# Development and Test Plans for the MSR EEV


**Robert Dillman[(1)], Bernard Laub[(2)], Sotiris Kellas[(3)], Mark Schoenenberger[(4)]**

*[(1)] NASA Langley, MS 472, Hampton VA 23681, USA; Email: Robert.A.Dillman@nasa.gov*
*[(2)] NASA Ames, MS 234-1, Moffet Field CA 94035, USA; Email: Bernard.Laub@nasa.gov*
*[(3)] General Dynamics, MS 495, Hampton VA 23681, USA; Email: S.Kellas@larc.nasa.gov*
*[(4)] NASA Langley, MS 365, Hampton VA 23681, USA; Email: Mark.Schoenenberger-1@nasa.gov*



ABSTRACT

The goal of the proposed Mars Sample Return mission is to bring samples from the surface of Mars back to Earth for thorough examination and analysis. The Earth Entry Vehicle is the passive entry body designed to protect the sample container from entry heating and deceleration loads during descent through the Earth's atmosphere to a recoverable location on the surface. This paper summarizes the entry vehicle design and outlines the subsystem development and testing currently planned in preparation for an entry vehicle flight test in 2010 and mission launch in 2013. Planned efforts are discussed for the areas of the thermal protection system, vehicle trajectory, aerodynamics and aerothermodynamics, impact energy absorption, structure and mechanisms, and the entry vehicle flight test.


## 1. INTRODUCTION

The overall Mars Sample Return (MSR) mission scenario [1] includes a Mars lander to place surface samples into a container with redundant seals, a small rocket to raise the container into low Mars orbit, and an orbiting spacecraft to capture this payload and insert it into the Earth Entry Vehicle (EEV). The Earth-return portion of the spacecraft then carries the EEV toward Earth on a near-miss trajectory; before passing Earth the EEV is released on an 11-12 km/s entry trajectory. The landing site has not been officially selected, but should include controlled ground- and air-space covering a large area of predominantly soft terrain, such as found at several military installations including the Utah Test and Training Range (UTTR) selected for Genesis and Stardust [2]. After landing and recovery, the EEV and the enclosed sample container are then transported to a dedicated sample handling facility, the design of which is under study by the JPL Mars Program.

As current plans do not call for sterilization of the samples before landing on Earth, containment of the returned materials is necessary for protection of the terrestrial environment. The NASA Planetary Protection Officer has established a draft containment assurance requirement calling for the probability of release of a Martian particle larger than 2.0 microns into Earth's biosphere to be less than $10^{-6}$. This is orders of magnitude beyond the reliability requirements levied on any previous planetary entry system [3], and has driven many aspects of the EEV design [4]. For example, the original forward thermal protection system (TPS) used state-of-the-art low density materials; however, these lacked sufficient flight heritage to achieve the desired vehicle reliability, and were replaced with fully dense carbon-phenolic – while much heavier, carbon-phenolic has extensive flight history and well-understood performance. Similarly, the traditional parachute for terminal descent was removed; to reach $10^{-6}$ the vehicle needed to maintain sample containment even during a hard landing after parachute deployment failure, so the EEV design was made robust enough to tolerate non-parachute landing as the nominal case. Removal of the parachute also deleted the associated deployment mortar, which had its own set of failure modes that contributed to the total system risk. The probabilistic risk assessment (PRA) used to track the overall probability of loss of containment assurance [5] was a vitally important tool for making these design decisions.

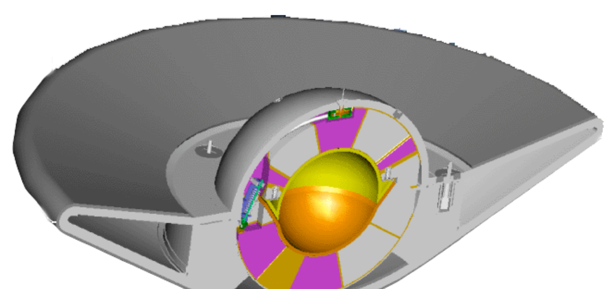

Fig. 1. Solid Model of the MSR EEV

The current MSR EEV is a 0.9 m diameter blunt body with an entry mass of 42 kg, including 0.5 kg for the Mars samples, and multiple layers of containment and protection. The vehicle forebody is a 60° half-angle cone with a spherical nose; the aft side is concave, with a central hemispherical lid that latches in place after insertion of the sample container. The JPL-produced sample container fits in the center of the vehicle, inside a flexible containment vessel that is sealed in Mars orbit before launch toward Earth, and both of these components are designed to accommodate higher loads than those seen during landing on clay, sand, or soil. Wrapped around the containment vessel and sample container is a spherical impact energy absorber, which limits their deceleration load if the vehicle lands on a harder surface. This vehicle geometry is preliminary and subject to change, as we are years away from launch, but the development and test plans have been laid out based on this configuration.

## 2. THERMAL PROTECTION SYSTEM

Fully dense carbon-phenolic (CP) was chosen for the EEV forward TPS based on its extensive flight heritage; it has been through thousands of tests and used on hundreds of flights, for missile re-entry heat shields, for the Shuttle solid rocket nozzle throats, and for the Galileo and Pioneer Venus probe heat shields. However, the heritage manufacturing processes that were used to fabricate the heat shield nose caps for the interplanetary missions are not fully documented. The methods used for the tape-wrapped CP used on the conical flank of the vehicles are well known, but there are gaps in the process information for the chopped-molded CP used at the stagnation point.

Upcoming TPS efforts focus on recreating the missing steps of the chopped-molded heritage processes. In 2005 several different chopped-molded CP samples will be fabricated, using different combinations and variations of the available processes, to see which approach reproduces the heritage capabilities. Sample performance will be tested in the Ames arc jet facilities, and the mechanical and thermal properties will also be compared. Once the heritage chopped-molded CP has been reproduced, the methods used will be thoroughly documented for future use.

An aft TPS capability and heritage survey is also planned for 2005, along with the selection of a preferred aft TPS material and identification of any associated design and test requirements. Earlier MSR EEV designs carried a nominal 10 mm aft TPS thickness instead of completing the selection of a specific TPS, as funding for this survey was previously unavailable.

In 2006 we plan to perform several TPS thickness studies for the forward and aft heat shields, based on updates to the entry trajectory and Monte Carlo estimation of the worst case heat load. We also plan to create detailed designs of the various TPS joints and penetrations on the vehicle. These TPS joints include those between the nose and flank CP materials, the flank CP to the aft TPS, and the seam in the aft TPS where the lid opens and closes. The penetrations are all in the aft TPS, and include those needed for the EEV mechanical attachment to the parent spacecraft as well as ones for the electrical cable bundles carrying sensor info and survival heater power.

In 2007 we plan to conduct arc jet tests for each of the TPS joints and penetrations, and also for vehicle locations of interest such as the forward and aft stagnation point and the vehicle shoulder. Each configuration will be tested at least four times: at least two samples will be exposed to their expected peak heat flux plus margin to prove material survival, and at least two more will see half that level, which better matches the energy absorption into the bulk of the TPS. At both of these test levels, the test duration will be calculated to input the full entry heat load into the sample coupons. Some 60-plus TPS coupons will be fabricated, tested, inspected, and analyzed as part of this test series, at a cost of over a million dollars.

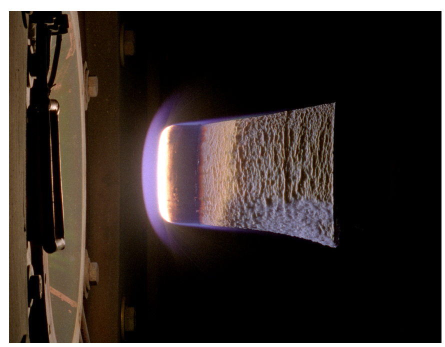

Fig. 2. TPS Arc Jet Testing

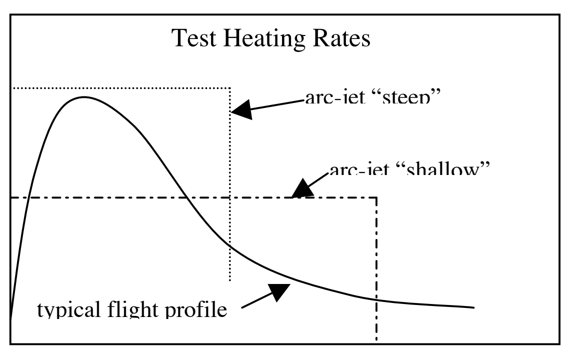


Fig. 3. Plot of Heat Flux (Q) v. Time

In 2008, a full scale engineering model of the TPS will be fabricated using the flight designs and heritage processes. After assembly to an engineering model of the EEV structure, it will undergo environmental testing and post-test inspection to look for any unexpected problems with the full-scale fabrication processes that were not evident on sample coupons.

Once detail designs using heritage materials have been tested and qualified for flight, TPS efforts will focus on preparation for the anticipated EEV flight test, discussed later in this paper. Redesign and TPS re-sizing to accommodate changes in mission requirements, vehicle configuration, and predicted heat load will likely continue until an eventual mission design freeze, but at a lower level of effort than the initial development tasks.

## 3. TRAJECTORY, AERO, AND AEROTHERMAL

There are no active systems on board the EEV during entry, descent, and landing, except for radio tracking beacons. An active guidance system was avoided due to the risks posed by potential failure modes; the vehicle relies instead on an accurate initial trajectory and the simple physical laws of ballistics, which still produce an acceptably small landing ellipse.

The concave aft shape of the EEV was sculpted to avoid the possibility of stable backwards orientations during re-entry. The vehicle is intended to be pointed nose-first at atmospheric interface, with a 2 rpm spin for stability, but possible failures of the spin-eject system on the parent spacecraft may lead to off-nominal entry conditions. Trajectory simulations of Earth entry performed using preliminary computational fluid dynamics and direct simulation Monte Carlo aero data show the vehicle to be self-reorienting from off-nominal entry states. Even from the extreme case of a backwards entry (180° angle of attack) with full spin-stabilization, the EEV pitches over to a forward orientation before the heat pulse.

Aerothermal calculations of the heat flux distribution around the EEV indicate that the coolest regions during entry will be the aft surfaces where the EEV lid joins the body [6], which simplifies the design of TPS joint and nearby mechanical penetrations. However, concerns about Mars dust reaching orbit with the sample container and possibly contaminating the outside of the EEV have led to an upcoming re-design. The aft body of the EEV will be re-examined to see if shape changes can raise the entry flux high enough to push the surface temperature past the 500°C sterilization level while maintaining the vehicle reorientation capability.

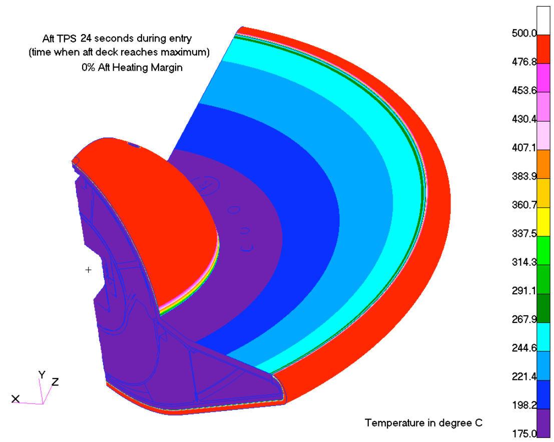


Fig. 4. EEV Entry Temperatures

Development plans for 2005 include updating heating and footprint calculations for a range of entry trajectories and the aerothermal calculations for the altered aft body shapes. In 2006 we plan to conduct several test runs in the Langley 20-inch Mach 6 air tunnel to anchor the aerothermal calculations for the most promising shapes, and some limited testing in larger facilities. Later in 2006 we plan to test the dynamic aero performance of the EEV, using a combination of ballistic range tests, spin tunnel tests, and drop tests of a full size EEV model. In 2007, we produce an updated aero database for the EEV and perform Monte Carlo trajectory runs across a range of possible entry conditions to confirm vehicle reorientation and evaluate the worst case entry conditions. Tasks in 2008-9 focus on updating the entry predictions as the vehicle design matures, and on providing analytic support for the EEV flight test.

## 4. IMPACT ENERGY ABSORBER

The EEV impact energy absorber has three main components: a relatively rigid inner shell, a crushable foam-filled cellular structure, and a tough outer shell for penetration resistance. In the nominal landing scenario, the vehicle will land in a well characterized region of mostly soft terrain, such as UTTR where the clay and sand are interrupted by only a few gravel roads and small concrete pads. For this case, the EEV's kinetic energy at impact is absorbed by deformation of the ground, as well as crushing and fracture of the vehicle structure and TPS. Full scale drop tests conducted at UTTR, using rigid, instrumented penetrometers and a rigid model of the EEV, showed a deceleration load of

1500 g's, well below the 2500 g requirement for preservation of the scientific value of the samples.

Fig. 5. UTTR Terrain

Fig. 6. Full Scale EEV Drop Model

For the off-nominal case of a hard surface landing, the impact energy absorber is designed to limit the loads at the interface to the sample container and containment vessel to less than 3500 g's [7]. In this case, some science degradation is expected, but sample containment is still maintained. The cell walls crush and tear to attenuate the impact loads, but the inner shell of the absorber remains intact. Full size impact tests at NASA Langley using a simulated sample container produced loads of under 3000 g's; a mechanical accelerator system was necessary to achieve the desired impact velocity of 41 m/s, which was higher than terminal velocity in the dense sea level air, but the impact absorbers repeatedly performed as intended.

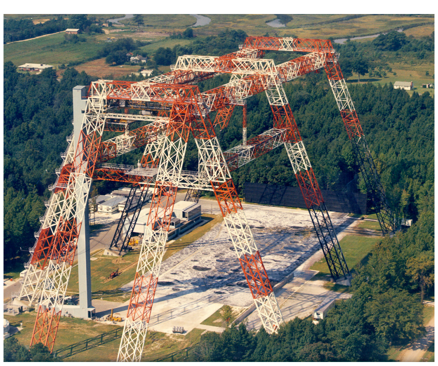

Fig. 7. Langley Impact Dynamics Research Facility

Fig. 8. Impact Absorber, Post-Test

Development plans for 2005 call for comparison of the existing composite energy absorber to an alternate metallic design and investigation of fabrication methods applicable to the metallic absorber. This is in response to an ongoing study on the risk effects of switching from a composite structure to a metallic one, due to earlier difficulties in analyzing the composite structure for $10^{-6}$ reliability – if the vehicle structure is switched from composite to metallic, it may be beneficial to change the energy absorber as well. In 2006 we will perform impact speed crush tests on our flight materials to generate data necessary for design of the absorber; the current engineering development models used non-flight materials to reduce expenses and simplify fabrication. In 2007 we begin conducting tests again of the full size impact absorber, this time with flight materials; two tests are planned with empty impact hemispheres, and two more are planned carrying flight-like sample containers. In 2008 we plan to impact test the absorber inside a full-size model of the vehicle structure, to verify analysis of the absorber-structure interactions and the impact behavior of the structure. This data is necessary to properly size the energy absorber; we need to know how much of the structural mass breaks free on a hard surface impact, and how much stays attached and must be decelerated by the energy absorber. Later in 2008 we plan to impact test another absorber inside a flight-like structure, with simulated TPS mass and a flight-like sample container. These full-up tests are designed to confirm that the absorber works as intended when assembled with the flight hardware, since the impact absorber will not be included in the EEV flight test.

Test results will be used to verify impact analysis models which could then be used for further design adjustments, and/or design modifications in the event of significant changes to the mission requirements. For example, if the Mars sample mass increases significantly, the overall EEV diameter will grow, and the earlier drop tests onto the ground at UTTR may be invalidated.

## 5. STRUCTURE AND MECHANISMS

The EEV structure supports the heat shield and maintains the vehicle drag area to achieve the desired terminal descent velocity. The structure must survive the 130 g atmospheric interface deceleration, but is not required to survive landing. The vehicle structure will thus be designed and tested to the entry loads, plus margin, rather than to the higher levels associated with ground impact.

Components intended to operate after launch from Earth, such as lid placement sensors, the lid latches, and the (retractable) launch lock bolts, will be vibration tested to the expected launch loads and then tested to verify proper operation across their expected thermal range. Mechanical components that must survive through entry and descent, such as the EEV body vents, will be tested to the atmospheric deceleration levels. The lid latches, which hold the lid and body of the impact absorber together, will see additional testing as part of the impact absorber. All EEV components will also go through vacuum thermal cycle testing to verify compatibility with the expected environment.

Preliminary design and analysis of a metallic structure for the EEV is expected in 2005 as part of the comparison between the current composite design and a metallic one. In 2005 we begin design of the lid latches and the launch lock bolts; they are planned for completion in 2006, along with the mechanical TPS penetrations, body vents, and lid closure sensors. In 2007 we will fabricate engineering models of the lid latches and EEV launch locks, for development work and environmental testing. In 2008 the engineering model of the EEV structure will be fabricated for integration and environmental testing with the TPS engineering model. In 2009 the development plans call for fabrication and assembly of flight versions of the EEV structure and mechanisms for use in the flight test in 2010.

Structural and thermal analysis of the various mechanical components will be conducted throughout the design process, using the relevant environments from launch, deep space, Mars orbit, trajectory maneuvers, and Earth entry. Structural and thermal analysis of the full EEV assembly will also be required, in the several different configurations experienced during the proposed MSR mission. Finally, as with the other subsystems, some redesign of the mechanical components will likely be required due to changes in mission requirements.




Fig. 9. EEV Cross Section

## 6. EEV FLIGHT TEST

The MSR EEV flight test is intended to functionally test critical aspects of the vehicle design that cannot be fully duplicated in ground testing and analysis. The flight will demonstrate the integrated TPS performance in the actual, time-varying flight environment, with the TPS interacting with the surrounding flow and with the underlying structure; ground testing in arc jets simulates only one heat flux level per test, and cannot match all the environmental variables. This test will also verify the EEV aero and aerothermal performance, confirm that critical risk parameters are within their design limits, and demonstrate that there are no unknown system-level issues.

The flight test is intended to validate the nominal vehicle performance, and as such will duplicate the EEV entry trajectory and vehicle size, shape, mass, and materials. This will allow the flight test to match the mission's entry trajectory, entry heating, deceleration, terminal velocity, and nominal landing at the chosen site. Rather than conducting a flight test using one particular set of extreme conditions, this test is intended to validate the performance models used in the PRA so that they can be reliably used for repeated Monte Carlo runs.

Given the low probability of landing on a hard surface, the flight test is unlikely to prove the performance of the impact energy absorber. In order to provide more useful data from the components under stress during this test, the sample container and impact absorber will be replaced, for this test only, with a high-g data recorder, an inertial measurement unit to track the vehicle trajectory, numerous thermocouples, and several pressure sensors. The impact sphere and sample container will receive sufficient testing outside the flight test for the overall test program to cover the entire mission scenario.

However, the future of the proposed EEV flight test is still uncertain. A 2001 study by Sandia looking at relevant launch vehicles concluded that the flight test would cost roughly $30 million, mostly to buy a suitable launch vehicle. The high cost, as well as the possibility that a launch failure during this flight test could delay the launch of the MSR mission, has led to an effort to quantify the benefits of the flight test and to see if the same results can be achieved through expanded ground testing.

## 7. CHANGES AFTER FLIGHT TEST?

Assuming a successful flight test in 2010, there is debate about whether to allow changes to the EEV design before launch to Mars. Some concerns exist that changing the proven vehicle would invalidate the flight test heritage; however, minor changes should be allowable, as long as they do not require alteration of the analytical methods verified by the flight test.

The most serious requirements changes from the test flight to the interplanetary mission would be the increased mission time and the addition of planetary protection requirements. The mission duration may have limited effect on the space-rated materials used on the vehicle; the Galileo probe structure and heat shield, for example, flew through space for years before reaching Jupiter. Planetary protection, however, is a very significant change: the flight test has no extraordinary concerns about the presence of microorganisms, but the mission to Mars must deal with the possibility of Earth organisms contaminating the Mars samples, and as such will have to implement stringent hardware cleaning processes. Since the flight test vehicle is intended to be a duplicate of the Mars mission hardware, these cleaning processes will also have to be imposed on the flight test hardware.

## 8. CONCLUSION

Plans for the development and testing of the MSR EEV were outlined here, from present tasks through a flight demonstration and 2013 mission launch. It should be noted that all of these plans are preliminary works in progress which are expected to continue to change as the design matures and as requirements and funding constraints vary.

It should also be mentioned that there are several other MSR components under development at JPL which interact closely with the EEV subsystems discussed here, including the sample container, the flexible containment vessel, the spin-eject mechanism that releases the EEV from the parent spacecraft, and the micrometeoroid shield needed to protect the EEV during flight to and from Mars. Interface requirements for these and other MSR systems will need to be developed in the future, but are beyond the scope of this paper.

## 9. REFERENCES:


1. Gershman, R., Adams, M., Mattingly, R., Rohatgi, N., Corliss, J., Dillman, R., Fragola, J., and Minarick, J., *Planetary Protection for Mars Sample Return*, COSPAR-PTP1-0011-02, 34th COSPAR Scientific Assembly, October 2001.
2. Desai, P. N., Mitcheltree, R. A., and Cheatwood, F. M., *Sample Return Missions in the Coming Decade*, IAF Paper IAF-00-Q.2.03, October 2000.
3. NPG 8020.12B, *Planetary Protection Provisions for Robotic Extraterrestrial Missions*, April 1999.
4. Mitcheltree, R. A., Hughes, S. J., Dillman, R. A., and Teter, J. E., *An Earth Entry Vehicle for Returning Samples from Mars*, AAAF Paper ARVS-102, March 2001.
5. Fragola, J. R., Minarick, J. W., and Putney, B., *Mars Sample Return Probabilistic Risk Assessment Final Report*, Science Applications International Corporation Report, September 2002.
6. Amundsen, R. M., Dec, J. A., Mitcheltree, R. A., Lindell, M. C., and Dillman, R. A., *Preliminary Thermal Analysis of a Mars Sample Return Earth Entry Vehicle*, AIAA Paper 2000-2584, June 2000.
7. Kellas, S., and Mitcheltree, R. A., *Energy Absorber Design, Fabrication and Testing for a Passive Earth Entry Vehicle*, AIAA Paper 2002-1224, April 2002.