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Abstract

The driving requirement for design of a Mars Sample return mission is assuring containment of the returned samples. The impact of this requirement on developmental costs, mass allocation, and design approach of the Earth Entry Vehicle is significant. A simple Earth entry vehicle is described which can meet these requirements and safely transport the Mars Sample Return mission's sample through the Earth's atmosphere to a recoverable location on the surface. Detailed analysis and test are combined with probabilistic risk assessment to design this entirely passive concept that circumvents the potential failure modes of a parachute terminal descent system. The design also possesses features that mitigate other risks during the entry, descent, landing and recovery phases. The results of a full-scale drop test are summarized.

Introduction

The Mars Sample Return (MSR) mission will return selected samples from Mars to Earth. The final phase of the mission requires an Earth entry, descent and landing capsule which is responsible for transporting the samples safely through Earth's atmosphere to a recoverable location on the surface. Preservation of the scientific value of these samples necessitates they remain isolated from Earth contaminants. In addition, the National Research Council's Task Group on Issues in Sample Return[1] determined that the potential for terrestrial contamination from Mars samples, while minute, is not zero. For these two reasons, requirements will be levied on the Earth entry capsule to assure containment of the samples to very high levels of reliability. It is anticipated that this reliability requirement will be orders of magnitude more stringent than those levied on any previous entry system.

The impact of this stringent reliability requirement on development and design of the Earth Entry Vehicle (EEV) is significant. Initial work performed under the auspices of the former MSR Project, indicated a factor of two increase in launch mass allocation and a factor of four increase in development cost to demonstrate adherence to this requirement. The design process must incorporate risk-based design strategies and probabilistic risk assessment at every stage. The concept itself must 1) decrease the number of failure modes by eliminating all nonessential subsystems and 2) utilize heritage systems with sufficient redundancy for each critical subsystem. This paper describes the simplest and most reliable option for the Mars Sample Return Earth Entry Vehicle and the probabilistic risk assessment undertaken to demonstrate the capsule's reliability.

The desire to obtain extraterrestrial samples for Earth-based analysis has spawned several upcoming sample return missions with destinations other than Mars[2]. The fourth discovery-class mission: Stardust [3,4](launched Feb. 7 1999), plans to return comet coma samples and interstellar dust in 2006. The fifth discovery class mission, Genesis[5], plans to collect samples of the solar wind for return in 2003. These missions, whose reliability requirements are less stringent, utilize direct entry capsules with parachute terminal descent.

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Sample return missions for Mars have been studied periodically for the past 30 years [6-9]. The Earth entry phase envisioned by previous studies involved either 1) an orbit insertion at Earth with Space Shuttle or Space Station rendezvous for recovering the samples or 2) direct entry with an Apollo-style entry vehicle utilizing parachutes, air-snatch or water recovery.

Earth Return Options

Returning Mars samples to the surface of Earth can be accomplished with a direct hyperbolic entry capsule as discussed in the present approach or via capture into Low Earth Orbit (LEO) for rendezvous with the International Space Station or the Space Shuttle. At first glance, a LEO rendezvous approach appears to decrease risk by eliminating the Earth entry capsule. However, this approach requires successful completion of a factor of 4 to 10 more critical events[9] and still involves the entry, descent and landing of the Space Shuttle (whose reliability may not meet the containment assurance requirements imposed upon the mission). A risk assessment performed on the present direct entry approach[10] concludes that the simplicity of the design achieves orders of magnitude increase in reliability relative to that demonstrated on other entry systems. This conclusion is consistent with those drawn by a previous study conducted by NASA Johnson Space Center[9] and an independent study conducted at NASA's Jet Propulsion Laboratory (Kohlhase, JPL, 1999).

There exists several options on how to accomplish entry, descent and landing for a direct entry approach. During entry, the capsule could include a lift generating aerodynamic shape with active guidance system to allow tailoring of the deceleration and heating rates and improve ground targeting. The complexities of the guidance system, however, introduce numerous failure modes. Failure of the power system, flight computer, connections or associated control authority system could lead to very large landing errors, excessive thermal and mechanical loads or uncontrolled flight. A passive ballistic entry, which relies solely on aerodynamics for atmospheric trajectory and attitude control, is much simpler and therefore more reliable. Targeting accuracy is controlled primarily by the accuracy of the initial position and flight vector delivered by the host spacecraft prior to capsule separation.

Terminal descent of an entry capsule typically includes a parachute deceleration system to decrease ground impact speeds. Unfortunately, parachute system reliability and that of their activation systems, while the highest of any deployable deceleration device, are not adequate to meet the reliability requirements anticipated for this mission. A capsule design with a parachute would have to assure containment of the samples even in the event of parachute failure. In addition, inclusion of a parachute system introduces other failure modes such as premature deployment. A parachute system increases the landing footprint, and increases capsule ballistic coefficient and thus heating rates. Packaging of the parachute system in a sample return capsule complicates the robotic transfer of the samples into the capsule since both systems require volume allocations in the capsule aft- centerline position. All of these risks introduced by a parachute system must be compared to the benefit derived from the decrease in landing speed provided. If higher impact speeds can be tolerated, the parachute system is unnecessary and should not be included.

Impact tests conducted on materials representative of Mars samples indicated that the mechanical loads associated with ground impact without parachute deceleration do not degrade the scientific quality of the samples. The simplest approach is then to omit the parachute terminal descent system and replace it with a passive energy absorbing material to cushion the Mars samples during ground impact.

Removing the parachute deceleration system introduces the challenge of assuring containment during the ensuing surface impact. This impact problem is simplified if water is selected as the target surface. This simplification however, is negated by the risks introduced relative to the recovery of the capsule from a water landing sight. The search area for a water landing grows with time due to water currents. Sinking in deep water would subject the sample containers to large pressure loads. Loss of the capsule must be assumed to be loss of containment of the samples. When the impact and the recovery problem are integrated, the challenges associated with the ground impact problem are easier to manage than the combined water impact and water recovery risks.

Direct entry and ground impact of a passive capsule that does not include a parachute terminal descent system but relies solely on aerodynamics for deceleration and attitude control represents the most reliable entry scenario. The samples, in such a design, are packaged in a set of diverse, hardened container(s) and surrounded by sufficient energy-absorbing material to limit loading during ground impact.

Entry, Descent, and Landing Risk Areas

A probabilistic risk assessment[10] of the direct entry of a passive entry capsule with no parachute terminal descent system identified six risk areas. In chronological order they are: 1) accuracy of the position, velocity vector and attitude with which the capsule is delivered to the edge of the atmosphere, 2) performance of the thermal protection system (TPS), 3) performance of the capsule's supporting structures under the aerodynamic deceleration loads, 4) aerodynamic stability in all flight regimes, 5) performance of the ground impact energy absorption system and 6) location and recovery of the landed capsule.

After a brief discussion of the EEV design drivers and a description of the suggested design, the features of the design relative to each of these risk areas are discussed.

System Requirements and Design Drivers

The driving requirement on the Earth-entry capsule is to assure containment of the Mars samples during the intense Earth entry, descent and landing phases of the mission. Shape, size, mass and strength of the sample canister are major drivers in the design of the Earth-entry system. For this study, the canisters are assumed to be spherical with diameter of 0.16 m and mass of 3.6 kg. These enclosures which consist of a 3 layer diverse seal and container set are capable of handling mechanical loads of 2500 g's without degrading the science quality of the samples and 3500 g's without loss of containment. (In this paper, canister refers to the hardened containers of the samples, while capsule refers to the entire entry vehicle.) At Earth return, the capsule is spun up and released from the host spacecraft just prior to Earth entry. After traversing hypersonic, supersonic, transonic and subsonic speed regimes the capsule will impact the ground traveling at subsonic, terminal velocity. During impact, the system must limit mechanical loads on the sample canisters below predetermined values.

The capsule's relative entry velocity at 125 km altitude is between 11 and 12 km/s, depending on the Earth return trajectory. This high-energy entry drives the design to a blunt aeroshell with an ablating heat shield to protect the vehicle from the heating environment expected in the first 30 seconds of the entry. Aerodynamic decelerations of 130 Earth g's occur during this portion of the entry. After 45 seconds, the capsule has decelerated to around Mach 1.0 and descended to 28 km altitude. For the remaining 310 seconds of the entry the capsule descends nearly vertically at subsonic speeds. Blunt aeroshell shapes which can survive the heating of the hypersonic heat-pulse, often suffer aerodynamic stability problems in the transonic and subsonic regimes. The conflicting requirements of minimizing heating while maximizing subsonic aerodynamic stability is a major design trade in selecting the aeroshell shape.

Surface winds are also a design driver. The capsule must be designed to accommodate sustained surface winds at impact. Winds increase the impact energy and can produce off-axis impact angles.

Suggested Design

Figure 1 presents a schematic of a design based on a 0.9 m diameter, spherically blunted, 60-degree half-angle cone forebody. The sample canister is inserted into the capsule via the removable hemispherical afterbody. Once inserted, the samples canister is encased within energy absorbing material. The primary structure is a 2.0 mm carbon-carbon shell. The forebody heatshield is 0.012 m thick carbon-phenolic.

Canister transfer access, attachment hard points, vents and electrical connections are positioned in the lower heating region of the afterbody. It is beyond the scope of this paper to discuss all of the trades that were examined in evolving this design. The remainder of this paper describes the features of the design relative to each of the six risk areas mentioned above.

State at Atmospheric Interface

The velocity vector, position and attitude of the EEV at atmospheric interface - the EEV's state - affects many aspects of the ensuing entry, descent and landing. This state is established by the host spacecraft prior to EEV ejection. The planetary protection requirements necessitate landing within a controlled recovery area. Errors in position or velocity vector lead to offsets in landing location. In addition, an entry that is too steep can lead to peak heating rates and peak aerodynamic decelerations which exceed the qualifications of the heatshield material or the aeroshell structure. An entry that is too shallow can lead to integrated heat load which produces heatshield backface temperatures in excess of design requirements.

It is not possible to design an entry system with a high degree of reliability in entry, descent, landing and recovery unless the state at atmospheric interface is accurate with similar degrees of reliability. In the present design, this reliability is obtained by appropriate operational constraints placed on the carrier spacecraft during the mission phases leading up to EEV separation.

The desired attitude of the EEV at atmospheric interface is nose forward with only a small angle between the velocity vector and the capsule's axis of symmetry. This attitude is obtained by the host spacecraft prior to separation and maintained for that portion of flight between separation and atmospheric interface by imparting a spin to the capsule during the separation event. A failure in this spin-eject mechanism could lead to errors in the capsule's attitude at atmospheric interface. The worst case leads to a backwards EEV at atmospheric interface. (Most blunt entry vehicles are stable in both forward and backward orientation such that an initial backward orientation may be maintained throughout entry.) To mitigate this concern, the present design incorporates a concave afterbody that is unstable backwards in the free molecular flow regime at the edge of the atmosphere. Tests were also conducted in the NASA Langley Mach 6 CF4 and Mach 20 Helium wind tunnels to establish the backward stability of the design in the lower altitude continuum flow regime. Figure 2 shows a 0.113 scale model during these tests. In this flow regime the capsule is stable backward but the stability is bounded to a small angle-of-attack region around the backward orientation. By combining the free-molecular aerodynamics with the continuum aerodynamics linked via a bridging function, a six-degree-of-freedom atmospheric flight dynamics analysis was performed. The analysis indicates that, with proper placement of the vehicle center-of-gravity, the capsule will reorient itself to a forward orientation prior to appreciable heating as shown in Figure 3.

Thermal Protection System

The function of the TPS is to protect underlying capsule structures from the entry heating environment. It accomplishes this by preventing direct exposure of those structures to the flowfield and by limiting the conduction of the associated thermal energy through its thickness. The thermal protection system (TPS) includes the forebody heatshield, the afterbody thermal protective layer and any penetrations to those coverings. For non-reusable ballistic capsules, ablative TPS materials are typically selected which provide the required protection while minimizing mass. Several low density developmental systems exist which possess sufficient performance capabilities for the EEV. However, the strict containment assurance requirements necessitate utilization of a system with significant test and fight heritage. This heritage is necessary to assure potential failure modes are well understood. In general, TPS failure modes can be classified as catastrophic burn through failures and bondline over-temperature failures. While the probability of bondline over-temperature can be reduced by increased TPS thickness, catastrophic burn through failures require sufficient test and heritage to demonstrate all possible failure mechanism have been revealed. None of the low density systems available possess the required heritage.

Fully dense carbon-phenolic is used on the nosecone of ballistic missile entry vehicles. It is used on the Space Shuttle Solid Rocket Motor nozzles as well as several other rocket nozzles. It was also used as the forebody heatshield on the successful Galileo and Pioneer Venus entry capsules. Its demonstrated performance capabilities far exceed those required for the EEV. It has been extensively tested in ground based facilities and in flight. Improper ply alignment relative to the surface is the only identified failure mechanism that may lead to burn through type failure modes. A substantial test program has been conducted and this mechanism is now well understood. The combined number of ground and flight test experience of the material is of order 10^4. This data is being collected and its relevancy to demonstrating the reliability relative to the catastrophic burn through is being assessed via a probabilistic risk assessment.

A carbon-phenolic heatshield is massive relative to available low-density ablators. Selection of carbon-phenolic is responsible for the factor of two increase in capsule mass necessary to attain the required reliability.

Selection of the optimal flight path angle for the entry is a trade among entry heat rate, integrated heat load, and deceleration loads. Steeper flight path angles include higher peak heating rate and deceleration loads but lower integrated heating. Since the integrated heat load determines the TPS thickness (and thus mass) there is a desire to enter steep (provided the TPS materials can handle the peak rate). Since appropriately manufactured carbon-phenolic can survive the rates associated with a 90 degree flight path angle (straight down), the steepness is not limited by the material's capabilities. Steep entries also decrease the landing dispersions caused by atmospheric density and drag uncertainties. Unfortunately, current ground test facilities are limited in their ability to simulate the heating rates associated with steep entry trajectories. At present, the flight path angle is constrained by the ability to flight qualify the heatshield material in ground-based facilities not by TPS performance or capability.

Sizing the TPS thickness for the EEV requires knowledge of 1) the heating environment, 2) the material properties of the TPS material, 3) the response of that material to the imposed environment, and 4) a bondline temperature criteria. Figure 4 presents the stagnation point, laminar entry heatpulse and aerodynamic deceleration pulse for a nominal entry of the EEV. Figure 5 presents the associated bondline temperature prediction from material response analysis for a 0.012 m layer of carbon-phenolic subjected to this environment. Since the underlying structure is high temperature capable carbon-carbon composite, the bondline temperature limit is defined by the RTV 560 adhesive (370 C). In fact, only 0.078 m of carbon-phenolic is required to limit bondline temperatures below the 370 C limit. A thermal analysis of the design is presented in Ref. 11.

As with any design problem, there exists uncertainty with respect to the predictive analysis of the environment and the material's response to that environment. Traditionally, an additional thickness margin is added to the TPS sizing in an attempt to cover these uncertainties and decrease the probability of failure. The numeric requirements of the probabilistic risk assessment necessitate increased rigor in establishing the proper margin. The challenge in sizing the TPS then becomes one of understanding these uncertainties and then combining them into a probabilistic simulation to determine the required thickness margin necessary to assure the specified degree of reliability.

The largest uncertainty in entry heating environment surrounds the question of boundary layer transition to turbulence. For the present carbon-phenolic heatshield, transition to turbulence would most likely be caused by roughness effects. An indicator of transition is then the Roughness Height Reynolds number. Computational Fluid Dynamics solutions can be utilized to extract values for this parameter associated with different roughness values for the material. The present assessment indicates that transition does not occur during the high heating portion of the heatpulse. However, there exists considerable uncertainty in the expected surface roughness, the accuracy of extracting values for the roughness height Reynolds number, and the appropriate critical value to indicate transition for this particular case. Therefore, the probabilistic approach to determining thickness margins discussed above will be used.

The afterbody thermal protection material must meet the same reliability requirements as the forebody heatshield material. For the present design, 1.0 cm of SLA-561V is designated for this material. SLA-561V was the forebody material for the Mars Viking and Pathfinder missions. It is also the afterbody material on the Stardust and Genesis Sample return capsule. This material has significant ground test and flight experience. However, the number of tests is much less than that for carbon-phenolic. In the present design, the SLA-561V covers a carbon-carbon composite structure. Afterbody heating rates for this design are predicted to be around 1-2 percent of the forebody. At a maximum they should remain less than 6 percent of the forebody. In this environment, the carbon-carbon composite possesses significant capabilities to perform as a redundant heatshield in the event of burn through failure of the initial SLA-561V covering.

Support Structures

For the present discussion, the support structures include the aeroshell structure beneath the forebody heatshield, the structure beneath the aftbody heatshield and the impact sphere shell that encases the energy absorber. The material currently selected for these structures is carbon-carbon composite. This selection is driven by the need to further mitigate the threat of bondline over temperature leading to collapse of the capsule shape. The design loads for this structure include launch loads and the 130 g aerodynamic deceleration loads associated with the selected flight path angle. These structures play no critical role during ground impact.

These structures have both strength and stiffness requirements. Their strength must be adequate to preserve the capsule's projected drag shape during flight. Collapse of the aeroshell during flight will decrease drag that will result in large landing dispersions. Collapse of the capsule structure during the intense heating phase could also subject internals (including the sample container) to extreme heating rates. The structures' stiffness requirements are driven by the need to limit structural loading to the overlying TPS layers and their bond joints.

The challenge to designing this composite structure to the required degree of reliability is again a challenge of determining the appropriate margins to be applied. The margins must cover uncertainties in the loading environment, the material's response to that environment and variations in the material's properties. This requires some understanding of the propagation of failures analogous to fracture mechanics that has been studied extensively for metals. A combination of data collection, analysis and test is underway to identify characteristic flaws in the manufactured structure, how these flaws can grow towards failure and how the combination of launch and entry loads could combine to lead to failure. These simulations and tests are combined into a probabilistic analysis to establish the required margins.

Aerodynamics and Flight Dynamics

In addition to the ability to reorient itself from any attitude at atmospheric interface, the EEV must possess sufficient stability to remain in controlled flight through hypersonic, supersonic, transonic and subsonic flow regimes. Of all these regimes, subsonic flow places the greatest restrictions on center-of-gravity (c.g.)[12]. The aerodynamic drag in each of these flow regimes must also be understood with sufficient accuracy to permit landing location determination.

The aerodynamic drag and stability of the candidate 60-degree half-angle spherically-blunted cone forebody has been studied extensively in the higher speed regimes. It's characteristics in subsonic flow is less well understood. A series of subsonic, free flight tests in the NASA LaRC 20 ft Vertical Wind Tunnel have been conducted. In addition, two full scale aerodynamic drop tests were performed. The selection of the 60 degree half-angle cone aeroshell was the result of a trade among hypersonic drag (heating), subsonic drag (impact velocity) and subsonic stability (available crush stroke).

Aerodynamic stability is a function of aeroshell shape and mass properties. A solid model of the design shown in Fig. 1 predicts the c.g. to be 0.155 m back from the nose. Aerodynamic stability is comprised of a static and dynamic component. For static stability, the slope of the moment curve at this c.g. location, $C_{m,\alpha}$ must be negative at the trim angle of attack (0^0 for this design). Static stability is highest in the hypersonic region (large negative $C_{m,\alpha}$). Static stability decreases below Mach 12 as the sonic line jumps from the nose to the shoulder of the vehicle. In addition to the decrease in static stability indicated by $C_{m,\alpha}$, dynamic stability decreases at lower speeds and can become unstable in the transonic and subsonic flight regimes. If a vehicle is stable in the low subsonic speed regime, it will typically be stable at higher speeds. Figure 6 shows the full-scale drop model tested at UTTR. Figure 7 shows the resulting attitudes measured from 3-axis accelerometers during the test. After the model accelerated to terminal velocity, the maximum amplitude of oscillations was 15 degrees.

Knowledge of the aerodynamics of the vehicle in all flight regimes is combined to form an aerodynamics database. This is then merged with Earth atmospheric and gravity models to form a 6-degree-of-freedom atmospheric flight dynamics simulation that calculates the trajectory from an initial state at the edge of the atmosphere to landing. When appropriate uncertainties are specified on each of these inputs, a Monte-Carlo simulation can be performed to statistically assess entry environments, flight dynamics and landing location.

Ground Impact

At landing, a complex interaction of events occurs whose sum is the removal of the capsule's remaining kinetic energy. Energy is absorbed by ground deformation, heatshield breakage, deformation and failure of the capsule structures and by crush of the energy absorbing material. The function of the impact energy absorption system is to limit mechanical loads on the sample canister during landing. In particular, the sample canister accelerations during impact must not exceed 2500 g's to preserve the scientific integrity of the samples and must not be subjected to greater than 3500 g's to prevent rupture of the container.

At present, the Utah Test and Training Range (UTTR) is being considered as a landing sight. This location is the largest combined ground and air space in North America controlled by the U.S. military. The site is also being used for the Stardust and Genesis Sample Return missions.

The EEV obtains very high reliability of containment during the impact event through a combination of the energy absorbing characteristics of the UTTR clay surface in conjunction with an on-board energy absorber. In addition, removal of the traditional parachute descent system, guidance and control system and other unnecessary systems decreases the mass of the design shown in Figure 1 such that ground impact occurs at the low subsonic speed of 40 m/s. The mass of the EEV is 42 kg. Ground characterization tests have been conducted at UTTR on four different occasions that included dropping instrumented penetrometers from cranes, hot-air balloons and helicopters at different impact speeds. In addition, a full-scale model of the design was dropped onto the UTTR surface. Figure 8 presents the accelerations experienced by this rigid model during ground impact. These tests reveal the surface will deform sufficiently during impact to limit loads to the sample container below the 2500 g limit. A dynamic finite element model of the impacting surface has also been created in DYTRAN. This model, which has been validated against the ground impact data collected at UTTR, will be used to examine off nominal impact conditions.

The dimensions of the current landing footprint are 33 km in downrange by 16 km in cross-range. This footprint is generated by variation of the initial state vector, the aerodynamic drag of the vehicle and the atmospheric density. The footprint is easily positioned within the 63 by 28 km ellipse of uniform clay surface available at UTTR for this approach azimuth. A site survey was conducted at UTTR which included numerous low speed impacts, rock distribution and size surveys and extensive photography of the surrounding area outside of the predominant clay. This data is being combined with USGS maps and aerial photography to produce a surface model of the areas surrounding the UTTR range. In the unlikely event the EEV lands outside of the clay surface area, an energy absorber is included that limits loads to the sample canister below 3500 g's even if the capsule impacts a concrete surface. This energy absorber is constructed of carbon-foam cells encased in carbon-fiber and Kevlar composite in an orientation which resembles a filled radial honeycomb. The performance of the absorber can be tuned to specific crush strengths by varying the lay-up and thickness of the web material. The absorber can also be adjusted to handle irregular surfaces such as rock by including a Kevlar shell. This concept has been tested at LaRC in accelerated drops onto concrete surface. Figure 10 compares the canister accelerations measured during a test of this energy absorber with pre-test numerical simulations during a concrete surface impact.

The statistical prediction of the landing location, the UTTR site survey, the ground characterization tests and the capabilities of the vehicle to survive concrete surface impacts are integrated into the probabilistic risk assessment to demonstrate very high reliability during the impact event.

Recovery

The challenge of assuring Mars sample containment does not end with the EEV ground impact event. The capsule must be located, recovered and transported to a receiving facility. This requirement discourages water landing as discussed previously.

Recovery begins prior to impact through tracking of the capsule during terminal descent. Radar and infrared tracking will be possible as the capsule approaches the landing site. In addition, analytic simulation will use the known location of the capsule from exoatmospheric host separation to predict a ground impact location.

After ground impact, the capsule will provide a detectable infrared signature for several hours. In addition, the capsule is equipped with a pair of independent 242 MHz ground location radio beacons that permit triangulation from multiple receivers. The expected duration of these beacons is several days following impact. Finally, if the featureless UTTR site is utilized, visual search should offer a simple means for locating the capsule. Once located, the capsule will be placed within a hardened container and transported to the receiving facility. Considerable attention is being given to selection of safe transport.

Conclusions

The driving requirement for design of a Mars Sample return mission is assuring containment of the returned samples to a very high degree of reliability. The impact of this requirement on developmental costs, mass allocations and design approach of the Earth Entry Vehicle is significant. A direct entry approach for this final mission phase requires successful completion of a factor of 4 to 10 fewer critical events than a low Earth orbit rendezvous approach. The capsule for this direct entry must be simple and reliable. Reliability is achieved by eliminating all nonessential subsystems and utilizing heritage systems with sufficient margin or redundancy for each critical subsystem.

The suggested capsule design has features to mitigate risks associated with each of the critical entry, descent, landing, and recovery phases. It is an entirely passive vehicle which relies solely on aerodynamics for deceleration and attitude control. It avoids the potential failure modes of a parachute terminal descent system by replacing that system with sufficient energy absorbing material to cushion the sample containers during ground impact. Full scale impact testing has revealed that this energy absorber is not needed if the capsule lands within the currently predicted footprint but is carried to mitigate possible errors in landing determination. The capsule has the ability to reorient itself in hypersonic flight in the event that there is a failure during spin-eject from the host spacecraft. The forebody heatshield is made of carbon-phenolic for which extensive ground test and flight data exists. Its structure is made of high-temperature carbon-carbon composite as mitigation against poor heatshield performance. The capsule also contains multiple layers of containment for the samples. Finally, recovery can be accomplished via infrared and radar tracking, infrared ground search, visual search and by triangulation of the onboard radio beacons.

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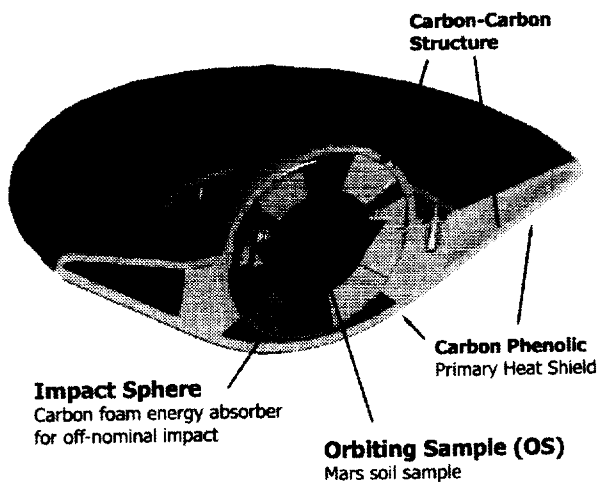

Figure 1: Schematic of entirely passive Earth Entry Vehicle.

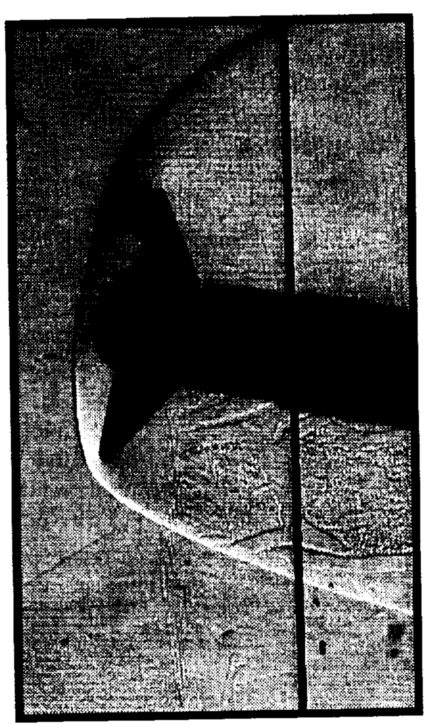

Figure 2: Hypersonic Reorientation aerodynamic test in the NASA LaRC Mach 6 CF4 tunnel (Model in backward orientation)

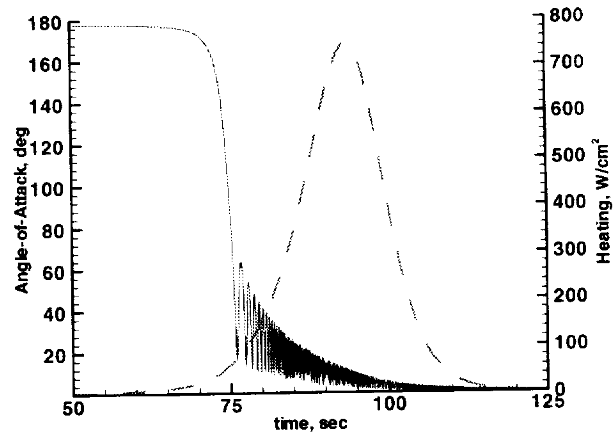

Figure 3: Six Degree of Freedom analysis illustrating the hypersonic reorientation capability of the capsule.

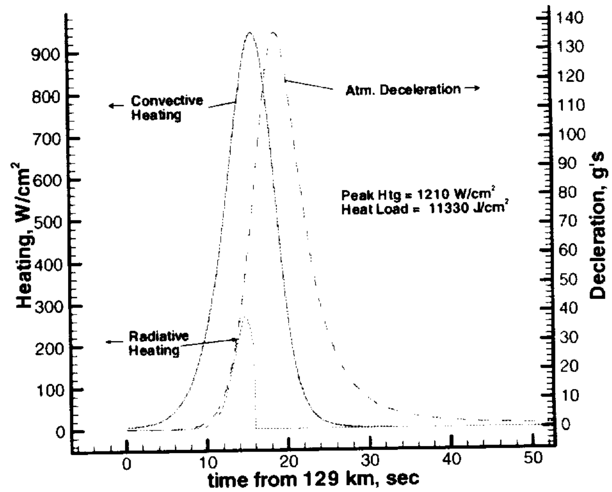

Figure 4: Nominal entry heating and deceleration pulses.

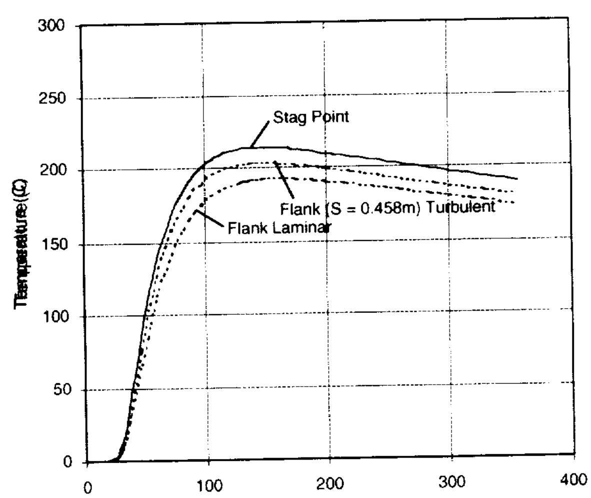

Figure 5: Bondline temperature from material response calculation on 1.2 cm carbon-phenolic heatshield.

Figure 6: The full scale drop model beside its impact crater at the Utah Test and Training Range

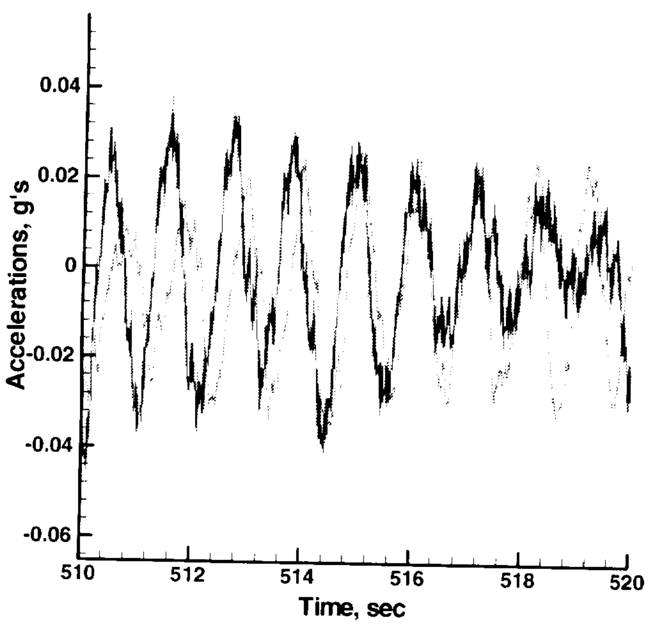

Figure 7: Three-axis accelerometer data during terminal descent flight of full scale drop model at Utah Test and Training Range.

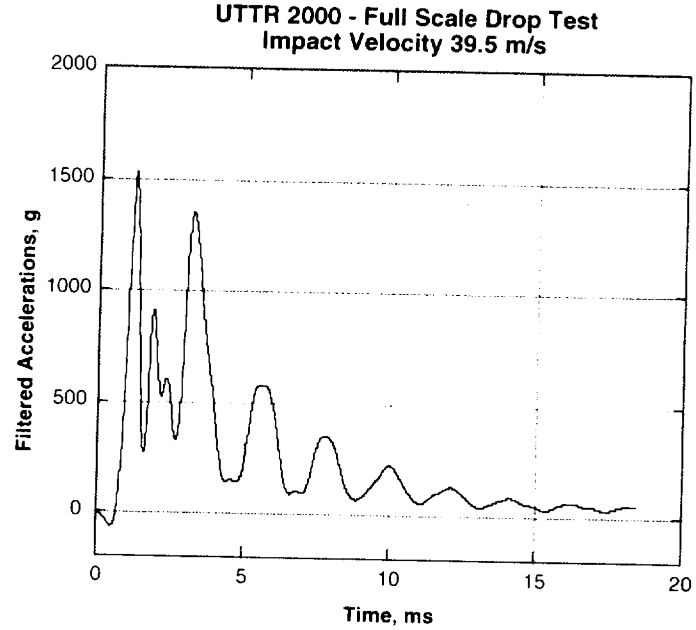

Figure 8. Impact accelerations measured during the ground impact of the full scale model at UTTR.

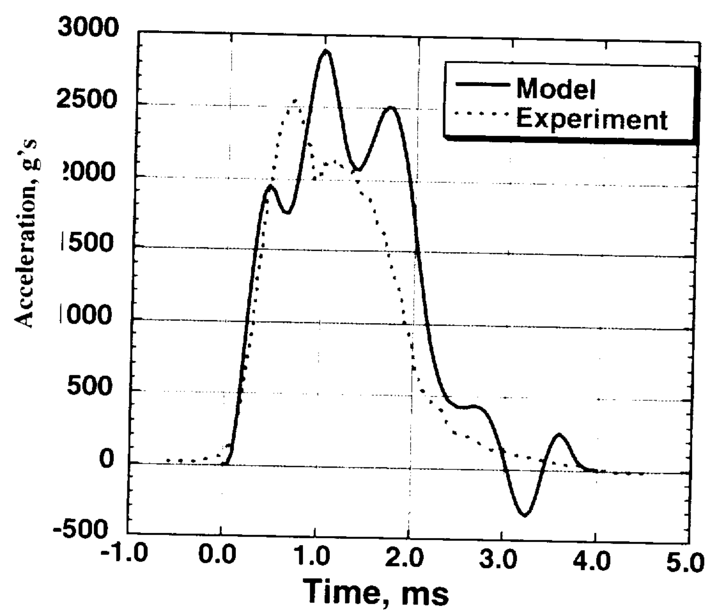

Figure 9: Comparison of dynamic finite element model prediction with measured impact accelerations associated with concrete surface impact.