Aerothermal Modeling for Entry and Aerocapture

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***Abstract*–The current status of aerothermal modeling for Earth and planetary entry missions is discussed. For such missions the accuracy of our simulations is limited not by the tools and processes currently employed, but rather by correctable deficiencies in the underlying physical models. Improving the accuracy of the models and reducing the uncertainties in these models will enable a greater understanding of the system level impacts of a particular thermal protection system and of the system operation and risk over the operational life of the system. A methodology is laid out by which key aerothermal uncertainties can be identified via mission-specific gap analysis. Once the gaps have been identified, the key uncertainties driving our ability to accurately and conservatively predict a given aerothermal environment are determined via sensitivity analyses. Examples of key modeling deficiencies and their impact on planetary entry missions will be shown.**

I. Introduction

Any aerocapture or direct entry vehicle will be subjected to significant heating as it dissipates its kinetic energy in the atmosphere of the destination planet. In either application, the primary purpose of the thermal protection system (TPS) is to protect the payload from this entry heating. The performance of the TPS is determined by the efficiency and reliability of this system throughout all anticipated operating environments. Therefore, for any rigid aeroshell, the material selection and design thickness of the selected TPS material define two of the key performance metrics of the entire entry system. The choice of the TPS material is typically governed by the peak heat flux, surface pressure, and shear stress encountered during the entry, with more robust (higher density) materials required to protect the vehicle from more severe entry conditions. Once a material has been selected, the thickness is governed primarily by the total integrated heat load during the entry (which can be large for aerocapture or high-lift entry missions, due to a long residence time in the atmosphere). The accurate determination of each of these quantities relies on high-fidelity aerothermodynamics modeling, including estimated uncertainties in the predicted values.

Although the tools and methodologies for performing this analysis have been around for many years, the In-Space Propulsion (ISP) program has recently invested in further development of the underlying physical models and new methodologies for uncertainty quantification in an attempt to better assess the performance advantage of aerocapture over traditional propulsive deceleration. This paper summarizes the current state of aerothermal models for key destinations of interest to NASA, including Mars and Titan, highlighting recent ISP-sponsored improvements.

II. What Is Aerothermodynamics?

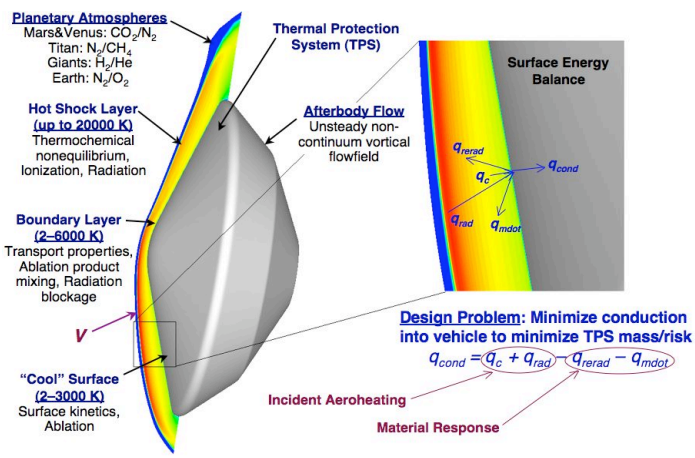


Figure 1. Principles of aerothermal models.

Spacecraft enter planetary atmospheres at high velocities, ranging from about 5 km/s for a low-speed Mars or Titan entry to almost 60 km/s for a polar probe to Jupiter. This energy must be safely dissipated in the atmosphere in order for the spacecraft to land successfully or perform the desired aerocapture maneuver. At these speeds, a strong shock wave forms in front of the entering probe that dissociates and ionizes the planetary atmosphere. The gas in the shock layer can reach temperatures in excess of 20,000 K and can be in thermochemical non-equilibrium, composed of many chemical species. A "cool" boundary layer forms at the surface that shields the spacecraft from the severe conditions behind the shock wave, however a small amount of energy reaches the surface in the form of convective heat transfer. The dissociated species in the shock layer can also recombine on the surface, an exothermic process that increases the heating. At high enough velocities the gas molecules even begin to emit radiation, which can further heat the surface of the vehicle.

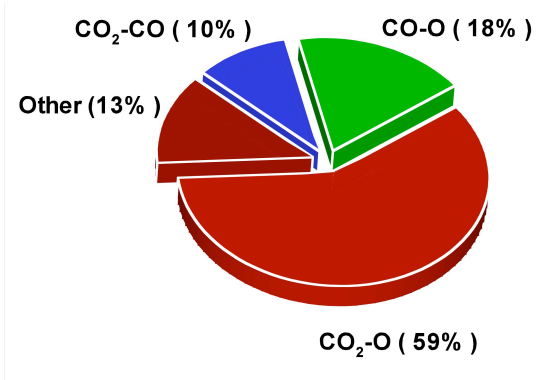
At its core, the discipline of aerothermodynamics is the analysis of this environment with a goal to accurately and conservatively predict the heating, pressure, and shear environment on the surface of the spacecraft as a function of time during entry. For many missions of interest, a further complication occurs due to the nature of the thermal protection system (TPS) material employed to protect the spacecraft from this extreme environment. At low entry velocities a reusable TPS system can be employed (such as on the Shuttle Orbiter) which acts primarily as a reradiator and insulator. However, at the conditions encountered by most planetary and high veloc-

ity Earth entry probes there is no reusable TPS material that can withstand the encountered heating environment. For these missions and ablative TPS material is employed. Ablative TPS materials still act as reradiators/insulators, but also pyrolize (outgas) and char during entry. These processes increase the efficiency of the material by injecting mass into the boundary layer (transpiration cooling) proportional to the heating environment encountered. However, this ablation process acts to couple the aerothermal environment to the response of the TPS material, which complicates analysis.

The discipline of aerothermodynamics consists of experimental techniques, employing wind tunnels, shock tunnels, and ballistics ranges, as well as computational methods, including computational fluid dynamics (CFD) and direct simulation Monte-Carlo (DSMC) methods. The resulting predictions are then margined by an amount which reflects the state of uncertainty in our ability to accurately predict the environment for a given mission and then used in the design and testing of the TPS system. Large, poorly defined uncertainties in the aerothermal predictions are not desired because they lead to unnecessarily heavy TPS, which can have a ripple effect throughout the spacecraft and may limit the payload mass allocation. In some cases, a better understanding of the aerothermodynamic environment may in fact be mission enabling. One example is the proposed Jupiter polar probe, for which current uncertainty levels would lead to a TPS mass fraction approaching 1.0. In addition, it is impossible to quantify the overall reliability of the entry system without a good physical understanding of the underlying uncertainties. It is important to note that no ground-test facility can fully replicate the flight environment of an entering spacecraft. Therefore, CFD analysis, founded on physics-based (rather than empirical) models, is required to enable traceability of the ground testing and to ensure that the resulting design analysis can be successfully extrapolated to a flight environment. While it is strongly desired to validate these aerothermal models with flight data, such data are sparse, and in fact nearly non-existent for non-Earth entries.

III. Methodology For Identifying Gaps

The primary tool for the analysis and simulation of aerothermal environments is CFD. Most of the basic methodologies that underlie current CFD codes have been around for 10-20 years. However, recent advances in parallel algorithms, coupled with the increasing affordability of commodity cluster based computers, have resulted in greatly increased injection of CFD into the TPS design process. While three-dimensional calculations were state-of-the-art only ten years ago, full 3D analysis, including geometrical singularities, are now routine practice early in the design cycle. At this point it is safe to say that, for TPS design, the "tall poles" are no longer process-oriented. While process improvements remain valuable, the current generation of tools are good enough for the task at hand. Rather, the largest areas for improvement are in the underlying physical models that are employed. Improvements to these models are required not only to improve the fidelity of the predictions, but also to quantify and reduce the basic uncertainty level to permit lower mass / higher reliability TPS designs for future missions.



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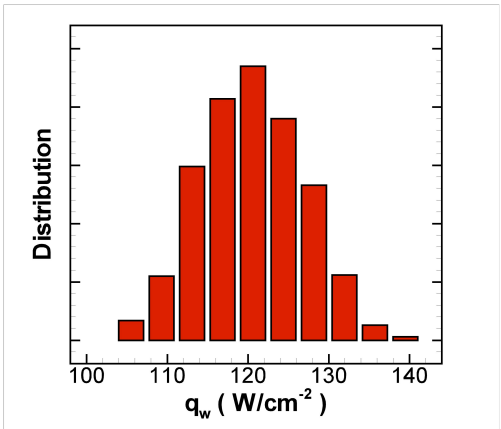

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Figure 2. Pathfinder uncertainty analysis: a) principle contributors to heating uncertainty and b) 2σ distribution of heat flux.

A survey of the state-of-the-art for a variety of planetary and Earth entry missions quickly reveals that there are large uncertainties in our ability to accurately predict the aeroheating environment in many instances. For example, the Mars Science Laboratory (MSL) currently carries a total uncertainty in forebody heating of nearly a factor of two. The details of these modeling deficiencies, or gaps, tend to be both destination and mission specific. For example, shock layer radiation tends to be a dominant gap for some missions and is completely irrelevant for others. However, while the details are mission specific, they tend to fall into several broad categories: gas-surface interactions, transition and turbulent heating, shock layer radiation, afterbody heating, and coupling between two or more of these modes. One additional modeling gap is aerothermoelastics, but this is specific to flexible decelerators (such as ballutes) and will not be specifically addressed in this paper.

The relative importance of each of these categories must be assessed for a given mission class using a mix of engineering judgment and sensitivity analyses. Recent advances

in computational performance have enabled much more rigorous analysis of the uncertainties associated with computational aeroheating predictions. One technique that has recently been introduced into the field is Monte-Carlo analysis. In this technique a large sample (~3000) CFD simulations are run on a given problem, varying the modeling parameters of interest over their uncertainty range, in order to determine which modeling uncertainties produce the largest uncertainty in the output quantity of interest (typically heat flux). For example, Bose et al. [1]-[2] demonstrated how parametric modeling uncertainties for a given reentry problem could be determined by directly coupling a non-equilibrium CFD solver to a Monte-Carlo based statistical analysis package. A detailed analysis of Mars Pathfinder convective entry heating was performed by Bose et al. [3]. The results (Fig. 2) demonstrated that the principal contributors to overall heating uncertainty were a few binary collision integrals, which are used to generate transport properties. Analysis like this can be used to quantify aeroheating uncertainties, identify key sensitivities, and prioritize research funding to target those items that are truly driving uncertainties for the given design problem.

IV. Modeling Status And Future Needs

This section briefly discusses the five classes of aerothermal modeling uncertainty identified in the previous section and outlines their relative importance to a variety of potential direct entry and aerocapture missions. In addition, suggestions are made for further research that could reduce these uncertainties further.

A. Gas-Surface Interactions

Catalysis is a gas-surface interaction (GSI) process by which dissociated boundary layer gases recombine on the TPS surface, releasing their heat of reaction in the process. This mechanism is of primary importance in the design of a non- or weakly ablating TPS material. Unfortunately, the physical mechanisms that govern catalytic recombination on the surface of a TPS material is poorly understood, in part due to difficulties in performing in-situ surface diagnostics at flight-appropriate gas-compositions, temperatures, and pressures. Therefore, most design analyses to date make conservative assumptions regarding this GSI. For example, a common assumption is that of a supercatalytic wall, in which it is assumed that the wall completely recombines the gas mixture to the lowest enthalpy state. The impact of wall catalysis is perhaps largest at Mars [3], for which the difference between a non-catalytic and supercatalytic wall can be as much as a factor of four in predicted heating (Fig. 3). Catalysis will also be a dominant contributor to aeroheating for giant planet and Titan entries, but is somewhat less important for entries to Earth and Venus.

Validated mechanisms for surface catalysis exist only for certain materials (mainly Shuttle heritage) for Earth entries; new models must be developed for other TPS materials and planetary destinations to avoid the current conservative design assumption. For ablating TPS, additional GSI reactions become important, including oxidation, nitridation, and sublimation of the surface layer. Finite rate reactions for all of these processes are material and destination dependent [4]. A mixture of ground-based testing to obtain basic rate data and improved analytical models incorporated into CFD codes is required to develop and validate GSI models for NASA missions of interest.

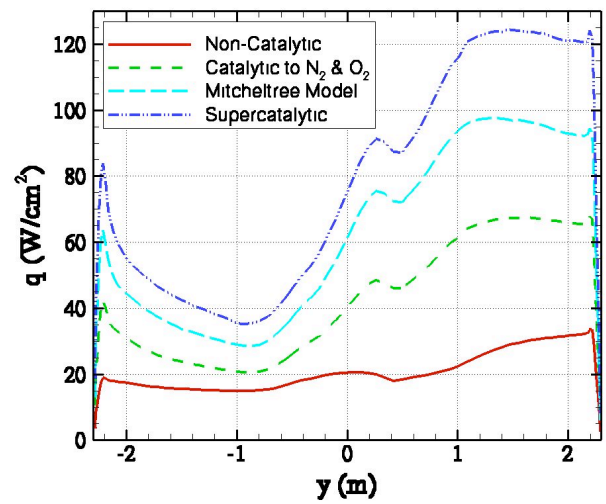


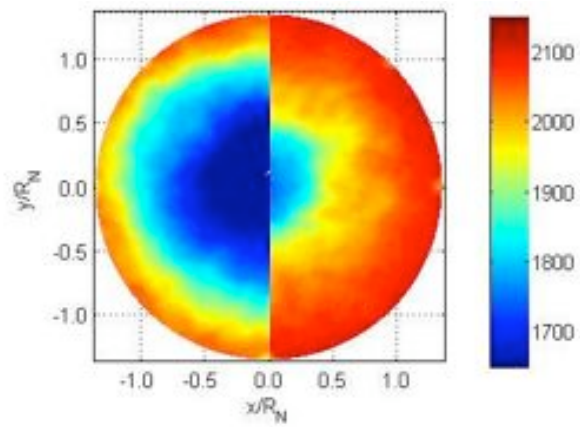
Figure 3. Impact of catalysis model on turbulent centerline heating for MSL.

B. Transition and Turbulence

Transition to turbulence is of paramount importance for many hypersonic cruise and entry vehicles, because turbulent heating levels are typically 3-4 times higher than laminar. For ablating entry systems the mechanism of transition will be governed by ablation induced roughness and blowing of pyrolysis gases into the boundary layer. In such cases simple smooth wall transition correlations may be non-conservative in their predictions. Roughness/blowing correlations have been studied in the past, but are highly dependent on both vehicle geometry and the specifics of the TPS material. Experimental techniques are under development in a hypersonic ballistics range [4] to measure the transition front and resulting turbulent heating level on realistic roughened models (Fig. 4a). Such experimental validation techniques, supported by test data from traditional wind tunnels as well as flight data where available, will be critical to developing realistic design transition criteria for future entry systems.

For many planetary entry vehicles, the combination of large size and ablating TPS will cause transition early in the entry. Once it has been determined that transition occurs prior to peak heating, the exact determination of the transition time and location are no longer important from a design standpoint. However, it is still extremely important to determine the resultant turbulent heating level. Turbulent heating is of primary concern for the MSL mission to Mars in 2009, where transition is predicted to occur early and the turbulent heating level is predicted to be a factor of three greater than the peak laminar level. Similar augmentation factors are predicted for Titan aerocapture and other low L/D lifting entries. This prediction,

made first with CFD tools, was later validated via an extensive aerothermal ground test program in support of Titan aerocapture [6] and later the MSL mission. Turbulence is also of great concern for the high lift entry vehicles proposed for large payloads at Mars or aerocapture to Neptune (Fig. 4b) [7]. For these geometries the large size and high angle of attack promote both axial and crossflow transition.



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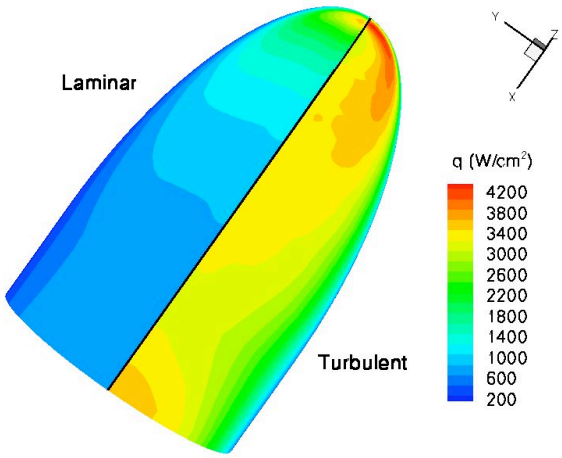

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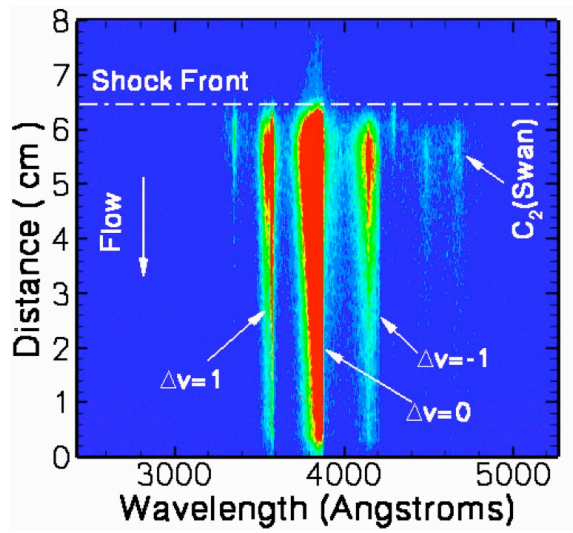
Figure 4. a) Hypersonic transition measurements in a ballistics range, and b) predicted heating for Neptune aerocapture ellipsled.

C. Shock Layer Radiation

When a gas mixture passes through a strong shock wave, it is first dissociated. At still higher velocities some of the atoms and molecules are electronically excited. When the excited electrons transition to a lower state a photon is emitted, resulting in shock layer radiation. Under certain entry conditions this radiation field can be strong enough to significantly impact the heating rate at the surface of the entry vehicle. This effect is directly proportional to the effective nose radius of the entry vehicle, and is a very strong (nearly exponential) function of the entry velocity. The threshold velocity at which radiation becomes important is also a strong function of the composition of the atmosphere. As a rule of thumb, shock layer radiation becomes important at Earth at entry velocities above 10 km/s, Mars and Venus above 8 km/s, Titan at 5 km/s, and at the giant planets above about 35 km/s. At the current time a validated radiation model exists only for moderate velocity Earth entries. For other planetary entries simplifying assumptions are typically employed, such as the assumption of a Boltzmann distribution of excited states. As a result, radiative heating uncertainties are typically quoted on the order of 50-100% for most missions of interest.



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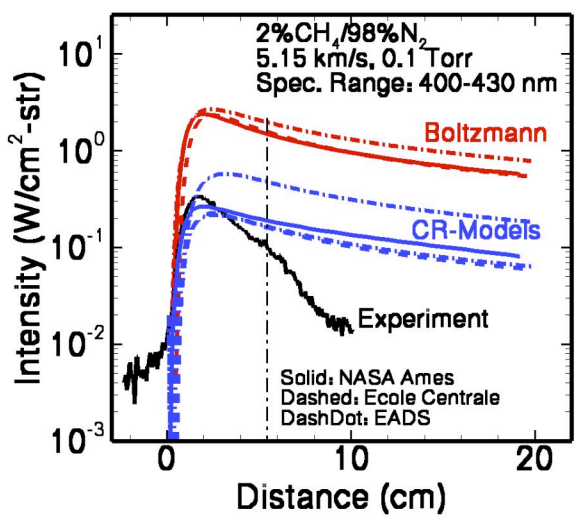

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Figure 5. a) Sample Titan spectrum from the EAST facility, and b) comparison of experimental data to several numerical models for CN radiation intensity.

Early predictions of the radiation environment for Titan aerocapture indicated that the total surface aeroheating was dominated by radiative heating due to the unique composition of the Titan atmosphere. As a consequence, the ISP program funded experimental studies in the Electric Arc Shock Tube (EAST) at NASA Ames aimed at collecting model validation data in the correct gas mixture, pressure and shock velocity.

These data were used to develop a new collisional-radiative model for Titan entries [8] and to demonstrate that the previous Boltzmann assumption was conservative by a factor of up to five (Fig 5b). The EAST data were also used during a pre-release assessment of the Huygens Titan entry probe to determine that the entry risk was within acceptable limits [9]. The EAST facility is currently being used to develop similar models for Mars entries (for the ISP program [10]) and to improve the fidelity of the Earth entry models (for CEV).

D. Afterbody Heating

The afterbody of a reentry vehicle is defined as the portion in the wake (Fig. 6). Although the afterbody typically experiences much lower heating rates than the heatshield, the associated uncertainty levels are much larger (~50-300%). This level can have a significant impact on TPS material selection and total mass. This conservatism will also shift the center of gravity aftward, which reduces the static stability of the probe and in some circumstances may necessitate the addition of ballast in the nose.

Afterbody flows much more difficult to simulate accurately than forebody (heatshield) flows. The separated wake can be a complex unsteady vortical flowfield, governed by extreme non-equilibrium and regions of embedded rarified flow. The strong flow expansion around the shoulder causes the various relaxation processes to freeze. Ablation products from the forebody can be entrained into the wake flow and impact afterbody environments. The simultaneous occurrence of all of these physical phenomena stresses the capabilities of current solvers.

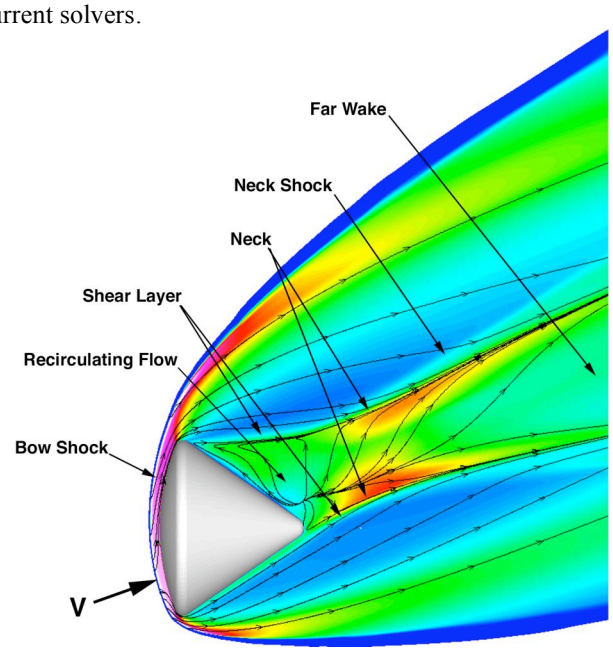

Figure 6. Afterbody flow schematic.

Given the complexity of afterbody flowfields, it is critically important that the CFD models be properly validated for such environments. Consequently, a primary reason for the large uncertainty in afterbody heating predictions is a perceived sparsity of relevant data for validation of the computational tools. Some flight data at appropriate velocities are available, but many of the most relevant flights occurred in the 1960's, and for the most part these data have not been critically evaluated or used for code validation purposes in more than 30 years. A recent review paper [11] collected the relevant data and prior simulation attempts and discussed their applicability for code validation purposes. The general conclusion was that a fair amount of data exist for Earth entries, but there are almost no relevant data for non-Earth applications. Because of this lack of non-Earth data it is expected that uncertainty levels will remain large until engineering flight data can be obtained from future science missions. Desired improvements to the state-of-the-art include the development of better models for the time-accurate simulation of unsteady wake flows (such as Detached Eddy Simulation [12]) and the development of hybrid CFD-DSMC codes that can simulate embedded regions of rarified flow with high accuracy.

E. Coupling

The current state-of-the-art for TPS design is to employ an uncoupled analysis. In this procedure, the convective heating environment is first computed using a CFD code and an assumption of a non-ablating TPS surface. The shock layer radiation component (if any) is then computed by a radiation transport code using the CFD data as input. Finally, the thermal response of the TPS material is computed using the combined convective and radiative heating as inputs. While this approach works well for many problems of interest, it neglects potentially important interactions between these various physical processes.

In particular, if the shock layer is radiating strongly, the energy converted to radiation is removed from the gas and must be accounted for in the fluid dynamic simulation. Neglecting this effect results in an overly conservative prediction of the emitted radiation. This effect was demonstrated for Titan aerocapture, where including the non-adiabatic loss of radiant energy reduced the total radiative heating by a factor of two (Fig. 7) [13].

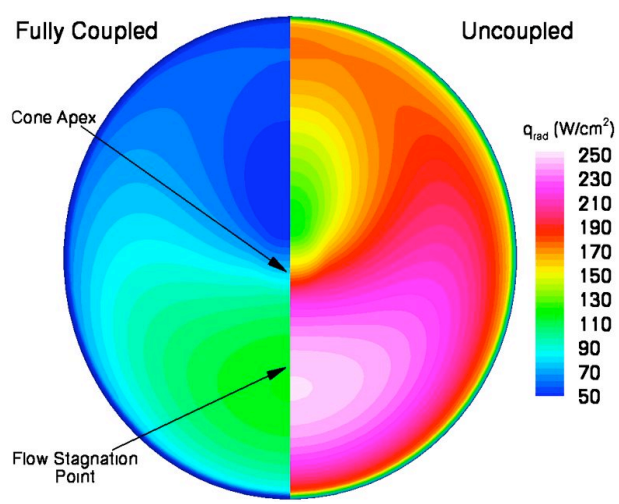

Figure 7. Impact of coupling on predicted radiative heating at Titan.

Another form of coupling occurs when the TPS begins to ablate and inject decomposition products into the boundary layer. The injected gases can have several effects, including reducing the heating due to transpiration, altering the chemistry in the boundary layer, and affecting transition to turbulence. The ablation products can also absorb shock layer radiation, or even emit their own radiation if the are sufficiently hot. Key destinations for which coupling effects are important include Titan aerocapture, Venus and giant planet entries, and high velocity entries at Earth and Mars.

IV. Model Validation

The "gold standard" for validating entry aeroheating and TPS material response models is with flight data. Two types of flight data can be employed for validation purposes. The first is recovered entry hardware. Examples include the Apollo flight test vehicles, which were recovered and cored to determine heatshield performance. More recently the Stardust and Genesis heatshields were cored and the TPS material was examined. Of course it is not typically possible to recover hardware from non-Earth entry vehicles, but a unique opportunity presented itself when the twin Mars Exploration Rovers (MER) lasted far beyond their planned lifetimes. At the urging of the engineering community MER-B (Opportunity) examined its ejected heatshield with the cameras and micro-imagers. The resulting images (Fig. 8) revealed the char-depth of the TPS material, information that can be compared to pre-flight predictions.

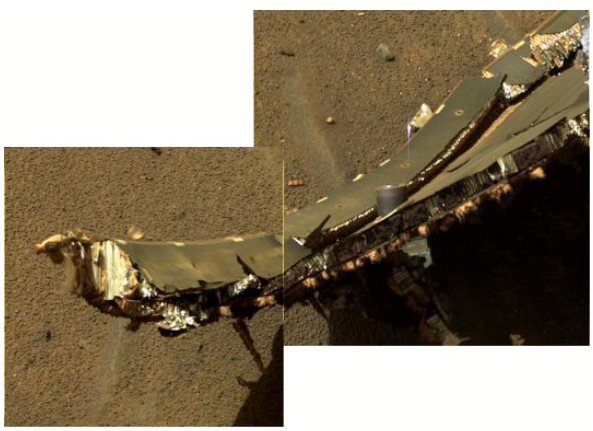

Figure 8. MER-B heatshield on the surface of Mars as imaged by the Opportunity rover.

The second type of flight data is in-situ engineering instrumentation. Unfortunately such data are scarce for non-Earth entries, and the recent trend has been to eliminate engineering instrumentation from planetary entry missions as a cost saving measure. For example, the MER landers included no instrumentation, and none is planned for Phoenix in 2007. The next opportunity to obtain flight aeroheating data is the MSL mission. A basic set of relatively mature low risk engineering instruments in the heatshield and backshell could go a long way toward answering some of the key questions discussed in previous sections of this paper. Fortunately, NASA has authorized a program called MEDLI (MSL Entry, Descent and Landing Instrumentation) to instrument the MSL forebody heatshield with seven TPS plugs, as well as seven pressure ports configured as a flush air data system (FADS). Figure 9 shows the planned location of all heatshield instrumentation. In Fig. 9, the white symbols labeled with a "T" are the thermocouple plug locations, while the black symbols labeled with a "P" are the pressure ports. No instrumentation will be included on the backshell due to MSL schedule constraints. This dataset will provide the first new non-Earth entry aeroheating data since the Pathfinder mission, and will hopefully help to answer some of the fundamental questions discussed in this review relating to leeside turbulent heating levels, forebody transition, and catalytic heating levels.

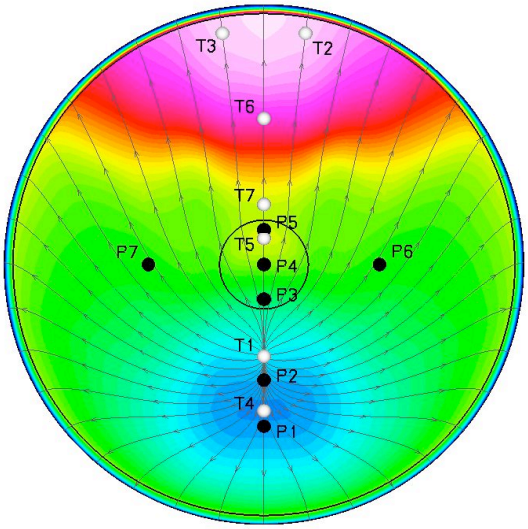

Figure 9. MEDLI instrumentation locations on the MSL heatshield.

V. Conclusions

The current state of the art for computational and experimental aerothermal analysis for entry missions is reviewed. The applicability of a rigorous Monte-Carlo methodology for identifying and quantifying the sensitivities and uncertainties inherent in the physical models employed to predict these phenomena is introduced. Five primary uncertainty areas are identified: gas-surface interactions, transition and turbulent heating, shock layer radiation, afterbody heating, and coupling between these modes. For gas-surface interactions, current design models assume a conservative upper limit for catalytic heat transfer, which implies that significant performance gains may be possible if this effect were better characterized. Upcoming missions, including the Mars Science Laboratory in 2009, will experience early transition to turbulence due to a combination of large size and high ballistic coefficient. Transition location and the resulting turbulent heating levels on the blunt, lifting cones employed for entry are another large source of uncertainty. Future missions, particularly crewed vehicles, will encounter additional heating from shock-layer radiation due to a combination of larger size and faster entry

velocity. At the current time, no validated model for shock layer radiation in a non-air environment currently exists. The need for fully coupled analysis that combines TPS material response and aerothermal prediction uncertainties is identified. Finally, the value of engineering flight data for final model validation is discussed.

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