1.32</td>
</tr>
</tbody>
</table>

The resulting trajectories and the variation of the drag coefficient for a selection of these cases are shown in Fig. 9 and 10, respectively.
Fig. 9 Velocity-Altitude trajectory plots showing the impact of varying rib stiffness. The undeformed and deformed entry vehicle geometries for case 8 are also shown.

Fig. 9 shows that the effect on the overall entry trajectory of increased rib flexibility is minor, with the majority of trajectories clustered close to the rigid trajectory. One example is Case 2, the near rigid case with stiffness of $84 \times 10^6$ Nm$^2$. An example rib that would provide this level of stiffness is of titanium alloy Ti-6AL-4V with an I-beam cross-section of width 25 cm, depth 50 cm, and flange thickness of 2 cm. The exceptions are the lowest stiffnesses considered ($7 \times 10^6$ Nm$^2$ and below), which do lead to significantly higher velocities due primarily to the lower drag coefficient, as seen in Fig. 10, as well as a small reduction due to change in cross-section area. An example rib with a stiffness of $4 \times 10^6$ Nm$^2$ is a titanium alloy Ti-6Al-4V rib with an I-beam cross-section of width 10 cm, depth 20 cm, and flange thickness of 2 cm. This lowest rib stiffness leads to a velocity at 10 km altitude that is approximately 25% higher than a rigid or close to rigid aero-decelerator of the same diameter. Fig. 9 also illustrates the large deformations to the entry vehicle shape for this maximum deflection case.
Fig. 10  Drag coefficient variation during the entry for varying rib stiffness

Fig. 10 shows that the rib deformation has a clear effect on the drag coefficient of the entry vehicle at altitudes below 70 km, where the atmospheric density starts to become significant. Note that the use of the Modified Newtonian method is applicable to hypersonic flow, and becomes less accurate as the Mach number falls below approximately Mach 5. Thus, drag coefficient results below approximately 15 km altitude should be considered with some caution. The variation of other parameters are explored in Fig. 11 to 12, including deflection error bars of 5% due to the rib mesh density used.
As expected, as the deployable rib stiffness decreases (i.e. as the compliance, $1/EI$, increases), the maximum deflection increases, thus reducing the drag coefficient. The lowest stiffness of $4 \times 10^6$ Nm$^2$ gives a maximum deflection of 1.3 m at the rib tip, which results in a drag coefficient reduction from 1.5 to 1.1, or 27%. The surface angle changed by up to 15°, which contributes up to 90% of this significant drag reduction. The remainder of the drag reduction comes from the slight reduction in reference surface area.

The reduced drag of the entry vehicle due to rib deflection leads to less effective deceleration, as seen in the trajectory plots of Fig. 9. This results in increased velocities at lower altitudes, and thus an increase in peak heat flux by up to 3 W/cm$^2$ or 7%. It also leads to higher velocities at the point at which supersonic retropropulsion initiation or parachute deployment should occur, thus requiring greater deceleration capability from the descent or landing
technologies. The peak g-load experienced by the entry vehicle decreases as rib stiffness decreases by up to 1 g or 13%, due to the reduced deceleration experienced by the deformed lower drag coefficient entry vehicles.

The effect of reduced stiffness on the attitude stability was also investigated. Fig. 13 shows the variation of the angle of attack for the same set of varying rib stiffness cases. The results show that reducing stiffness – i.e. increasing flexibility – of the ribs leads to more effective damping of the attitude oscillations. This is expected, as entry vehicle shapes with greater sphere-cone angles, i.e. 45° or 60°, have better stability than the conventional Mars sphere-cone angle of 70° [28]. Thus, as the deployable ribs are pushed back to greater angles, a new and more stable shape results. This effect could prove advantageous near the end of the entry phase when stability becomes more of an issue.

![Figure 13: Angle of attack stability with varying stiffness.](image)

The damping effect begins to be significant at about 85 s in the entry trajectory. At this point, the altitude is approximately 37 km, the Mach number is in the range of 25, and the dynamic pressure is around 8 kPa.

The results thus far indicate that flexibility can have some detrimental effects, including increased peak heat flux, that are undesirable – with the exception of a damping effect on attitude oscillations. However benefits from possible mass savings resulting from flexible ribs have not yet been captured and quantified.
B. Effect of Entry Vehicle Diameter Variation with Tapered Deployable Ribs

Varying the diameter of an entry vehicle has a significant effect on the entry trajectory, as seen in Fig. 5. In addition, a tapered rib presents significant advantages, allowing mass savings and some flexibility while retaining sufficient strength at the location of maximum principal stress near the support strut. For these reasons, in this section we investigate the potential impact on performance of reallocating mass savings gained by using tapered rib designs towards an increase in the entry vehicle diameter – thus decreasing the ballistic coefficient.

The baseline configuration is the Human Cargo reference mission, with a 16 m diameter and 12 deployable ribs of length 6.12 m, with a single support strut for each rib. Assuming initially a non-tapered Ti-6Al-4V rib with an I-beam cross-section of width 20 cm, depth 40 cm, and flange thickness of 2 cm, the total mass of each rib is 412 kg. This results in a mass of 4,944 kg for 12 ribs. A total entry mass of 50,000 kg for the entry vehicle was maintained, and assumptions about the flexible TPS mass were made to ensure the decreased rib mass and increased TPS masses were balanced. A TPS areal mass of 4 kg/m² was assumed based on datasheets for a 7-layer 3D woven carbon material of IM7-GP-12K fibres from supplier Sigmatex [29]. The additional TPS mass was calculated for a set of entry vehicle diameters, and then the taper ratio of the ribs was optimized to give a total rib mass saving that matched the TPS mass increase for each diameter. Diameters larger than 21 m were omitted as the tapered rib mass saving becomes less than the TPS mass increase at that point. The results are shown in Fig. 14 and Table 4.
With decreasing stiffness, both the peak g-load and peak heat flux loci shown in Fig. 14 move to occur at a higher altitude and at lower velocities. The diameter variation has a clear effect on the entry trajectory, with the largest diameters having more effective decelerations at higher altitudes and incurring lower peak heat fluxes (30 W/cm$^2$ compared to 42 W/cm$^2$). These relatively small differences could have important implications that enable the use of a different class of flexible TPS material, such as Nextel BF-20, which has a peak heat flux limit of 30 W/cm$^2$ [30]. The peak g-load, however, does not exhibit a clear trend. This suggests that varying flexibility of the deployable aero-decelerator is not a good tool for controlling maximum acceleration.

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**Fig. 14** Velocity-Altitude plots of varying taper ratio and diameter, with peak-load and heat flux loci.

**Table 4 Entry trajectory parameters with varying diameter.**

<table>
<thead>
<tr>
<th>Diameter [m]</th>
<th>16</th>
<th>17</th>
<th>19</th>
<th>21</th>
</tr>
</thead>
<tbody>
<tr>
<td>Taper Ratio</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Max Deflection [m]</td>
<td>0.168</td>
<td>0.305</td>
<td>0.729</td>
<td>1.640</td>
</tr>
<tr>
<td>Peak Dynamic Pressure [Pa]</td>
<td>12,125</td>
<td>11,217</td>
<td>9,895</td>
<td>9,453</td>
</tr>
<tr>
<td>Peak Heat Flux [W/cm$^2$]</td>
<td>41.64</td>
<td>38.5</td>
<td>33.51</td>
<td>30.28</td>
</tr>
<tr>
<td>Peak G-Load [g]</td>
<td>7.50</td>
<td>7.61</td>
<td>7.72</td>
<td>7.50</td>
</tr>
<tr>
<td>Ballistic Coefficient at EIP [kg/m$^2$]</td>
<td>161.0</td>
<td>147.6</td>
<td>114.2</td>
<td>93.4</td>
</tr>
</tbody>
</table>

With decreasing stiffness, both the peak g-load and peak heat flux loci shown in Fig. 14 move to occur at a higher altitude and at lower velocities. The diameter variation has a clear effect on the entry trajectory, with the largest diameters having more effective decelerations at higher altitudes and incurring lower peak heat fluxes (30 W/cm$^2$ compared to 42 W/cm$^2$). These relatively small differences could have important implications that enable the use of a different class of flexible TPS material, such as Nextel BF-20, which has a peak heat flux limit of 30 W/cm$^2$ [30]. The peak g-load, however, does not exhibit a clear trend. This suggests that varying flexibility of the deployable aero-decelerator is not a good tool for controlling maximum acceleration.
VI. Conclusions

A 6 DOF aero-structural entry trajectory simulator has been developed, which enables assessment of next generation entry vehicle concepts. The results have been validated against NASA Spirit and Phoenix flight data, and correlated against an industry tool. The simulator can model the aerodynamics of various deployable aero-decelerator designs for each timestep of the trajectory, rather than relying on existing aerodynamic databases for specific shapes, which represents an improvement over current industry tools. Aero-structural modelling has been coupled into the 6DOF simulator, to model the deployed rib deflections and update the mesh at each timestep.

The coupled aero-structural modelling demonstrates that some flexibility in deployable aero-decelerators is beneficial if the resulting mass savings can be reallocated towards increasing the entry vehicle diameter. This reduces peak heat flux significantly. In addition, more flexible ribs damp out attitude oscillations more effectively, which indicates enhanced stability. The coupled tool can be used to optimize the rib design for any particular mission, balancing flexibility and mass saving through varying dimensions, cross-sections and material properties of the ribs.

This tool allows significantly improved mass estimates for deployable aero-decelerators, to contribute to the selection of the preferred entry technology for the next major Mars missions. Although we have performed a simplified analysis thus far, ongoing work will enable more detailed investigation, for example the calculation of the maximum principal stress to ensure the selected material yield strength is not exceeded, and the location of the support strut to be varied and optimized as required. Future work at Imperial College London is planned to focus on validation of the aero-structural effects for deployable elements via experimental test campaigns.

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