Ultra Lightweight Ballutes for Return to Earth from the Moon

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Ultra lightweight ballutes offer revolutionary mass and cost benefits along with flexibility in flight system design compared to traditional entry system technologies. Under funding provided by NASA's Exploration Systems Research & Technology program, our team was able to make progress in developing this technology through systems analysis and design, evaluation of materials and construction methods, and development of critical analysis tools. Results show that once this technology is mature, significant launch mass savings, operational simplicity, and mission robustness will be available to help carry out NASA's Vision for Space Exploration.

Nomenclature

CAD	=	Computer Aided Design
CEV	=	Crew Exploration Vehicle
CFD	=	Computational Fluid Dynamics
ESR&T	=	Exploration Research & Technology
FEA	=	Finite Element Analysis
ISA	=	Integrated Systems Analysis
ISP	=	In-Space Propulsion
ISS	=	International Space Station
JPL	=	Jet Propulsion Laboratory
LEO	=	Low Earth Orbit

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MSFC = Marshall Space Flight Center
PBO = Phenylene-benzobisoxazole
TPS = Thermal Protection System
TRL = Technology Readiness Level
ULWB = Ultra Lightweight Ballute

I. Introduction

Ultra lightweight ballutes (ULWB) offer the potential to provide aerodynamic deceleration for NASA's Exploration and Science missions that require entry or aerocapture at Earth (or other destinations with atmospheres) without high heating rates on the host spacecraft, and with much lower mass than traditional technologies. This innovative concept, illustrated in Fig. 1, involves deployment prior to entry of a large, lightweight, inflatable aerodynamic decelerator (ballute) whose large drag area allows the spacecraft to decelerate at very low densities high in the Earth's atmosphere with relatively low heating rates. Because the vehicle using the ballute decelerates at much higher altitude, the peak heating rate for return to Earth from the moon is two orders of magnitude lower than for entry using a traditional lunar return capsule, negating the need for an ablative thermal protection system.

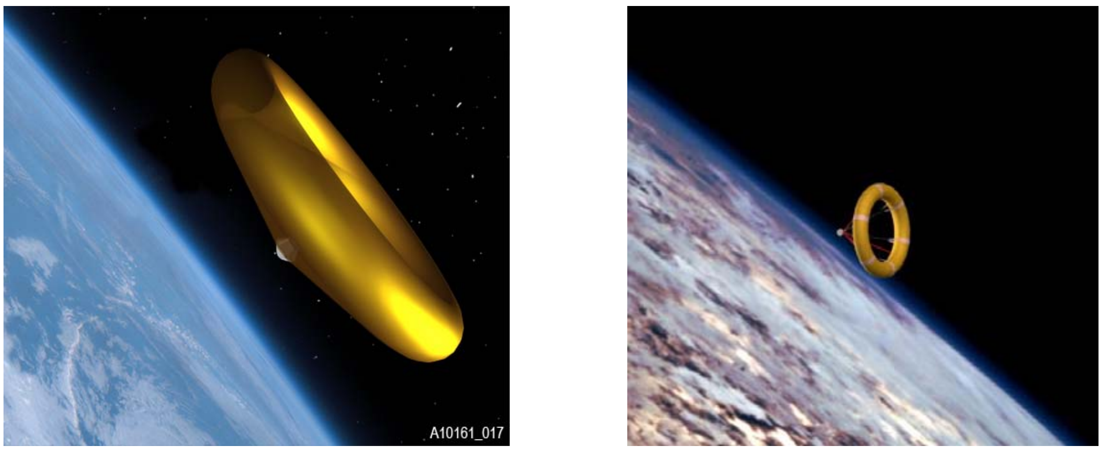


Figure 1. Attached (left) or trailing (right) ultra lightweight ballutes provide deceleration for entry without high heating rates on the host spacecraft.

The lower heating rates experienced during atmospheric entry allows the use of light-weight construction techniques for the ballute, resulting in significant mass performance advantages. Ultra lightweight ballutes potentially provide a mass fraction (deceleration system mass divided by total mass at entry) of 10 to 15 percent, compared to 25 to 35 percent for aeroshells with ablative thermal protection systems (TPS), resulting in a large payload mass or operational benefit. Since the spacecraft with an ultra lightweight ballute system is not subjected to high heating, a high degree of spacecraft reusability results. Subsequent missions require replacing only the ballute, similar to providing a new parachute. Ultra lightweight ballutes are stowed in a small volume and inflated shortly before use. Thus, use of ultra lightweight ballutes frees the spacecraft design from constraints imposed by an aeroshell, including packaging, strict control of the center of gravity, heat rejection, and antenna or sensor viewing.

Ball Aerospace and its partners have been developing ULWB technology for use in aerocapture at Titan and Neptune under funding provided by NASA's In-Space Propulsion (ISP) Office.[1,2,3] Ref. 4 presents a survey of these concepts as well as other ballute concepts applied to aerocapture. Through these prior efforts, key technical challenges associated with ultra lightweight ballute technology have been identified, including materials survivability, aeroelastic response, and techniques for ballute construction, packaging, and deployment. A team consisting of Ball Aerospace, ILC Dover, Georgia Tech, NASA Langley, NASA MSFC, and the Jet Propulsion Laboratory has recently been addressing these technical issues under funding provided by NASA's Exploration Systems Research and Technology (ESR&T) Program. The team's efforts under the ESR&T project were focused on advancing the Technology Readiness Level (TRL) of ultralightweight ballutes for return to Earth from the Moon through integrated systems analyses anchored with data obtained in a series of ground-based tests. The project was initiated in April 2005. Due to budget reductions in NASA's ESR&T program, the project was prematurely ended in October 2005.

This paper provides a brief summary of the results from the team's ESR&T efforts, including requirements and mission concepts, ballute systems analysis and design, aeroelastic modeling and analysis, and development of integrated systems analysis tools. A comprehensive report of the team's efforts can be found in Ref. 5.

II. Requirements and Mission Concepts

A. Requirements

The mission objective is to provide deceleration for the Crew Exploration Vehicle (CEV) upon return to Earth from the Moon. Deceleration at high altitude is desired to reduce heating and increase mass performance. The performance goal was for the ballute system to be 10 to 15% of the total mass at entry, with the packaged ballute system less than 5% of the CEV volume. A systems engineering approach was applied to define system level requirements and analyze the requirements' impact on the design. This is illustrated in Table 1, which shows the system requirements used for development of reference ballute system designs during this project.

Table 1. Top-Level System Requirements and Impact on Design

Parameter	Requirement	Impact to Design / Comment
Entry Velocity	11.1 km/s at 125 km altitude (assume 2 day min trip time)	Peak heating and peak dynamic pressure on ballute are proportional to V^3 and V^2, respectively.
Entry flight path angle delivery error	±0.1 deg, 3-sigma	As dispersions increase, ballute system mass performance decreases because the system is designed to handle the 3-sigma worst-case conditions.
Vehicle mass to be decelerated	7500 kg	Ballute size is directly proportional to total mass to be decelerated. Interest in range of 5000 to 10000 kg.
Maximum ballute temperature	500, 800, 1000 C for all film, film/laminate, fabric construction respectively	Constrains maximum aerodynamic heating rate allowed on ballute, which in turn determines maximum allowable ballistic coefficient (minimum size of ballute). Also influences ballute geometry details.
Trajectory profile and timeline	Nominal peak acceleration of 6 to 8G for entry.	Determines aero/aerothermal loading profile that ballute must withstand. Materials can potentially be used outside of their normal qualified operational ranges because loading is a transient event.
Ballute deployment time	Less than 1 hour	Drives inflation system design, as well as induced dynamic loads and effects. Desire to minimize deployment time, but also deployment dynamics.

B. Mission Concepts

ULWB technology results in a family of general purpose, high altitude, hypersonic decelerators that can be used in many mission scenarios for deceleration of the CEV or other assets. Return to Earth from the moon mission scenarios considered during the project include the following: aerocapture into Low Earth Orbit (LEO); direct entry; and hybrid entry.

For aerocapture, the spacecraft approaches Earth on a hyperbolic trajectory. After spacecraft configuration for entry, the ballute is deployed and inflated before entering the atmosphere. As the spacecraft enters the atmosphere, the ballute provides deceleration from lunar return speeds. Peak heating on the ballute occurs first, followed by peak dynamic pressure and deceleration. At the desired time, the ballute is separated from the spacecraft, allowing the spacecraft to exit the atmosphere and achieve the desired orbit. A small propulsive maneuver is used to adjust the final orbit.

The sequence for direct entry is the same as that for aerocapture, except that the ballute is not separated early enough to allow the spacecraft to exit the atmosphere. The ballute provides deceleration for entry from a lunar return down to lower altitudes. At that point, the ballute could be retained for landing, or the ballute could be separated, allowing terminal descent to be provided by another means (e.g., parachute, parafoil, or wings).

For the hybrid entry, the ballute is inflated prior to entry just as it is for aerocapture and direct entry missions. The ballute then provides deceleration from lunar return speeds at high altitudes with low heating rates, and once LEO velocity is obtained, the ballute is separated. From this point, the vehicle performs a traditional LEO entry. Deceleration at higher altitude results in a lower peak heating rate upon return form the moon, and then after ballute separation, heating rates are similar to that experience for entry from LEO. Thus, the hybrid entry option allows the

CEV thermal protection system to be designed and qualified for the LEO aerothermodynamic environment, but be used for lunar return missions.[6]

ULWBs can also be used for missions other than return to Earth from the Moon. These include:

- Entry and deceleration at Mars (for Aerocapture, direct entry, or hybrid entry)
- Return to Earth from Mars (for Aerocapture, direct entry, or hybrid entry)
- CEV Ascent Abort – The ballute provides high altitude deceleration with low heat rates in abort modes.
- Returning mass from the International Space Station (ISS).[7]

The primary focus during this technology development project was to investigate return to Earth from the Moon applications. However, systems assessments of alternative mission concepts for the hybrid entry, CEV ascent abort, and ISS down-mass were completed and are documented in Ref. 5, 6, and 7.

III. Ballute Systems Analysis and Design

A. Analysis and Trade Studies

The objective of the ballute systems analysis and design effort was to develop ballute system concepts that provide the context for ground testing activities. The work included trajectory, aerodynamic, thermal, and structural analyses. The analysis and design efforts investigated several ballute critical design parameters including: ballute geometry and dimensions, mass, maximum stagnation temperature, and dynamic pressure. Factors involving ballute packaging, deployment, fabrication, TRL, applicability to other missions, and ease of integration with the CEV were also considered.

Parametric trajectory analysis was performed first to investigate the trade space and determine drag area and initial ballute sizing required to maintain material temperature constraints. Once this initial ballute sizing was completed, more detailed analysis was performed to refine the concepts. Computational Fluid Dynamics (CFD) analysis was completed to determine pressure and heating distributions. The aeroheating distributions were applied to 3-dimensional thermal models to predict temperatures and verify adequate temperature margin against the material limits. Pressure distributions were applied to nonlinear structural finite element models to determine material thickness, verify adequate strength margin against the material allowables at temperature, and to estimate deflections. Mass and packaging analysis was performed to determine how the various concepts being considered met the mass and volume performance goals. Details of the analyses completed can be found in Ref. 5.

Two reference ballute configurations have been developed. One configuration was an attached (clamped) ballute configuration made of fabric and film materials. The other configuration was an attached ballute made of all thin-film construction. The material temperature limits dictate the sizes of each ballute. For a 7500 kg payload, the fabric/film ballute requires a diameter of 20 m, while the all thin-film ballute requires a diameter of 80 m to keep temperatures within the respective material limits. Due to the large size of the all thin-film ballute, the fabric/film approach was selected as the preferred approach.

B. Reference Ballute Configuration

The reference ballute concept is shown in Fig. 2. The overall outside to outside diameter is 20 m, with 8.75 m major radius and 1.25 m minor radius (i.e., an aspect ratio of 7). The ballute consists of four major subcomponents, namely the inflatable torus, thermal protection system, inflatable booms, and local reinforcement. These subcomponents are fully

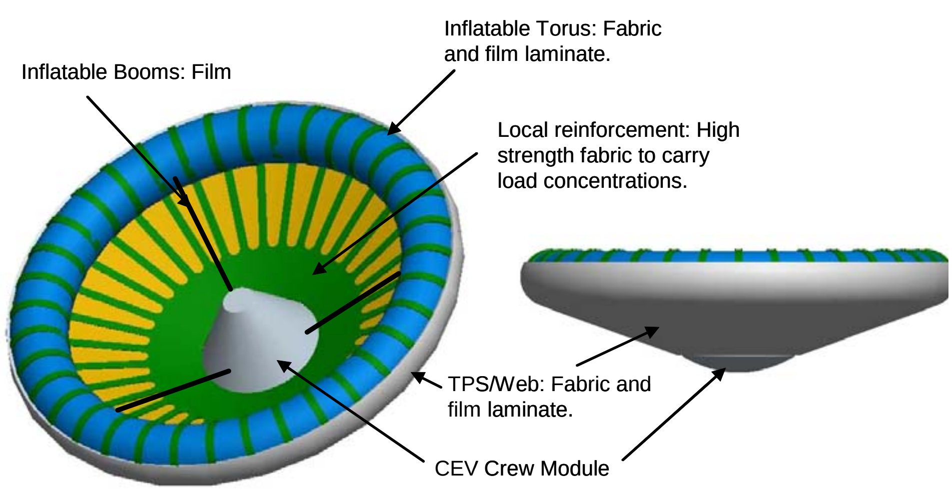


Figure 2. Reference ballute configuration for CEV is an attached configuration fabricated from a combination of fabric and film materials.

integrated after assembly and will be packed and deployed as a single system.

A mass analysis was performed for the reference concept using computer aided design (CAD) tools. The total system mass (not including inflation system) with 20% mass contingency is approximately 550 kg, which is about 7.3% of the entry mass (7500 kg). The same CAD model used for mass analysis was also used for packing volume evaluation. The estimated packing volume is 1.2 m^3, which is less than 5% of the CEV crew module volume.

In the future, additional shape design and construction options should be investigated. The current torus baseline design was chosen to be circular with 7 to 1 aspect ratio, but this selection can be further refined. The construction of the torus, e.g., number of segments should be optimized for shape accuracy, but stay within the desired mass limit. The structural attachment from the torus to the spacecraft is currently a non-inflated reinforcement integrated with the TPS/web. This design must be analyzed and traded with potential inflated options that can possibly provide higher stiffness with lower mass. Materials should be assessed against the number of expected packaging and deployment cycles. Finally, interface designs such as TPS/web to crew module attachment, TPS/web to torus attachment, TPS/web reinforcement, deployment boom to CEV interface, deployment boom to torus interface, and packing design options have not been thoroughly defined.

IV. Materials Testing

A. Introduction

The driving constraint for ultralightweight ballutes is the material property limits at the ballute operating temperature. Testing efforts were completed to assess a variety of materials' performance under the expected operating conditions. Two separate but related test efforts were conducted by the team. Teammates at ILC Dover focused on testing existing film and fabric materials and adhesives. These efforts were based on work previously completed under contract to NASA's In-Space Propulsion Office for ballute aerocapture at Titan and Neptune.[1,2,3] Teammates at JPL focused on higher temperature laminates made from lower TRL materials. These efforts were based on work previously completed at JPL for planetary balloon and ballute programs.

B. Materials Testing at ILC Dover

In the area of thin-film materials testing, the team performed testing of Upilex and PBO films at high temperature (up to 600 C) to characterize performance and determine maximum feasible temperatures for ballute applications. The results established non-linear stress/strain curves for the materials as well as upper use temperature limits for Upilex and PBO film. Tests were conducted in oxidative and non-oxidative environments to assess affects on performance.

Because the team was interested in pushing the operating temperature of the ballute to as high as possible, concepts involving film with flexible fabrics for TPS were considered. To support these efforts, several fabric materials were tested in a furnace to assess their insulating properties and potential to be used as TPS. The tests identified several candidate materials, but further test and evaluation is required before conclusions can be reached.

Adhesives for film materials were also tested. From previous work, ILC Dover had identified an adhesive that could be used for Upilex film, but had not found a suitable adhesive for PBO film. During the ESR&T project, ILC developed a custom adhesive for the PBO film. The team demonstrated the strength of this new adhesive for PBO film at high temperature. They also demonstrated successful use of this same adhesive for Upilex film. Thus, a new adhesive suitable for use at high temperatures with both Upilex and PBO film was demonstrated.

Results from all of the material tests were documented in Ref. 5 and included in the finite element models being used for system design. Thus, the ballute system concepts being developed include as realistic materials data as possible. Further testing will be required to develop a statistical basis for material properties, and additional tests to fully characterize material response throughout the expected environment will be required.

C. Materials Testing at JPL

The JPL concept for the composition of a high temperature ballute material is a multi-component laminate composed of materials that are very low weight and capable of retaining their strength at temperatures of up to 700 or 800 C. The current concept, illustrated in Fig. 3, consists of a three-element laminate. The outer layer (fabric) is capable of tolerating the highest operating temperature and provides the primary

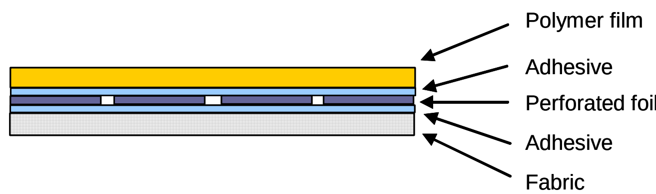


Figure 3. Illustration of high temperature ballute material concept.

strength for retaining the integrity of the ballute. The middle metal foil layer is also designed to tolerate the highest operating temperature while acting as a gas barrier. Perforations are included in the metal foil layer to provide enough flow area to vent gas from the ballute at the desired rate as the gas is heated and expands. The inner polymer film layer is sacrificial at the elevated temperatures, but does provide a flexible substrate to facilitate packaging, storage, and deployment of the other two layers. It also covers the holes in the metal foil layer until burned off, thereby preventing significant gas leakage between the time of ballute deployment and the high temperature phase of the atmospheric flight. These three layers will be bonded together with a high temperature adhesive.

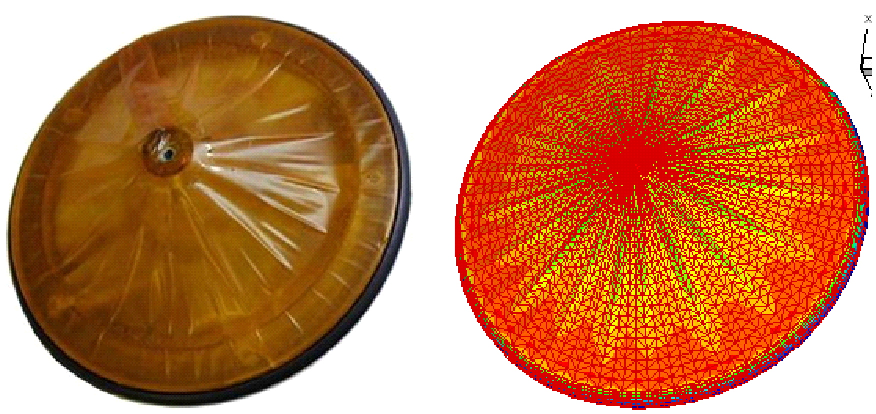
Multiple options were identified for each of the three layers of the laminate, and candidate material samples were acquired and tested. Materials testing included quartz and carbon fiber fabrics, stainless steel and titanium meshes, Molybdenum, Titanium, and Nickel foils, and various adhesives at temperatures up to 800 C. The team also completed an initial assessment of ballute fabrication and construction issues. The results of these tests indicate that a laminate can be produced that meets the requirements of 700 to 800 C operation with a low areal density. Further work is required to refine the concept and perform testing at larger scales needed for ballute construction. More details of the testing and results can be found in Ref. 5.

V. Aeroelastic Modeling

Due to the lightweight construction of the ballute, there are significant structural deflections, and therefore aeroelastic modeling and analysis is required. For the mission concepts under consideration, an aeroelastic tool set that integrates nonlinear structures, hypersonic rarified aerodynamics, and thermal analysis is needed. Currently, tool sets to address this problem are just beginning to be developed. The difficulties expected to be encountered in developing these tool sets are described in Ref. 8.

During the ESR&T project, the team pursued two different approaches for aeroelastic modeling and analysis: one by Georgia Tech, and one by NASA Langley. The primary aerothermodynamic codes to be used within the NASA Langley Research Center effort for aerothermoelastic analyses are LAURA, FUN3D, and CFL3D. Since the coupling of several computationally intensive analyses such as Navier-Stokes CFD and an FEA code will be prohibitively expensive for routine flutter analyses, several levels of code fidelity will be required. Initial aerothermoelastic computations can be performed using real gas Euler fluid dynamics. However, because of the complex vehicle ballute shape and high Mach number, laminar and possibly transitional viscous analysis of selected conditions will, at some point, be required. The aeroelastic modeling effort at Georgia Tech is aimed at developing a loosely coupled interface utilizing existing codes to determine both static and dynamic behavior of ballutes. The static method iterates between analyses until a converged solution is obtained, while the dynamic solution uses a time-accurate stepping procedure. The interface code is designed to couple LS-DYNA for structural dynamics, NASCART-GT for continuum aerothermodynamics, DAC for transitional and rarefied aerothermodynamics, and SINDA for thermal response. In addition to these full physics models, a set of impact based aerodynamic methods is included to provide a lower computation time option.

The team made substantial progress in developing tools for coupling of nonlinear structures, hypersonic aerodynamics, and thermal analysis for aeroelastic analysis of ULWB systems. The team completed an initial version of a loosely coupled tool-set with simplified aerodynamics, and developed a preliminary version of the tool-set with an



Hypersonic Thin Film Test Model

Simulation Results

Figure 4. Aeroelastic simulation results accurately model wrinkles induced by hypersonic flow field in wind tunnel tests.

embedded CFD analysis. The team also validated the initial version of the loosely coupled tool-set against existing wind tunnel test data of thin-film ballutes. The results, shown in Fig. 4, show good agreement between the wind tunnel test data and predictions from the analysis tool set. Several options for a tightly coupled tool-set were investigated and the team completed initial aeroelastic analysis to demonstrate feasibility of that approach as well. Work is ongoing in this area at Georgia Tech and results from the validation study and static aeroelastic solutions will be presented in Ref. 9 using both low- and high-fidelity aerodynamics.

VI. Integrated Systems Analysis Capability

The Integrated Systems Analysis (ISA) tool has been developed within the Space Systems Design Laboratory at the Georgia Institute of Technology to enable rapid design and analysis of entry systems for entry, descent, landing, and aerocapture applications. The tool integrates relevant disciplinary analyses including aeroshell geometry, atmospheric modeling, vehicle aerodynamics, atmospheric flight mechanics, aerothermodynamics, and thermal protection system analysis within a single multi-disciplinary framework. The tool is intended for application to conceptual design and analysis.

During the ESR&T project, students at Georgia Tech worked on extending this tool to include ballute entry system analysis capabilities to enable rapid assessment of various ballute configurations and material options as the project progressed. Although the project was discontinued before this was complete, significant progress was made. The team completed an initial version of the tool and used it to perform assessments of alternate ballute missions. This included preliminary analysis of ballute application for ISS down-mass,[7] CEV ascent abort, and CEV hybrid entry missions.[6] Future work is planned to continue the development of this tool, gradually increasing the fidelity of the various analysis codes. A detailed description of the tool along with preliminary ballute analysis results is presented in Ref. 5.

VII. Summary and Conclusions

Although the ultralightweight ballute project was ended early due to reductions in NASA's exploration technology budget, several conclusions can be drawn from the efforts completed to date. It can be concluded that ultralightweight ballutes are applicable and offer substantial benefits to CEV and Exploration missions, providing general purpose deceleration at high altitude and high velocities with relatively low heating rates. One finding, however, is that as the mass to be decelerated increases, the size (and mass) of ballute increases rapidly for a fixed ballute operating temperature. Therefore, external ballute peak operating temperatures of 800 to 1000 C are required to maintain the ULWB mass performance benefit for masses in the CEV range. This temperature is outside of the range of operation of thin-film materials. However, thin-film materials, with peak operating temperatures of 500 – 650 C, can still be used because the entire ballute does not see the peak temperature, and films can be used as bladders with other materials providing local thermal protection. For smaller payload masses, all thin-film construction is feasible because the required size of the ballute does not have to be as large to maintain temperatures in the 500 – 650 C range. Future analysis should investigate this trade space more thoroughly to identify appropriate ranges of masses, mass fractions, and temperature.

The team began converging on an ULWB material lay-up concept for use on CEV-class missions that makes use of a combination of films and fabrics. An assessment of manufacturing methods and materials shows that these material lay-up concepts are feasible. Initial analysis and testing of the ballute material lay-up concepts shows potential to achieve 800 C or higher operating temperatures that are required for high performance CEV-class missions. Further material lay-up testing is required to validate these concepts and collect data to reduce uncertainty in analysis models.

Progress to date on aeroelastic modeling of ULWBs is promising. Initial tool-sets have been developed where none existed before. The tools have been used on the ballute concepts and shown that reasonable solutions can be obtained. Preliminary validation of these tools with existing hypersonic test data of thin-film ballute models shows good agreement. However, further validation of these tools is required. There is very limited test data for which to validate these analysis tools. The project had the specific objectives of performing additional tests to obtain validation data, but this will not be completed in the near future due to the aforementioned constraints in NASA's Exploration technology budget. Further development of these tools is recommended, along with validation through tests.

Acknowledgments

The work summarized in this paper was carried out by a team of people from Ball Aerospace, ILC Dover, the Georgia Institute of Technology, NASA Langley, the Jet Propulsion Laboratory, and NASA MSFC. The authors wish to acknowledge the significant contributions from Chris McLean, Rand Donohoo, Fred Hausle, and Maureen

Riesco from Ball Aerospace; Ian Clark, Kevin Flaherty, Scott Francis, Jin Lee, Steve Ruffin, John Theisinger, and Grant Wells from Georgia Tech; Denise Lawless, Jim Stein, and Cliff Willey from ILC Dover; Jack Jones, Steve Jones, and Andre Yavrouian from JPL; Dave Piatak, Rob Scott, and Barmac Taleghani from NASA Langley; and Joe Bonometti from NASA MSFC.

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