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<td>Greg Swanson</td>
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Mars Sample Return Earth Entry Vehicle Design:
Can the Reliability Requirement be met by Emerging New Technologies?

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Planetary missions to Mars and Other Icy worlds such as Enceladus, Europa and Titan have the burden of not contaminating the planetary environment in the course of exploration (1). Missions that collect and bring back samples from these destinations have added the requirement of accidental release of the samples into earth atmosphere. Planetary protection from forward and backward contaminations impose stringent reliability requirements on sample return missions, especially on the design of the earth entry system and in the engineering of it. This proposed presentation will review past efforts and assess what emerging technologies could be used to address the reliability requirements.

Between 1997 and 2002, NASA working in partnership with CNES performed an end-to-end mission design study including earth entry phase. Probabilistic risk analysis was performed to lay the foundation for establishing a basis for the design elements to meet the overall planetary protection requirement (Ref 2, 3). Efforts such as chute-less entry capsule that are designed to withstand direct impact were focused on improving the reliability of the overall mission by making the entry segment entirely passive (4).

“ As has been indicated by the National Academy of Sciences for scientific reasons, the possibility that the samples might be harmful, therefore, all the elements of the mission architecture must assure containment of the samples from the time of their acquisition through final disposition. The probability of containment not being assured (CNA) is 1.0E-06,” from ref (2). The study performed in 2002 and reported out in 2004 looked at direct entry using an earth entry vehicle (EEV) and a Shuttle transfer system where the sample will be brought down to earth using the Shuttle. Since the Space Shuttle program no longer exists, the feasible options are direct earth entry vehicle or using the Orion capsule instead of the Space Shuttle orbiter. For this proposed presentation, we will focus on the former and specifically the EEV.

Table 1 and Figure 1 (included at the end) are taken from Ref. 2 and they illustrate why the EDL Segment needs to be robust. Earth targeted Cruise mission phase and earth targeting are two important mission segments that have the two largest contributions (54% and 15%), and there are many ways to achieve the required reliability. The next biggest contributor is the entry segment (7.6%) and the remaining EDL segments are not far behind, including the post impact or recovery segment. The focus of this presentation is to both assess the state of the art entry system and erroneous assumptions made in the past studies, and address options that may become available to design and demonstrate the entry system that could possibly meet the stringent requirement of CNA not higher than 2.6E-07 for the entry system.

In the (1997 – 2002) study, the heritage Carbon Phenolic (CP) material, utilized on Pioneer Venus and Galileo, was baselined as the ablative thermal protection material. The EEV was a direct impact, parachute-less design as shown in Figure 2.

The heat-shield material that was baselined for the MSR-EEV was heritage Carbon-Phenolic (HCP). The heuristic argument that led to the selection of the heat-shield material HCP are problematic due to the assumptions made and also the
fact that the manufacturing of HCP has atrophied. The argument for baselining HCP went something like this: “Given that the Venus and Jupiter entry environments are much more severe than the Sample Return Environments and the large amount of flight data for non-civilian uses it has been assumed that one can demonstrate that a CP system can meet the reliability requirements.”

However HCP has its own challenges and risks associated with it. A HCP heat-shield consists of two versions of CP, one of which has not been manufactured for entry missions since Galileo and the flight heritage of that material is much more limited to just two missions (Galileo and Pioneer-Venus) or 5 probes. Given that there are two versions of HCP to make one heat-shield, it requires the heatshield to have a seam between the two components which introduces additional verification challenges. HCP is also susceptible to inherent failure modes due to its 2D laminated structure.

Recent SMD and STMD funded activities have looked at the use of 3D Weaving to develop TPS materials with material architectures that are inherently more robust than 2D systems and also they are not simply one material but a family of materials. In 2012 and 2013, under the 3-D Woven TPS project, a family of materials, single and multi-layer materials were woven, some resin infused and others were not, and these materials were arc jet tested to understand the performance limit and failure modes. The family of ablative TPS materials manufactured and tested are shown in Figure 3 and also shown on this figure are three heritage materials, namely Carbon-Phenolic, Avcoat and PICA.

Recent work under the Heat-shield for Extreme Entry Environment Technology (HEEET) project have demonstrated acreage materials that are robust to entry environments of 5000+ W/cm² and above 5 atm. pressure, well in excess of that for sample return missions which are around 1500 W/cm² (margined) of peak heat-flux and under 0.4 atm of peak pressure during earth entry. In Figure 4, the thermal testing performed in comparison to a range of Venus and Saturn missions are shown. It is very exciting to note that the 3-D Woven acreage material selected for Venus and Saturn missions under the HEEET project has not shown any type of failure at any of these test points.

Knowledge gained during the HEEET project has provided insights into the complexities associated with weaving, resin infusion and other steps involved in making a robust heat-shield.

This proposed presentation will discuss in detail the challenges facing the EDL community in designing an Aero-shell that can meet the planetary protection requirements, and the opportunities the 3D woven heat shield offers in terms of improved heat-shield and back-shell TPS compared to a traditional CP heatshield, and some challenges that would need to be overcome to develop such a system.

References:

Table E-1. EEV Mission Phase Contribution to CAN (From Ref 2)

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<tr>
<th>Phase</th>
<th>5%</th>
<th>50%</th>
<th>mean</th>
<th>phase cont.</th>
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<tr>
<td>Break-the-Chain (sample isolation)</td>
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<td>/MAV Launch</td>
<td>1.1E-10</td>
<td>5.4E-10</td>
<td>1.1E-09</td>
<td>0.1%</td>
<td>2.8E-09</td>
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<td>Mars Orbit Rendezvous</td>
<td>6.7E-11</td>
<td>1.2E-09</td>
<td>6.9E-09</td>
<td>0.5%</td>
<td>1.9E-08</td>
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<td>Mars/Earth Transit</td>
<td>2.6E-10</td>
<td>3.3E-09</td>
<td>3.1E-08</td>
<td>2.4%</td>
<td>1.1E-07</td>
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<td>Earth Targeting</td>
<td>1.6E-07</td>
<td>5.0E-07</td>
<td>7.2E-07</td>
<td>54.2%</td>
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<td>Spin Eject</td>
<td>7.4E-11</td>
<td>3.5E-10</td>
<td>6.2E-10</td>
<td>0.0%</td>
<td>2.1E-09</td>
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<td>Earth Targeted Cruise</td>
<td>7.5E-09</td>
<td>8.9E-08</td>
<td>2.0E-07</td>
<td>15.1%</td>
<td>1.3E-06</td>
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<td>Entry (Thermal)</td>
<td>2.8E-09</td>
<td>2.5E-08</td>
<td>1.0E-07</td>
<td>7.6%</td>
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<td>Descent (Structural)</td>
<td>1.0E-08</td>
<td>5.5E-08</td>
<td>8.8E-08</td>
<td>6.7%</td>
<td>3.0E-07</td>
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<td>Impact</td>
<td>7.3E-09</td>
<td>3.5E-08</td>
<td>7.5E-08</td>
<td>5.7%</td>
<td>2.3E-07</td>
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<td>After Impact</td>
<td>2.5E-09</td>
<td>2.5E-08</td>
<td>1.0E-07</td>
<td>7.6%</td>
<td>3.1E-07</td>
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<td>Residual (unclassified)</td>
<td>2.9E-10</td>
<td>7.1E-10</td>
<td>4.4E-10</td>
<td>0.0%</td>
<td>2.1E-09</td>
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<tr>
<td><strong>Total</strong></td>
<td><strong>4.6E-07</strong></td>
<td><strong>1.0E-06</strong></td>
<td><strong>1.3E-06</strong></td>
<td></td>
<td><strong>2.7E-06</strong></td>
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Figure 1. EEV MSR PRA Scenario Results by Phase (Taken from Ref 2)
Figure 2. Mars Sample Return Earth Entry Vehicle

Figure 3. 3-dimensionally woven, single and multi-layer, ablative TPS materials manufactured and tested in (2012-2013). Density of the family of materials are compared with heritage ablative TPS (HCP, AVCOAT, PICA).
Figure 4. Thermal test conditions the HEEET material has been tested at in comparison to the mission profile in terms of heat-flux and pressure. One test was conducted at 2000 W/cm² and at 14 atm. stagnation pressure. Another test was conducted at 8000 W/cm² at the LHEML laser test facility with no flow. The HEEET acreage material not only survived these extreme conditions, post-test inspection showed no failure modes. In comparison, heritage CP showed anticipated failures such as spallation and ply separation.
EDL MISSION ANALYSIS AND DESIGN FOR HIGH ENERGY SAMPLE RETURN MISSIONS

Gabriele De Zaiacomo\textsuperscript{1}, Cristina Parigini\textsuperscript{1}, Davide Bonetti\textsuperscript{1}, Simone Centuori\textsuperscript{1}, Juan Luis Cano\textsuperscript{1} \textsuperscript{1}DEIMOS Space S.L.U., Tres Cantos, Madrid, Spain (gabriele.dezaiacomo@deimos-space.com)

In the most recent years, space exploration missions from worldwide agencies dramatically increased, or are on route to, the knowledge of the solar system, collecting information about comets, asteroids, proto-planets in the asteroid belt, and dwarf planets in the Kuiper belt. Nevertheless, only in a few cases these missions were able to successfully bring back to Earth samples of solar wind particles (Genesis, NASA, partially successful), comet and cosmic dust particles (Stardust, NASA), and asteroid material (Hayabusa, JAXA). A sample return mission represents an extremely challenging task due to the increase in complexity of the overall mission and system design, and, in particular, to the need of assuring a safe return to Earth of a sample container, coming from an interplanetary orbit at very high energy.

The Atmospheric Flight Competence Centre (AFCC) of DEIMOS Space S.L.U. has been working for the last 13 years in the area of Mission Analysis for atmospheric flight, leading and supporting the Mission Engineering and Flight Mechanics for several studies and projects, covering different scenarios, from hypersonic and space transportation to exploration, with a wide range of vehicles (from capsules to space planes), environments (Earth, Mars, Titan), and flight phases (ascent, coasting, and entry, descent, and landing - EDL) \textsuperscript{1}. The methodologies and atmospheric flight design tools developed within the AFCC, flight validated by the IXV mission in 2015 \textsuperscript{2}, are expected to be further validated by the ExoMars16 mission, scheduled to land on Mars on October 19\textsuperscript{th}, 2016.

In this paper a design approach developed for the Mission Analysis of the EDL phase of return missions is presented. The objective was to define and implement a method to allow the robust design of the EDL phase, in case of high energy entry scenarios, assuring that the entry vehicle is able to fulfill the mission objectives and at the same time assuring its integrity by respecting the aerothermodynamic constraints. To achieve this goal, three building blocks are needed: the entry vehicle (aerodynamics, Mass, Centre of Gravity and Inertia, control surfaces – if any), the environment (density, temperature, winds), the initial conditions (time, position, velocity, attitude) and the events (triggering of new phases and GNC modes).

The process starts with an analysis of the aerodynamics of the re-entry vehicle and an assessment of its flying qualities (FQ: trim, stability and controllability indicators). Through a parametric analysis of the FQ, coupled with thermo-mechanical constraints, an optimization of the Centre of Gravity location is performed. Based on the initial conditions (usually part of an end-to-end trajectory optimization and landing site targeting process) an analysis of the environment is performed for the expected range of entry conditions. The above steps are necessary to set up a trajectory simulation of the entry vehicle.

With proper worst cases combinations of the variability expected along the trajectory it is possible to perform an Entry Corridor analysis, Local or Global, depending on the degrees of freedom considered, to support system trade-off and design. Based on these results, sizing trajectories are identified, usually on the steep or shallow entry corridor bounds, to derive specifications for relevant subsystems (e.g. for the detailed thermal characterisation and Thermal Protection System design that is critical for high energy return missions). Finally, a reference trajectory is designed within the entry corridor, and the mission performance and margins are assessed through Monte Carlo analysis.

Examples of application of this design approach are presented in this paper focusing on two sample return missions currently under development by ESA and the European industry, aiming to return samples of material from the Moon pole (Lunar Polar Sample Return) and from the Mars moon Phobos (Phobos Sample Return). In both projects DEIMOS Space S.L.U. is responsible of the EDL mission analysis bringing in its expertise and the design approach that is presented in this paper.

References:

\textsuperscript{1} Bonetti D. et al (2016) “PETbox: Flight Qualified Tools for Atmospheric Flight”, 6\textsuperscript{th} ICATT.


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Introduction: Sample return missions offer the opportunity for a far more thorough investigation of material from the target than can be made by any in-situ probe. Paramount in the design of the sample handling and storage systems for such missions is the maintenance of the sample itself. The ideal situation is to ensure that the sample is brought back unmodified from the point of collection. Unfortunately, this is often not feasible, because the conditions at the point of collection are so different from the conditions during the return to Earth and on Earth. Thus, it will usually be the case that we must identify important scientific goals for the sample collection, and ensure that the conditions of the return do not modify the sample in a way that destroys the scientific information of interest.

In this study we focused on the last stage of a sample return mission, the re-entry through the Earth’s atmosphere. We used, as a re-entry vehicle, the Multi-Mission Earth Entry Vehicle (MMEEV) design [1-3], because it is a promising design for multiple future missions, and because its very unconventional terminal-velocity impact landing may have the potential to modify samples in ways that a parachute descent would not. We focused on two main scientific parameters of interest: the chemistry and stratigraphy of the sample. These two parameters are of key scientific interest for a range of targets, including comets, asteroids, and planetary surfaces. It is important to maintain both in their initial form to extract maximum scientific value from the sample return mission. Our models and experiments demonstrate that the re-entry could have a small, but detectable, effect on both the chemistry and the stratigraphy of the sample. The magnitude and nature of these changes is such that the impact on the scientific goals of the mission is likely to be small.

Modeling and Experiments: In order to determine the conditions at the sample during re-entry, we used a simulated re-entry initiated with a high-speed atmospheric entry of 14 km/sec and flight path angle of -10°, as is likely to be required for a comet surface sample return mission. These conditions resulted in roughly 90 seconds of deceleration in the range of 80g, followed by roughly 10 minutes of terminal-velocity descent, and ending with an impact with the ground. The simulations showed that the vibration at the interface between the MMEEV shock-absorbing material and the metal sample canister is relatively mild, at ~0.05g for the duration of the re-entry. Temperature at this interface was also mild, never rising above 50 °C.

In order to gauge the effects of impact with the ground, experimental data from NASA drop-tests of the Mars sample return mission EV [4,5], on which the MMEEV design is based, were used to determine the temporal profile of the acceleration at the sample chamber interface. Thus, the effects of both the initial impact and of the post-impact vibrations of the vehicle were tested.

A variety of sample simulants were used in centrifugation, vibration, and drop-shock tests designed to match the conditions derived as described above. The primary simulant was the mineral olivine ground in a ball mill to a variety of sizes, from gravel to powder. Vermiculite, topsoil, coarse sand, and clay were also used. Finally, to elucidate the effect of re-entry on fragile molecules that could be present in the samples, olivine sand (75 µm – 500 µm grain size) was coated with amino acids by soaking the olivine sand in an amino acid solution and evaporating the liquid. Also, molecular dynamics simulations were conducted to show the effect on such fragile molecules of impacting hard surfaces under the expected impact conditions.

To simulate the effect of re-entry on samples made of these materials, they were stacked in a variety of configurations in transparent glass or plastic cylinders. They were subjected to centrifugation at 80g, and vibration at up to 0.1 g amplitude that matched the frequency profile predicted by the simulations. To simulate the effect of impact with the ground, a drop-shock machine was used and the conditions tuned until the acceleration at the sample chamber interface was of the appropriate amplitude (1500g and 2000g), with a temporal profile similar to that of the Mars sample return EV as described above. The experiment assumed that the MMEEV and the sample inside do not tumble after impact with the ground, consistent with the observations in the Mars EV drop tests. A 10,000 frame per second camera was used to capture the effects of the drop-shock tests on the sample.

Results: In all cases, the effect of in-flight vibration on the stratigraphy was negligible. We surmise this is due to the fact that the 90 seconds of 80g deceleration that preceded the vibration packed the sample in such a way that it was not susceptible to the relatively mild vibrations. However, the simulated impact with
the ground did have a measurable effect on the stratigraphy, but a relatively small one. After the impact, stratigraphic interfaces within the sample were typically spread by a few % of the total sample height. Analysis of the amino acid samples showed that the drop-shock tests caused a slight reduction in the total amount of amino acid present. No new amino acids were formed and all amino acids were depleted to the same degree.

**Conclusions:** These tests demonstrate that the MMEEV is a promising return vehicle for a number of potential sample return missions, including from comets, asteroids, and planets. We chose a particularly high-speed re-entry to simulate for this work, and thus the results are likely to be applicable to a variety of missions that have milder descents to Earth.

VERSATILE LANDING BY CRUSHABLE STRUCTURE FOR SMALL SEMI-HARD IMPACT PROBE CONCEPT. Tetsuya Yamada1, Hideyuki Tanno2, Koichi Kitazono1, Tatsuaki Hashimoto2, 1Japan Aerospace Exploration Agency, (3-1-1 Yoshinodai, Chuo-ku, Sagamihara, Kanagawa 229-5210, JAPAN, yamada.tetsuya@jaxa.jp) 2Tokyo Metropolitan University, 3 Japan Aerospace Exploration Agency.

Introduction: Recently, target bodies of the solar exploration have extended to distant heavenly bodies beyond the Mars, and even to more distant ones. Because the overall mission duration tends to be longer and longer in such high-energy missions, reliability requirements for the subsystems become severer. In case of sample return missions from the distant asteroids, the earth entry velocity of the sample return capsule is often estimated to exceed 14 km/s, or reach to 15 km/s, and the overall mission duration also exceeds 10 years, or approaches to 20 years in some cases such as sample return from distant comets. The high-speed compact entry probe is one of the key technologies to accomplish such challenging missions. Two of the important issues concerned with the high-speed entry capsules are severe thermal protection in the high-speed entry and secure functional reliability of the overall slow-descending sub-systems at the very final phase of the mission. The present study shows state-of-the-art technology development of the crushable structure together with a design example for the small lunar semi-hard impact probe mission.

Crushable Structure and Ground Tests: The crushable structure realizes not only chuteless landing on the planets with atmosphere but also airless bodies by absorbing landing shock energy and protects the instrument modules against the landing shock within a prescribed deceleration level. These versatile landing is advantageous for secure decent system in that ignition electronics, parachute-triggering sensors, and triggering-control logic circuits etc. are no longer required. Mechanical characteristics, mainly stress vs. strain characteristics, have been investigated for various material such as porous metals and carbon structures. It is crucial to control the mechanical characteristics parametrically according to the mission requirement (allowable maximum deceleration, instrument weight, etc) material properties are now under parametrical investigation. In order to verify validity of the analysis and extracting other issues concerned with crushable material, the landing impact tests were carried out by using the ballistic range in JAXA Kakuda Space Center by impacting a small test model with onboard measurement systems onto the target. (Fig. 2). Measured impact data are compared to the analytical prediction and discussed.

References: Use the brief numbered style common in many abstracts, e.g., [1], [2], etc. References should then appear in numerical order in the reference list, and should use the following abbreviated style:


Fig. 1 Artist’s impression of a small low-ballistic coefficient capsule with instrument module surrounded by the crushable material.

Fig. 2 The crushable-ball for the ballistic range impact test: An example of the crushable material is mounted in the center of the forebody shell (upper left in the photo). The data-logger is on the bottom of the aftbody shell (upper right).
Autonomous High Altitude Package Return System. C. M. Bailey\textsuperscript{1} and J. A. Sutton\textsuperscript{2}, \textsuperscript{1}University of Idaho, Mechanical Engineering, email: bail6232@vandals.uidaho.edu, \textsuperscript{2}University of Idaho, Mechanical Engineering, email: sutt9939@vandals.uidaho.edu.

Introduction: Students at the University of Idaho have taken on the problem of high altitude package return. The purpose of this project is to create a reliable guided parafoil made with inexpensive, off the shelf parts. Previously the Idaho RISE high altitude ballooning program has allowed for an unguided descent. Weather prediction software has been used to generally predict the landing site. To better this system a guided parafoil will direct the package safely to a prespecified location. The techniques that are used in this project could be also be used to help guide small payloads to the surfaces of extra-terrestrial bodies.

Equipment: The project will be constructed using three DC motors connected to high ratio worm gear boxes to allow for an auto locking setup to minimize power consumption. The output shafts of the gear boxes will turn spools attached to control lines. Two independent motors will be used to control the left and right trim, while the third motor will control the angle of attack. The motors will be controlled using two md08a motor drivers which will be controlled by a 5V Arduino Pro mini microcontroller. The angle of rotation of each of the gear boxes will be monitored by 10 turn potentiometers. The position will be measured using a uBLOX MAX-M8Q GPS unit through a serial connection. The dynamics of the capsule will be measured using GY-85 nine axis IMU. The entire apparatus will be housed within the existing capsule as well as all communication equipment, and other engineering projects and science experiments. The parafoil, to ensure inflation, will contain sealed Mylar packets that are filled air at atmospheric pressure. In addition to being a guided parafoil the duty of cutting down of the package from the high altitude balloon that the package will be attached to, is also being taken over by this apparatus. This is done using a custom designed bracket and a Firgelli micro linear actuator.

Controller: There will be four stages of flight programmed into the flight controller. The first will be to disconnect from the high altitude balloon that it will be attached to. A multiple redundancy coding system will ensure that the parachute and package will disconnect from the balloon. There will also be a mechanical backup to allow the package to return to the ground safely in the systems fails. The second stage of flight will be to descend to a layer of atmosphere that the parafoil will be more easily controlled. This will be done by putting the package into a spiral to increase downward velocity while minimizing drift. The third stage of flight will be to seek and revolve around the vertical axis of the destination as the package starts its decent. The radius from the vertical axis will grow as a function of elevation to widen as it reaches the ground in order to set up for the fourth stage of flight which will be landing. Once the package reaches a set altitude based on decent rate. It will break its tracking around the axis of descent and start towards its final landing zone. The package will then head on a linear path towards it final destination. Once it reaches its destination the motors will fully pull on both trim lines causing it to stall and fall the short distance to earth.

Discussion: The proposed system will be an active feedback control loop that will allow the microcontroller to continually monitor its dynamics and position and adjust its response accordingly. The parameters for the loop will change based on elevation of the package and the changing environment that it will be traveling through. The controller will also be able to redefine environment variables based on elevation to adjust its dynamic responses. These techniques could be implemented on the atmospheric stages of planetary probes to help them land safely on the surface.
Low Cost Innovative Atmospheric Entry Probes Combining CubeSat and HIAD Technologies. S. J. Hughes¹, N. A. Miller², G. A. Bailet³, and Dr. F. McNeil Cheatwood⁴,¹HIAD-2 Flight Aeroshell Lead, Engineering Directorate (ED)/Mechanical Systems Branch, NASA Langley Research Center/MS432 Stephen.J.Hughes@nasa.gov, ²Aerospace Engineer, Engineering Directorate (ED)/Mechanical Systems Branch, NASA Langley Research Center/MS432 Nathanael.A.Miller@nasa.gov, ³PhD Candidate, Ecole Centrale Paris - Laboratoire EM2C - CNRS UPR 288, France BailetGilles@gmail.com, ⁴HIAD-2 Principle Investigator (PI), Engineering Directorate (ED)/Atmospheric Flight & Entry Systems Branch, NASA Langley Research Center/MS489 F.M.Cheatwood@nasa.gov

Introduction: Recent developments in Hypersonic Inflatable Aerodynamic Decelerators (HIAD) and CubeSat technologies make possible a variety of missions capable of meeting important science and technology needs. Small spacecraft require all of the functionality of larger spacecraft, and atmospheric entry is an enhancement that would expand the mission set for which the CubeSat might be applicable. HIAD offers the opportunity to stow the heatshield required to protect a spacecraft from atmospheric entry in a compact volume and then deploy to its entry configuration just prior to utilization. Using a HIAD in conjunction with a CubeSat can create a low cost spacecraft with reentry capability. Such a capability can open opportunities for mission designers tackling three specific areas of study. The first area of study, anchoring design uncertainties for all HIAD missions, is achieved by providing data on entry flight environments and performance metrics on the response of the aeroshell system. Small sat platforms could provide HIAD technologists this crucial data at a fraction of the cost of traditional flight test opportunities. Second is enabling in-space assembly applications that require welding. By enabling the return of experiment specimens from low earth orbit, more thorough investigation of the test materials in ground based laboratories could be achieved. Such an approach would relieve the requirement to validate welded joints while on-orbit. Finally, the reduced mass, volume and cost of the combined HIAD/CubeSat architecture could open opportunities for atmospheric entry probes. Multiple individual probes could be released at several points around a planetary body to gather data over a global scale rather than extrapolating to a global scale from a single sampling point. Alternatively, a suite of planetary entry probes could enter in place of a single large expensive probe reducing operational risk of a given mission because of redundancy of measurements at lower total cost.

In this paper we discuss the impact of the conceptual design of atmospheric entry probes, utilizing HIAD and CubeSat technologies that will enable new mission architectures to meet important science and technology objectives.
HUMAN MARS ENTRY, DESCENT AND LANDING (EDL) STATUS UPDATE AND INVESTMENT PLANNING.  M. M. Munk¹ and R. A. Lillard², ¹NASA-Langley Research Center (1 N. Dryden Street, Mail Stop 489, Hampton, VA, 23681, Michelle.M.Munk@nasa.gov), ²NASA-Johnson Space Center (Mail Code XA1, Houston, TX, 77058, Randy.Lillard-1@nasa.gov).

Introduction: Over the past year, NASA’s Evolvable Mars Campaign’s (EMC) Entry, Descent, Landing and Ascent (EDL&A) Team has been joined by technologists developing various EDL concepts for landing humans on Mars in the 2030’s. The expanded team has been studying four concepts for integrated landing vehicles, each with the same payload complement, but utilizing a different hypersonic deceleration system for Mars EDL. All concepts utilize supersonic retropropulsion (SRP) and precision landing sensor and algorithm technologies to land safely and precisely, meeting EMC requirements.

This presentation will status the two main activities of the EDL team: (1) risk identification for a particular concept of Mars Aerocapture and EDL operations, for each configuration, and (2) detailed design of the integrated vehicles. The risk identification activity included an assessment of the risk mitigation activities needed, including ground and flight testing. A key question was whether a Mars precursor mission was needed to reduce the risk of the first large-scale flight. Near-term activities common to multiple concepts were provided as inputs to Space Technology Mission Directorate (STMD) planning and future funding requests. The design assessment puts the simulations and models in place so that the team can analyze a range of future vehicle types, and performs the next level of detailed design on each concept. Near the completion of the design assessment, independent experts will review the work, and various criteria will be considered, to find discriminators that indicate which system will be more suitable. Since NASA’s funding is too limited to mature multiple technologies, it is important to narrow the technology choices as soon as possible.
MANUFACTURING CHALLENGES AND BENEFITS WHEN SCALING THE HIAD STACKED-TORUS AEROSHELL TO A 15 METER CLASS SYSTEM. G. T. Swanson¹, F. M. Cheatwood², R. K. Johnson², S. J. Hughes², A. M. Calomino², ¹AMA Incorporated, NASA Ames Research Center, Moffett Field, CA 94035 USA, ²NASA Langley Research Center, Hampton, VA, 23681, USA

Abstract: Over a decade of work has been conducted in the development of NASA’s Hypersonic Inflatable Aerodynamic Decelerator (HIAD) deployable aeroshell technology. This effort has included multiple ground test campaigns and flight tests culminating in the HIAD project’s second generation (Gen-2) aeroshell system. The HIAD project team has developed, fabricated, and tested stacked-torus inflatable structures (IS) with flexible thermal protection systems (F-TPS) ranging in diameters from 3-6m, with cone angles of 60 and 70 deg. To meet NASA and commercial near term objectives, the HIAD team must scale the current technology up to 12-15m in diameter. Therefore, the HIAD project’s experience in scaling the technology has reached a critical juncture. Growing from a 6m to a 15m-class system will introduce many new structural and logistical challenges to an already complicated manufacturing process.

Although the general architecture and key aspects of the HIAD design scale well to larger vehicles, details of the technology will need to be reevaluated and possibly redesigned for use in a 15m-class HIAD system. These include: layout and size of the structural webbing that transfers load throughout the IS, inflatable gas barrier design, torus diameter and braid construction, internal pressure and inflation line routing, adhesives used for coating and bonding, and F-TPS gore design and seam fabrication. The logistics of fabricating and testing the IS and the F-TPS also become more challenging with increased scale. Compared to the 6m aeroshell (the largest HIAD built to date), a 12m aeroshell has four times the cross-sectional area, and a 15m one has over six times the area. This means that fabrication and test procedures will need to be reexamined to account for the sheer size and weight of the aeroshell components. This will affect a variety of steps in the manufacturing process, such as: stacking the tori during assembly, stitching the structural webbing, initial inflation of tori, and stitching of F-TPS gores. Additionally, new approaches and hardware will be required for handling and ground testing of both individual tori and the fully assembled HIADs.

There are also noteworthy benefits of scaling up the HIAD aeroshell to a 15m-class system. Two complications in working with handmade textile structures are the non-linearity of the material components and the role of human accuracy during fabrication. Larger, more capable, HIAD structures should see much larger operational loads, potentially bringing the structural response of the material components out of the non-linear regime and into the preferred linear response range. Also, making the reasonable assumption that the magnitude of fabrication accuracy remains constant as the structures grow, the relative effect of fabrication errors should decrease as a percentage of the textile component size. Combined, these two effects improve the predictive capability and the uniformity of the structural response for a 12-15m HIAD.

In this presentation, a handful of the challenges and associated mitigation plans will be discussed, as well as an update on current 12m aeroshell manufacturing and testing that is addressing these challenges.
SYSTEM LEVEL AEROTHERMAL GROUND TESTING FOR THE ADAPTIVE DEPLOYABLE ENTRY AND PLACEMENT TECHNOLOGY. Alan Cassell\textsuperscript{1}, Sergey Gorbunov\textsuperscript{2}, Bryan Yount\textsuperscript{3}, Dinesh Prabhu\textsuperscript{1}, Maxim de Jong\textsuperscript{4}, Tane Boghozian\textsuperscript{5}, Frank Hui\textsuperscript{1}, Y.-K. Chen\textsuperscript{1}, Carl Kruger\textsuperscript{3}, Paul Wercinski\textsuperscript{1}

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Introduction: The Adaptive Deployable Entry and Placement Technology (ADEPT), a mechanically-deployable entry vehicle technology, has been under development at NASA since 2011. As part of the technical maturation of ADEPT, designs capable of delivering small payloads (~10 kg) are being considered to rapidly mature sub 1 m deployed diameter designs. The unique capability of ADEPT for small payloads comes from its ability to stow within a slender volume and deploy to achieve a mass-efficient drag surface with a high heat rate capability. The low ballistic coefficient results in entry heating and mechanical loads that can be met by a revolutionary three-dimensionally woven carbon fabric supported by a deployable skeleton structure. This carbon fabric has test-proven capability as both primary structure and payload thermal protection system.

In order to rapidly advance ADEPT’s technical maturation, the project is developing test methods that enable thermostructural design requirement verification of ADEPT designs at the system level using ground test facilities. Results from these tests are also relevant to larger class missions and help us define areas of focused component level testing in order to mature material and thermal response design codes. The ability to ground test sub 1 m diameter ADEPT configurations at or near full-scale provides significant value to the rapid maturation of this class of deployable entry vehicles. This paper will summarize arc jet test results, highlight design challenges, provide a summary of lessons learned and discuss future test approaches based upon this methodology.
Heat-shield for Extreme Entry Environment Technology (HEEET)  
Development Status

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The Heat shield for Extreme Entry Environment Technology (HEEET) Project is a NASA STMD and SMD co-funded effort. The goal is to develop and mission infuse a new ablative Thermal Protection System that can withstand extreme entry. It is targeted to support NASA’s high priority missions, as defined in the latest decadal survey, to destinations such as Venus and Saturn in-situ robotic science missions. Entry into these planetary atmospheres results in extreme heating. The entry peak heat-flux and associated pressure are estimated to be between one and two orders of magnitude higher than those experienced by Mars Science Laboratory or Lunar return missions. In the recent New Frontiers community announcement [1] NASA has indicated that it is considering providing an increase to the PI managed mission cost (PIMMC) for investigations utilizing the Heat Shield for Extreme Entry Environment Technology (HEEET) and in addition, NASA is considering limiting the risk assessment to only their accommodation on the spacecraft and the mission environment [1].

The HEEET ablative TPS utilizes 3D weaving technology to manufacture a dual layer material architecture. The 3-D weaving allows for flat panels to be woven. The dual layer consists of a top layer designed to withstand the extreme external environment while the inner or insulating layer by design, is designed to achieve low thermal conductivity, and it keeps the heat from conducting towards the structure underneath. Both arc jet testing combined with material properties have been used to develop thermal response models that allows for comparison of performance with heritage carbon phenolic. A 50% mass efficiency is achieved by the dual layer construct compared to carbon phenolic for a broad range of missions both to Saturn and Venus.

The 3-D woven flat preforms are molded to achieve the shape as they are compliant and then resin infusion with curing forms a rigid panels. These panels are then bonded on to the aeroshell structure. Gaps exist between the panels and these gaps have to be filled with seams. The seam material then has to be bonded on to adjacent panels and also to the structure. The heat-shield assembly is shown in Figure 1. One of the significant challenges we have overcome recently is the design, development and testing of the seam. HEEET material development and the seam concept development have utilized some of the unique test capabilities available in the US. The various test facilities utilized in thermal testing along with the entry environment for Saturn and Venus missions are shown in Figure 2. The HEEET project is currently in it’s 3rd year of a four-year development. Figure 3 illustrates the key accomplishments to-date and the challenges yet to be overcome before the technology is ready for mission infusion. This proposed presentation will cover both progress that has been made in the HEEET project and also the challenges to be overcome that is highlighted in Figure 3. Objective of the HEEET project is to mature the system in time to support the next New Frontiers opportunity and we believe we are well along the way to mission infuse HEEET.

Reference:

(1) http://newfrontiers.larc.nasa.gov/announcements.html
Figure 1. HEEET panels and seams integrated on to a structure has to be demonstrated to withstand > 5000+ W/cm² peak heating and stagnation peak pressures greater than 6 atmospheres.

Figure 2. The numerous facilities, both arc jet and laser heated (LHEML) where tests have been conducted to assess the performance of and understand failure mode if any of the dual layer, 3-D woven, ablative TPS material along with seam concepts. These tests include stagnation point coupons as well as testing flat panels in a wedge configuration to test at combined (heat-flux, pressure and shear).
Figure 3. HEEET project went through a year of formulation and is currently in its 3rd year of a
**Exomars Radar Doppler Altimeter: A doppler radar for accurate attitude measurements in support to landing phase.** Ornella Bombaci, ThalesAleniaSpace Italia (ornella.bombaci@thalesaleniaspace.com), Marco Iorio, ThalesAleniaSpace Italia (marco.iorio@thalesaleniaspace.com), and Pasquale Pepe, ThalesAleniaSpace Italia (pasquale.pepe@thalesaleniaspace.com).

**Introduction:** ExoMars programme, led by the European Space Agency, consists in two missions scheduled in the year 2016 and 2018. Thales Alenia Space Italy (TAS-I) is the Industrial Prime Contractor for both missions. TAS-I is also responsible of the Mission 2016 Entry Descent Landing Demonstrator Module (EDM) designed to demonstrate key technologies for Entry Descent and Landing (EDL) capability. Among these key technologies, TAS-I has specifically designed and manufactured a Ka-band Radar Doppler Altimeter (RDA) that is capable to support during the landing phase the autonomous determination by the EDM of its velocity and altitude with respect to the Mars surface. The RDA is requested to provide range and velocity measurement with a very high accuracy during the landing phase at 20Hz rate. In order to exploit to the maximum extent the capabilities of the RDA and its performance during its development, several verification stage have been planned at different level and on different model stage to provide the full evidence and confidence that instrument is able to operate as expected. The performance verification has been conducted at radar electronic equipment level and at radar system level using dedicated Electrical Ground Support Equipment (EGSE) which included a dedicated MARS environment simulator, and outdoor tests to exploit the verification and validation tasks in real environment to cope with the need to investigate measure conditions not fully reproducible by the ground simulation equipment. The present paper aims to describe the key performance parameters and the related design drivers, the overall radar design, its development process up to PFM delivery to EDM, and most of all the tests campaign conducted to show the evidence of the radar full compliance to the measure’s performances. Finally an outlook to the possible uses of such a kind of design to other applications is presented.
THE MARS 2020 LANDER VISION SYSTEM. J. Montgomery, Y. Cheng, A. Katake, N. Trawny, B. Tweddle, J. Zheng, A. Johnson, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109 (james.f.montgomery@jpl.nasa.gov).

Human and robotic planetary lander missions require accurate surface relative position knowledge to land near science targets or next to pre-deployed assets. In the absence of GPS, accurate position estimates can be obtained by automatically matching sensor data collected during descent to an on-board map. The Lander Vision System (LVS) being developed for the Mars 2020 mission generates landmark matches in descent imagery and combines these with inertial data to estimate vehicle position, velocity and attitude. Given this position estimate, the Mars 2020 EDL system will repurpose its existing powered descent divert to avoid 100m scale hazards identified in the landing ellipse map prior to landing. This has enabled the selection of landing sites that have scientifically interesting terrain relief and were not selectable in the original implementation of MSL EDL [1]. Mars scientists see great value in adding this capability to the Mars 2020 lander mission [3].

LVS technology development at JPL has been ongoing for more than a decade. Recent work has focused on development of a real-time flight-like prototype which was flown in a helicopter field test conducted in February and March 2014 [2]. During this test, less than 40m position knowledge errors were demonstrated over a wide variety of terrain, illumination conditions, and attitude dynamics. Subsequent to the field test, the LVS prototype was reengineered into a smaller self contained package and integrated with a terrestrial vertical take off and landing rocket to demonstrated a closed loop Mars EDL scenario [5]. Follow on work focused on LVS algorithm improvements and an analysis, using MSL and other data sources, that showed the LVS could tolerate large changes in vertical motion [3]. The Mars 2020 flight design of the LVS occurred in parallel with this field testing and performance analysis and culminated in a preliminary design completed in November 2015. The LVS technology was baselined on the Mars 2020 mission on January 27th 2016.

This presentation will describe the preliminary design of the LVS. It will touch on all aspects of the design including the Vision Compute Element (VCE) processor, the LVS Camera (LCAM), the Map Relative Localization algorithms and firmware, the map used for landmark matching, the flight software implementation, system interfaces (Figure 1, top) and LVS processing timeline (Figure 1, bottom).

Acknowledgments: This work was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.


Figure 1: The LVS system block diagram (top) and LVS processing phases (bottom).
IN FLIGHT QUALIFICATION OF THE GNC OF THE INTERMEDIATE EXPERIMENTAL VEHICLE (IXV) R. Haya¹, V. Marco² and M. Kerr³.
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**Introduction:** The Intermediate eXperimental Vehicle (IXV) is an ESA re-entry lifting body demonstrator built to verify in-flight the performance of critical re-entry technologies. The IXV was launched on February the 11th, 2015, aboard Europe’s Vega launcher. The IXV’s flight and successful recovery represents a major step forward with respect to previous European re-entry experience with the Atmospheric Re-entry Demonstrator (ARD), flown in October 1998. The increased in-flight manoeuvrability achieved from the lifting body solution permitted the verification of technologies over a wider re-entry corridor. IXV has represented an opportunity to increase the TRL level of technologies and to validate design methodologies and tools.

In order to fulfil the technology experimentation and validation objectives, a number of experiments have been flown on-board the IXV. These experiments include the Thermal Protection System (TPS), the TPS instrumentation, the aerothermodynamics and aerodynamics. In addition to these, the IXV provides a basis to mature, consolidate and prove European GNC technologies for re-entry vehicles, including the guidance algorithms for the unique lifting body, the use of the inertial measurement unit measurements with GPS updates for navigation, and the flight control by means of aerodynamic flaps and reaction control thrusters.

**Overview of the Mission:** IXV is a 5 m long lifting body with active elevons that was injected by Vega launcher in an equatorial suborbital trajectory to reach an entry velocity at the EIP of 7.44 km/s. After injection, the IXV flight was comprised of three main phases: orbital, re-entry and descent. The re-entry phase for the IXV vehicle is a guided lifting entry with a flown range beyond 7000 km up to the deployment of a 3 stages parachute system at 25.5 km, with the IXV flight terminating with splashdown on the Pacific Ocean with a flotation system maintaining IXV in conditions suitable for the ship recovery.

**Guidance, Navigation and Control:** The IXV GNC and Flight Management (FM) subsystem is responsible for controlling the vehicle after separation, and taking it to the desired location for parachute triggering. The GNC&FM makes use of an Inertial Measurement Unit (IMU) and GPS measurements to estimate the state vector of the vehicle along the flight, from pre-launch to splashdown, as well as four 400 N thrusters and two aerodynamic flaps as actuators to control it. The GNC is composed of the core functions Guidance, Navigation and Control, and the Flight Management function which manages the changes of the G, N & C modes.
GNC Qualification before flight: In the frame of IXV, the GNC&FM subsystem validation and verification approach went through an incremental verification process which was optimized by ensuring the maximum representativeness and reuse through all stages. It permits to easily identify any GNC algorithm malfunctioning and to isolate and trace those errors through all stages of the GNC design, from the development until the qualification, locating the source of the deviation. These incremental steps cover from the functional verification (ex: Monte Carlo simulations) up to the processor and hardware in the loop subsystem verification within the Avionics Test Facility (ATF) and the Proto Flight Model (PFM). An innovative verification chain has been setup in IXV which includes the retrofitting of the GNC application software into the functional simulation chain as main novelty.

GNC Qualification at post-flight: The final stage of the GNC verification process is the post-flight analyses, in which the performance is assessed. The mission was successful and the vehicle was recovered as planned. The vehicle separated, performed the controlled orbital and hypersonic/supersonic flight and descended under parachutes to reach the designated landing point. An initial post-flight analysis has been conducted using several inspection, simulation and reconstruction techniques. On one side, the flight performance has been compared with the variability band (Monte Carlo) characterized for the System Qualification and Acceptance Review. On the other, a flight prediction using the high fidelity 6DoF Functional Engineering Simulator has been created using the environment and initial conditions as measured during the flight and compared with the flight performance. Some examples of the post-flight results are presented in the following figures. Post-flight results confirm the in-flight qualification of the GNC for a lifting body re-entry vehicle using elevons and RCS as actuators.

Lessons Learned: Lessons learned from analysis, tests and correlation, useful for other present and future Space Projects, will be provided in the paper.

Conclusions: The flight has shown that the GNC subsystem has successfully performed its main tasks: to control the vehicle trajectory and attitude from Separation up to the DRS deployment. Initial post-flight analysis show a flight performance within the expected variability band. The GNC verification during post-flight shows that the GNC has been robust against different departures, compensating the deviations accumulated in orbit to steer the vehicle towards the expected target safely, ensuring nominal fulfilment of the mission objectives.

Acknowledgements: This work has been carried out in the frame of the IXV programme of the European Space Agency with Thales Alenia Space Italia as Prime contractor. In this context, SENER, DEIMOS as part of the core team and GMV have been responsible for the GNC subsystem design, development and verification. The authors wish to thank the complete GNC IXV team (SENER: R. Contreras, R. Sánchez, G. Rodríguez and D. Serrano; DEIMOS: V. Fernández, J. Ospina, G. De Zaiacomo; GMV: J.A. Béjar-Romero), the IXV-GNC and system interface at Thales Alenia Space Italia (R. Yague, E. Zaccagnino, R. Angelini) and the ESA GNC and System responsible (J.P. Preaud, S. Mancuso). In addition the authors wish to thank Mr. Daniele Gherardi, Dr. Luis Peñin and Mr. Augusto Caramagno for their relevant contribution to this project.
ENABLING MARS EXPLORATION USING INFLATABLE PURDUE AERODYNAMIC DECELERATOR WITH DEPLOYABLE ENTRY SYSTEMS (iPADDLES) TECHNOLOGY. M. Sparapanay†, T. Antony‡, H. Saranathan§, L. Klug¶, B. Libben¶, E. Shibata¶, J. Williams¶, M. J. Grant¶, S. J. Saikia†, †PhD Student, msparapa@purdue.edu, ‡PhD Student, t Antony@purdue.edu, §Ph.D. Candidate, hsaranat@purdue.edu, ¶Graduate Student, lklug@purdue.edu, ‡Graduate Student, bliben@purdue.edu, §Graduate Student, e shibata@purdue.edu, ¶Graduate Student, jdrwilliams2014@gmail.com, †Assistant Professor, mjgrant@purdue.edu, ‡Visiting Assistant Professor, ssai kia@purdue.edu, §School of Aeronautics and Astronautics, Purdue University, 701 W. Stadium Ave., West Lafayette, IN, 47907-2045.

Introduction: In this paper, an example iPADDLES system is considered in the context of the National Institute of Aerospace—managed NASA’s Breakthrough, Innovative and Game-changing (BIG) Idea Challenge [3]. The challenge had teams design and analyze Hypersonic Inflatable Aerodynamic Decelerator (HIAD) concepts that could achieve a modulated L/D of 0.2 to 0.5 in the hypersonic regime, be scalable to diameters of 15 to 20 meters, possess a smooth outer mold line, and be aerodynamically-stable from 6.5 km/s to 0.6 km/s for a payload of 50 metric tons. At present, the iPADDLES concept has been chosen for a presentation at NASA Langley Research Center to compete against teams from three other universities in the BIG Idea Challenge.

Concerns with Martian Missions: The Martian environment has a number of hazards and concerns for a crewed mission. Radiation becomes an issue with a non-existent magnetic field, which requires human missions to transport or build shielding. Without a readily available source of food and water, all consumables must be brought from Earth. All of these problems increases the entry mass required for a Mars mission. Having a density that is only a fraction of that of Earth’s atmosphere, the Martian atmosphere is dense enough to create concerns for heating, while not dense enough to provide sufficient drag to decelerate a high mass entry vehicle using traditional aeroshells [1]. The recent Mars Curiosity mission represents the limit on the size rigid aeroshells that can be used on Mars with current launch vehicle volume constraints. Deployable decelerators allow for a larger area to slow down the vehicle, while being small enough to fit into current launch vehicles while stowed. This new paradigm of entry vehicle design using volume and mass efficient designs is required to overcome volume constraints. In the EDL Systems Analysis (EDL-SA) study, flexible inflatable decelerators were identified as an enabling technology for landing large payloads on Mars [1].

Another challenge facing crewed Mars missions is the requirement for pin-point landing. The atmospheric variabilities during the hypersonic regime require the consideration of the landing error ellipse in-flight. Current state-of-the-art technology involves using a rigid aeroshell with a center-of-gravity (CG) shift to generate lift that provides control authority. This allows for guidance and control to be performed during the hypersonic phase to decrease the landing error ellipse.

Despite the heritage behind rigid materials, high mass payloads to Mars forces a redesign in entry technology. The recent Inflatable Reentry Vehicle Experiment (IRVE) program shows that a HIAD with a CG shift is a feasible design, and represents an upcoming game-changing technology [2]. A shape-morphing aeroshell would also be able to provide the modulating lift necessary to decrease landing error. Combining this with an inflatable design allows the ballistic coefficient to be low enough to allow for a low terminal velocity. To this end, a HIAD with control surfaces is proposed.

Vehicle Design: Using a design philosophy of simplicity, low mass, and controllability, a standard symmetrical stacked toroid HIAD is expanded upon by adding linearly actuated control flaps. The four flaps are placed symmetrically around the circumference of the top toroid (two on the horizontal axis, and two on the vertical axis) to allow for independent angle of attack and sideslip control. The inflatable control surfaces allow for nonzero trim angles of attack (AoA) that can create a maximum L/D of 0.5. The asymmetric deployment of these flaps allow for crossrange control by initiating a sideslip angle.

![Fig. 1: Cut-away view of the proposed iPADDLES design. A symmetrical stacked toroid HIAD is the base, and four linear actuation systems are added to the top-most toroid to modulate the control flaps.](image)

Guidance: A Sideslip Augmented Apollo Guidance (SAAG) algorithm is proposed that takes advantage of flight-proven heritage Apollo reentry guid-
ance algorithm. In theory, the aerodynamic forces vectors acting on an axisymmetric vehicle resulting from a bank maneuver (velocity vector roll) and an angle-of-attack and side-slip maneuver are the same provided the vehicle’s axis of symmetry points in the same direction after either maneuvers. Consequently, the bank command from Apollo guidance is converted into an AoA and sideslip sequence. The tabs are deflected suitably to follow this command. The variation in lift and drag coefficients resulting from deployment of the tabs are accounted by introducing a bias in the sensed drag acceleration. The roll angle of the vehicle is held at 0 deg. using reaction control thrusters.

The chosen vehicle configuration can generate sufficient drag while maintaining a high lift-to-drag ratio. A Monte-Carlo simulation was performed to analyze the robustness of the vehicle and the guidance algorithm to errors in entry, environment, and vehicle parameters. The variations tested are similar to those used for the analysis of MSL by Striepe et al [4]. The atmospheric data was also varied by using different datasets from MARSGRAM2005. The dispersions were obtained by running the simulation for 2000 iterations, starting at the entry interface (6 km/s at 80 km altitude) and terminating at Powered Descent Initiation (PDI) at 600 m/s.

Results: The aerodynamic properties are studied using panel methods and modified Newtonian flow theory. The maximum trim angle-of-attack attainable using the selected configuration is found to be 30.9º, corresponding to a lift-to-drag ratio of approximately 0.5. At this trim AoA, the ballistic coefficient is found to be 132 kg/m² for a vehicle mass of 50 metric tons which is comparable to that of MSL (140 kg/m²) [4].

The 99% footprint at Powered Descent Initiation (PDI) is found to be around 20 km along the downrange direction and 0.6 km along the crossrange direction. If only 95% of the points are considered, the footprint shrinks further to 12 km x 0.4 km. This is comparable, if not better than what the previous Mars missions have accomplished and enabled pin-point landing after the powered-descent phase.

TENSION ADJUSTABLE NETWORK FOR DEPLOYING ENTRY MEMBRANE (TANDEM) CONCEPT
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Abstract: The Tension Adjustable Network for Deploying Entry Membrane (TANDEM) is a specially configured tensegrity rover that is designed to also be the primary structure of a deployable heat shield. TANDEM combines the infrastructure used for the Entry, Decent, and Landing sequence as well as on-the-ground locomotion into a single efficient multi-functional system. This efficiency translates directly to longer mission lives, increased scientific payload, or a reduction in mission cost and total mass of the whole mission. TANDEM provides the same benefits as the well-established ADEPT concept (larger payloads, lower entry G's, smaller launch vehicles, low ballistic coefficient, etc) while adding the benefit of surface locomotion into a single efficient multi-functional system. This efficiency translates directly to the-ground locomotion into a single efficient multi-functional system. The added controllability can be used to maximize the time spent in different layers of the atmosphere where the vehicle can collect more data samples than classical entry vehicle.

After peak heating, TANDEM can release the heat shield. The vehicle may be equipped with parachutes for some missions. Instead, one option is to use the back shell as a flexible drag plate. For Venus specifically it will be advantageous to slow descent while in the upper atmosphere and cloud layer and then descend quickly through the hot dense lower atmosphere. TANDEM can easily preform this by tensioning specified cables to opening and closing its back shell. The same mechanism can also be used to perform accurate pinpoint landing.

When the vehicle is above its desired landing site, the back shell can be released allowing the rover to free fall to the surface. Depending on the size and mass required for the specific mission, the terminal velocity for Venus applications will be roughly 20 m/s. Due to TANDEM’s large shock absorbing capability, this can be survived without the payload exceeding a 100 g’s limit. Like most tensegrity concepts, TANDEM inherently provides omni-directional protection on impact. However, due to the unique shape of the TANDEM lander, it can actively adjust its shape for the optimal landing configuration. If unforeseen circumstances cause the lander to turn upside-down, TANDEM can be programmed to change its aerodynamic center and reorient itself. Alternatively, it can adjust its outer circumference to provide a safer landing based on its current orientation. This improves reliability and safety to any mission using the TANDEM concept.

Figure 1 on the following page shows the baseline TANDEM design, with a 70 degree cone angle and no heat shield. Figure 2 shows a baseline EDL sequence for a mission to Venus.

References:
Figure 1: TANDEM Design

Figure 2: EDL Sequence