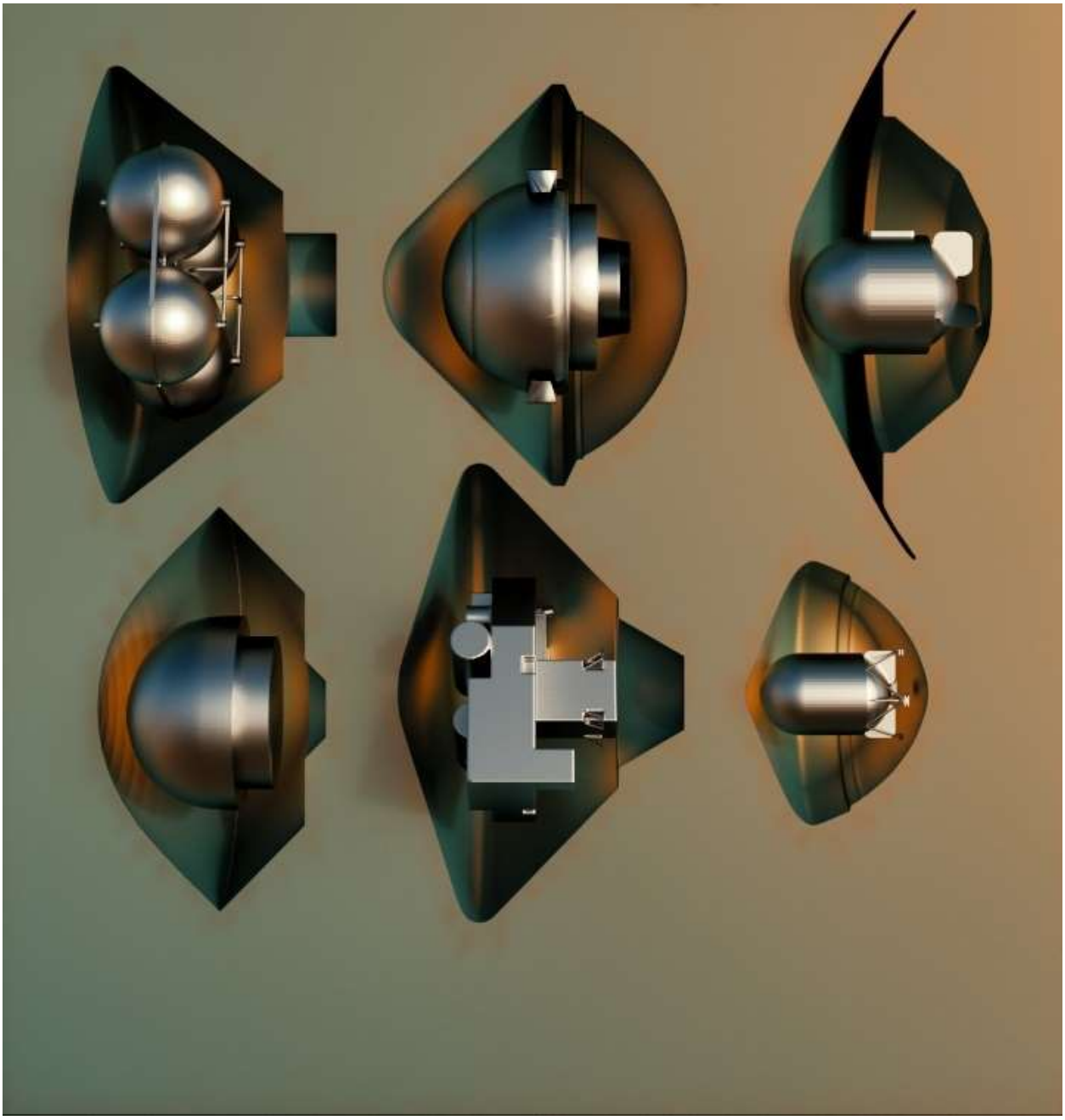
10th INTERNATIONAL PLANETARY PROBE WORKSHOP – BOOK OF ABSTRACTS



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Part I: Oral

Session 2: Mars Science Laboratory

MARS SCIENCE LABORATORY ENTRY, DESCENT, AND LANDING SYSTEM OVERVIEW AND FLIGHT PERFORMANCE RESULTS

Adam Steltzner,Allen Chen, Miguel San Martin

The Mars Science Laboratory project recently placed the Curiosity rover safely on the surface of Mars. With the success of the landing system, the performance envelope of entry, descent, and landing (EDL) capabilities has been extended over the previous state of the art. This paper will present an overview of the MSL entry, descent and landing system design and flight performance results.

Entry System Reconstructed Performance for the Mars Science Laboratory Mission

Allen Chen, Robin Beck, Paul Brugarolas, Karl Edquist, Gavin Mendeck, Mark Schoenenberger, David Way

While the Sky Crane portion of Mars Science Laboratory's (MSL) Entry, Descent, and Landing (EDL) system draws well-deserved attention as the iconic image of the landing, the entry segment is at least an equally harrowing part of the journey where a multitude of opportunities for failure exist. Over 99% of the spacecraft's kinetic energy is dissipated during entry as the spacecraft goes from over 5.8 km/s to approximately 400 m/s; all before the use of the supersonic parachute or the descent engines in powered flight. The entry vehicle had to survive aerothermal heating and considerable aerodynamic deceleration (up to 15 Earth g's), all while autonomously executing hypersonic and supersonic maneuvering to target the landing site and increase altitude and timeline margins. This paper summarizes Curiosity's as flown entry performance on the night of August 5th as reconstructed by the flight team and compares the performance to preflight predictions and design environments.

Reconstruction of the Mars Science Laboratory Parachute Performance and Comparison to the Entry, Descent, and Landing Simulation

Juan R. Cruz, David W. Way, Jeremy D. Shidner, Jody L. Davis, Douglas S. Adams, Devin M. Kipp

The Mars Science Laboratory used a single mortar-deployed disk-gap-band parachute of 21.35 m nominal diameter to assist in the landing of the Curiosity rover on the surface of Mars. The parachute system's performance on Mars has been reconstructed using data from the on-board inertial measurement unit, atmospheric models, and terrestrial measurements of the parachute system. In addition, the parachute performance results were compared against the end-to-end entry, descent, and landing (EDL) simulation created to design, develop, and operate the EDL system. Mortar performance was nominal. The time from mortar fire to suspension lines stretch (deployment) was 1.135 s, and the time from suspension lines stretch to first peak force (inflation) was 0.635 s. The reconstructed aerodynamic portion of the first peak force was 153.8 kN; the median value for this parameter from an 8,000-trial Monte Carlo simulation yielded a value of 175.4 kN – 14% higher than the reconstructed value. Aeroshell dynamics during the parachute phase of EDL were evaluated by examining the aeroshell rotation rate and rotational acceleration. The peak values of these parameters were 69.4 deg/s and 625 deg/s2, respectively, which were well within the acceptable range. The EDL simulation was successful in predicting the aeroshell dynamics within reasonable bounds. The average total parachute force coefficient for Mach numbers below 0.6 was 0.624, which is close to the pre-flight model nominal drag coefficient of 0.615.

Powered Flight and Sky Crane Maneuver Reconstructed Performance Summary for the Mars Science Laboratory Mission

Steven Sell, Miguel San Martin, Frederick Serricchio, Christopher White

The task of the Powered Flight segment of Mars Science Laboratory's (MSL) Entry, Descent, and Landing (EDL) system is to dissipate the final 0.1 % of the kinetic energy with which the spacecraft entered the Martian Atmosphere. This task begins approximately 1.6 km above the surface descending at just under 90 m/s with the Powered Descent Vehicle (the Curiosity Rover along with it's Descent Stage 'jetpack' separating from the parachute and backshell and beginning a propulsive descent. Approximately 20m over the surface, the Rover is lowered on a bridle to 7.5m below the Descent stage and continues a slow descent until the Rover is supported entirely by the Martian Surface. This "Sky Crane Maneuver" was a new an innovative approach to landing a payload on planetary surface. This paper provides an overview of the powered flight design, key features, and event timeline. It also summarizes Curiosity's as flown performance on the night of August 5th as reconstructed by the flight team with specific focus on the Sky Crane Maneuver.

The MSL Navigation Filter

Fred Serricchio, Miguel SanMartin, Edward Wong

The Mars Science Laboratory (MSL) Entry Descend and Landing (EDL) system delivered a rover named “Curiosity” at Gale crater (Mars) on August 5th, 2012. The MSL EDL system landed Curiosity within the targeted ~20 km landing ellipse. The EDL system used a guided atmospheric entry capsule to achieve this targeting performance. The entry capsule guidance, navigation and control system employed a set of ejectable balance masses, a descent inertial measurement unit (DIMU), and a propulsive reaction control system (RCS). The navigation filter (NavFilter) integrated the DIMU measurements to inertially propagate the position and attitude of the capsule. The NavFilter also used the Terminal Descent Sensor (TDS) to obtain surface relative velocity and altitude corrections to the state of the vehicle. The NavFilter provided all of the dynamic state information to the entry controller, entry guidance, and powered descent guidance and control algorithms. Attitude information was passed from the Cruise Attitude Control System (ACS) to the NavFilter for EDL before the Cruise Stage is jettisoned. During the phase just prior to EDL, the NavFilter estimated the z-axis scale factor and the 2–axis (x,y) misalignments that were observable from the spinning cruise stage. The alignment estimate was necessary to minimize attitude errors during EDL that directly translate into position and velocity knowledge errors at touchdown. The inertial position and velocity state of the vehicle was sent to the spacecraft to initialize the NavFilter using ground-based navigation. The exo-atmospheric phase starts after separation from the Cruise stage and it last until the capsule entered the Martian atmosphere. Once the NavFilter reported that we sensed the appropriate acceleration from the atmosphere, the entry phase was triggered. After guided entry is complete, the parachute deployed when the appropriate dynamic conditions of the vehicle were met. While on the parachute the heatshield was deployed which enables the TDS to start seeing the ground for the first time. The six beams of the TDS were cycled through at 50 Hz to provide line-of-sight velocity and slant-range measurements. During this time the NavFilter uses those measurements to determine 3-axis velocity, acceleration and altitude errors. The NavFilter used the TDS measurements in a background process that employed an iterative batch filter to enable robust data editing of bad measurements, while still being able to provided accurate estimates. The data editing method used a moving window and a fixed number of iterations to determine which TDS measurements should be incorporated into the solution for the given window. The final solution for each estimate was validated by a number of different tests to determine the quality of that solution before incorporating that into the estimate. The NavFilter algorithm had parameters that define the window length, the beams to include in a given solution, error thresholds, and uncertainty levels. Each group of these parameters were set specifically for each phase of EDL to determine the most robust and accurate solution throughout all of EDL until the Curiosity rover touched down safely on the surface of Mars. This paper will describe the design and the as-flown performance of the navigation filter during each phase of flight.

Mars Science Laboratory: Deviations from Pre-Flight Predictions

David Way

On August 5, 2012, the Mars Science Laboratory (MSL) Curiosity rover successfully landed on the floor of Gale Crater, only 2.39 km from the target point and within 6.9 s of the predicted mean landing time. End-to-end Entry, Descent, and Landing (EDL) simulation predictions were used throughout the project lifecycle: to inform EDL design choices, to compare and certify candidate landing sites, to verify EDL system performance, to select parameters, and to evaluate operational decisions. Overall, this simulation performed exceptionally well in correctly predicting many EDL performance metrics. For example, as-flown values for peak deceleration during entry and at parachute inflation were within -0.471? and -0.806? of simulation predictions, respectively. Likewise, the peak attitude rate and angular acceleration were predicted well at 0.964? and 0.216?, respectively. However, this paper focuses on the three specific instances where as-flown performance deviated from pre-flight simulation predictions. First, the entry vehicle experienced higher than expected deceleration at altitudes above 58 km above the MOLA reference ellipsoid. Likely due to lower temperatures and higher density in the upper portions of the atmosphere, this anomaly resulted in a very short -3.836? time in Guidance, Navigation, and Control (GN&C) Mode 8, ”WAIT FOR GUIDANCE START”, an initial bank angle 8.595 deg away from the expected pre-bank angle (3.050?), and a 2.509? long time in GN&C Mode 9, ”RANGE CONTROL SLEW TO COMMANDED BANK ANGLE”. Second, low supersonic deceleration was observed during the “Straighten-Up and Fly Right” (SUFR) maneuver. It is possible that the vehicle encountered a moderate tail wind during this portion of the flight, which contributed to a combined 3.23? time in GN&C Modes 14 and 15. Finally, a 2.58? long time in GN&C Mode 35, ”PD READY FOR TOUCHDOWN”, was determined to be due to an error in estimating the local gravitational acceleration at the landing site.

Performance of the MEDLI Integrated Sensor Plug (MISP) Hardware and TPS Response Reconstruction

Todd White, Deepak Bose, Milad Mahzari, Karl Edquist, Chris Kuhl, Alan Little, Michelle Munk

NASA's Mars Science Laboratory (MSL) entered the atmosphere of Mars on August 5th, 2012 and landed the 900 kg Curiosity rover. MSL represented a significant advancement in planetary entry, descent, and landing capability; with an entry mass of 3200 kg and a 4.5 m diameter heatshield, MSL was the heaviest and the largest Mars entry vehicle. The MSL heatshield was instrumented to acquire important flight data for aerodynamics, aerothermodynamics, and thermal protection system response. The instrumentation suite that captured this data was the Mars Science Laboratory Entry, Descent, and Landing Instrumentation (MEDLI). The MEDLI suite consisted of 7 pressure transducers, 24 thermocouples, and 6 ablation sensors. It successfully acquired and returned surface pressure, in-depth temperatures, and material decomposition characteristics at various locations on the heatshield. The MEDLI suite on MSL represents the most extensively instrumented Mars entry heatshield. This submission presents an assessment of the thermal data received from the MEDLI Instrumented Sensor Plug (MISP), including some comparisons of in-depth temperatures with model predictions, and provides estimates of the aerothermal environment using the flight data.

RECONSTRUCTED AERODYNAMIC PERFORMANCE OF THE MSL ENTRY VEHICLE USING DATA FROM THE MEADS EXPERIMENT AND ONBOARD INERTIAL MEASUREMENTS

Mark Schoenenberger, John Van Norman, Chris Karlgaard, Prasad Kutty, Michelle Munk

On August 5th 2012, the Mars Science Laboratory entry vehicle successfully entered Mars' atmosphere, flying a guided entry until parachute deploy. The Curiosity rover landed safely in Gale crater upon completion of the Entry Descent and Landing sequence. A significant effort to reconstruct the entry trajectory has been completed and the aerodynamic performance of the capsule as been estimated. This presentation first provides an overview of the reconstruction instrumentation used for that reconstruction, including the Mars Entry Air Data System (MEADS) experiment and the inertial measurement unit (IMU) used to navigate the vehicle during entry. A brief summary of the trajectory reconstruction results will be presented followed by a description of the reconstructed aerodynamic performance. Overall the MSL capsule flew very much as expected, well within the preflight uncertainty models. CFD and experiment data predicted the trim characteristics and effectiveness as a decelerator quite well. The results presented will focus on the areas that deviated from the nominal predictions. It was observed that the axial force experienced during entry was generally greater than predicted over much of the entry trajectory by a small percentage. Also, as the capsule decelerated through low supersonic speeds approaching parachute deploy conditions, data suggest the capsule flew threw a moderate crosswind. The vehicle controller reacted appropriately crabbing the nose of the vehicle into the wind to maintain proper ground track. These details are described with supporting data.

Mars Science Laboratory Entry, Descent, and Landing Atmosphere Reconstruction

Allen Chen, Alicia Cianciolo, Ashwin Vasavada, David Kass, Dan Tyler, Jeff Barnes, Scot Rafkin, Bruce Cantor, Chris Karlgaard

Atmosphere reconstruction for Mars entry systems has always been plagued by a lack of data. Unlike previous missions, the flight of Mars Science Laboratory and successful landing returned a wealth of in situ data that can be combined with orbiter remote sensing data to understand the state of the atmosphere. No single data set collected is sufficient to understand the atmosphere state along the path flown by the spacecraft; rather, the atmosphere must be cobbled together from the various data and with some assumptions and inferences. The combination of data sets requires a clear understanding of each source's strengths and weaknesses. While there are still significant limitations to the reconstruction approach, the various data sets appear self-consistent and reinforce each other. The assembled best estimate of the as flown atmosphere compares favorably to preflight predictions made with the aid of historical atmosphere observations and substantial modeling efforts.

MSL EDL Lessons Learned Summary

Adam Steltzner

Session 3: Future Mars I

The SEIS Instrument on the NASA INSIGHT mission to Mars: overview of the development

Rene Perez, S. Deraucourt, K. Hurst, L. Kerjean, Ph. Laudet, D. Mimoun, P. Lognonné

The Seismic Experiment for Interior Structure (SEIS) instrument is the major experiment on-board the NASA INSIGHT mission to Mars (Interior Exploration using Seismic Investigations, Geodesy and Heat Transport). This mission has been selected by NASA, in the frame of the DISCOVERY program, for a launch in March 2016, and will deploy the first geophysical observatory on Mars, providing scientific knowledge essential to understand the fundamental processes of telluric planet formation and evolution. SEIS, measuring seismic activity, the meteorites impact flux, and the Phobos tide, will characterize the deep interior structure of Mars, giving information about the thickness and structure of the crust, the composition and structure of the mantle, and the size of the core. The SEIS instrument, developed since one decade by IPGP with support of CNES and CNRS, will be provided by the French Space Agency, who coordinates a wide set of international hardware contributors (Institut de Physique du Globe de Paris (IPGP), Swiss Federal Institut of Technology of Zurich (ETHZ), Max-Planck Institute of Lindau (MPS), Imperial College (IC) from London and Oxford University, Jet Propulsion Laboratory) and the Institut de l'Aéronautique et de l'Espace (ISAE) from Toulouse. After a summary of the scientific needs and of the main technical requirements, this presentation will focus on the development of the instrument, with respect to the constraints applicable. Development overview: With the INSIGHT mission selected lately in August 2012, and a launch programmed in March 2016, the main constraint on the SEIS development is clearly the schedule, as the instrument delivery is expected at JPL no later than January, 9th, 2015. This development timeline is very tight, even shorter than our previous experience at CNES on Mars Science Laboratory, with the ChemCam instrument. The SEIS project is not starting from scratch, and many preparation activities have been performed before the mission decision for all European sub-systems, with a long inheritance of R & T actions, phase A studies and breadboard models, and the anticipation by CNES, during the competitive phase A of INSIGHT, with the manufacturing of an EM model of the Very broad Band seismometer (VBB). These actions lead to reach a Test Readiness Level TRL-6 status (so, in relevant environment) as early as May 2013, but also to obtain an increase of the design margins of the Leveling sub-system (LVL) and the sensors (VBB, SP (Short Period)). This was possible thanks to a strong effort of all the partners to have a global architecture respecting all the technical constraints and presenting adequate performances. Even with this tight and challenging schedule, it has been decided to work with a rich-hardware development plan, including STM (Structural & Thermal model), EM (Engineering model), QM (Qualification model), FM (Flight model) and Spare. This plan is the key to consolidate, with an iterative and progressive process, our knowledge of the Instrument, of its performances, and its capabilities to withstand the severe environment of a martian mission. An other important benefit of this development plan is to have all the partners learning how to work together, and to optimize and prepare all the test plans and procedures, which is the way to the success for delivering a SEIS FM model in a very

short time. Conclusion: SEIS development is presenting a challenge to the SEIS team, due to its very tight schedule, imposed by the launch-to-Mars 2016 slot, and many technical and performances constraints. We think this rich-hardware development plan, experimented by CNES on ChemCam on MSL, is a good model to follow, and will be the good lead to success.

Mars 2020 Entry, Descent, and Landing Overview

Aaron Stehura, Erisa Hines, Ravi Prakash, Allen Chen

In December 2012, NASA announced it would pursue a Mars rover for the 2020 launch opportunity. Referred to as Mars 2020 (M2020), it would be a follow-on to the Mars Science Laboratory (MSL) mission, heavily leveraging MSL heritage to reduce cost and schedule risk while accommodating an as-yet-undetermined set of science instruments. M2020's Entry, Descent, and Landing (EDL) system will use the approach successfully pioneered by MSL, but a small number of changes are being considered to improve performance or demonstrate targeted technologies. This paper presents an overview of the preliminary EDL design for M2020 and summarizes current status on a series of trade studies investigating proposed changes from the MSL baseline. These possible changes include tuning Cruise and EDL flight dynamics for the 2020 opportunity, revisiting a number of issues that were discovered during the development, testing, and flight of MSL, and accommodating a variety of prospective EDL technologies.

Mars Sample Return Using Commercial Capabilities

Lawrence G. Lemke, Jeffrey Bowles, Andrew Gonzales, Loc Huynh, Carol Stoker

A Mars Sample Return (MSR) mission is the highest priority science mission for the next decade as recommended by the recent Decadal Survey of Planetary Science[1]. This paper describes a study that examined use of emerging commercial capabilities to land the sample return elements, with the goal of reducing mission cost. We examined the feasibility of the following scenario for MSR : A Falcon Heavy launcher injects a SpaceX Dragon crew capsule and trunk onto a Trans Mars Injection trajectory. The capsule is modified to carry all the hardware needed to return samples collected on Mars including a Mars Ascent Vehicle (MAV), an Earth Return Vehicle (ERV) and Sample Collection and Storage hardware. The Dragon descends to land on the surface of Mars using SuperSonic Retro Propulsion (SSRP) as described by Braun and Manning[2]. After samples are obtained and stored in the ERV, the MAV launches the sample-containing ERV from the surface of Mars. We examined cases where the ERV is delivered to either low Mars orbit (LMO), C3 = 0 (Mars escape), or an intermediate energy state. The ERV then provides the rest of the energy (delta V) required to perform trans-Earth injection (TEI), cruise, and insertion into a High Earth Orbit (HEO). A later mission, possibly a crewed Dragon launched by a Falcon Heavy (not part of the current study) retrieves the sample container spacecraft from Earth orbit. The retrieving spacecraft packages the sample and performs a controlled Earth re-entry to prevent Mars materials from accidentally contaminating Earth. The key analysis methods employed a set of parametric mass estimating relationships (MERs) and standard aerospace analysis software codes modified for the MAV class of launch vehicle to determine the range of performance parameters that produced converged spacecraft designs capable of meeting mission requirements. Subsystems modeled in this study included structures, power system, propulsion system, nose fairing, thermal insulation, actuation devices, and GN&C. Best practice application of loads and design margins for all resources were used. Both storable and cryogenic propellant system were examined. The landed mass and lander capsule size provide boundary conditions for the MAV design and packaging. We estimated the maximum landed mass of a Dragon capsule using SSRP. This and the volume capability to store the MAV was deduced from publically available data from SpaceX as well as our own engineering and aerodynamic estimates. Minimum gross-liftoff mass (GLOM) for the MAV were obtained for configurations that used pump-fed storable bi-propellant rocket engines for both the MAV and the ERV stage. The GLOM required fits within our internal estimate of the mass that Dragon can land at low elevation/optimal seasons on Mars. While this study is still in progress, it appears that a single Mars launch sample return mission is feasible using current commercial capabilities to deliver the return spacecraft assets. [1] Visions and Voyages for planetary Science in the decade 2013-2022, Nat'l Academy Press, 2010. [2] Braun, B. and R. Manning, Mars Entry Descent and Landing Challenges, IEEEAC paper 0076, 2005).

Sample Acquisition and Caching Architecture for the Mars Sample Return Mission

Kris Zacny

The Mars Sample Return is a campaign consisting of at least three separate missions. In the first step, a caching mission would consist of rover with an arm-deployable drill. The drill would acquire rock and regolith samples and put them into a cache. After acquiring the required number of samples the cache would be placed on the ground. The next step in the campaign would consist of a fetch rover and the Mars Ascent Vehicle (MAV) mission. The fetch rover would find the cache, pick it up and return it to the MAV. The MAV would then launch the cache into the Mars orbit. The third step would consist of an Earth Return Vehicle (ERV) that would capture the orbiting cache and bring it back to Earth. In the current architecture, the Mars orbiter with the ERV would be launched from Earth ahead of the MAV mission, this is in order for the orbiter to guide the fetch rover and act as a communication relay link. NASA has committed itself to launching a Curiosity-size rover in 2020. This so called Mars2020 mission is required to have its mission goals aligned with the Mars Sample Return goal. Although the exact mission details are not known yet, it is very likely that the rover will have a drill for acquiring approximately 20-40 rock cores, ~1 cm diameter by ~5-7 cm long. These samples would be placed inside a cache and hermetically sealed. The reduction of mission risk was chosen as having greater priority for MSR than reduction in mass or volume. For this reason, we selected a “One Bit per Core” approach. Each drill bit is used to acquire a single rock core. The core is then sheared off at the base using Honeybee Robotics’ patented Core Breakoff technology which simultaneously breaks the core and captures it within the Bit. Once the core acquisition process is complete, the Bit is encapsulated within a sleeve using a threaded connection. This particular connection also allows for hermetic sealing. All of the encapsulating sleeves are housed within the Sample Cache. As rock cores are collected, the Sample Cache is gradually populated with Bits. Once full, the Sample Cache can be removed from the rover body and placed within the Orbital Sample for return to Earth. Having dedicated bits allows a reduction in the number of core transfer steps and actuators. It also alleviates the bit life problem, eliminates cross contamination, and aids in hermetic sealing. An added advantage is faster drilling time, lower power, lower energy, and lower Weight on Bit (which reduces Arm preload requirements). In order to select the appropriate sample for earth return, the rock should be thoroughly interrogated in-situ. For that reason, our architecture includes additional 3 tools: The Rock Abrasion and Brushing Bit (RABBit), The PreView, and the Powder and Regolith Acquisition Bit (PRABit). The Rock Abrasion and Brushing Bit (RABBit) uses the same approach as the Rock Abrasion Tool on the Mars Exploration Rovers to brush and abrade rocks. It creates a surface ~35 mm in diameter, weights ~ 360 grams, and fits within a 30 mm diameter and 55 mm tall cylinder. An estimated life of the bit based on the RAT-data and increasing the grinding bit volume is 45 operations in hard rocks and 120 operations in softer rocks. The Core PreView Bit acquires short cores for in-situ analysis by non-contact instruments such as LIBS, Raman, and Microscopic Imager. The PrewView Bit has side slots in the break-off tube and the auger tube. The bit takes

advantage of the Honeybee break-off system utilizing two rotary-systems: an inner rotating sleeve to shear and capture the core and outer auger/bit. During drilling, the internal break-off tube is rotated to open the side slot. During the break-off stage, the inner tube is rotated so as to close the side slot (and at the same time shears and captures the core). Hence the captured core cannot fall out. During the core dispensing stage, the break-off tube is rotated to open up the slot. The advantage of this bit is that the core can be viewed while inside the bit i.e. in a known location. The Powder and Regolith Acquisition Bit (PRABit) can be used to acquire rock cuttings during the drilling process, and it can also be used to acquire regolith. Once the bit acquires rock powder or regolith, the sample can be either dispensed into an observation tray or cached for earth return along with rock cores. Bit operation is very similar to that of the PreView bit in that there are slots within the break-off tube and the auger. When the slots are aligned, the regolith or rock powder can flow into the inner cavity. When the slots are closed (the inner tube is rotated with respect to the outer auger) the powder inside the bit is retained. The PRABit has integrated sieves and can either acquire particles below sieve size or retain particles above certain sieve size, depending upon mission requirements. If an original, unsorted sample is desired, the bit could then dispense a sample having various particle sizes.

EDL and GNC Optimization of a Network of Small Mars Landers

Davide Bonetti, Gabriele De Zaiacomo, Irene Pontijas Fuentes, Rodrigo Haya Ramos, Gabriele Bellei, David Riley

This paper presents the results of the EDL and GNC optimization of a network of three Small Mars Landers. This study is performed in the frame of the ESA MREP (Mars Robotic Exploration Preparation) Programme, as preliminary activity towards the ESA INSPIRE mission which is planned to be launched in the 2022-2024 time frame. The main objective of the study presented is to produce, for a given payload of 130 kg, an optimum EDL and GNC design. A critical extra requirement is imposed to the problem: the solution must be identical for the three landers that separate from the same orbiter and reach the planet surface in different sites. Given the wide Mars variability in terms of atmosphere and topography this results into an additional robustness required to the common design solution. Several steps are followed to identify the best EDL and GNC solution: past missions review, preliminary trade-offs, numerical trade-offs and detailed design. To support to the last two steps, ESAT (EDL&GNC Sizing and Analysis Tool) has been used to obtain a multi-disciplinary optimum design. Starting from a given payload and arrival conditions, ESAT allows to trade-off multiple figures of merit of an end to end mission analysis (from Mars arrival to Mars touchdown), coupling them with the robust optimization of the main events (separation sequence, triggering of EDL phases), of the EDL elements (aeroshell, parachute, retro-rockets and airbags) and the of GNC equipment (sensors, actuators and triggering logics). From the above process, two candidate configurations are obtained: the first one is similar in concept to the Mars Pathfinder while the second one is similar in concept to the Mars Exploration Rovers. Both have been optimized in terms of EDL elements and GNC equipments to guarantee the highest mission success with the minimum entry mass. Finally, the results obtained supported the identification of critical components and mission phases for which a low Technology Readiness Level is detected at European level. This critical review allows to draw a preliminary technology development roadmap with the objective of paving the way for Europe towards this challenging Mars network mission.

Landing Site Reconnaissance, EDL Imaging, and Coordinated Science with the High Resolution Imaging Science Experiment (HiRISE)

Kristin M. Block, Ingrid J. Daubar, Anjani T. Polit

The High Resolution Imaging Science Experiment (HiRISE) on the Mars Reconnaissance Orbiter (MRO) is critical to both the science and safety of current and future Mars missions. HiRISE does landing site reconnaissance and science aiding in site selection, as well as descent imaging during the entry, descent, and landing mission phase, and imaging of and coordinated science with landed assets. The high-resolution images (up to 25 cm/pixel) HiRISE provides were key in the landing site selection process for both the Phoenix Mars Lander and Mars Science Laboratory (MSL Curiosity) (see Golombek, Huertas, et al. 2012; Golombek et al. 2012; Golombek et al. 2008; Grant et al. 2011; Spencer et al. 2009). HiRISE is now imaging potential landing ellipses for InSight through cooperation between MRO and InSight teams (Golombek et al. 2013). After an initial characterization phase consisting of sample images in the proposed ellipses, HiRISE will continue with a systematic imaging campaign to extend coverage in select areas and begin acquiring stereo coverage. Digital elevation models (DEMs) constructed from the stereo pairs will be used to investigate the potential landing ellipses for rock abundance and other hazards, with the ultimate goal of certifying the selected landing ellipse. HiRISE data will also no doubt be critical for landing site certification for the recently announced 2020 Mars science rover. Twice HiRISE has successfully imaged spacecraft during their descent through the martian atmosphere (Phoenix and MSL). These images require considerable advance cooperation between operations teams and special sequence builds and were important in their potential for troubleshooting any unexpected events during descent and landing, e.g. problems with parachute deployment. Additionally, HiRISE provides the imagery necessary for post-EDL assessment by capturing the landing zone in high resolution soon after landing. This is again of high importance in the case of landing anomalies, fortunately not experienced in these two cases. HiRISE also supports tactical science planning (detailed traverse planning using HiRISE DEMs in the case of both MSL and MER) and coordinated science with landed assets (e.g. Tamppari et al. 2010). Imaging of defunct assets (e.g. Viking) is also performed by HiRISE, as well as searches for "lost" spacecraft such as Mars Polar Lander, Beagle, and soviet missions such as Mars 3, in attempts to understand and learn from past mission anomalies. References Golombek, M., A. Huertas, D. Kipp, and F. Calef (2012), Detection and Characterization of Rocks and Rock Size-Frequency Distributions at the Final Four Mars Science Laboratory Landing Sites, Mars, 7, 1–22, doi:10.1555/mars.2012.0001. Golombek, M. et al. (2012), Selection of the Mars Science Laboratory Landing Site, Space Science Reviews, 170 (1-4), 641–737, doi:10.1007/s11214-012-9916-y. Golombek, M. P. et al. (2008), Size-frequency distributions of rocks on the northern plains of Mars with special reference to Phoenix landing surfaces, Journal of Geophysical Research, 113, E00A09, doi:10.1029/2007JE003065. Grant, J. A., M. P. Golombek, J. P. Grotzinger, S. A. Wilson, M. M. Watkins, A. R. Vasavada, J. L. Griffes, and T. J. Parker (2011), The science process for selecting the landing site for the 2011 Mars Science Laboratory. Planetary and Space

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Mars atmosphere reconstruction using FADS on ExoMars EDM

Bart Van Hove, Ozgur Karatekin

We will present our methodology for angle of attack, side slip and stagnation pressure reconstruction from the Flush Air Data System (FADS) instrumentation on the heat shield of the 2016 ExoMars Entry, Descent and Landing Demonstrator Module (EDM). The atmospheric density during hypersonic entry can be derived from stagnation pressure measurements by modeling the flow through the bow shock. We will present a simple approach to reconstruct atmospheric density, taking into account real gas high temperature effects. The benefits of heat shield FADS instrumentation with regard to reconstructing atmospheric profiles will be discussed. As opposed to previous Mars missions, the ExoMars EDM will land on Mars in the dust season. The impact of atmospheric dust loading on the reconstructed profiles will be investigated. Similarly, the presence of winds on Mars might affect FADS reconstructed quantities such as the angles of attack and side slip. We will present a preliminary uncertainty analysis on simulated ExoMars EDM trajectories and associated simulated FADS data, with regard to atmospheric profile reconstruction.

Filling Mars Human Exploration Strategic Knowledge Gaps with Next Generation Meteorological Instrumentation

Scot Rafkin, Don Banfield, Rich Dissly

NASA's Human Exploration and Operations Mission Directorate (HEOMD), together with the Mars Exploration Program (MEP), have identified Strategic Knowledge Gaps (SKGs) that must be filled to enable future human exploration of Mars. Many of these SKGs are related to reducing risk of and increasing precision for Entry, Descent, and Landing (EDL) of spacecraft. Within this subset of EDL SKGs, the tasks of better characterizing atmospheric properties, and validating and improving atmospheric numerical models are heavily represented. Previous surface meteorological stations have been flown to Mars, beginning with Viking and most recently with the Rover Environmental Monitoring Station (REMS) on the Mars Science Laboratory. To date, Viking remains the best meteorological data set. While additional data similar to Viking, obtained from fully functional and appropriately accommodated instrumentation, would be incrementally useful to further validate atmospheric models, such data does little to identify why the models are in error, nor does such data provide the confidence to extrapolate model results from one site to other locations. More of these same data will do little to make significant progress on the atmospheric SKGs. To make meaningful headway on the HEOMD SKGs, a new class of meteorological sensors is needed. These sensors provide not only the basic meteorological parameters previously measured (e.g., temperature, horizontal wind, and pressure), but also provide information about the physical mechanisms responsible for producing the observed meteorological values. By measuring turbulent fluxes of heat, and momentum, and the radiative fluxes, the energy inputs that collectively drive the atmosphere can be measured for the first time. With these additional data, the actual forcing of the Martian atmosphere can be compared with the model-predicted forcing. Different models give different results and give different forcing under nearly identical conditions, and there is presently insufficient information to understand or resolve these differences. Importantly, given the appropriate information about forcing, it becomes possible to not only validate models, but understand model errors and then fix those errors. By improving the underlying model physics, the models can then better simulate the atmosphere at any given location; this extends the models utility beyond just one landing site with data to future landing sites where little or no data are available. A suite of modeling studies are presented to show the importance of measuring the forcing as well as the more traditional meteorological parameters. Using data from previous missions and from the numerical simulations, the impact of errors in model forcing are shown to contribute significantly to model error and uncertainty in atmospheric predictions. For example, an error of just a few W/m^2 in turbulent energy flux is estimated to produce greater than a 2% density error in the lowest 40 km of atmosphere at Gale Crater, where MSL landed. Density changes of 5% or more were considered important for MSL, so errors just in fluxes can contribute significantly. Given the importance of the forcing terms, a suite of sensors to measure these terms on the surface of Mars is presented. These sensors include an acoustic anemometer, tunable laser spectrometer, visible and

infrared radiometers, and a pressure sensor. All of these sensors have been under development for as much as ten years and are now mature and ready for final flight development. The implementations are low risk, low power, robust, and cost effective. The proposed sensor suite will provide substantial return on investment by addressing the SKGs and reducing the cost and effort needed to engineer around the atmospheric uncertainty and unknowns. Concomitantly, the instrumentation will provide a wealth of novel scientific climate data, much of which is given the highest priority by the MEP.

Session 4: Giant Planets

The MAss Spectrometer for Planetary Exploration (MASPEX)

Tim Brockwell, J Hunter Waite

The MAss Spectrometer for Planetary EXploration (MASPEX) is a high sensitivity, variable resolution, time of flight mass spectrometer designed to meet the science requirements of a variety of missions. The versatility of the instrument is demonstrated by its application to the requirements of the Europa Jupiter System Mission (EJSM) and most recently the Jupiter Icy Moons Explorer missions (JUICE). The key characteristics of sensitivity, scan rate, and resolution are related to the requirements of different mission phases and extrapolated to other future missions. The science requirements of the JUICE mission allied with the origin objectives of the EJSM mission flow down into a science traceability matrix with four science objectives and on into eight required measurements. The driving requirement for each measurement is identified and is; mass resolution, sensitivity, or spatial resolution (related to scan rate). The design of MASPEX enables each of these parameters to be varied allowing the instrument to fulfill all of the required measurements by changes in its methodology. Resolution: The measurement of the D/H ratio in water is important in determining the formation conditions of the Galilean satellites and the analysis requires a resolution of 12,283 to separate the H217O and HD16O isotopologues. MASPEX's ability to change the length of the ion flight path, as required to measure specific ions, enables customized methodology for specific ions. This flexibility is relevant to all missions as in any gas mixture there are a range of resolutions required for complete analysis. By limiting the regions of the spectrum that are measured at high resolution more data can be taken in the other regions leading to lower detection limits and higher precision in measurement. Sensitivity: The measurement of noble gases and trace organic species provides information about the formation and habitability of the satellites but requires great sensitivity. The use of a cryotrap can concentrate noble gases by factors of 107 and the increased sample size and high sensitivity storage source enable ppm level organic materials to be measured. The low detection limits for noble gases offered by MASPEX's cryotrapping system makes noble gases accessible even in cometary comas. The combination of high sensitivity and high resolution provides the greatest insight into mixtures of organic materials making MASPEX suitable for planetary probes and landed missions to targets such as Titan. Spatial resolution: The requirement to observe spatial features as small as 400km of orbital track needs high scan rates to collect the data. The ability of MASPEX to operate at 5000 extractions per second with each extraction resulting in a complete spectrum leads to higher precision measurements than can be achieved by a scanning instrument taking seconds to measure the mass range. MASPEX's high extraction rate provides the spatial resolution required for flyby / orbital and planetary probe mission where the sample composition may be changing rapidly but it also increases the detection limit for trace species and improves measurement precisions overall, aiding suitability for lander missions as well.

Atmospheric Entry Studies for Saturn Missions: 45° Sphere-Cone Rigid Aeroshells and Ballistic Entries

Dinesh K. Prabhu, Gary A. Allen, Thomas R. Spilker, Gelsomina Cappuccio, Helen H. Hwang, Robert W. Moses

To further understand the entry technology needs of a New Frontiers class mission to Saturn recommended by the National Research Council's Planetary Science Decadal Survey [1], NASA's In-Space Propulsion Technology (ISPT) program sponsored a rapid study, the results of which form the basis of the proposed paper. The study team analyzed direct ballistic entries of 45° sphere-cone rigid aeroshell probes with a well-characterized thermal protection material, entry-grade carbon phenolic [2], to establish performance baselines so that future comparisons could be made with emerging materials technologies. The proposed paper will present results of an exploration of an entry trajectory space parameterized by heading angle, entry velocity, ballistic coefficient, and entry flight path angle. The exploration is guided in part by the Galileo mission experience, and in part by the work of Spilker [3]. Equatorial and high-latitude entries are considered for inertial entry velocities ranging from 35 to 38 km/s. Various entry masses – ranging from 200 to 350 kg – and diameters – ranging 0.8 to 1.3 m – are considered in the development of 3DoF flight trajectories [4] for entry flight path angles between skip out and ?30°. From these trajectories, the entry flight path angle windows are determined through the use of mechanical and thermal performance limits on the thermal protection material and science payload. For mechanical performance, which determines the steepest entry angle, a 200 g limit is placed on the peak deceleration load experienced by the science instruments, and 10 bar is assumed as the pressure limit for entry-grade carbon-phenolic material. For thermal performance, which determines the shallowest entry angle, inflection points in the total heat load distribution are used as cut off criteria. The proposed paper will explore the use of these entry flight path angle windows to specify performance parameters for a new generation of ablative thermal protection materials, with entry-grade carbon phenolic used as a reference. References: 1. National Research Council, Visions and Voyages for Planetary Science in the Decade 2013-2022, National Academies Press, 2012. 2. Gasch, M., "Re-Entry Grade Carbon Phenolic for NASA Planetary Probes - Past History and Assessment of Current Availability," 60th JPM/9th MSS/7th LPS/6th SPS/Joint Subcommittee Meeting (ITAR Restricted), Colorado Spring, CO, Apr.-May 2013. 3. Spilker, T. R. and Atkinson, D. H., "Saturn Entry Probe Potential for Uranus and Neptune Missions," 9th International Planetary Probes Workshop, Toulouse, France, June 2012. 4. Gary A. Allen, Jr., Michael J. Wright, and Peter Gage, "The Trajectory Program (Traj): Reference Manual and User's Guide," NASA/TM-2004-212847, 2005.

CONCEPT OF A SATURN PROBE MISSION FOR THE FUTURE L2/L3 ESA CALL

Olivier Mousis, Leigh N. Fletcher, Nicolas André, Michel Blanc, Athéna Coustenis, Daniel Gautier, Wolf Dietrich Geppert, Tristan Guillot, Patrick Irwin

In response to ESA's call for science themes for the L2 and L3 missions in the coming decades, we intend to submit a white paper describing the scientific rationale and potential mission architectures for in situ exploration of our giant plyanets. This white paper will describe in particular the case for a Saturn entry probe presented as the next logical step beyond the Galileo Jupiter probe, the Juno mission and Cassini's orbital remote sensing. The results from Galileo have informed our understanding of the atmospheric properties (temperatures, winds, composition, clouds, waves, etc.) of Jupiter as a template for gaseous worlds throughout our galaxy, and have provided crucial constraints on theories of the origins and evolution of planets in our solar system. The Galileo probe provided a step-change in our understanding of Jupiter, but was it truly representative of our diverse collection of gaseous giants? We present the concept of a single atmospheric probe dedicated to the chemical and isotopic analysis of the atmosphere of Saturn that will be submitted to the L2-L3 2014 ESA call.

Characterizing Uranus with an Ice giant Planetary Origins Probe (Ice-POP)

Mark Marley, Jonathan Fortney, Nadine Nettelmann, Kevin Zahnle

We now know from studies of planetary transits and microlensing that Neptune-mass planets are ubitquitous and may be the most common class of planets in the Galaxy. As such it is crucial that we understand the formation and evolution of the ice giant planets in our own solar system so that we can better understand planet formation throughout the galaxy. An entry probe mission to Uranus would help accomplish this goal. In fact the Planetary Decadal Survey recommended a Uranus orbiter with entry probe but did not explore in detail the specifications for the entry probe. NASA Ames is currently studying thermal protection system requirements for such a mission and this has led to questions regarding the minimum interesting science payload of such an entry probe. The single most important in-situ measurement for an ice giant entry probe is a measurement of atmospheric composition. For Uranus this would specifically include the methane and noble gas abundances. An in situ measurement of the methane abundance, from below the methane cloud, would constrain the atmospheric carbon abundance, which is believed to be roughly 30 to 50 times solar. There are hints from the transiting planets that extrasolar ice giants show comparable or even greater enhancements of heavy elements compared to their primary stars. However the origin of this carbon enhancement is controversial. Is Uranus a “failed core” of a larger gas giant or was the atmosphere enhanced by accretion of icy planetesimals? Constraining atmospheric abundances of C and perhaps S or even N from below 5 bars would provide badly needed data to address such issues. A measurement of the N abundance would provide clues on the origin of the planetesimals that formed Uranus. Low N-abundance indicates planetesimals from 'warmer' regions where N was mainly in form of NH3, whereas a strong enrichment could indicate planetesimals / cometary material from the colder outer regions of the nebula. Furthermore CO and HCN have been detected in Neptune but not in Uranus. A measurement of the abundance of either would constrain the source mechanisms for these molecules (exogenic or internal). A major surprise from the Galileo Entry Probe was that the heavier noble gases Ar, Kr, and Xe are enhanced in Jupiter’s atmosphere at a level comparable to what was seen for the chemically active volatiles N, C, and S. It had been generally expected that Ar, Kr, and Xe would be present in solar abundances, as all were expected to accrete with hydrogen during the gravitational capture of nebular gases. Enhanced abundances of Ar, Kr, and Xe is equivalent to saying that these noble gases have been separated from hydrogen. There are several mechanisms that could accomplish this but these hypotheses require further testing. Measurement of noble gas abundances in an ice giant would constrain the planetary formation and nebular mechanisms responsible for this enhancement. Standard three-layer models of Uranus find that the outer, predominantly H/He layer of Uranus does not reach pressures high enough (~1 Mbar) for H2 to transition to liquid metallic hydrogen. However, valid models can also be constructed with a smaller intermediate water-rich layer, with hydrogen then reaching the metallic hydrogen phase. If this occurs, He should phase separate from the hydrogen and ``rain out,” taking along a

substantial abundance of Ne, as suggested for Jupiter (and likely also for Saturn). Hence He and Ne depletions could be probes of the planet's structure in the much deeper interior. A determination of Uranus' atmospheric abundances, particularly of the noble gasses, is thus critical to understanding the formation of Uranus, and giant planets in general. These measurements can only be performed with an entry probe. The second key measurement would be a temperature-pressure sounding to provide ground truth for remote measurements of atmospheric temperature and composition and to constrain the internal heat flow. This would also establish that the methane abundance measurements have indeed been made below any possible methane cloud. Finally an ultra stable oscillator would measure wind speeds and constrain atmospheric dynamics. In our presentation we will discuss the importance of all of these measurements and argue that an entry probe is a crucial component of any ice giant mission.

Synergistic Science With an Entry Probe and Carrier Vehicle at Uranus

Thomas R. Spilker, Mark D. Hofstadter, Neil Murphy

Discussions over the years have pointed to potential synergistic observations and measurements between an atmospheric entry probe and a vehicle outside the atmosphere. Measurements by instruments on the entry probe could greatly aid in the interpretation of remote sensing measurments made by instruments on an extra-atmospheric vehicle such as an orbiter or flyby spacecraft, or vice-versa. Examples include some associated with commonly considered probe and remote sensing instruments: thermal profile measurements by the probe could greatly reduce interpretation uncertainties in radiometric measurements by exo-atmospheric instruments such as an infrared mapping spectrometer or a microwave radiometer; imaging from the exo-atmospheric vehicle could supply valuable context information about the probe entry location. Development of new instruments and techniques is expanding the range of potential synergistic measurements. Notably, Doppler spectrographic imaging instruments are beginning to open the door to a new type of investigation for the giant planets: planetary normal mode seismology. When applied to our sun, the technique of helioseismology provides great inroads concerning the sun's interior structure. The same technique applied to the giant planets could significantly advance our understanding of giant planet interior structure. Recent ground-based observations of Jupiter appear to confirm that there are dectectable seismological signals present there. So far no signals have been detected from the other giant planets, probably due to the combination of lower signal intensities at those planets and larger distances from Earth that decrease the SNR attainable with ground-based instruments. Moving the instrument from Earth to the near vicinity of a target planet would significantly help with that SNR problem and could enable interior structure measurements at the ice giant planets. This paper will discuss the potential for synergistic measurements at Uranus between normal mode seismological observations from an exo-atmospheric platform and measurements from an atmospheric entry probe.

Session 5: Airless Bodies

AN OVERVIEW OF SMALL BODY SAMPLE RETURN MISSION CANDIDATES AT ESA

David Agnolon, Denis Rebuffat, Jens Romstedt

ESA's Rosetta spacecraft is reaching its final destination, the comet 67P/Churyumov-Gerasimenko in May 2014. Building on the development resulting from this major undertaking, ESA is planning a number of follow-up airless body missions. The ESA's Cosmic Vision M3 (Medium class #3) Programme features several candidate missions from which one will be selected for a launch in the early 2020's. It includes a sample return mission from a Near Earth Asteroid (NEA) called Marco Polo R. Moreover, ESA's Mars Robotic Exploration Preparation (MREP) Programme is preparing candidate missions to Mars to be launched after the 2016 and 2018 ExoMars missions. MREP includes a Phobos Sample Return mission, called Phootprint. Both of these small body sample return missions aim at returning a sample in the order of hundreds of grams and therefore many synergies can be exploited. This paper presents these two mission concepts. The overall architecture and mission scenario and design will be introduced and emphasis will be given on the key sample return capabilities/technologies of the mission concepts, i.e. the asteroid descent and touchdown GNC and operations, the landing and/or touchdown system, the sampling mechanism as well as the associated sealing and sample transfer system and the Earth re-entry capsule. A dedicated paper is proposed on the European re-entry capsule technologies in a separate session, covering both of these missions, so the main focus of this paper will be on the near-asteroid operations and the sampling spacecraft design. Even though both missions target a low-gravity body, Phobos is over 20 km while the target of MarcoPolo-R, 2008 EV5 is only 400 m. Besides, the specifics of the mission environment as well as programmatic boundaries have a non-negligible influence on the design. These mission-dependent aspects altogether will be highlighted wherever relevant and design differences between a Phobos sample return mission and a smaller near-Earth asteroid will be discussed throughout. In addition, a number of technology pre-development have been completed, ongoing or initiated supporting the definition of these two missions with respect to the sampling tool, the GNC hardware and software for near-asteroid operations, the touchdown and/or landing gear, etc. The purpose of these development is to bring about the maturity of these innovative capabilities to a maturity sufficient to enter the implementation phase, and therefore to reach TRL 5 with the demonstration of a breadboard in a relevant environment. We will see how this is being pursued through the description of the breadboard design, the test environment and set-up and the test facilities which are being used. Needless to say, for a low-gravity body, this is not straightforward and need to involve specific and innovative test procedures such as parabolic flights, "rendezvous and docking"-like test benches, microgravity-simulated touchdown facilities, etc. This also increases the understanding of what needs to and can be done throughout the verification and validation cycle of the project leading to the flight model development, allowing to build a robust and well-identified development approach.

MarcoPolo-R: Mission and Spacecraft Design

Lisa Peacocke, Steve Kemble, Héloïse Scheer, Jean-Marc Bouilly

The MarcoPolo-R mission is a candidate for the European Space Agency's medium-class Cosmic Vision programme, with the aim to obtain a 100 g sample of asteroid surface material and return it safely to the Earth. Astrium is one of two industrial contractors currently studying the mission to Phase A level, and the team has been working on the mission and spacecraft design since January 2012. Asteroids are some of the most primitive bodies in our solar system and are key to understanding the formation of the Earth, Sun and other planetary bodies. A returned sample would allow extensive analyses in the large laboratory-sized instruments here on Earth that are not possible with in-situ instruments. This analysis would also increase our understanding of the composition and structure of asteroids, and aid in plans for asteroid deflection techniques. In addition, the mission would be a valuable precursor for missions such as Mars Sample Return, demonstrating a high speed Earth re-entry and hard landing of an entry capsule. Following extensive mission analysis of both the baseline asteroid target 1996 FG3 and alternatives, a particularly favourable trajectory was found to the asteroid 2008 EV5 resulting in a mission duration of 4.5 to 6 years. In October 2012, the MarcoPolo-R baseline target was changed to 2008 EV5 due to its extremely primitive nature, which may pre-date the Sun. This change has a number of advantages: reduced DeltaV requirements, an orbit with a more benign thermal environment, reduced communications distances, and a reduced complexity propulsion system – all of which simplify the spacecraft design significantly. The single spacecraft would launch between 2022 and 2024 on a Soyuz-Fregat launch vehicle from Kourou. Solar electric propulsion is necessary for the outward and return transfers due to the DeltaV requirements, to minimise propellant mass. Once rendezvous with the asteroid is achieved, an observation campaign will begin to characterise the asteroid properties and map the surface in detail. Five potential sampling sites will be selected and closely observed in a local characterisation phase, leading to a single preferred sampling site being identified. For the sampling phase, the spacecraft will perform a touch-and-go manoeuvre. A boom with a sampling mechanism at the end will be deployed, and the spacecraft will descend using visual navigation to touch the asteroid for some seconds. The rotary brush sampling mechanism will be activated on touchdown to obtain a good quality sample comprising regolith dust and pebbles. Low touchdown velocities and collision avoidance are critical at this point to prevent damage to the spacecraft and solar arrays. The spacecraft will then move away, returning to a safe orbit, and the sample will be transferred to an Earth Re-entry Capsule. After a final post-sampling characterisation campaign, the spacecraft will perform the return transfer to Earth. The Earth Re-entry Capsule will be released to directly enter the Earth's atmosphere at a high speed of 11.6-12.8 km/s. The capsule is designed to survive a hard landing with no parachute deceleration, as would be required for Mars Sample Return, and the thermal protection system and crushable material used will be described. No beacons will be included in the capsule, so detection and recovery will occur from the ground. The asteroid sample would then be extracted in a sample curation facility in preparation for the full analysis

campaign. This presentation will describe Astrium's MarcoPolo-R mission and spacecraft design in detail, with a focus on the innovative aspects of the design.

Philae-Lander Touchdown Dynamics Revisited – Tests for the Upcoming Landing Preparations

Lars Witte, Silvio Schroeder, Reinhard Roll, Stephan Ulamec, Jens Biele, Joachim Block, Tim van Zoest

In view of the upcoming landing operations end of 2014 a partial retesting of the Philae-lander touchdown system has been done in spring 2013 by a consortium of DLR Institutes and the Max-Planck-Institute of Solar System Research. Whilst intensive testing was done as part of Philae's design development and verification program these new test series addresses touchdown conditions which have been out of capability of the pendulum test facility used at that times. These primary tests made use of a pendulum facility, letting the test object swing against a vertical wall to separate the Earth-weight from the forces of inertia. However a limitation of this concept is a severely constrained motion of the test object and the inability to touch down on loose granular material. This disadvantage are overcome by using an active weight-offloading device which is provided for the recent new tests by DLR's Landing & Mobility Test Facility (LAMA). The retesting aims at first at the intercomparability of the different test facility concepts and to identify and separate vehicle motions from facilities induced motions. The follow-on tests then consequently focus on touchdown conditions such as asymmetric load cases, effects from terrain undulation and the understanding of granular soil mechanics which could not be addressed sufficiently by the original pendulum tests. This paper recalls briefly the Philae-lander touchdown system, introduces the test facility concept and its operating mode. The effects of the different facility concepts are highlighted using test data from both the heritage and the re-testing campaign. The paper also presents and discusses the preliminary results from this recent campaign and gives an outlook to its further use in the upcoming landing preparations.

Europa Penetrator Design and Survivability Testing

Sanjay Vijendran, Marie-Claire Perkinson, Lester Waugh, Mike Williams, Tom Kennedy, Phillip Church, Jeremy Fielding, Nick Taylor

Planetary penetrators are hard landers, designed to survive a high-g impact into a planetary body and operate from the subsurface. The concept is well established and has been taken to the level of flight programmes by the US, Russian and Japanese space agencies. The Japanese Lunar-A probe, the Russian Mars-96 and NASA's Deep Space 2 concepts all included penetrators of various sizes and complexities. Sadly none achieved mission success due to cancellation, launch failure and loss of contact after Mars entry respectively. The European Space Agency started a Penetrator technology development programme in 2009, completing its first phase of development in 2010 with a feasibility study and design of penetrator concepts for missions to the Jovian moons Ganymede and Europa. The study provided promising concepts which are now being pursued in Phase 2 of the development where a design of an astrobiology-focused penetrator for Europa has been made and is now undergoing validation through small and full-scale high-velocity impact testing. The key science focus for the current penetrator is Astrobiology, the key science goals can be achieved within the first day of operation but a longer lifetime is required for the transmission of the science data to the orbiter. The approach taken is to design the Penetrator with two separate bays. The front bay will be a short lifetime bay which will sample the surface and complete all analysis and data transfer within 10 hours. The rear bay is a warm bay which will house the penetrator support systems required to transmit all collected data to the orbiter. Small scale trails have been undertaken at the University of Cambridge Cavendish Laboratory to validate the impact modelling techniques and the robustness of critical components. A range of trials have been carried out to assess survivability of key elements of the design, including the sampling mechanism, potting compounds, accelerometers, shell, batteries and Torlon suspension springs. Full scale trials are planned to test the overall structural integrity of the system and the penetration profile, this programme will be carried out in June 2013 at the MoD test range in Pendine South Wales. Two targets (sand and ice) will be used to test the survivability of the design for missions to different planetary surfaces. This paper will present the results of the small scale tests and the planning for the full scale test programme.

Geophysical Measurements on Small Bodies using Explosive Surface Probes

Shane Roark, Daniel Scheeres, Richard Dissly, William Frazier

Simple and inexpensive explosive surface probes are being developed for small-body rendezvous missions. The key components are a charge of high-energy explosive to perform active cratering experiments and accelerometers or a geophone to detect impact and seismic response from the blasts. The system is housed in an orientation-independent structure with all of the required support subsystems. Flyby and rendezvous missions to asteroids and comets have been important for understanding the nature and diversity of small solar system bodies. Sample return missions extend our knowledge to the micro-scale by enabling rigorous laboratory analyses that can be used to deter-mine the connection between meteorites and asteroid types, and the detailed processes of planetary formation. Geophysical experiments using active cratering will compliment past and future rendezvous and sample return missions by determining the mechanical properties at the surface and subsurface of small bodies in their natural environment, which cannot be replicated on Earth. In addition, geophysical measurements will provide crucial precursor information for human missions. Multiple copies of the probes would be deployed from an orbiting spacecraft to the surface of a small solar system body, forming a temporary “net” of sensors that would measure fundamental properties associated with the strength, cohesion, porosity and compaction, and seismic wave transmission of the regolith. High-resolution imaging from the orbiting spacecraft would observe the resulting crater size, crater formation time, characteristics of the ejecta plume, and morphology of the ejecta blanket. The crater size and formation time provide a sensitive measurement of the strength and cohesion of the surface material by extrapolating from gravity scaling curves generated from decades of laboratory and field cratering experiments over a wide range of conditions. Images of the topography of the ejecta blanket can be used to determine if crater formation is dominated by the strength of the regolith or gravity. If crater formation is dominated by gravity, the ejecta blanket will follow a known geometric profile. The cratering experiments also will enable current space weathering hypotheses to be directly tested by using remote spectral observations of the surface before and after it has been excavated by the blast. Inclusion of appropriate seismic sensors will help constrain the seismic properties of small solar-system bodies. Detection and measurement of the pressure waves by probes away from the blast, combined with remote observations of changes in the surface, will enable characterization of the efficiency of seismic shaking and measurement of the wave speed through the surface. Computational and experimental work [1] indicate that the arrival time, amplitude, and width of the first peak in the pressure wave emanating from a blast can be used as strong indicators of the degree of consolidation and strength of the medium. Arrival times for analogous seismic experiments during the Apollo program were used very successfully to determine the degree of regolith cohesiveness and layering on the moon [1]. Impact accelerometers are incorporated to detect initial contact with the surface, indicate the strength of the regolith upon impact, determine if or when the probes come to rest after impact, and determine if the probes are separated

from the surface as a result of high-amplitude shock waves. The breadboard probe system currently is at a Technology Readiness Level (TRL) of 4. The system consists of a spherical external structure, a cubic internal housing, the explosive, an impact three-axis accelerometer, three single-axis seismic accelerometers, batteries, two patch antennas, thermal control, electronics cards to support the avionics, power conditioning and communications, and a release mechanism that interfaces to the host spacecraft. The requirements were derived assuming a mission to a near-Earth asteroid, and the complete package is expected to have a diameter of roughly 25 cm, a mass less than 15 kg, and require an average power near 3 W. Reference: [1] J.D. Walker and W.F. Huebner (2004) Seismological Investigation of Asteroid and Comet Interiors; in Mitigation of Hazardous Comets and Asteroids, Cambridge University Press, Cambridge, UK, 234-265.

Session 6: Venus and Titan

Titan : Scientific Wonderland and Engineering Playground

Ralph Lorenz

Titan has perhaps the richest range of accessible meteorological, oceanographical, chemical, geological and astrobiological processes in our solar system. The wide pantheon of interactions between Titan's interior, surface, atmosphere and space environment have drawn the attention of several Flagship mission studies (2007 Titan Explorer and TandEM, and 2008/9 TSSM) and are the subject of ongoing observations by Cassini which is not only widening mapping coverage during its solstice mission, but is also observing dramatic seasonal changes. I will review some recent discoveries and the scientific motivations for future missions. The range of surface textures on Titan (sand dunes, seas, rugged mountains) motivate a range of lander and mobility options, and the thick atmosphere facilitates soft landing and both heavier- and lighter-than-air flight options. Thus engineers can reasonably propose almost any vehicle - airplane, balloon, helicopter, rover, hovercraft. The Titan atmosphere also is attractive for orbital insertion via aerocapture and/or aerobraking, and has some unique aerothermodynamic aspects. Titan exploration offers the widest canvas on which engineers can paint imaginative solutions to these scientific challenges.

Tensegrity Based Probes for Planetary Exploration: Entry, Descent and Landing (EDL) and Surface Mobility Analysis

George Gorospe, Jonathan Bruce, Atil Iscen, Adrian Agogino, David Atkinson, Vytas SunSpiral

Low-cost yet highly capable planetary entry probes have the potential to revolutionize science return even at today's modest space budgets. However, it is nearly impossible to imagine such probes being built with traditional rigid robotics and traditional landing systems due to the high weight, complexity, and high mission control costs of these systems. A novel probe concept based on tensegrity structures can overcome many of these limitations. The core tensegrity structure is entirely formed out of tensile components (cables) and compression components (light-weight rods) to create a tensile network. Our concept is to develop a tensegrity probe in which the tensile network can be actively controlled to enable compact stowage for launch followed by deployment upon landing. Due to their natural compliance and structural force distribution properties, tensegrity probes can safely absorb significant impact forces, enabling high speed Entry, Descent, and Landing (EDL) scenarios where the probe itself acts much like an airbag. However, unlike an airbag which must be discarded after a single use, the tensegrity probe can actively control its shape to provide compliant rolling mobility while still maintaining its ability to safely absorb impact shocks that might occur during exploration. This combination of functions from a single structure enables compact and light-weight planetary exploration missions with the capabilities of traditional wheeled rovers, but with the mass and cost similar or less than a stationary probe. In this paper we evaluate tensegrity probes on the basis of the EDL phase performance of the probe in the context of a mission to Titan. Titan's atmosphere, stable bodies of surface liquid and complex organics make it one of the most complex and Earth-like environments in the solar system. This tensegrity probe mission will build on the science returns from Huygens and answer many of the new and unresolved questions surrounding Titan's ongoing organic processes, geologic history, atmosphere, and surface-atmosphere interactions. Upon arrival at Titan, the tightly packed tensegrity probes will separate from the spacecraft, either individually or simultaneously to target specific regions or environments on the surface, and begin atmospheric descent. Without requiring parachutes, each probe is projected to impact the surface at about 11 m/s, absorbing and distributing impact stresses while protecting its science payload. As the tensegrity probes bounce, roll, and finally come to a rest on the surface of Titan, the actuated tensegrity structure will then begin to function as the primary mobility system for these mobile probes. Once on the surface, a notional science payload containing an atmospheric package, an analytical chemistry package, and an imaging package can begin the probes' science mission. Since each probe is equipped with a science payload, mission success is not entirely dependent on the survival of a single probe, and the deployment of only a few probes enables the distributed exploration of a planetary surface. An individual tensegrity probe has a significantly lower EDL hardware overhead and increased science payload mass percentage when compared to similar planetary surface probe missions, increasing the potential for science return. Compared with the Mars Exploration Rovers (MER) and Mars Science

Laboratory (MSL) which employed complex EDL systems to support roving vehicles with a science payload mass fraction of 1.6% for MER and 3.33% for MSL, a tensegrity probe may be able to operate on Titan as a mobile probe with a science payload mass fraction of up to 50% due to the dual use of the tensegrity structure as for both EDL and surface mobility. For applications on other targets such as Mars, and given an EDL scenario similar to MER with a 100 km/h (27.7 m/s) impact velocity, a tensegrity probe may be able to survive EDL and carry a higher percentage science payload by mass. A key quality of these probes is their ability to shock-protect core systems (science payloads, avionics, and actuators), even when landing at relatively high speeds. We will discuss analytical, simulated, and experimental results of terminal velocity landing events to show how the impact forces are distributed through the structure and reduce the internal stress at any one point within the probe. We will then discuss how these results are guiding engineering choices for actuation and material properties of the tensegrity probe and our ongoing efforts to develop and test physical prototypes. Given the multi-function use of the tensegrity probe, we will then describe how the structure can also enable mobility for exploration of a landing site. Finally, the tensegrity probe is directly compared to other EDL and mobility solutions with a focus on cost, complexity, delivered science payload, and robustness.

TITAN MONTGOLFIERE BALLOON ANALYSIS AND DESIGN USING COMPUTATIONAL FLUID DYNAMICS SIMULATIONS

Yuri Feldman, Jeffery L. Hall, Andre Vargas, Tim Colonius, Walter Dieudonne, Jack A. Jones, Kim Reh, Julian Nott

In this paper a multiple-layer heated balloon is considered for future Titan missions. We describe computational fluid dynamics (CFD) studies aimed at predicting the buoyancy for double-, triple-, and quadruple-walled balloons, and determining the sensitivity to the location of an external heat source. The buoyancy predictions from CFD show that the effective gap conductivity is higher than what is predicted by engineering correlations. Direct and large-eddy simulations (DNS/LES) are carried out for an idealized concentric spherical annulus with isothermal walls in order to investigate the source of the discrepancy and the resulting data is then used to design a new correlation. Results for multiple-layer balloons show the allowable scientific payload grows rapidly with increasing number of walls. Next, the location of the heat source is analyzed, and we consider locations internal and external to the balloon. It is found that regardless of its location, an externally located heat source generates too much wasted heat and balloon performances is degraded substantially. A second design study deals with the external payload (gondola) and its effect on the balloon filling (blockage effects). It is found that even though important reductions in filling rates take place, it does not affect the overall descent rate, primarily due to the low gravity on Titan.

Recent Advances in Planetary Balloon Technology

Kim Reh, Jeff Hall, Andre Vargas

This paper summarizes recent advances in Titan and Venus balloon technology resulting from ongoing work by JPL, CNES, Caltech and a collection of industrial partners. Computational fluid dynamics models of Titan Montgolfiere (hot air) balloons have been validated with small scale cryogenic experiments. These validated models have been used to explore numerous aspects of the design space including payload mass versus balloon size and heat input, the effect of different RTG locations on the buoyancy production, and the effect of gondola to balloon separation distance on the filling efficiency of the balloon during initial deployment and inflation. Cryogenic testing of a 1.5 m diameter helium superpressure Titan balloon verified the ability of the construction materials to tolerate the extreme cold environment and quantified the helium leak rate over a multi-day period. Continued prototyping and laboratory testing of materials for Venus superpressure balloons yielded an upgraded laminate with improved acid resistance and helium permeation properties. A new 7 m diameter prototype Venus balloon based on this upgrade laminate material is under construction at the current time with buoyancy and leakage testing planned for the summer of 2013. The paper concludes with a discussion of how these technology advances impact the risk posture of planetary balloon technology for Titan and Venus and the implications for future mission opportunities.

Science Questions and Needed Measurements in the Neighborhood of the Cloud Region of Venus

S.S. Limaye, David Crisp, Tibor Kremic, Lori Glaze

All in situ measurements of the Venus atmospheric composition and dynamics by Venera, VEGA, and Pioneer Venus entry probes were obtained at altitudes below ~ 62-65 km. The two VeGa balloons measured atmospheric properties at a level of about 54 km above the surface. From the few measurements of the cloud properties of Venus on day and night side, it is be-lieved that the bottom of the cloud layer is at approxi-mately 48-50 km. Both remote sensing and entry probe measurements indicate that the cloud top was above the highest altitudes where the measurements were obtained.Analysis of remote sensing observations from Pioneer Venus Orbiter Cloud Photopolarimeter (OCPP) showed that the cloud tops extend to as high as 71 km at latitudes below 70 degrees, and a haze layer of smaller particles extended as high as 90 km above the surface. Observations from the Pioneer Ve-nus Orbiter [2,3] and entry probes [4] indicate that the unknown absorber that causes the cloud-top ultraviolet contrasts is confined within the upper cloud (57.5 – 71 km) and is responsible for absorbing almost half of the solar radiation deposited on the planet[5]. How-ever, its global distribution is not well characterized. Venus Express observations made since its arrival at Venus in April 2006 have provided new information about the cloud top level of Venus from near infrared observations from VIRTIS, from solar occultation ob-servations from SPICAV/SOIR and from imaging ob-servations from the Venus Monitoring Camera (VMC). These recent observations indicate that the cloud tops are at about 73-75 km in equatorial latitudes and dip to about 65-67 km in polar latitudes. Meanwhile, numerical simulations of the atmos-pheric circulation continue with the goal of reproduc-ing the observed superrotation of the atmosphere with its~4 day period at the cloud top levels. Near infrared measurements from the ground [6] and from fly-by's and orbiters [6,7] provide insights into the circulation, within the cloud deck, but much less is know about the the winds in the lower atmosphere (e.g. the altitude of the peak kinetic energy and angular momentum densi-ties). The recent efforts by many groups have been able to achieve some resemblance to the observed cir-culation, but only by modifying the heating rates in the models. Indeed, one of the dominant problems in the pro-gress of such efforts is the uncertainty in the knowl-edge of where in the atmosphere is being heated by absorbed solar energy and by the absorption of the thermal radiation emitted from the lower atmosphere and surface [8]. These and other questions have motivated a com-plete of review of our current understanding, and the measurements needed to characterize the thermal structure, composition, and dynamics of cloud-top regions. To conduct this review,, a one day meeting is being held on 24 January 2013 at the NASA Glenn Research Center in Cleveland, Ohio. The goal of this meeting is to consider whether existing tools and tech-nologies are capable of yielding any new measure-ments required and identify technology needs as nec-essary. The outcome of this workshop will be summa-rized at this conference. References: [1] Knollenberg. et al. (1980) JGR,85, A13, 8039-8058. [2] Travis, L. D., et al., (1979) Science, 203, 781–784; Lane and Opstbaum, 1983, Icarus, 54, 48-58. [3] Tomasko et al. (1980) JGR,85,A13, 8167-8786. [4] Tomasko et al. (1985)

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Comparison of Aerial Platform Options for investigating the Upper Atmosphere of Venus

James A. Cutts, Patricia M. Beauchamp

In situ exploration of the upper atmosphere of Venus from approximately 57 to 72 km could provide the answers to many important questions about Venus. This region contains a time variable ultraviolet absorber whose composition is unknown but which controls many aspects of the heat balance on Venus. Temperature and pressure conditions at the base of this zone are not dissimilar to those at the Earth's surface. Understanding the composition and dynamics of this unknown UV absorber is an important science goal for Venus. Various options for in situ exploration of this region have been identified, some of which are described in companion papers: low ballistic coefficient entry probe, Lighter-than-Atmosphere (LTA) vehicles (balloons), solar-powered Heavier- than Atmosphere (HTA) vehicles (airplanes) as well as hybrid LTA/HTA options (blimps). The options differ in terms of the altitude range in the atmosphere they cover, the range in times of day that are accessible, the ability to investigates variations with latitude and the sampling efficacy. This paper compares the effectiveness of each of these platforms for addressing the fundamental scientific questions about this region of the Venus atmosphere. It also discusses issues of technology readiness for each of the prime options including both the vehicle and the instrument technologies.

Venus Atmospheric Maneuverable Platform (VAMP)

Kristen Griffin, Daniel Sokol, Dean Dailey, Gregory Lee, Ronald Polidan

Over the past year we have explored a possible new approach to Venus upper atmosphere exploration by applying recent Northrop Grumman (non-NASA) development programs to the challenges associated with Venus upper atmosphere science missions. Our concept is a low ballistic coefficient (we initiated a feasibility study for a semi-buoyant maneuverable vehicle that could operate in the upper atmosphere of Venus. In this presentation we report preliminary results from the ongoing feasibility study and plans for future trade studies, analyses, and prototyping to advance and refine the concept. We will discuss the overall mission architecture and concept of operations from launch through Venus arrival, orbit, entry, and atmospheric science operations. We will present a strawman concept of VAMP, including ballistic coefficient, planform area, percent buoyancy, inflation gas, wing span, vehicle mass, power supply, propulsion, materials considerations, structural elements, subsystems, and packaging. The interaction between the VAMP vehicle and the supporting orbiter will also be discussed. In this context, we will specifically focus upon four key factors impacting the design and performance of VAMP: 1. Feasibility of and options for the deployment of the vehicle in space 2. Entry into the Venus atmosphere, including descent profile, heat rate, total heat load, stagnation temperature, control, and entry into level flight 3. Characteristics of flight operations and performance in the Venus atmosphere: altitude range, latitude and longitude access, day/night performance, aircraft performance (aerodynamics, power required vs. power available, propulsion, speed, percent buoyancy), performance sensitivity to payload weight 4. Science payload accommodation, constraints, and opportunities We will discuss interdependencies of the above factors and the manner in which the VAMP strawman's characteristics affect the CONOPs and the science objectives. We will show how the these factors provide constraints as well as enable opportunities for novel long duration scientific studies of the Venus upper atmosphere that support VEXAG goals 2 and 3. We will also discuss how the VAMP platform itself can facilitate some of these science measurements.

Aerodynamic Characteristics of a Venus Lander as Obtained from Static and Dynamic Wind Tunnel Tests

Anita Sengupta, Juan R. Cruz

The Surface and Atmosphere Geochemical Explorer (SAGE) mission proposed placing a lander on the surface of Venus. Because of the high density of Venus' atmosphere (~65 kg/m3 at the surface), a rigid decelerator of modest size was sufficient to obtain the required landing velocity. For the SAGE lander the rigid aerodynamic decelerator took the form of a drag plate. As part of the lander development effort, a set of static and dynamic wind tunnel tests was conducted to select the size and shape of the drag plate. The basic requirements to be met were: 1) sufficient drag area to yield a landing velocity of less than 10 m/s, 2) static stability over a large range of angles of attack with a stable trim point at or near zero degrees total angle of attack, and 3) acceptable dynamic behavior in free flight (i.e., low oscillation angles and rates). The first two requirements were addressed in a static (force and moment) wind tunnel test conducted at the NASA Ames Research Center Fluid Mechanics Laboratory with an 11.58 percent scale model. The dynamic requirement was addressed in a test with a 25-percent Froude-scaled free flight model conducted in the NASA Langley Research Center Vertical Spin Tunnel. Results for three drag plates will be presented in this paper: a small drag plate, a large flat drag plate, and a large drag plate with 45° dihedral. These data show that the large flat drag plate and the large drag plate with 45° dihedral configurations met the descent rate requirement of 10 m/s or less, but the small drag plate configuration did not. The pitching moment coefficient curve for the small drag plate, as obtained from the static wind tunnel test, indicated that this configuration had the desired stable trim point at an approximately zero degree angle of attack. However, the dynamic test indicated that this configuration tended to eventually pitch to a large total angle of attack. This behavior was consistent with the pitching moment curve, which showed a weak restoring moment at large angles of attack. The pitching moment coefficient curve for the large flat drag plate, as obtained from the static wind tunnel test indicated that this configuration had stable trim points at angles of attack of approximately zero and seven degrees. This dual set of stable trim points was due to a hysteresis loop around zero degrees angle of attack. During initial dynamic testing this configuration was mostly stable around zero degrees angle of attack, with oscillations bounded by angles of attack from approximately -5 to 5 degrees. However, occasionally the model would pitch about 7° and start moving rapidly sideways. This behavior was consistent with the hysteresis loop. The pitching moment coefficient curve for the large drag plate with 45° dihedral, as obtained from the static wind tunnel test, indicated that this configuration had a stable trim point at an angle of attack of approximately two degrees. During dynamic testing this configuration was stable around its trim point with oscillations bounded by angles of attack from approximately -10 to 10 degrees. The rate at which the model pitched was greater than that for the large flat drag plate.

Ballistic Entries at Venus

Dinesh Prabhu, Gary Allen, Thomas Spilker, Gelsomina Cappuccio, Helen Hwang, Robert Moses

To further understand the entry technology needs of a New Frontiers and/or Flagship class missions to Venus recommended by the National Research Council's Planetary Science Decadal Survey [1], NASA's In-Space Propulsion Technology (ISPT) program sponsored a rapid study, the results of which form the basis of the proposed paper. The study team analyzed direct ballistic entries of probes to establish performance baselines so that future comparisons could be made with emerging technologies. The proposed paper will present results of an exploration of an entry trajectory space parameterized by entry velocity, ballistic coefficient, and entry flight path angle. The exploration is guided in part by the Pioneer Venus Multiprobe mission experience, and in part by the mission concepts studies [2-4] in support for the Decadal Survey. Atmospheric entries are considered for inertial entry velocities ranging from 10.8 km/s to 11.6 km/s. Ballistic coefficients ranging from 25 kg/m2 to 300 kg/m2 are considered in the development of 3DoF flight trajectories [5] for entry flight path angles between ?8.5°and ?30°. From these trajectories, the entry flight path angle windows are determined through the use of mechanical and thermal performance limits on the thermal protection material and science payload. For mechanical performance, which determines the steepest entry angle, a 200 g limit is placed on the peak deceleration load experienced by the science instruments, and 10 bar is assumed as the pressure limit for entry-grade carbon-phenolic material. For thermal performance, which determines the shallowest entry angle, inflection points in the total heat load distribution are used as cut off criteria. The proposed paper will explore the use of these entry flight path angle windows to specify performance parameters for a new generation of ablative thermal protection materials, with entry-grade carbon phenolic used as a reference. References: 1. National Research Council, Visions and Voyages for Planetary Science in the Decade 2013-2022, National Academies Press, 2012. 2. Baker, C. L., and Karpati, G., "Venus Mobile Explorer," a Mission Concept Study in support of the NRC Planetary Science Decadal Survey, December 2009. 3. Adams, M. L., and Baker, C. L., "Venus Intrepid Tessera Lander," a Mission Concept Study in support of the NRC Planetary Science Decadal Survey, April 2010. 4. Adams, M. L., and Balint, T., "Venus Climate Mission Study," a Mission Concept Study in support of the NRC Planetary Science Decadal Survey, June 2010. 5. Gary A. Allen, Jr., Michael J. Wright, and Peter Gage, "The Trajectory Program (Traj): Reference Manual and User's Guide," NASA/TM-2004-212847, 2005.

Arcjet Testing of Woven Carbon Cloth for Use On Adaptive Deplyoable Entry Placement Technology

James O. Arnold, Benard Laub (Deceased), Y-K Chen, Dinesh Prabhu, Molly Bittner, Ethiraj Venkatapathy

This paper describes arcjet testing and analysis that demonstrates the viability of three dimensional woven carbon cloth for multi-use in the Adaptive Deployable Entry Placement Technology (ADEPT). ADEPT is an umbrella-like entry system that is folded for stowage in the launch vehicle's shroud and deployed in space prior to reaching the atmospheric interface. A key feature of the ADEPT concept is a lower ballistic coefficient for delivery of a given payload than seen with conventional, rigid body entry systems. The benefits that accrue from the lower ballistic coefficient include factor-of-ten reductions of deceleration forces and entry heating. The former enables consideration of new classes of scientific instruments for solar system exploration while the latter enables the design of a more efficient thermal protection system. The carbon cloth now base lined for ADEPT has a multi-use in that it serves as the thermal protection system and as the "skin" that transfers aerodynamic deceleration loads to the umbrella-like substructure. The arcjet testing described in this paper was conducted for some of the higher heating conditions for a future Venus mission using the ADEPT concept [1], thereby showing that the carbon cloth can perform in a relevant entry environment. The testing and analysis reported here played a key role in achieving a major project milestone and was highly significant to the success and possible adoption of ADEPT for future NASA missions. Importantly, this paper also describes a preliminary engineering level code, based on the arcjet data, that can be used to estimate cloth thickness for future ADEPT missions and to predict carbon cloth performance in future arcjet tests. [1] Smith, B., Venkatapathy, E., Wercinski, P., Yount, B., Prabhu, D., Gage, P., Glaze, L., Baker, C. "Venus In-Situ Explorer Mission Design using a Mechanically Deployed Aerodynamic Decelerator", Big Sky IEEE Aerospace Conference, Big Sky Montana, March 2-9, 2013.

Session 7A: Cross Cutting Technologies I

EUROPEAN EARTH RE-ENTRY CAPSULES FOR SAMPLE RETURN MISSIONS: DESIGN AND TECHNOLOGIES

Denis Rebuffat, David AGNOLON, Heiko RITTER, Luca FERRACINA

The ESA's Cosmic Vision M3 (Medium class #3) Programme features several candidate missions from which one will be selected for a launch in the early 2020's. It includes a sample return mission from a Near Earth Asteroid (NEA) called Marco Polo R. Moreover, ESA's Mars Robotic Exploration Preparation (MREP) Programme is preparing candidate missions to Mars to be launched after the 2016 and 2018 ExoMars missions. MREP includes a Phobos Sample Return mission, called Phootprint, while the Mars Sample Return mission is considered as the long term objective of the programme. This paper briefly describes the Earth Re-entry Capsules (ERC) that will bring back the collected samples for these three missions, and the associated ongoing technology. The Phootprint and Marco Polo R missions currently foresee a similar ERC design, as the amount of sample (around 100g), the re-entry conditions, and the science requirements are comparable. The capsule will weight around 30 Kg and will perform a high speed re-entry at about 11-12 Km/s. A high performance Thermal Protection System (TPS) is accommodated that can sustain heat fluxes up to 15 MW/m2. The capsule performs a fully passive entry and descent, then performs a hard landing without parachute to minimise the design complexity and demonstrate the technology for a future Mars Sample Return capsule. This requires the use of a crushable material in order to limit the g-loads on the sample container. Despite similar re-entry conditions, the Mars Sample Return mission, on the other hand, features a larger ERC of more than 100 Kg, due to the size of the Biocontainer that contains around 500g of Martian samples. One important specificity of this mission are the Planetary Protection requirements that are very stringent, since it must be demonstrated that the Earth biosphere will not be contaminated with uncontained and unsterilized Martian material. This explains the need for a large Biocontainer with more difficult g-loads requirement, driving the ERC size, but also has several implications linked to the ERC reliability. Related areas are the resilience of the TPS to micrometeorids impacts, the choice of materials in particular for the TPS, the development and qualification approach, the ERC release and recovery operations, the prevention to use a parachute for reliability reasons. These three missions have been or are being studied for ESA, and numerous associated technology developments are on-going. This paper will provide an overview of the system and technological activities results that are related to the ERC. In particular the following European developments will be addressed: • Light density TPS material development and testing • High density & highly reliable TPS material testing • Crushable material development and testing • Ablators catalytic properties • Shock tubes and high enthalpy wind tunnel test facilities • ERC aerodynamic characterization by tests from supersonic to subsonic regimes • Specific impacts on the MSR ERC design of the Planetary Protection requirements

Technology Development for High-Speed Reentry Small Capsules ~ Hayabusa-II SRC and more ~

Tetsuya Yamada

The present paper overviews Japanese technology development for high-speed reentry small capsules: Hayabusa-2, iBall and future high-speed sample return missions. The objectives of the Hayabusa-2 mission is to explore the C-type asteroid 1999JU3 and to take back to the earth the asteroid sample which is considered to contain plenty of organic substances. Hayabysa-2 spacecraft is planned to be launched in 2014 and return in 2020. Hayabusa-2 sample return capsule (SRC) is designed to deliver the asteroid sample to the earth passing through the excessively severe aerodynamic heating environment at entry velocity of 12 km/s. As is known, Hayabusa-1 SRC successively returned to the earth in 2010, and post flight analyses were carried out for the sake of the Hayabusa-2 and future reentry missions [1]. The present paper describes the design review and verification tests of the Hayabusa-2 in addition to the results of the post flight analysis. Hayabusa-2 SRC enters earth atmosphere and deploys the parachute at a given altitude after passing through the aerodynamic heating corridor. In the Hayabusa-2 mission, SRC is required to have much more system reliability within almost same weight resource as of the Hayabusa-1 SRC. During over 10 years since the development of the Hayabusa-1 SRC, some devises in the onboard electronics have become out of manufacture line, the onboard electronics and the system of the Hayabusa-2 SRC need to be designed to accomplish all the functions described above. In order to increase the reliability, several verification tests have been carried out; Separation mechanism from the mothership has been improved to diminish separation errors and the verification tests are carried out. Heatshield design was reconfirmed based on the post flight analysis of the Hayabusa-1 SRC and results of the heating tests of the heatshield material manufactured according to the prescribed process. As for the descending system, the functional safety of the parachute is confirmed taking account of the material long-term decay in the space environment during the orbital flight. Reentry environment measurement module (REMM) is mounted on SRC in order to measure the heatshield temperature, acceleration and attitude motion of the SRC during the reentry. The objectives of i-Ball (IHI Aerospace and JAXA) is to demonstrate small capsule technologies and rapid recovery together with the reentry flight measurements during its reentry; i-Ball-1 is a small capsule with total weight of 17 kg and 40 cm in diameter, which entered the earth atmosphere and recovered on the seas, after being released from the HTV-3 during its destructive entry phase in July 2012. It has complete sphere shape with slightly-shifted center of gravity for aerodynamic stability. Because it is difficult to control attitude condition at the (destructive) release from the HTV-3, the sphere shape and its aerodynamic stability worked well. Regardless of its small size, the low altitude GPS is mounted on i-Ball for the quick recovery, which demonstrated its functional High-speed reentry missions are now under investigation in ISAS/JAXA. Several sample return missions which require over 9 km/s V-inf (infinite asymptotic approach velocity), where the entry velocity goes beyond 14 km/s with over . Sample return from extinct core of comets belongs to such kinds of high-speed/high-energy mission, in which

mission the weight resources allocated to the sample return capsule is severely limited. One of the supporting technologies of the missions is certainly light weight heatshield technology, efficient slow-descending system and recovery (localization) system. JAXA has been developing low-density (density is controllable within 0.3 to 0.4 g/cc) heatshield material (ablator) with 3-dimension fiber structure. The present paper also development status the heatshield material for future high-speed reentry missions.

Re-entry Platform for studying radiation

Gilles Bailet, Thierry Magin, Jean Muylaert, Christophe Laux

The current experimental data obtained in-flight by means of optical emission spectrometry are scarce. Only two program exist: the Flight Investigation of the Reentry Environment I&II (1964/1965) conducted under hyperbolic reentry conditions at very high speeds (v?11.5 km/s) and the Bow Shock Ultra Violet I&II (1990/1991) for suborbital flight regimes (v?3.5 km/s and 5.1 km/s). These two programs have permitted to cover the extremes of the reentry missions of interest at their time (lunar return and suborbital flight) and to provide time-resolved emission spectra along the reentry trajectory. They have shown and corrected major unknowns in the physico-chemical models, but the limited set of data leaves important gaps and partially validated models for the range of conditions needed for nowadays expectations on Low Earth Orbit (LEO, v?7.5 km/s) return. The von Karman Institute for Fluid Dynamics (VKI) is developing a reentry vehice named QARMAN (QubeSat for Aerothermodynamic Research and Measurement on AblatioN) focusing on reentry phenomena such as radiation withing LEO return trajectory conditions that will permit to respnd to the interest of the scientific community. Investigations are needed to select and calibrate the emission spectrometer as long as the specific optical path development onboard QARMAN in its constraining form and mass factor (340x100x100 mm3 for 3kg). The development methodology has to actualize the numeric tools with up-to-date chemical and radiation models and the plasma torch test procedure has to be adapted to compare spectra collected from the instrumented sample and simulation results. A deeper understanding of the radiations occurring during the reentry of the vehicle will permit to acquire flight data for numerical code and ground test methodologies validation. The ground to flight extrapolation models will also highly benefits from these data as the full duplication of flight condition is not possible on ground. The contribution of radiation on the phenomena of reentry is one of the most challenging and important phenomena as it has a direct effect on the thermal protection system of the vehicle by increasing the total heat flux with the radiative component reaching a maximum of twenty percent for a LEO reentry. In conclusion, the low-cost QARMAN vehicle will permit to conduct breakthrough advances for a better understanding of the radiative process during reentry.

Mission, GNC and System Overview of the Intermediate eXperimental Vehicle (IXV)

Rodrigo Haya Ramos, Luis F. Peñín, Victor Marco, Roberto Angelini, Davide Bonetti, Jordi Freixa, Jorge Serna, Murray Kerr, Gabriele de Zaiacomo

The Intermediate eXperimental Vehicle (IXV) is a re-entry demonstrator whose objective is to tackle the basic European needs for re-entry from Low Earth Orbit (LEO), consolidating the knowledge and expertise necessary for the development of future European re-entry systems. It is actually under Phase D after a successful CDR and launch is foreseen for summer 2014. The IXV is a technology platform that represents a step forward with respect to the Atmospheric Re-entry Demonstrator (ARD), flown in October 1998, with an increase in-flight manoeuvrability permitting the verification of technologies over a wider re-entry corridor and with aerodynamic surfaces employed actively to trim and manoeuvre the vehicle. The IXV mission and system objectives are the design, development, manufacturing, assembling and onground to in-flight verification of an autonomous European lifting and aerodynamically controlled reentry system, integrating critical re-entry technologies at system level. Among such critical technologies of interest, special attention is paid to aerodynamic and aerothermodynamics experimentation including advanced instrumentation for aerothermodynamics phenomena investigations, thermal protections and hot-structures, guidance, navigation and flight control through combined jets and aerodynamic surfaces (i.e. flaps), in particular focusing on the technologies integration at system level for flight. The IXV vehicle will be injected by the Vega launcher in an equatorial suborbital trajectory to reach an entry velocity at the Entry Interface Point (EIP, 120 km altitude) above 7.4 km/s. After injection, the IXV flight will be comprised of three main phases; orbital, re-entry and descent. The re-entry phase for the IXV vehicle corresponds to a guided lifting entry during 8000 km ground track up to the deployment of a 3 stages parachute system, with the IXV flight terminating with splashdown on the Pacific Ocean. The suborbital nature of the mission strongly couples all of the phases and hence phase by phase analyses are not applicable. To cope with this challenge, an end to end approach has been implemented in Mission Analysis and flight mechanics in order to properly identify the design values applicable to all phases and to consider the coupling effects. A strong interaction with the System, AEDB/ATD and GNC teams is also a key element for the maturation of the mission scenario. This end to end approach has been used both for design and for performance assessment. This paper describes the IXV mission design and performances in terms of trajectory, Flying Qualities, visibility and safety aspects. The IXV is a 5 m long lifting body with a lift-to-drag ratio of 0.7 in the hypersonic regime and, distinct from other re-entry vehicles such as ARD and Orion, is actuated through the combination of two body flaps mounted at the aft windward side of the vehicle and RCS thrusters. The challenges for the IXV re-entry GNC system design are common to those for re-entry vehicles performing a first flight, with the re-entry GNC required to be robust to high levels of uncertainty in the vehicle dynamics. This paper describes the IXV re-entry G&C and DRS triggering, including the assessment of the re-entry GNC performance using the IXV functional engineering simulation developed for the IXV programme. In order to be able to execute

this type of mission and to achieve the mission objectives the overall Mission, System and Vehicle architecture has been developed to achieve the Critical Design Review level of maturity and moving from an engineering design phase to the early development status. The System activities performed the global coordination of the project through the management of the program requirement flowing from the ESA M&SRD and declined through the different levels of specifications, defined the overall margin policy to be adopted at system and subsystem level in order to achieve the optimum balancing between design robustness and programmatic and technical requirements. This paper shows an overview of the main subsystems, covering from aerodynamics/aerothermodynamics to avionics, flap control system, RCS, cold structure, mechanisms, thermal control, descent and recovery system, TPS, in flight experiment, operations and Ground segment as well as the test campaigns. The authors wish to thank ESA IXV System team, led by Salvatore Mancuso, for their valuable comments on the Mission and System design and performance. It is also acknowledged the contribution from Telespazio as subcontractor for the link budget aspects.

SHEFEX-II Flight Instrumentation and Discussion on Selected Flight Data

Ali Guelhan, Frank Siebe, Thomas Thiele, Dominik Neeb, John Turner

In the frame of the SHEFEX program DLR launched the SHEFEX-II hypersonic flight experiment from Andoya rocket range in Norway on 22.06.2012 consisting of an extensively instrumented scientific payload on top of a two-stage rocket configuration. In total 140 sensors for the measurement of pressure, temperature and heat flux were integrated into the scientific payload. With an apogee of about 177 km, the vehicle achieved flight velocities up to 2790 m/s corresponding to Mach numbers up to apprx. 10. Almost all heat flux, temperature and pressure sensors provided very clear data during ascent and descent phases. Measured pressure data from sensors at different locations of the scientific payload show consistent results concerning the aerodynamic behavior on the vehicle along the complete trajectory. Pressure fluctuations measured during flight, show a good correlation to angle of attack variations. The comparison between the measured pressure data and calculated pressure coefficients from a CFD analysis at selected trajectory points show good agreement and demonstrate that the results depend also on the atmospheric model used for the calculations. The agreement between measured and computed heat flux data show still some discrepancies. A parametric study has been carried out to extract the main source of uncertainties in the measured flight data and calculations.

Strategies for Mars Network Science Missions via Innovative Aerocapture and EDL Architectures

Sarag J. Saikia, Blake A. Rogers, James M. Longuski

We propose flexible strategies for Mars network science missions via innovative Aerocapture and EDL (AEDL) architectures with a variable number of network stations with a single launch from Earth. The architecture can deliver four or more landers with a minimum separation of 3,000 km. The major instruments onboard each lander could include seismometer, precision tracking instruments; and secondary instruments to measure heat flow, electromagnetic sounding, and boundary layer meteorology. The concept relies on the proven technologies from Pathfinder, MERs, and Phoenix missions and the new AEDL architectures to minimize risk and cost. Each flight system consists of a cruise stage and an entry system for each landers. Like the JPL's Mars Geophysical Network concept study, this concept will help carry two such identical flight systems (leading and trailing) that contain two landers each. This architecture's main emphasis is to land four (or more) landers with a simple cruise stage and using new AEDL architectures. A Centaur upper stage propels the stacked flight systems into a trajectory towards Mars. Each flight system consists of two landers (primary and secondary spacecraft) which are stacked on top of each other. The same Centaur stage (or an ion propulsion system) can provide the necessary to achieve the required 7 days separation between Mars entry of the two flight systems. Since the science requirements do not necessitate a precise landing, MER- and Phoenix-type unguided entry is used. If precise landing is required, guided entry can be employed. On arrival at Mars by the leading flight system, the secondary spacecraft will be released to land downrange from the primary spacecraft. The primary space with a low ballistic coefficient continues to descend. The same AEDL sequence is then replicated by the trailing flight system in 7 days at a geographically distant location predetermined by the scientific community. This concept has potential applications in the exploration of other atmospheric bearing bodies of the solar system.

The Multi-Tier Mission Architecture and a Different Approach to Entry, Descent and Landing

Jeremy Straub

Planetary missions are generally very well planned out. Where the spacecraft will be deployed, what it will do there and in what order are generally determined before launch. While some allowance is made for greater depth exploration of scientifically interesting items identified during the investigation, a successful mission is (generally) one that doesn't deviate significantly from its planning. When sending an initial mission to an unsurveyed planet or moon, however, this approach is not suitable. Current space technology provides the capability to send a combined survey and lander mission (instead of conducting an initial survey mission and following it up with a separate lander-based mission). Removing the ability for conducting pre-planning (as the exact mission approach will remain unknown until the survey is complete), however, increases mission risk – and this risk cannot be effectively mitigated. The spacecraft and landing system have only the capabilities they were dispatched with, which may be non-optimal given the now-known nature of the body. Equipment capabilities may also not be ideally matched with surface conditions and desired experimentation (though equipment capabilities will be considered when making this selection). The Multi-Tier Mission Architecture is an approach to conducting this type of a mission. Instead of a single large interplanetary craft and a single large lander, a collection of multiple heterogeneous small craft is proposed. These small craft are deployed from the interplanetary craft to locations of scientific interest, based on the results of the initial orbital survey. Craft can be deployed to multiple regions of the body to collect concurrent investigations. Also, the use of multiple craft disperses mission risk (as mission success parameters should be achievable without every craft being successful). This paper considers the types of craft that can be deployed as part of a multi-tier mission on each planet in the solar system and the impact of these limitations on the multi-tier architecture's capabilities. Short mission scenarios are presented for each planet and several moons. The benefits of the multi-tier architecture in each scenario (versus a conventional projected or historical mission) are evaluated. The paper concludes with a discussion of what technological and other advancements are required to facilitate the use of the multi-tier architecture for planetary missions.

IRENE Preliminary Study– (Italian Re-Entry NacellE Preliminary Study)

Raffaele Savino, Renato Aurigemma, Pasquale Dell'Aversana, Luciano Gramiccia, Jose Longo, Lionell Marraffa, Francesco Punzo

International Planetary Probe Workshop 10 17-21 June 2013 – San Jose State University, California, USA IRENE Preliminary Study– (Italian Re-Entry NacellE Preliminary Study) R. Aurigemma (ALI Scarl), P. Dell'Aversana (ALI Scarl), L. Gramiccia (ALI Scarl), J. Longo (ESA), L. Marraffa (ESA), F. Punzo (ALI Scarl), R. Savino (DII). ABSTRACT The Italian and European Space Agencies (ASI and ESA) are supporting a research programme, developed in Campania region (South of Italy) by a cluster of industries, research organizations and universities, on a low-cost re-entry capsule, able to return payloads from the ISS to Earth and/or to perform short-duration scientific missions in Low Earth Orbit (LEO). The ballistic capsule is characterized by a deployable "umbrella-like" heat shield, that allows relatively small dimensions at launch and a sufficient exposed surface area in re-entry conditions, reducing the ballistic coefficient and leading to acceptable heat fluxes, mechanical loads and final descent velocity. The TPS materials have already been tested in the SPES hypersonic wind tunnel at DIAS (Aerospace Engineering Department) of the University of Naples, and in the SCIROCCO PWT (Plasma Wind Tunnel) at CIRA (Centro Italiano Ricerche Aerospaziali), of Capua, Italy. ESA is supporting a preliminary study to develop a small flight demonstrator of the capsule to be embarked as a secondary payload in the inter-stage adapter of a sub-orbital Maxus sounding rocket. This capsule will be ejected, during the ascent phase of the payload, after its separation from the booster, to demonstrate deployment, re-entry and recovery capabilities. The deployable thermal protection system concept may be applied to any ESA mission in the framework of the Science and Robotic Exploration Program, requiring planetary entry and, possibly also to ESA missions in the framework of Human Spaceflight, requiring planetary entry or re-entry. The technology offers also an interesting potential for aerobraking/aerocapture and for de-orbiting. This paper summarizes the results of these activities, including the preliminary definition and analysis of the mission scenario, the identification of the mission and system requirements and of different possible solutions. The results include aerodynamic, aerothermodynamic, mechanical and structural analyses and preliminary technical definition of avionics, instrumentation and main subsystems. The system is based on low-cost components and commercial hardware and its payload includes sensors and onboard memories to measure and store relevant properties during the flight, in particular temperature, pressure, accelerations, angular rates, strains, etc.

Ground Testing of In-Flight Re-Entry Experiments of Re-Entry CubeSat QARMAN

Isil Sakraker, Roger Torras Nadal, Roger Torras Nadal, Olivier Chazot

The extreme aerodynamic heating and the exothermic chemical reactions due to the gas-surface interaction at hypersonic free stream velocities make the thermal protection system design and testing a challenging process. The chemically reacting hypersonic flight regime cannot be fully duplicated in ground facilities and the design methodologies, experimental and numerical, have to be validated by real flight data which come with high costs. It is the objective of QARMAN CubeSat, QubeSat for Aerothermodynamic Research and Measurements on AblatioN, to perform an atmospheric entry experiment to Earth and collect real flight data for highly affordable costs. QARMAN is a triple unit CubeSat respecting the CubeSat standards; a rectangular prism with dimensions 30x10x10 cm. It has a low density/high porosity ablator of carbon fiber preform impregnated with phenolic resin in the front unit. The mission is representative of real atmospheric entry missions with 7.7 km/s velocity at 120 km of altitude and a stagnation heatflux between 2 - 2.5 MW/m2. It will be launched together with the other QB50 satellites at 350 km in mid-2015. During the atmospheric entry, starting at the rarefied non-continuum altitudes, there are many interesting complex physical phenomena that deserve attention. The standard CubeSat platform, besides many advantages on the available components and flexibility of the launch opportunities, does not allow testing all the interesting events at once due to the limitation of mass, volume, power and data size. The aerothermodynamical in-flight experiment subjects are determined as; TPS ablation, recession and efficiency, aerothermal environment and species radiation, stability and trajectory rebuilding, shear force and side panel heating and insulation efficiency. It is important to state that all the chosen phenomena can be extracted or rebuilt by simple but meticulous temperature and pressure measurements that fit in QARMAN vehicle in terms of mass, volume, power and datalink. On top of these, a miniature spectrometer will also be onboard for radiation measurements. The measurement chains and concept of operations need to be carefully designed since the re-entry phase of the mission will start at 120 km; it will last 10 minutes and at 50 km of altitude the vehicle will be disintegrated. During the re-entry phase the batteris will not be recharged and since the vehicle will not be recovered, the telecommunications blackout will be prevented by data linking backwards to the IRIDIUM constellation. The CubeSat standards allow having components off the shelf for most of the subsystems however a careful design is currently being done for the data acquisition system on board which is unique for QARMAN concept. The entire payload and acquisition system will be tested at VKI Plasmatron with the aim of autonomous measurements. VKI Plasmatron is an inductively coupled plasma generator where the aerothermochemical environment in front of the spacecraft during the re-entry can be duplicated.

Acceptance of Autonomy in the Planetary Science Community: Current Attitudes and What Can Be Done to Improve It

Jeremy Straub

As a spacecraft moves progressively further from the Earth, round-trip communications take longer, available bandwidth and (presuming the mission is towards the outer planets) power generation capabilities decline. In short, the mission can either accept these limitations (resulting in comparatively less science data as opposed to a mission to a closer planet, such as Mars), increase the size of the spacecraft to support higher gain antennas and increased surfaces for solar panel mounting or implement an alternate mission model. One such model is to allow the spacecraft to act more autonomously in the control and decision making of data collection and to make decisions (based on mission objectives) related to what data to transmit back to Earth (and at what level of data product: raw, summary, conclusion and defense, etc.). A survey was performed to gauge the acceptance of autonomous data gathering and analysis. For this purpose, 230 planetary scientists were surveyed. This paper presents an overview of the results of this data collection and analysis process and its implications for interplanetary space missions. Specific focus is given to the areas where the respondents indicated that autonomy would or would not be acceptable for use.

Session 7B: Future Mars II

ExoMars 2016 Entry Descent and Landing Demonstrator Module EDL overview

Leila V. Lorenzoni, Thierry Blancquaert, Olivier Bayle, Thoams Walloschek, Stephane Langlois, Stefano Portigliotti, Maurizio Capuano

ExoMars 2016 Entry Descent and Landing Demonstrator Module: EDL overview Leila V. Lorenzoni*, Thierry Blancquaert, Olivier Bayle, Thomas Wallosheck, Stephane Langloise ESTEC *leila.lorenzoni@esa.int Stefano Portigliotti, Maurizio Capuano Thales Alenia Space The ExoMars 2016 mission will be launched in January 2016 by a Roscosmos provided rocket. The ExoMars 2016 spacecraft is defined as a Spacecraft Composite composed by a Mars Orbiter and a 600 kg Entry Descent and Landing Demonstrator Modue (EDM). The launch will take place from Baikonur and, after a direct injection and a T2 interplanetary transfer, the spacecraft composite will approach Mars around mid-October 2016. Prior to arrival to Mars the EDM will be released from the orbiter and it will travel three days prior to reach a specific point in the Mars atmosphere targeted to land in the predefined 100kmx15km landing ellipse located in the Meridiani planum. During entry, the vehicle will lose most of its kinetic energy and, when the proper velocity conditions are reached, the parachute opening will be triggered. The supersonic pachute will decelerate the vehicle down to subsonic speed. The opening of the parachute initiates the Descent phase and after the EDM flight condition is stabilized, the front-shield will be jettisoned allowing the Radar Doppler altimeter to measure altitude and velocity with respect to the Martian terrain. Once the vehicle reaches the terminal velocity of approximately 90 m/s, the back-cover separation will be triggered and the EDM Surface Platform (ESP) will initiate the descent phase. This final phase will be controlled by using the data provided by the IMU and by the Radar Doppler Altimeter and by modulating the pulses 9 400N hydrazine engines housed in 3 clusters mounted every 120 degree along the external edge of the ESP. In proximity of the surface, nominally at 1.5 m of altitude, the thrusters will be switched off and the ESP will drop on the surface subjected only to Mars gravity and the impact loads will be absorbed by crushable structures. During EDL, essential telemetry will be sent to the Relay Orbiter and during surface operations the EDL collected telemetry together with the EDM payload data will be transmitted to the Relay Orbiter within the first 4 sols after landing. This paper intends to describe the EXM approach in the definition of the EDM delivery at Entry point, the EDL sequence and the preliminary results in terms of performance of key parameters.

Overview of the InSight Entry, Descent, and Landing System

Christine Szalai, Myron R. Grover, David Skulsky

The Interior Exploration using Seismic Investigations, Geodesy and Heat Transport (InSight) is the 12th selection in NASA's series of Discovery Class missions, and will land on Mars in 2016. This mission builds on spacecraft technology used in the highly successful Phoenix lander mission, which landed in the northern polar region of Mars on May 25, 2008. The entry, descent, and landing (EDL) phase of the mission is made up of the hypersonic, parachute, and powered descent sub-phases. The basic EDL architecture remains unchanged from the Phoenix design, and includes an entry vehicle that protects the payload during the extreme heating conditions during the hypersonic phase and a Viking heritage disk-gap-band (DGB) supersonic parachute. Once the heathshield is separated from the entry vehicle, the lander legs are deployed and the radar is activated to determine the correct velocity and altitude state in which to separate the lander from the backshell. A system of 12 pulsed thrusters on the lander are used to perform a gravity turn, and engine cut-off occurs when one of the three landing legs contacts the Martian surface. An overview of the InSight EDL system architecture, comparisons to Phoenix, and unique challenges associated with the InSight mission design will be discussed.

ENTRY, DESCENT AND LANDING TECHNOLOGY DEVELOPMENTS FOR ESA'S FUTURE MARS MISSIONS

Kelly Geelen, Jonan Larranaga, Sanjay Vijendran

ESA's Mars Robotic Exploration Preparation (MREP) Programme, is preparing candidate missions to Mars to be launched after the 2016 and 2018 ExoMars missions. MREP established the Exploration Technology Programme for reinforcing Europe's position in Mars robotic exploration. The main candidate missions to Mars are the Phobos Sample Return mission, called PHOOTPRINT, and the Mars Network Science mission, also called INSPIRE. Later mission concepts include also Mars precision landers and rovers. This paper briefly describes the Mars landing missions being considered and the ongoing technology developments for the Entry, Descent and Landing system. The Mars Network Science mission, INSPIRE, addresses key science goals on Mars that can only be achieved by simultaneous measurements from a number of landers, which are spaced across the planet. The primary objective concerns the planet's internal geophysics and meteorology. The 2016 NASA Insight mission will implement a comparable payload and would be a valuable pre-cursor mission for this. The ESA architecture currently includes a carrier spacecraft with 3 landers, performing ballistic entries at Mars and airbag landings. A Mars Precision Lander mission has been studied by ESA as part of a Mars Sample Return (MSR) campaign. The mission has been designed to deliver a payload of ~100kg onto the surface of Mars with a landing precision of 10 km (3?), and a goal of 7.5km (3?). This landing precision, which is an order of magnitude better than current ESA Mars missions is necessary to deliver a Sample Fetch Rover in close proximity to other elements of a potential MSR mission. A precise and safe landing is non-trivial, and requires precise navigation and targeting during the Mars approach phase, a highly accurate guided entry and a powered descent phase capable of surface hazard avoidance. Technology developments are ongoing for both mission concepts within the MREP technology programme. This paper will describe the key results of the activities related to the entry, descent and landing part of these missions: • Unvented airbag design, breadboarding and drop testing • Development of a miniaturized radar and laser altimeter • Subsonic parachute development and balloon and wind tunnel testing together with new parachute testing methods such as the use of pressure sensitive paint • Entry decent and landing system and GNC simulation campaigns, including testing of various guided entry algorithms • Miniaturized accelerometer and Inertial Measurement Unit (IMU) development • Throttleable engine development for Mars landings • Vision and lidar based hazard avoidance technologies and data fusion

Feasibility of a Dragon-derived Mars lander for scientific and human-precursor missions

John S. Karcz, S. M. Davis, G. A. Allen, B. J. Glass, A. A. Gonzales, J. L. Heldmann, L. G. Lemke, C. P. McKay, C. R. Stoker

A minimally-modified SpaceX Dragon capsule launched on a Falcon Heavy rocket presents the possibility of a new low-cost, high-capacity Mars lander for robotic missions. We have been evaluating such a "Red Dragon" platform as an option for the Icebreaker Discovery Program mission concept. Dragon is currently in service ferrying cargo to and from the International Space Station, and a crew transport version is in development. The upcoming version, unlike other Earth-return vehicles, exhibits most of the capabilities necessary to land on Mars. In particular, it has a set of high-thrust, throttleable, storable bi-propellant “SuperDraco” engines integrated directly into the capsule that are intended for launch abort and powered landings on Earth. These thrusters provide the possibility of a parachute-free, fully-propulsive deceleration at Mars from supersonic speeds to the surface, a descent approach which would also scale well to larger future human landers. We will discuss the motivations for exploring a Red Dragon lander, the current results of our analysis of its feasibility and capabilities, and the implications of the platform for the Icebreaker mission concept. In particular, we will examine entry, descent, and landing (EDL) in detail. We will also describe the modifications to Dragon necessary for interplanetary cruise, EDL, and operations on the Martian surface. Our analysis to date indicates that a Red Dragon lander is feasible and that it would be capable of delivering more than 1000 kg of payload to sites at elevations three kilometers below the Mars Orbiter Laser Altimeter (MOLA) reference, which includes sites throughout most of the northern plains and Hellas.

MECHANICALLY-DEPLOYED HYPERSONIC DECELERATOR TECHNOLOGIES FOR MARS MISSIONS

Paul Wercinski

Summary: The concept of a mechanically deployable hypersonic decelerator, developed initially for high mass (~40 MT) human Mars missions, is currently funded by OCT for technology maturation. The ADEPT (Adaptive, Deployable Entry and Placement Technology) project has broad, game-changing applicability to in situ science missions to Venus, Mars, and the Outer Planets. Combined with maturation of conformal ablator technology (another current OCT investment), the two technologies provide unique low-mass mission enabling capabilities otherwise not achievable by current rigid aeroshell or by inflatables. We will present results that illustrate the mission enabling capabilities of the mechanically deployable architecture for: (1) robotic Mars (Discovery or New Frontiers class) in the near term; (2) alternate ap-proaches to landing MSL-class payloads, without the need for supersonic parachute or lifting entry, in the mid-term; and (3) Heavy mass and human missions to Mars in the long term.

Mars Sample Return Using Commercial Capabilities: ERV Trajectory and Capture Requirements

Nicolas Faber, Cyrus Foster, David Willson, Andrew Gonzales, Carol Stoker

Mars Sample Return was presented as the highest priority planetary science mission of the next decade [1]. Lemke et al. [2] present a Mars Sample Return mission concept in which the sample is returned directly from the surface of Mars to an Earth orbit. The sample is recovered in Earth Orbit instead of being transferred between spacecraft in Mars Orbit. This paper provides the details of this sample recovery in Earth orbit and presents as such a sub-element of the overall Mars sample return concept given in [2]. We start from the assumption that a Mars Ascent Vehicle (MAV), initially landed on Mars using a modified SpaceX Dragon capsule, has successfully delivered the sample, already contained within an Earth Return Vehicle (ERV), to a parking orbit around Mars. From the parking orbit, the ERV imparts sufficient Delta-V to inject itself into an earthbound trajectory and to be captured into an Earth orbit eventually. We take into account launch window and Delta-V considerations as well as the additional constraint of increased safety margins imposed by planetary protection regulations. We focus on how to overcome two distinct challenges of the sample return that are driven by the issues of planetary protection: (1) the design of an ERV trajectory meeting all the requirements including the need to avoid contamination of Earth's atmosphere; (2) the concept of operations for retrieving the Martian samples in Earth orbit in a safe way. We present an approach to retrieve the samples through a rendezvous between the ERV and a second SpaceX Dragon capsule. The ERV executes a trajectory that brings it from low Mars orbit (LMO) to a Moon-trailing Earth orbit at high inclination with respect to the Earth-Moon plane. After a first burn at Trans-Earth Injection (TEI), the trajectory uses a second burn at perigee during an Earth flyby maneuver to capture the ERV in Earth orbit. The ERV then uses a non-propulsive Moon flyby to come to a near-circular Moon-trailing orbit. To perform the Earth Orbit Rendezvous (EOR), a second Dragon capsule is then launched from Earth and a similar lunar flyby is performed to rendezvous with the ERV. The requirements for rendezvous, close proximity operations and capture of the sample canister are described. A concept of operations for sample retrieval is presented along with design specifications of the ERV, the required modifications to the Dragon capsule, as well as the hardware, software, sensors, actuators, and capture mechanisms used. In our concept, a container is mounted to the front hatch of Dragon, capable of accommodating the sample canister and sealing it from the rest of the capsule. The sample canister is captured using a robotic arm with a magnetic grappling mechanism. Dragon then performs a propulsive maneuver to return to Earth for a controlled re-entry while the ERV (sans sample container) is left in the Moon trailing orbit. Contingency cases and related mitigation strategies are also discussed, including the advantages and disadvantages of performing the ERV rendezvous with a crew. [1] Visions and Voyages for planetary Science in the decade 2013-2022, Nat'l Academy Press, 2010. [2] Mars Sample Return Using Commercial Capabilities, L.G. Lemke, J. Bowles, A. Gonzales, L. Huynh, and C. Stoker, to International Planetary Probes Workshop, June 17-21, San Jose, CA 2013

A concept for the autonomous landing shock attenuation, self-leveling and lowering of a robotic landing platform

Robert Buchwald, Erich Werth, Anton Böljes, Arne Schramm

One of the most challenging tasks of a successful robotic sample return mission to Mars is the safe and precise landing of a rover or a re-ascent stage on the surface. Besides landing systems with landing legs, different types of airbag systems or the sky crane concept have successfully been used in the past. Airbags can properly attenuate the landing shock but cannot avoid uncontrolled roll over during the touch down phase and do not guarantee a sufficient precision. The sky crane allows the precise delivery of a rover in challenging terrain but requires a highly complex system design. Landing systems with landing legs have a much simpler design with a great heritage and are able to land with the same precision but the distance to the ground after landing imposes challenges for the rover egress. The paper will present an alternative concept based on a landing system with landing legs. During Mars atmospheric entry, the legs are folded at the side of the platform in order to fit in the capsule. During the parachute phase and after heat shield separation, the legs are deployed and latched in their landing position by a robotic deployment arm. After touchdown, this arm allows to lower the platform to the ground for rover egress or to level the platform in the gravitational field in order to provide an adequate launch base for an ascent stage. The paper will give a look inside the achieved results with special focus on the relevant breadboarding activities.

MARS HOPPER: SCIENCE CASE AND ENABLING TECHNOLOGIES

Michael Williams, Katherine Bennell, Marie-Claire Perkinson, Richard Ambrosi, Nigel Bannister, Hugo Williams, Jon Sykes, Hilde Schroeven-Deceuninck

A Mars Hopper vehicle and enabling technologies are currently being studied by Astrium Ltd and the University of Leicester, under ESA contract within the General Studies Programme. The “Proof of Mars Hopper Concept Technologies” study will allow the key disruptive technologies to be further investigated and matured, and the baseline vehicle design to be updated and refined. The current study builds on previous studies undertaken by Astrium and the University of Leicester in recent years. Mars has a vast land area of a similar size to Earth, yet very little of it has been explored until now. Rover vehicles are our primary method of in-situ Mars exploration, but due to the constraints of the descent system and wheeled vehicles, they are limited in their traverse capabilities to relatively short distances over flat plains with few rocks. Mars contains immense canyons, mountains and craters that are of very high scientific interest – yet cannot be visited by our currently available vehicles. A Mars Hopper would open up the most scientifically interesting areas and is the logical next step in exploration of the Martian surface. The Mars Hopper is a legged vehicle that uses an in-situ resource – carbon dioxide extracted from the atmosphere – to fuel a radioisotope-based thruster that allows the vehicle to perform ballistic ‘hops’ of around 1 km range, landing safely each time on a robust landing system. This hopping capability allows scientific instruments to characterise potentially hundreds of sites, limited only by the lifetime of the subsystems. The vehicle could also hop up and down slopes to visit mountains and craters, and could even hop into the canyons that are expected to contain the most ground-breaking science. Three key subsystems have been identified as vital to both the Mars Hopper and future exploration missions. 1. ISRU Gas Compression and Storage system. Repeatedly extracting and isolating the selected propellant from the Martian atmosphere and compression and storage present unique challenges in terms of cryogenics, thermal, propulsion, chemical processes and as well as in technology areas. 2. Thermal Management, Power and Propulsion System. The application of a thermal energy source to provide both platform power and propulsion provides a multidisciplinary challenge, where thermal design, power and propulsion system performance are integrated and linked in a unique fashion. 3. Landing System. Landing safely and precisely and then taking off again is the single point failure feature of exploration and sample return missions and the design of a lightweight and reliable system allowing access to a variety of terrain is a key system driver. These technologies matured as part of the Mars Hopper study will be applicable not just in the framework of an expanded space exploration programme to difficult planetary bodies such as Titan, but also to many alternative terrestrial technologies. This paper will focus initially on the science that can be achieved when removing the current constraints imposed by wheeled rovers, leading to an outline mission for the Mars Hopper to be used as a case study for critical subsystem and vehicle design. The enabling technologies required to meet these demands are then discussed in light of driving requirements, covering

the range of possible technologies and concepts, trade-offs and subsystem architecture definitions.

Atromos: A Nanosat-Class Mission For The Surface of Mars

M.Murbach, P. Papadopoulos, D. Atkinson

The future scientific exploration of Mars can be greatly enhanced by complementing currently planned large surface missions with a set of smaller ones deployed from the same carrier spacecraft. In that Since many of the most scientifically interesting locations on Mars are out of reach of current Entry/Descent/Landing technologies - a complementary capability that would have such access - would be compelling. Such a mission concept is proposed, using entry technologies currently under development. The entry probe is a Tube Deployed Re-entry Vehicle (TDRV) which has the critical attributes of being self-stabilizing, and of sufficiently low ballistic coefficient that the convective heating rates are significantly reduced. The initial probe concept would be deployed from a large version of a standard cubesat launcher (a large version of the P-POD - Poly PicoSat Orbital Deployer) with dimensions of 30x30x100cm (a '90U' launcher).

Planetary Probes as Precursors for a Human Mars Mission

Rob Grover

NASA has established a Mars exploration policy that all robotic Mars missions contribute to knowledge and experience toward a human mission to Mars. Mars Design Reference Architecture 5 (DRA 5) is NASA's current reference architecture for a landed human Mars mission, and illustrates specific EDL challenge areas for a landed human mission, such as flexible TPS, supersonic retropropulsion, and the transition of vehicle configurations during EDL. Efforts to roadmap Mars EDL technologies to connect today's robotic missions to an eventual human landing provide potential pathways for developing the necessary capabilities to land a crewed mission. Future Mars robotic missions and the planetary probe community serve an important role in providing precursor opportunities for a human mission to Mars. This presentation will stimulate discussion on the role of robotic spacecraft in developing capability for an eventual human Mars mission.

Session 8A: Cross Cutting Technologies II

Dual Body Hypersonic Inflatable Decelerator for Atmospheric Entry

Sajjad Reza, Suraj Rawal, Maxim de Jong, Brian Aiken, Essam Sheta, Sami Habchi

The NASA' vision is to develop Hypersonic Inflatable Aerodynamic Decelerators (HIADs) for future robotic and human missions requiring atmospheric entry. These inflatable decelerators with increased surface-area footprints reduce heat flux, and induced temperatures, and are stowable in a compact volume within the launch vehicle payload fairing and spacecraft. During a NASA sponsored conceptual design development effort of 10 meter HIAD, the team of Lockheed Martin Space Systems Company / Thin Red Line Aerospace, and CFDR Inc. has developed an innovative inflatable structure (IS) based on an existing Ultra High Performance Vessel (UHPV) design. The UHPV based design offers low mass design simplicity with reduced manufacturing complexity and unlimited scalability. It also allows more simple integration of flexible TPS into the inflatable fabric structure. Leveraging upon developmental hardware validated by test and analysis, UHPV design architecture can produce highly predictable, scalable performance configurations that exhibit stable in-flight performance. During the concept study effort a Dual Body IS was developed with an annular primary inflatable supporting a secondary sphere cone inflatable. The annular structure maintained at a higher pressure of ~5 psi (35kPa) provides a large diameter footprint for the lower pressure (~1.5 psi, 10.3kPA) secondary sphere cone volume that is the primary UHPV drag surface. This paper presents an overview of the development of the "Dual body" UHPV based design concept, and its key advantages for use as part of a planetary entry system.

Simulator for orbital observation of atmospheric entries phenomena

A.Bouquet, D. Baratoux, J. Vaubaillon, M.I. Gritsevich, D. Mimoun, O. Mousis

The determination of the exact characteristics of the present flux of interplanetary matter entering the Earth's atmosphere from ground-based or space-based optical observations is challenged by the complexity of the mechanics of atmospheric entry of small bodies. Spectroscopic analysis in particular shows great variations in lines from both atmospheric components and body ablated components from meteor to meteor, depending both on characteristic of body and its entry trajectory. The METEOR project aims at putting in orbit one or several detectors dedicated to the observations of atmospheric entries of small bodies (cm-sized to 10m-sized), with optimized and continuous conditions of observation and no impediment from lower layers of atmosphere. In preparation of this project, we are developing a simulator with two objectives. The first objective is the determination of the number of expected detections in function of the observations parameters. For this purpose, we have considered the existing data about the size repartition of meteoroids, and of the characteristics influencing their luminosity. Reasonable assumptions could be made on the number of events occurring in our detector's field of view and whether they would be bright enough to be spotted. This assessment was made considering an off-the-shelf CAMERA developed for this purpose: the SPOSH camera. Possible directions for improvements will be also discussed. The second objective is to assess the feasibility of predicting the characteristics of a meteoroid from the observations, and which constraints should be set on our detectors to achieve sufficient accuracy in this regard. Nature (rocky/icy or metallic), mass and velocity are the parameters to be determined in priority from the trajectory and luminous behaviour during the entry phase. If possible, these parameters may be used then to infer the orbital characteristics of the meteoroid and trace it back to its parent body. It appears that such a mission would allow for unprecedented number of detection of meteors, permitting numerous analyses of entries of bodies with different characteristics and thus a vast sampling of objects, which would complement in-situ analysis of parent bodies. Especially, spectroscopic analysis including UV domain would bring a wealth of data on the mechanics of atmospheric entry on Earth, which could be used to understand similar phenomena in atmospheres of other bodies in the solar system. This knowledge would help preparing future missions including atmospheric entries and using meteor phenomena as natural probing of atmospheric layers.

A System Analysis Tool for Design of Earth Entry Vehicles

Jamshid Samareh, Rick Winski, Anjie Emmett, John Theisinger, Lou Glaab, Michelle Munk, Parul Agrawal, Steve Sepka, Jose Aliaga

A System Analysis Tool for Design of Earth Entry Vehicles Jamshid Samareh*, Rick Winski, Anjie Emmett, John Theisinger, Lou Glaab, and Michelle Munk NASA Langley Research Center *e-mail: Jamshid.a.samareh@nasa.gov Parul Agrawal, Steve Sepka, and Jose Aliaga NASA Ames Research Center A significant step in space exploration is the return of sample materials from Mars to Earth for analysis. Missions that directly return sample material to Earth share one common element: an Earth entry vehicle (EEV). A planetary entry, descent, and landing system for an EEV typically consists of a heatshield for entry, a parachute for descent, and either retro rockets or airbags for landing. The current EEV design, shown in Fig. 1, is a simple passive entry system that replaces the parachute and landing system with a hardened container surrounded by sufficient energy absorbing material to assure containment during ground impact. The EEV outer mold line (OML) is a 60o sphere-cone with the spherical nose designed to control the maximum stagnation heat rate. The OML is designed to provide hypersonic re-orientation capability in the event of entry attitude failures due to spacecraft separation or meteoroid impact. The backshell is concave, and it is connected to the heatshield at the shoulder with the appropriate shoulder radius to control the maximum heat rate. The options for forward TPS are fully dense carbon phenolic or Phenolic Impregnated Carbon Ablator (PICA). Acusil and Silicone Impregnated Reusable Ceramic Ablator (SIRCA) are the TPS options for the backshell. The internal structures are made of carbon-carbon, designed to withstand only the launch and entry loads. The potential for terrestrial contamination from returned sample material is a major driver for EEV design. In this example, the payload is stored in an Orbiting Sample (OS) container, which has a spherical shape and is designed to withstand entry and ground impact loads. An impact sphere with a cellular structure surrounds the OS. The analysis and design of an EEV is multidisciplinary in nature, requiring the application of many disciplines. Integration of a multidisciplinary problem is a challenging task; the execution process and data transfer among disciplines should be automated and consistent. Integrated multidisciplinary analysis tools improve the performance of the systems analysis team by automating and streamlining the process, and this improvement can reduce the errors resulting from manual data transfer among discipline experts. In addition, the use of an integrated tool improves and speeds up the design activities such as trade studies, sensitivity analyses, Monte Carlo analyses, and vehicle optimization providing a quantum improvement in the understanding of the trade space involved. The paper describes an integrated system for the analysis and design of an EEV for sample return missions. The system includes the following modules: geometry, mass sizing, impact analysis, structural analysis, flight mechanics, and TPS.

Inexpensive Sensing for Full State Estimation of Spacecraft

Benoit Pigneur, Kartik B. Ariyur

Inexpensive Sensing for Full State Estimation of Spacecraft Benoit Pigneur (student)* and Kartik B. Ariyur School of Mechanical Engineering, Purdue University *e-mail: bpigneur@purdue.edu Key topics: sensor, measurement, autonomous guidance and control, inertia, center of gravity Over the years, several autonomous control algorithms have been studied and developed for space exploration and planetary or asteroid landing. At the Autonomous Systems Laboratory at Purdue University, we believe that the autonomy of a spacecraft can be dramatically improved by direct measurements of the full state. This paper starts with the premise that better measurements will improve the autonomy of the GNC algorithms. The purpose of autonomy ultimately is to reduce the costs of these vehicles during their operational life. To that end, we choose to use inexpensive, low SWAP (size, weight, and power) sensing elements, such as MEMS accelerometers on every moving part in the space vehicle along with CMOS imagers for celestial body tracking on the outer surface of the vehicle. The novelty does not reside in the development of new costly instruments but in a different way to exploit well characterized, light-weight, and inexpensive sensors. Through the use of multiple accelerometers and multiple imagers, we are able to obtain a broader range of measurements, and thus greater information on the state of the vehicle. In this paper we lay emphasis on the estimation of the real time variation of the inertia, the mass and the center of gravity of the spacecraft by measurement of accelerations. Then we analyze its effects on the autonomous control algorithms of the spacecraft. We isolate the acceleration terms related to specific behaviors of different systems and parts of the spacecraft. Each of them contributes to the variation of the inertia, mass and center of gravity of the spacecraft. For example, in the case of the displacement of a thruster nozzle, the thrust vector orientation and the application point can be determined from the acceleration vector. The contributing terms to the changes in spacecraft's inertia and the center of gravity are evaluated. This concept can be extended to other moving parts of the spacecraft (momentum control gyros, solar panel and deployable sails). Better knowledge of the inertia, mass and center of gravity of the vehicle permit reliable autonomous control algorithms that need little human supervision. New guidance and control algorithms are developed based on accurate knowledge of the real-time variation of the inertia, the mass and the center of gravity of the spacecraft. This paper presents this technique for a simple scenario in the case of an airless body landing. The benefits of this new approach are also discussed. Finally, similar concepts are extended to an atmospheric environment. Several studies identified guidance and control during atmospheric entry as key technologies for future Mars and other planetary explorations. The two most popular technologies are bank angle modulation and CoG/Inertia variation. We believe that this paper will contribute towards the development of better CoG/Inertia control for atmospheric entry. In addition, better knowledge of the state of the spacecraft due to improved measurement techniques will be beneficial not only for entry, descent and landing but also for other scenarios such as station keeping, in-orbit control, and collision avoidance maneuvers. Increased autonomy will

help to decrease the cost of exploitation of spacecraft in different mission configurations (LEO, orbital transfers, and planetary exploration).

SUBSONIC DYNAMIC STABILITY TESTS IN SUPPORT OF PARAMETRIC MULTI-MISSION EARTH ENTRY VEHICLE ANALYSIS

Lou Glaab, Mike Fremaux

Multi-Mission Earth Entry Vehicles (MMEEVs) are blunt-body vehicles designed with the purpose of transporting payloads from outer space to the surface of the Earth. To achieve high-reliability and minimum weight, MMEEVs avoid use of limited-reliability systems, such as parachutes, retro-rockets, and reaction control systems and rely on the natural aerodynamic stability of the vehicle throughout the Entry, Descent, and Landing (EDL) phase of flight. The Multi-Mission Systems Analysis for Planetary Entry (M-SAPE) parametric design tool is used to facilitate the design of MMEEVs for an array of missions and develop and visualize the trade space. Testing in NASA Langley's Vertical Spin Tunnel (VST) was conducted to significantly improve M-SAPE's subsonic aerodynamic models. Vehicle size and shape can be driven by entry flight path angle and speed, thermal protection system performance, terminal velocity limitations, payload mass and density, among other design parameters. The objectives of the VST testing were to define usable subsonic center of gravity limits, and aerodynamic parameters for 6-degree-of-freedom (6-DOF) simulations, for a range of MMEEV designs. The range of MMEEVs tested was from 1.8m down to 1.2m diameter. A backshell extender provided the ability to test a design with a much larger payload for the 1.2m MMEEV.

Combining Laser Communications and Power Beaming for use on Planetary Probes

Liz Hyde, James Grady, Marcus Murbach, Periklis Papadopoulos

Two aspects of any planetary mission is the ability to transmit and receive data, and generate power. Laser communication has the ability of increasing data received by a factor of 100, with a small power and mass penalty. This technology has been demonstrated on the ground as well as in space, with an image successfully transmitted to the Lunar Reconnaissance Orbiter (LRO). Laser power transfer has the potential to be a way to “recharge” a probe along side, or instead of traditional power generation methods. For planetary missions, this can supplement existing solar arrays, or provide bursts of high-voltage power that Radioisotope Thermoelectric Generators (RTGs) fail to provide. This technology has also been demonstrated on the ground, but has yet to be attempted in space. The core of these technologies is the laser itself. Creating a system that fulfills two roles has the potential to save power, mass and volume in an individual system. Using base stations, a network of probes has a centralized location for data and power storage and transmission. Creating an infrastructure that can provide power and communications coverage into deep space opens a new door into solar system exploration. This paper investigates the use of both laser communications and laser power transmission, their applicability for planetary missions, and models a small satellite that acts as a “base station” for a laser communications/power demonstration. This mission is designed as a “technology demonstration”, which can be expanded upon for future use as a base station in a Lunar or Martian environment.

The Doppler Wind Experiment on Titan in the Optical Communications Era

Kamal Oudrhiri, Sami Asmar, Bruce Moision

The Doppler Wind Experiment (DWE) was one of the six scientific investigations conducted on the Huygens probe during its mission to Titan [Lebreton et al., 2005]. During its descent thru Titan's atmosphere, Doppler shift of the transmitted radio signal from the probe were detected by several large radio telescopes on Earth and were successfully used to compute the velocity of the probe in the direction to Earth. In generating the profile of the zonal wind velocity, it was determined that the standard deviation of the Doppler residuals is 0.11 Hz and the measurement noise accuracy provided the best estimate of ±16 mm/s in radial velocity for wind profile [Folkner et al., 2006]. In line with the current NASA optical communications technology initiatives to develop capabilities for supporting interplanetary mission, we will examine the power-limited error term of the DWE that would be achievable with an optical communications link and compare it with the radio frequency error term.

Entry, Descent, and Landing Statistical Trajectory Reconstruction using Inertial Navigation and Monte Carlo Techniques

Rafael Lugo, Robert Blanchard, Robert Tolson, Lynn Huh

A data-driven navigation approach to entry, descent, and landing trajectory reconstruction that includes a statistical formulation is presented. The process, called Inertial Navigation Statistical Trajectory and Atmosphere Reconstruction (INSTAR), utilizes inertial navigation techniques to deterministically reconstruct a trajectory without the use of aerodynamic, atmospheric, control, or noise models. Statistics are introduced into the INSTAR process by using Monte Carlo dispersion techniques that are often used in pre-flight design and development of EDL missions. The major advantage of inertial navigation is that it requires no models of the aerodynamics, atmosphere, non-gravitational forces, or observation equations. A model of the gravitational forces is required, but for Mars, the gravity field is known with more than adequate precision and certainly orders of magnitude better than the aerodynamics and atmospheric models. If the initial conditions, sensors, and gravity model were perfect, the trajectory reconstruction would be perfect. In fact, without some redundant information, this is the only method to reconstruct the trajectory. In addition to minimal reliance on pre-flight atmospheric modeling, wind tunnel testing, and computational fluid dynamics (CFD) simulations, there are other advantages to the navigation approach. These advantages include: (1) results can be obtained relatively quickly and (2) there are no convergence issues. A significant disadvantage of inertial navigation is that there are no straightforward mechanics for generating valid statistics of the resulting state and attitude errors, something that the INSTAR process rectifies. In the INSTAR process, the initial state conditions (position, velocity, and orientation) and as IMU parameters (scale factors, biases, and misalignments) are dispersed in a Monte Carlo sense to within specified uncertainties. The observed IMU accelerations and rates are then integrated as in inertial navigation to obtain a set of dispersed reconstructed trajectories. Redundant data not normally used in inertial navigation such as landing site location, altimetry, air data system observations, etc. are then used to refine the dispersed trajectories to the subset that satisfies those observations. Trajectory parameter uncertainties are then obtained from this subset of dispersed trajectories. Thus, statistics are obtained using inertial navigation, without relying on the models required in typical statistical reconstruction techniques. The INSTAR process is demonstrated on flight data from the recent MSL mission, using landing site location as the redundant data source. The differences between the a priori initial conditions and the INSTAR solution were uniformly below the a priori 3-sigma uncertainties, with most parameter differences below 1-sigma. Wind angles and associated uncertainties were obtained from the dispersed trajectories as an example of how statistics of derived variables can be generated using INSTAR. It was found that including the landing site as an observation did not have a discernible effect on the nominal wind angle histories, although the uncertainties were reduced significantly. Acceleration biases on the order of tens of micro-g's were observed in the flight data. The differences between these biases and those obtained from the INSTAR solution were attributed to solving only for

acceleration biases. It is expected that solving for additional IMU errors such as scale factors and misalignments will bring the solution biases into closer agreement. Further work using the INSTAR process will include solving for additional IMU errors and incorporating other forms of redundant observations, such as altimetry. Pressure measurements from MEADS will also be included to aid in atmosphere reconstruction. Eventually, INSTAR will provide trajectory and atmosphere statistics that incorporate multiple sources of redundant data.

Challenges of pin-point landing for planetary landing : the LION absolute vision-based navigation approach and experimental results

VOIRIN Thomas, Jeff Delaune, Clément Bourdarias, Challenges of pin-point landing for planetary landing : the LION absolute vision-based navigation approach and experimental results. After a first phase of mostly orbital exploration, next European planetary exploration projects will aim at performing either in-situ investigations (through rovers or static ground stations) such as Lunar Lander, Marco Polo and Mars Precision Landing missions. These contemplated projects require accurate, safe and autonomous de-orbiting, entry, descent & landing capabilities, with challenging requirements towards the navigation subsystem. After ExoMars in 2016 and 2018, future ESA missions to Mars, the Moon, or asteroids will require safe and pin-point precision landing capabilities, with for example a specified accuracy of typically 200m at touchdown for a Moon landing, encompassing control, guidance & navigation dispersions. Such accuracy will be required in future exploration missions, for example to deliver cargo supply to assets previously disposed on Lunar Surface (in which case the 200m error can then be easily recovered by a surface mobility vehicle), or to land within a site of major scientific interest with valuable samples for a mars mission. State-of-the art technology in Europe, based on Terrain-Relative Navigation (TRN) sensors such as Wide-Field-of-View vision-based camera (so-called “VisNAV” concept) or Imaging Lidar, enable to meet the “safe precision” landing requirement, by providing an accurate ground-relative velocity estimate, and full hazard detection functionality. ESA-led technology activities in the past ten years have raised the TRL level of such sensors to typically 4/5. To reach the pin-point precision requirement however, on-board absolute navigation with respect to the landing site is mandatory, with a typical accuracy better than 100m at touch-down for a Lunar mission, or below 10 and 1 km (respectively along and cross-track) at Entry interface for a Mars landing missions. Various techniques are being currently explored by ESA and partners, based on vision-based recognition of known landmarks. Such options rely on the a priori identification of visual landmarks from previous orbital missions imagery (e.g. craters), co-registered with a high resolution Digital Elevation Map (also available from previous orbital missions) to generate a landmarks database, which will be used by thon-board vision-based navigation system as a surface map. So-far, most of the developments have focused on the use of craters as visual landmarks ; however, such landmarks may not be found on all planetary surfaces, and may limit the use of such algorithms to areas where at least 3 to 4 craters are in visibility ; for the Moon, this limits its altitude of use to several km. The present paper focuses on an alternative vision-based navigation system, called LION, co-funded by ESA, ASTRIUM and ONERA, which is fully generic and therefore applicable to all future landing missions. LION relies on the use of generic visual landmarks (Harris points), and is able to cope with any 3D topography. Scale invariance is brought by the use of the scale of each landmark as an additional matching parameter. The LION activity includes the design and prototyping of the navigation system and then its incremental experimental validation through simulation (virtual images), hardware-in-the-loop testing on a planetary mock-up (hosted at ESTEC), and then test flight onto ONERA’s

RESSAC UAV. This incremental process implemented in this activity has confirmed the ability of LION to meet the demanding requirements of precision landing for the Moon application, with a Technology Readiness Level of 3 – 4. In this paper, the LION navigation system will be first presented, as well as the implemented incremental validation and verification approach ; experimental set-up and end-to-end tests results will be summarized. Finally, applicability to various planetary landing probes, way forward and lessons learned will be discussed.

Session 8B: Cross Cutting Technologies III

RASTAS SPEAR : Radiation-Shapes-Thermal Protection Investigations for High Speed Earth Re-entry

Jean-Marc Bouilly, Aurélien Pisseloup, Olivier Chazot, George Vekinis, Alexis Bourgoing, Bruno Chanetz

An important step for Space Exploration activities and for a more accurate knowledge of the Earth, universe and environment is to develop the capability to send vehicles into space, which collect and return to Earth samples from solar system bodies. To return these samples, any mission will end by high-speed re-entry in Earth's atmosphere. This requires strong technological bases and a good understanding of the environment encountered during the Earth re-entry. Investment in high speed re-entry technology development is thus appropriate today to enable future Exploration missions such as Mars Sample Return. Rastas Spear project started in September 2010, with the main objective to increase Europe's knowledge in high speed re-entry vehicle technology to allow for planetary exploration missions in the coming decades. The research leading to these results has received funding from the European Union Seventh Framework Programme (FP7/2007-2013) under grant agreement n° 241992. The project's main objective can be derived in sub-objectives as follows: - OBJ1: To better understand phenomena during high speed re-entry enabling more precise Capsule sizing and reduced margins. - OBJ2: To identify the ground facility needs for simulation - OBJ3: To master heat shield manufacturing techniques and demonstrate heat shield capabilities. - OBJ4: To master damping at ground impact and flight mechanics and thus ensure a safe return of the samples. This study has been carried out by a consortium of European companies and institutes : VKI (B), Kybertec (Cz), Demokritos (Gr), IoA (Pl), CIRA (I), CFS (CH), MSU (Ru), CNRS and ONERA (F), and coordinated by Astrium (F). After shortly reminding the organisation and objectives of the RASTAS SPEAR project, the scope of this paper is to present its main achievements, aiming at enhancing the basic capabilities on some specific topics such as: - Aeroshape stability - High speed aerothermal environment - Sub-system / equipment : Thermal protection, Crushable material REFERENCES [1] www.rastas-spear.eu [2] J-M. Bouilly et al, RASTAS SPEAR: Radiation-Shapes-Thermal Protection Investigations for High Speed Earth Re-entry, 9th International Planetary Probe Workshop IPPW9, June 18 - 22, 2012, Toulouse

Demonstrated Modular Manufacturing of H/C Reinforced Elastomeric 2.65-m Heatshield -- 9-Module System 300 W/cm2 Capable

William M. Congdon

June 2013 Demonstrated Modular Manufacturing of H/C Reinforced Elastomeric 2.65-m Heatshield – 9-Module System 300 W/cm2 Capable William M. Congdon ARA Ablatives Laboratory (ABL) Centennial, Colorado 80112 Charring ablator heatshields for NASA planetary missions, especially Mars entry and Earth return, are commonly reinforced with composite honeycomb (H/C) as well as internal chopped fibers. Use of H/C can significantly increase the overall system robustness because, among several advantages, it reinforces the char layer and compartmentalizes the surface against char separation. This paper presents results from a multi-year development effort sponsored by NASA SMD's In-Space Propulsion Technologies Project to enable modular manufacturing of large-scale, honeycomb-reinforced ablative heatshields. Large monolithic heatshields of 4.0-m diameter and greater can pose significant manufacturing challenges when conventional honeycomb packing is used. However, modular manufacturing has now been demonstrated to greatly facilitate production of such systems. The figure below shows some processing steps for ABL's 2.65-m modular SRAM-20 aeroshell that was produced under ISPT. It shows flank module molding, milling and fit checking, as well as final 5-axis CNC milling of the fully-bonded modular system (1 nose, 8 flanks). The structure was a sandwich composite with graphite facesheets and titanium H/C core. SRAM-20 used for the heatshield demonstration is an elastomeric, silicone-based ablator material with a density of 0.32 g/cm3 (20 lb/ft3) and a tensile strain-to-failure of 4 percent. Over the past ten years it has been tested extensively in the IHF, AHF, and ARMSEF arc-jets and has shown good performance to heating rates up to and beyond 300 W/cm2. The figure below shows stagnation test results for 5.0-in. diameter samples of this ablator exposed to heating rates of 128, 254, and 304 W/cm2. Funded Under NASA Contract NNM07AA93C W.M.Congdon April 2013

Ballistic Range Tests of Saturn and Uranus Atmospheric Entry Probes: Heat Transfer Augmentation due to Surface Roughness

Michael C. Wilder, Dinesh. K. Prabhu, Daniel. C. Reda

Experiments in support of Saturn and Uranus atmospheric entry probe mission studies are being conducted in the NASA Ames Hypervelocity Free Flight Aerodynamic Facility (HFFAF) ballistic range to determine the effects of surface roughness on convective aerodynamic heat transfer to an atmospheric entry vehicle geometry. Such effects are of particular importance when ablative Thermal Protection System (TPS) materials are required. As the TPS material ablates on entry, a surface roughness characteristic of the material forms. Interactions between this roughness and the wall-bounded shear layer can significantly increase the surface heat transfer and skin friction, with the degree of augmentation depending on the roughness size relative to the shear-layer thickness. Experiments are underway on a 45o sphere-cone geometry at conditions relevant to Outer-Planet atmospheric entry probe missions. Several key parameters of full-scale flight will be matched in the experiments, including the roughness scale relative to the boundary-layer scale, the smooth-wall turbulent convective heating rates, and, for certain trajectories, the free stream Reynolds number. The full paper will include these results and comparisons with published data Similar data were recently obtained on a 30o sphere-cone geometry flown at an average speed of 3.4 km/s through quiescent room-temperature air at 0.3 atm pressure (M = 10, ReD = 2 x 106). The average stagnation-point heat flux was 4500 W/cm2 ± 200 W/cm2. A model is shown in flight in Fig. 1(a). Heat transfer rates were determined from global surface temperature measurements obtained using high-speed thermal imaging cameras stationed along the flight path. The ratio of rough-wall to smooth-wall turbulent heat transfer (q/q0) at a point midway along the conic frustum is shown in Fig. 1(b). The data correlate well with the logarithm of the roughness Reynolds number, Rek, here based on the computed smooth-wall turbulent boundary-layer properties and the measured mean roughness height. Results showed increases in heat transfer as high as 60% for the roughness heights examined. Surface roughness was produced by grit-blasting, and characterized using non-contact optical profilometry. The roughness distributions qualitatively resemble those of several ablated TPS materials, including chop-molded carbon-phenolic. Details will be included in the full paper.

Conformal Ablative Thermal Protection System for Planetary and Human Exploration Missions

Robin A. S. Beck, James O. Arnold, Matthew Gasch, Mairead Stackpoole, Christine Szalai, Paul Wercinski, Ethiraj Venkatapathy

The Office of Chief Technologist, NASA identified the need for research and technology development in part from NASA's Strategic Goal 3.3 of the NASA Strategic Plan to develop and demonstrate the critical technologies that will make NASA's exploration, science, and discovery missions more affordable and more capable. Furthermore, the Game Changing Development Program is a primary avenue to achieve the Agency's 2011 strategic goal to "Create the innovative new space technologies for our exploration, science, and economic future." The National Research Council (NRC) Space Technology Roadmaps and Priorities report highlights six challenges and they are: Mass to Surface, Surface Access, Precision Landing, Surface Hazard Detection and Avoidance, Safety and Mission Assurance, and Affordability. In order for NASA to meet these challenges, the report recommends immediate focus on Rigid and Flexible Thermal Protection Systems. Rigid TPS systems such as Avcoat or SLA are honeycomb based and PICA is in the form of tiles. The honeycomb systems are manufactured using techniques that require filling of each (3/8" cell) by hand, and in a limited amount of time all of the cells must be filled and the heatshield must be cured. The tile systems such as PICA pose a different challenge as the low strain-to-failure and manufacturing size limitations require large number of small tiles with gap-fillers between the tiles. Recent investments in flexible ablative systems have given rise to the potential for conformal ablative TPS. A conformal TPS over a rigid aeroshell has the potential to solve a number of challenges faced by traditional rigid TPS materials. The high strain-to-failure nature of the conformal ablative materials will allow integration of the TPS with the underlying aeroshell structure much easier and enable monolithic-like configuration and larger segments (or parts) to be used. By reducing the overall part count, the cost of installation (based on cost comparisons between blanket and tile materials on shuttle) should be significantly reduced. The conformal ablator design will include a simplified design of seams between gore panels, which should eliminate the need for gap filler design, and should accommodate a wider range of allowable carrier structure imperfections when compared to a rigid material such as PICA. The Conformal TPS development project leverages the past investments made by earlier projects with a goal to develop and deliver a TRL 5 conformal TPS capable of ~250 W/cm2 for missions such as MSL or COTS missions. The capabilities goal for the conformal TPS is similar to an MSL design reference mission (~250 W/cm2) with matching pressures and shear environments. Both conformal and flexible carbon-felt based materials were successfully tested in stagnation aerothermal environments above 500 W/cm2 under earlier programs. Results on a myriad of materials developed during FY11 were used to determine which materials to start with in FY12. In FY12, the conformal TPS element focused on establishing materials requirements based on MSL-type and COTS Low Earth orbit (LEO) conditions (q ~ 250 W/cm2) to develop and deliver a Conformal Ablative TPS. This involved down selecting, manufacturing and testing two of the best candidate materials, demonstrating uniform infiltration of resins into baseline

~1-in thick carbon felt, selecting a primary Conformal material formulation based on arc jet and material property testing, developing and demonstrating advanced instrumentation for felt-based materials and based on the material property and arc jet data, developing low fidelity material response model so that the conformal TPS thickness for missions can be established. In addition, the project began to develop Industry Partnerships. Since the nominal thickness of baseline carbon felts was less than 1 inch, a partnership with a rayon felt developer was made in order to upgrade equipment, establish the processes required and attempt to manufacture ~4-in thick white goods. A partnership with a processing house was made to develop the methodology to carbonize large pieces of the white goods into ~3-in thick carbon felt. In FY13, development and refining metrics for mission utilization of conformal ablator technology along with assessment for potential mission stakeholders will be carried out. By the end of the second year, the project will develop and deliver CA250 to TRL ~5 by continue material characterization testing that includes further thermal, arc jet and structural properties and a mid-fidelity thermal response model. In addition, developing and demonstrating joining technique/seams in aerothermal environment will lead to system level technology maturation and demonstration.

Testing of Advanced Conformal Ablative TPS

Matt Gasch, Parul Agrawal, Robin Beck

Novel low-density (0.28 g/cm3) ablative thermal protection materials have been developed under a project called CA250 at NASA Ames Research Center (ARC). These new materials are not only capable of withstanding high heat fluxes and shear stresses, they can also be made to conform to/comply with the underlying aeroshell without the use of any strain isolation because of their high strain-to-failure values. These materials, therefore, hold the promise of being able to overcome a number of technical challenges faced by traditional rigid ablative materials such as, PICA (Phenolic Impregnated Carbon Ablator) used in the heatshield for the Mars Science Laboratory (MSL), or honeycomb based Avcoat used in the heatshield for the Multi-Purpose Crew Vehicle (MPCV) currently under development. Furthermore, these materials can be scaled such that a single piece monolithic construction could be used for small size capsules, or cover the entire heatshield of a large entry capsule with a significantly reduced part count compared to a heatshield tiled with material of equivalent performance. The new materials, however, have to be tested and qualified in aerothermal environments that are flight representative so that they can be infused into future US missions that have an atmospheric entry component. The proposed paper will present recent results of a test campaign conducted in the arcjet facility at NASA ARC. In keeping with the novelty of the thermal protection material, tests were conducted using a novel geometry called SPRITE (Small Probe Re-entry Investigation for Tps Engineering).1 SPRITE enables simultaneous replication of pressure, heat flux and shear stress, and departs from the traditional method of arcjet testing in which combinations of pressure and heat flux, or heat flux and shear, are replicated in either stagnation mode using a blunt cylinder, or a shear mode using a blunt wedge. Furthermore, since the aerothermal environments along the conical flank are nearly constant, a sufficiently large test article ensures enough run length to make pressure and heat flux measurements under constant shear stress. The SPRITE test geometry used in the tests of conformal thermal protection materials is a 55° blunt cone, roughly 8-in diameter. Four gores of candidate materials – either low density (0.28 g/cm3) material similar to PICA or SIRCA (Silicone Impregnated Refractory Ceramic Ablator – were bonded on to the SPRITE shell using RTV-560 adhesive. The conformal version of PICA was developed with a carbon felt substrate infiltrated with phenolic resin, and the conformal version of SIRCA uses silicone infiltration into the carbon felt substrate. Both materials were successfully tested with the SPRITE configuration at heat fluxes ranging from 250 to 500 W/cm2 and for pressures ranging from x kPa to y kPa. The shear stresses experienced by the materials were estimated to be in the range of z Pa to t Pa. The measured recession and temperature profile for these materials were comparable to PICA.

Flexible Thermal Protection Systems Trade Studies for High Energy Atmospheric Reentry Test Vehicle

Joseph A. Del Corso, Ali R. Mazaheri, Aaron D. Olds, Nathanael J. Mesick, Walter E. Bruce III, Stephen J. Hughes, Henry S. Wright

High Energy Atmospheric Re-entry Test (HEART) flight project has investigated several thermal protection system (TPS) candidates with four Outer Mold Line (OML) cone angles and re-entry trajectories, to be used for an International Space Station return mission using an inflatable aeroshell. This trade study investigated four aerodynamically stable OML cone angles with different spherical nose cap diameters, for characterization and sizing of several TPS candidates for survivability down to Mach 0.7. The TPS sizing was performed using Fully Implicit Abalation and Thermal Response Program (FIAT) and the aerothermodynamic environments calculated using Langley Aerothermodynamic Upwind Relaxation Algorithm (LAURA), a reacting viscous flow solver. The intent of the trade study was to investigate a range of large deployable aeroshell, which increase the drag performance while significantly reducing the ballistic coefficient of high-mass entry vehicles, to survive re-entry conditions and allow for TPS down-selection. This paper provides an overview of the trade study performed and TPS sizing results. Included in this paper is an overview of, and results from, trajectory development, CFD analysis of OML, TPS sizing using FIAT, and preliminary ground test results, of three high-potential TPS candidates.

Woven Thermal Protection System (WTPS) a Novel Approach to Meet NASA's Most Demanding Reentry Missions

Donald Ellerby, Ethiraj Venkatapathy, Margaret Stackpoole, Ronald Chinnapongse, Adam Beerman, Jay Feldman, Keith Peterson, Dinesh Prahbu, Michelle Munk

NASA's future robotic missions utilizing an entry system into Venus and the outer planets, namely, Saturn, Uranus, Neptune, result in extremely severe entry conditions that exceed the capabilities of state of the art low to mid density ablators such as PICA or Avcoat. Therefore mission planners typically assume the use of a fully dense carbon phenolic heat shield similar to what was flown on Pioneer Venus and Galileo. Carbon phenolic (CP) is a robust TPS material however its high density and relatively high thermal conductivity constrain mission planners to steep entries, with high heat fluxes and pressures and short entry durations, in order for CP to be feasible from a mass perspective. The high entry conditions pose challenges for certification in existing ground based test facilities and the longer-term sustainability of CP will continue to pose challenges. In 2012 the Game Changing Development Program (GCDP) in NASA's Space Technology Mission Directorate funded NASA ARC to investigate the feasibility of a Woven Thermal Protection System (WTPS) to meet the needs of NASA's most challenging entry missions. This project was highly successful demonstrating that a Woven TPS solution compares favorably to CP in performance in simulated reentry environments and provides the opportunity to manufacture graded materials that should result in overall reduced mass solutions and enable a much broader set of missions than does CP. Building off the success of the WTPS project GCDP has funded a follow on project to further mature and scale up the WTPS concept for insertion into future NASA robotic missions. The matured WTPS will address the CP concerns associated with ground based test limitations and sustainability. This presentation will briefly discuss results from the WTPS Project and the plans for WTPS maturation into a heat-shield for extreme entry environment.

Performance of Woven TPS Ablative Materials

Mairead Stackpoole, Ethiraj Venkatapathy, Jay Feldman

Woven Thermal Protection Systems (WTPS) is a new TPS concept that is funded by NASA's Office of the Chief Technologist (OCT) Game Changing Division. The WTPS project demonstrates the potential for manufacturing many TPS architectures capable of performances demanded by the many potential solar system exploration missions. Currently, missions that encounter heat fluxes in the range of 1500 – 4000 W/cm2 and pressures greater than 1.5 atm have very limited TPS options - only one proven material, fully dense Carbon Phenolic, is currently available for these missions. However, fully dense carbon phenolic is only mass efficient at heat fluxes greater than 4000 W/cm2, and current mission designs suffer this mass inefficiency for lack of an alternative mid-density TPS. WTPS not only bridges this TPS gap but also offers a replacement for carbon phenolic, which itself requires a significant and costly redevelopment effort to re-establish its capability for use in the high heat flux missions recently prioritized in the NRC Decadal survey, including probe missions to Venus, Saturn and Neptune. This presentation will introduce some woven TPS architectures considered in this project and summarize some recent arc jet testing to evaluate the performance of fully dense and mid density WTPS. Performance comparisons to heritage carbon phenolic will be drawn where applicable.

In-situ material response characterization of lightweight ablators in high enthalpy plasma flows

Bernd HELBER, Thierry MAGIN, Olivier CHAZOT, Annick HUBIN

The success of spacecraft entering a planetary atmosphere relies to a great extent on the thermal protection system (TPS) and the associated thermal protection material (TPM), which protects the vehicle from the severe heating during hypersonic flight. Ablative materials are able to accommodate high heat loads by transforming the energy of the high enthalpy gases into degradation of the material through pyrolysis and ablation, whilst the remaining solid material insulates the vehicle substructure. They are the only choice for Earth reentries at velocities higher than 10km/s (Stardust, 12.9km/s), typical for return missions from extraterrestrial bodies, and are used for planetary probes, as currently demonstrated by the Mars Science Laboratory (August 5, 2012). A critical problem for safety of the astronauts and embarked payload is the associated material response model, required for prediction of the material performance and the design of the heat shield thickness. A sound methodology for the determination of the design margins is still missing, demanding an important effort in basic theoretical, numerical and experimental research to understand the complex gas-surface interaction and material response in plasma flows for advancement of the prediction tools used for the TPS design of tomorrow's space vehicles. Following the current effort at the von Karman Institute to characterize porous low-density ablators, we applied a comprehensive experimental setup to the 1.2MW Plasmatron facility for in-situ material response investigation. Three emission spectrometers recorded the highly reacting boundary layer along the stagnation line, allowing for spatially and temporally resolved signals. A high-speed camera tracked the hot sample surface in order to estimate ablation rates and to determine the post-test volume of ablated samples. We performed additional post-test examination with the support of microscopic and spectroscopic tools to address the surface modification and internal degradation due to pyrolysis and ablation. Three different ablative materials were tested: the two European ablators AQ61 and ASTERM, in addition to a carbon fiber preform, similar to PICA's precursor fiberform. We performed tests in air and nitrogen plasmas at pressures between 1.5 - 20kPa and heat fluxes of 1MW/m2 and 3MW/m2. We identified species originating from internal pyrolysis and char layer ablation such as C2, CH, CN, NH, and OH. Recorded emission intensities of different carbonic molecules correlated very well for several tests when reaching identical probing distances from the ablating surface. A preliminary analysis of the CN violet spectrum in the range 370-390nm using the simulation software SPECAIR and applying a two temperature model (Trot=Ttrans, Tvib=Telect) yielded a higher deviation from thermal equilibrium at low chamber pressures. The shape of the ablated test sample yielded the post-ablation volume under the assumption of an axisymmetric body. If the char density gradient inside the material is known, we can estimate the mass loss due to pyrolysis outgassing and hence, the gas mass blowing rate. Current ablation models use non-dimensional ablation tables, usually referred to as B'- tables, for determination of the char consumption rate as a function of the pyrolysis-gas mass flux (B'g and B'c). The coupling strategy of material codes with the flow environment is based on

considering a thin control volume close to the wall where mass balance and equilibrium chemistry are satisfied. The benefit of experimentally estimated pyrolysis gas blowing rates will be twofold: 1) the data will enable a direct comparison with the pyrolysis mass flux obtained from material response codes and 2) provide the boundary condition for stagnation line descriptions including B'-table generation if no material model is used. Micrographs depicted strong thinning of the carbon fibers at the surface due to oxidation from the fiber tips, resulting in an icicle shape. Charred, partially ablated resin became visible in a depth of ?50 µm below the surface. We refer to this zone, between fiber tips and charred resin, as ablation zone, in which oxygen leads to a fast ablation of all charred resin, due to its high reactivity, before the carbon fibers start to oxidize. This behavior changed drastically when nitrogen was used for plasma generation, where ablated fibers showed strong corrosion along the whole fiber length and charred resin was visible at the surface. Moreover, a mesh of thin “cross filaments” was visible at the surface probably assignable to very long chains of interconnecting C-C bonds, called catenation. Those bonds are strong, and stable, allowing carbon to form an almost infinite number of compounds. It seems plausible that, in a nitrogen atmosphere, the available carbon from gasified hydrocarbons becomes solid again to form such carbon chains.

THE CHALLENGES OF CREDIBLE THERMAL PROTECTION SYSTEM RELIABILITY QUANTIFICATION

Lawrence Green

The talk will address the questions: • What does it take to connect Thermal Protection System (TPS) design margins with quantified reliability? • Where we are and what do we still need to develop? Although the talk will leverage issues related to TPS design, most of the talk will be much broader than this – it will discuss the nuances of many of the uncertainties affecting very highly reliable systems. The talk will essentially present a top-ten list of the things most likely to contribute to uncertainty in reliability estimates for high reliability systems. The talk will describe four types of uncertainties (bias, dispersion, sampling and model form) that are recurrent throughout the current TPS reliability assessment process. It will then quantify the potential impact of these uncertainty sources on the reliability estimate and provide guidance as to where investments might be made to reduce these uncertainties. It will cover areas of data measurement and data fitting uncertainties, problem (model) formulation, statistical limitations of measured data sets, and statistical interpretation of solution results.

Session 9: Cross Cutting Technologies IV

Curiosity's Landing Dynamics as Observed at the CSIRO Parkes Radio Telescope

Sami Asmar, John Sarkissian, Kamal Oudrhiri, Daniel Kahan, Stephan Esterhuizen, Suzy Jackson, Brett Preisig, Özgür Karatekin, Hannes Griebel

The Mars Science Laboratory spacecraft, carrying the Curiosity rover, landed on the surface of Mars on 6 August 2012. The complex mission-critical seven-minute landing phase required prime as well as back-up communication methods. The prime method was via a UHF link to two Mars orbiters and an X-band link to the Deep Space Network. The back-up method was to attempt receiving the UHF link carrier via a large radio telescope on Earth. Given that this link was not designed for reception at Earth, the low expected received signal-to-noise ratio, the high signal dynamics, as well as the potentially overwhelming interference in the band on Earth, the chances of success were considered marginal. However, the 64-meter radio telescope at Parkes, Australia, received the signal solidly for the entire period utilizing a portable NASA Radio Science Receiver. The vital success at Parkes did not receive adequate notice, understandably, since the prime communication methods were also successful. The experience, however, proved to be instructive and paved the way for future missions to utilize the method. This paper describes the preparation process by CSIRO staff in building new instrumentation and minimizing the interference, the real time reception process, the reconstruction of the spacecraft dynamics from the received signal and the validation/correlation with the X-band profile as well as inertial measurements onboard the spacecraft.

Hypersonic Inflatable Aerodynamic Decelerator (HIAD) Technology Development Overview

Stephen J. Hughes, Dr. F. McNeil Cheatwood, Dr. Anthony M. Calomino, Henry S. Wright, Mary Elizabeth Wusk, Monica F. Hughes, Alan M. Cassell, Dave M. Bose

The successful flight of the Inflatable Reentry Vehicle Experiment (IRVE)-3 has further demonstrated the potential value of Hypersonic Inflatable Aerodynamic Decelerator (HIAD) technology. This technology development effort is funded by NASA's Space Technology Mission Directorate (STMD) Game Changing Development Program (GCDP). This paper provides an overview of a multi-year HIAD technology development effort, detailing the projects completed to date and the additional testing planned for the future. The effort was divided into three areas: Flexible Systems Development (FSD), Mission Advanced Entry Concepts (AEC), and Flight Validation. FSD consists of a Flexible Thermal Protection Systems (FTPS) element, which is investigating high temperature materials, coatings, and additives for use in the bladder, insulator, and heat shield layers; and an Inflatable Structures (IS) element which includes manufacture and testing (laboratory and wind tunnel) of inflatable structures and their associated structural elements. AEC consists of the Mission Applications element developing concepts (including payload interfaces) for missions at multiple destinations for the purpose of demonstrating the benefits and need for the HIAD technology as well as the Next Generation Subsystems element. Ground test development has been pursued in parallel with the Flight Validation IRVE-3 flight test. A larger scale (6m diameter) HIAD inflatable structure was constructed and aerodynamically tested in the National Full-scale Aerodynamics Complex (NFAC) 40by80 test section along with a duplicate of the IRVE-3 3m article. Both the 6m and 3m articles were tested with instrumented aerodynamic covers which incorporated an array of pressure taps to capture surface pressure distribution to validate Computational Fluid Dynamics (CFD) model predictions of surface pressure distribution. The 3m article also had a duplicate IRVE-3 Thermal Protection System (TPS) to test in addition to testing with the aero-cover configuration. Both the aero-covers and the TPS were populated with high contrast targets so that photogrammetric solutions of the loaded surface could be created. These solutions both refined the aerodynamic shape for CFD modeling and provided a deformed shape to validate structural Finite Element Analysis (FEA) models. Extensive aerothermal testing has been performed on the TPS candidates. This testing has been conducted in several facilities across the country. The majority of the testing has been conducted in the Boeing Large Core Arc Tunnel (LCAT). HIAD is continuing to mature testing methodology in this facility and is developing new test sample fixtures and control methodologies to improve understanding and quality of the environments to which the samples are subjected. Additional testing has been and continues to be performed in the NASA LaRC 8foot High Temperature Tunnel, where samples up to 2ft by 2ft are being tested over representative underlying structures incorporating construction features such as sewn seams and through-thickness quilting. With the successful completion to the IRVE-3 flight demonstration, mission concepts such as the HIAD Earth Atmospheric Reentry Test (HEART) are being developedwhich will demonstrate a relevant

scale vehicle in relevant environments. HEART would employ a large-scale aeroshell (10m) entering at orbital velocity (~7km/sec) with an entry mass on the order of 4MT. Also, the Build to Print (BTP) hardware (built as risk mitigation for the IRVE-3 project to have a “spare” ready to go in the event of a launch vehicle delivery failure) is now available for an additional sub-orbital flight experiment. Mission planning is underway to define a mission that can utilize this existing hardware and help the HIAD project further mature this technology.

Design and Execution of the Hypersonic Inflatable Aerodynamic Decelerator Large-Article Wind Tunnel Experiment

Alan M. Cassell

The testing of 3- and 6-meter diameter Hypersonic Inflatable Aerodynamic Decelerator (HIAD) test articles was completed in the National Full-Scale Aerodynamics Complex 40 ft x 80 ft Wind Tunnel test section. Both models were stacked tori, constructed as 60 degree half-angle sphere cones. The 3-meter HIAD was tested in two configurations. The first 3-meter configuration utilized an instrumented flexible aerodynamic skin covering the inflatable aeroshell surface, while the second configuration employed a flight-like flexible thermal protection system. The 6-meter HIAD was tested in two structural configurations (with and without an aft-mounted stiffening torus near the shoulder), both utilizing an instrumented aerodynamic skin. All of the primary test objectives were met, which included the measurement of the global and local deflected surface, surface static pressures, and structural strap loads under various inflation pressures, angles of attack, and wind tunnel dynamic pressures. Results from this test series reveal several advantages for conducting aerodynamic decelerator design, development and structural qualification work in a large-scale testing environment. In this paper, the test objectives, design and execution of the test are presented along with examples of the research data collected for simulation model development and validation.

Flight Performance of the Inflatable Reentry Vehicle Experiment 3 (IRVE-3)

Robert A Dillman, John DiNonno, Richard J Bodkin, Nathanael Miller, Aaron Olds, Walt Bruce

The Inflatable Reentry Vehicle Experiment 3 (IRVE-3) launched July 23, 2012, from NASA Wallops Flight Facility and successfully performed its mission, demonstrating the survivability of a hypersonic inflatable aerodynamic decelerator in the reentry heating environment and also illustrating the effect of an offset center of gravity on the inflatable's flight L/D. The reentry vehicle launched on a Black Brant XI sounding rocket and autonomously inflated its aeroshell to the reentry configuration, reoriented itself to the desired attitude for atmospheric entry, and mechanically shifted its center of gravity before reaching atmospheric interface. During reentry the inflatable aeroshell experienced a peak deceleration of 20 G's and a peak heat flux of approximately 14W/cm2. Performance data from the entire mission was telemetered to the ground for analysis. This paper discusses the IRVE-3 mission scenario, reentry vehicle design, and predicted versus as-flown performance of the on-board systems.

Development and Testing of an Exploration-Class Supersonic Inflatable Aerodynamic Decelerator for Planetary Entry

David Jurewicz, Brian Johnson, Ben Tutt, Daniel Coatta

The Exploration-class Supersonic Inflatable Aerodynamic Decelerator (SIAD-E) is an 8m diameter attached isotensoid. The isotensoid structure is a unique family of pressure vessels that produces uniform stress in the structure when inflated. Isotensoids were studied heavily from the 1950's – 1970's but were virtually abandoned with the successful supersonic Disc Gap Band (DGB) parachutes developed for the Viking mission. The SIAD-E program leverages the historical designs and thorough documentation from these early development programs. Advances in textile science have increased the thermal capability and material strength of aramids such as Technora and Kevlar. Additionally, the flex fatigue properties of Technora enable the decelerator to be tightly packed around a 4.7m rigid aeroshell. The decelerator increases the drag area of the aeroshell by initiating Gas Generators (GGs). The GGs expel the gas into diffusers that control the deployment of the device in the supersonic flow. Ram-air inlets are then used to maintain the inflation pressure. Deployment of the aeroshell above Mach 2, the standard deployment condition for supersonic DGB parachutes, slows the vehicle early in the entry trajectory. The early energy dissipation leads to increased landed mass, improved landing accuracy, and/or landing at higher altitudes. The planned SIAD-E test campaign will mature the attached isotensoid design to Technology Readiness Level (TRL) 5. The decelerator will undergo subsonic rocket sled testing to subject the unit to flight-like aerodynamic loads. This will be followed by a rocket-assisted balloon launch allowing the aeroshell to reach Mach 4 at an altitude of 50km, mimicking the atmospheric conditions on Mars. Increasing the TRL of the isotensoid architecture, through extensive documentation of the testing, allows this technology to be an important resource for future planetary exploration missions.

Conceptual Modeling and Analysis of Drag-Augmented Supersonic Retropropulsion for Application in Mars Entry, Descent, and Landing Vehicles

Michael A. Skeen, Ryan P. Starkey

The successful landing of the Mars Science Laboratory (MSL) in August 2012 represents the current state-of-the-art in Mars entry, descent, and landing (EDL) vehicles. The development of new decelerator technologies will be required as the desired payload mass for future robotic missions and potential crewed landing missions increase beyond the capability of the MSL EDL architecture. Supersonic retropropulsion (SRP) is one such candidate technology. SRP systems use rocket thrust directed into the free stream flow to decelerate the vehicle during descent. Previous investigations of aerodynamic-propulsive interactions resulting from SRP systems have shown that under certain conditions the aerodynamic drag on the entry vehicle can be preserved or even augmented. The manipulation of the flow field using SRP with multiple nozzles located near the periphery of the aeroshell face combines both the thrust force and augmented drag force to decelerate larger mass vehicles for Mars EDL while requiring less propellant mass than central nozzle configurations, which eliminate drag. These flow field interactions have been investigated in previous literature using both wind tunnel tests and CFD simulations for a limited set of vehicle configurations and flow parameters. This work describes the development of a drag model characterizing the drag augmentation capabilities of SRP flow interactions. The model combines empirical results from CFD simulations available in previous literature with quasi-one-dimensional flow to estimate the drag coefficient of a 70° sphere-cone aeroshell as a function of free stream Mach number. While CFD simulations and wind tunnel tests are essential for capturing all relevant flow physics to fully characterize the aerodynamic-propulsive interactions for SRP flows, they are time and labor intensive. The model developed in this work was designed to capture the dominant flow physics of pressure conservation through various shock cascade structures. The resulting model can run on the order of minutes with minimal user input. This allows conceptual level systems analysis to be performed across a wide range of input parameters beyond the current parameter space that has been evaluated through test or CFD simulations for drag-augmented SRP. The drag coefficient model developed was validated against previous results from wind tunnel tests and CFD simulations available in literature to within 10% for Mach numbers greater than 2. In addition, the sensitivity of the computed drag coefficient to the empirically determined model inputs was analyzed. A study of drag-augmented SRP operation concepts for use in Mars EDL is also presented. The feasible entry and payload masses for ballistic entries were determined for a range of SRP operation intervals. The maximum dry mass of the EDL vehicle was increased by 34% over the ballistic reference trajectory by using drag-augmented SRP. The effects of acceleration limits and the constraints of real engines were also examined. The trajectory analyses performed illustrate the flight regimes which drag-augmented SRP is most effective. In addition, the feasibility of two concepts combining supersonic decelerator concepts was investigated: a combination of drag-augmented SRP and high-thrust SRP, and a combination of drag-augmented SRP and inflatable aerodynamic

decelerators (IADs). These ‘hybrid’ solutions have the potential to combine some of the benefits of each technology by using them in the appropriate flight regime.

Clustered Performance of Conical Ribbon Drogue Parachutes in Support of Earth Re-Entry Missions

Anita Sengupta, Brooke Goree, Ed White, Gary Bourland, Jose Laguna, Robert Sinclair, Elsa Hennings, Richardo Machin, John Guthery

Since the recent retirement of the Space Shuttle, the American space program has been without a vehicle to launch astronauts into space. Ushering in a new era of spacecraft technology is NASA's Orion Multi-Purpose Crew Vehicle, the newest spacecraft to be developed for human spaceflight. Designed to carry astronauts beyond Low Earth Orbit (LEO), the Orion Vehicle features multiple parachute assemblies for reentry, known as the CEV (Crew Entry Vehicle) Parachute Assembly System (CPAS). The reentry phase of the vehicle's flight includes two drogue parachutes, two pilot parachutes, and three main parachutes. The drogue chutes' primary objectives are to begin the initial deceleration of the command module, as well as to orient the vehicle so that it descends to the earth's surface with the crew in an upright position. Ten percent of full-scale Orion conical ribbon drogue parachutes in the wake of a 10% command module were tested in a 3x2.1 m (10'x7') subsonic wind tunnel. The motivation behind the test program was to provide a cost-efficient aerodynamic performance database that captured the "wake-effect" of the command module on the parachute for Earth Re-entry missions. The subscale parachutes are geometrically scaled to the full-scale configuration in terms of geometric porosity, number of gores and ribbons, and trailing distance. The program explored the aerodynamic parametric space including Command Module angle of attack, Reynolds number, and dynamic pressure. The test program also allowed comparison of the performance of various parachute configurations including reefing and clustering. Non-intrusive diagnostics were utilized including a three-axis load cell, force balance, high speed imaging, and particle image velocimetry. Data to be presented includes average and time varying drag and parachute shape of the clustered configurations including mixed reefing stages to indicate the various failure scenarios.

IMPACT FOAM TESTING FOR MULTI-MISSION EARTH ENTRY VEHICLE APPLICATIONS

Lou Glaab, James Hawbaker, Parul Agrawal

Multi-Mission Earth Entry Vehicles (MMEEVs) are blunt-body vehicles designed with the purpose of transporting payloads from outer space to the surface of the Earth. To achieve high-reliability and minimum weight, MMEEVs avoid use of limited-reliability systems, such as parachutes and retro-rockets, instead using built-in impact attenuators to absorb energy remaining at impact to meet landing loads requirements. The Multi-Mission Systems Analysis for Planetary Entry (M-SAPE) parametric design tool is used to facilitate the design of MMEEVs and develop the trade space. Testing was conducted to characterize the material properties of several candidate impact foam attenuators to enhance M-SAPE analysis. In the current effort, two different Rohacell foams were tested to determine their thermal conductivity in support of MMEEV design applications. These applications include thermal insulation during atmospheric entry, impact attenuation, and post-impact thermal insulation in support of thermal soak analysis. Results indicate that for these closed-cell foams, the effect of impact is limited on thermal conductivity due to the venting of the virgin material gas and subsequent ambient air replacement. Results also indicate that the effect of foam temperature is significant compared to data suggested by manufacturer's specifications.

Part II: Poster

Session: Airless Bodies

In Situ Water Extraction System for Asteroids and Comets

Kris Zacny

In-Situ Resource Utilization (ISRU) facilitates planetary exploration by drawing needed resources, such as water, from the local environment. Many approaches to extracting water from frozen soil follow a 'terrestrial' mining approach; they consist of mining ice or ice-bearing soil, transferring the feedstock to a water recovery plant, and then extracting and storing of water. The Mobile In situ Water Extractor (MISWE) eliminates the transfer or processing of feedstock steps and in turn simplifies the water extraction process. The MISWE approach is to integrate both the mining and the water extraction systems into a single unit, integrated with the drill. The water extraction process follows three steps: 1) mining the soil using deep fluted auger, 2) extracting the water from soil within the flutes, and 3) discarding the soil. Hence only the water is transported back, while dry soil is left behind. A single MISWE reactor consists of the Icy-Soil Acquisition and Delivery System (ISADS), and the Volatiles Extraction and Capture System (VECS). The ISADS is a deep fluted auger that drills into the ice or icy-soils and retains material on its flutes. Upon material acquisition, the ISADS is retracted into VECS and sealed. The VECS consists of a cylindrical heat exchanger and volatiles transfer system (a reactor). The material on the deep flutes is initially heated and resultant water vapor is bled into a water collection canister by a one way valve where it condenses. The heat from the water collection canister can be circulated back into the reactor via a heat exchanger. After water extraction is complete the ISADS is lowered towards the ground and spun at high speed ejecting the dry soil via centrifugal action. At the same time, the collected water is pumped from the canister into a holding tank. The MISWE then moves to the next location and the operation is repeated. Once the water tanks are full, the spacecraft is launched back towards Earth or other destination. Since the regolith is not actually transferred, there is no need for a transfer system and associated mechanisms. Also, if a spacecraft is powered using Radioisotope Thermal Generators (RTG) or more efficient the Advanced Stirling Radioisotope Generator (ASRG), the heat generated by the unit can be transferred to the reactor. For example, the RTG on the 2011 MSL Curiosity rover generates ~120 Watts electrical power and > 1 kW heat. To determine feasibility of this water extraction approach, approximately 50 tests were completed in vacuum chamber. The MISWE breadboard, in the optimum configuration, has been demonstrated to extract 19 grams of water from icy soil. The water extraction efficiency was 92% and the remaining 8% of water was lost and never captured. The power usage during the 30 minute process was 34 Watts. This translates to the energy usage of 17 Wh or 0.9 Wh/gram of water, or ~80% energy efficiency. During the tests it was observed that a soil temperature can be used to monitor the drying cycle. Once the temperature starts increasing it indicates that the soil is dry and the heat is no longer absorbed by the water sublimation process. Hence the heating process can be terminated making the extraction system more efficient. To make the process even more efficient, the power and duration of the applied heat and dwell time after the heating cycle could also be varied. A MISWE reactor with a 1 meter long and 20 cm diameter auger will be able to

recover ~3 kg of water every 40 minutes from regolith with ~10 wt% water. Assuming that it takes another 20 minutes for the remaining tasks (drilling to 1 meter depth, discarding of dry soil), a total of 3 kg of water can be recovered every hour. Thus the mass of water that the Asteroid MISWE system with 8 reactors can recover per hours is 8 x 3 kg per hour = 24 kg per hour or 16 tons per month. At a very conservative cost of water at EML1 of $10K/kg, this translated to a value of extracted water of $160M.

Suborbital Experiment to Observe Gentle Collisions of Dust Particles in Microgravity

Brad Hoover, J.E. Colwell, A. Dove, J. Howard, R. Cannon, H. Nagler

Understanding the dynamics of dust particle aggregation in deep space is an important factor in the success of missions that aim to land on small solar system bodies. These missions must be cautious when disturbing loose regolith upon landing as mechanical issues can arise. This poster presents the development and ground testing of an experiment that will observe gentle collisions of the lunar regolith simulant JSC-1 in microgravity. Ground testing was performed with a drop tower providing approximately 0.75 seconds of a microgravity environment. Particle collisions are encouraged by a vibrating motor which will pulse at given intervals throughout the duration of the microgravity environment. Following sufficient drop tower testing, this experiment will use a suborbital flight to achieve a longer duration of microgravity. We aim to observe the conditions that lead to the formation of particle aggregates. Future iterations of this experiment should be performed in order to better constrain the conditions that lead to particles forming agglomerates. To more accurately represent an asteroid or comet surface environment, this experiment can be depressurized to a vacuum in the future. This experiment could also be expanded to include multiple regolith simulants.

Penetration of Cryogenic Water Ice using Ballistic Impact

Lee W. Kohlman, Vikas Prakash, J. Michael Pereira, Kelly S. Carney, Duane M. Revilock, Charles R. Ruggeri, Charles Lawrence, Xianqian Wu

A potential method for placing instrumented probes beneath the icy surfaces of planets and moons is by moderate speed ballistic impact. Successful insertion and data acquisition will require a good understanding of the mechanics of the penetration process. In some cases, such as with Jupiter's moon Europa, the surface to be penetrated is water ice at very low temperature. The ice on Europa is estimated to be between 50K at the poles and 110K at the equator. Physics-based and empirical models of the penetration process are extremely useful in developing optimum penetrator designs, understanding the expected accelerations and loads on the penetrator, and designing the penetrator for desired penetration depth. In this paper, we describe ongoing research to develop accurate models for penetration of cryogenic water ice. Because ice on most planetary surfaces exists at ambient temperatures far below 263 K, the approach focuses on the high strain rate as well as shock response of low pressure phase, ice Ih, at 250 K, 150 K and 100 K temperatures. These studies will enable critical information of temperature effects on dynamic strength, shock behavior, and failure of Europa (equator) ice to be determined. These experimental data will be then used to perform computer simulations using finite element analysis to compare with the results of ballistic experiments of penetrators into cryogenic ice. The first component of this research focuses on the high strain rate, temperature dependent response of water ice relevant to penetrating impact. High strain rate compression and fracture experiments were previously performed at temperatures of -10C and -30C and the results of various studies in the literature have documented different aspects of the high rate ice behavior that are related to the constitutive behavior of ice. In this study, the dynamic strength properties of ice at cryogenic temperatures will be obtained using a modified Split Hopkinson compression bar. Dynamic material parameters of interest are the material dynamic yield strength, strain hardening, strain-rate sensitivity, failure stress/strain levels, and unloading behavior of the fragmented/pulverized ice samples. To obtain the shock properties of ice, an 84 mm single stage gas gun flyer-plate impact facility will be employed. Ice disks approximately 30 mm in diameter and 3-4 mm in thickness will be grown. The ice disk targets will be impacted by planar 40-mm diameter polycarbonate or metal (aluminum or steel) flyer plates, launched via compressed air. The data collected will yield the shock states (stress versus volume relationship), key shock parameters, as well as the spall strength as a function of ice temperature. The second component of this research involves ballistic impact of penetrators with varying mass, density and nose shape, and with varying velocity into cryogenic water ice targets. Measurements will be made of the projectile attitude before and after impact as well as the penetration depth and ice fracture characteristics. This data will be used to develop empirical relationships for predicting penetration depth and for validating our computational models. The third component of this research consists of computational model development using explicit finite element analysis. The analysis incorporates deformation and failure models of ice which have been developed and implemented for use in modeling ice

impactors. Preliminary models will be used to define the parameter space and guide the constitutive and ballistic impact experimental efforts. Once temperature dependent, high rate mechanical properties are measured, updated material models will be used to model the penetrator impacts. Two analytical codes will be used, LS-Dyna and CTH. In the current presentation, details of the approach will be presented including high strain rate and shock behavior already obtained on ice at higher temperatures. In addition, results of some preliminary ballistic impact tests and computational modeling of the experiments on ice near the freezing point will be shown. Predictions of heat transfer rates and temperature distribution in ice blocks, used to design a cryogenic water ice ballistic target, will also be presented.

Collection of Regolith Under Low-Gravity, Airless Conditions

Kelly Lai, Dr. Joshua Colwell, Dr. Adrienne Dove

Asteroids are primordial material leftover from the formation of the Solar System, and as such, they can provide insight into our Solar System's past. Additionally, these low-gravity, airless bodies contain a variety of useful elements, such as hydrogen, helium, and iron. Expanding upon our knowledge and understanding of these small bodies is essential for understanding the history of our solar system, and for future planetary exploration. Due to the nature of asteroids, visitation to these small bodies generates a wide variety of technical challenges that are not typically found on larger planetary bodies. The proposed payload, the Collection of Regolith Experiment (CORE), uses hardware and technology designed to operate in a dusty regolith environment that is similar to conditions found on asteroids. The mission's objective is to test a mechanism for the collection of asteroid regolith. The payload, which is designed to operate autonomously through the use of a microcontroller, plunges a spring-loaded regolith reservoir and a prototype retrieval mechanism to sample approximately 10 grams of replica lunar regolith, JSC-1. CORE has been tested in laboratory conditions at the University of Central Florida in the Center for Microgravity Research and Education (CMRE) labs. Unfortunately, the behavior of regolith varies greatly between the low-gravity conditions found on asteroids and 1-g laboratory conditions. Even the smallest upward acceleration in low gravity will stir up a cloud of regolith dust that might hamper spacecraft operations. CORE addresses these technical challenges and is slated to fly on a suborbital rocket to provide gravity conditions more similar to those that would be encountered at an asteroid.

Uranian Moon Investigation with "Hitchhiker" Small Spacecraft

Jeremy Straub, Tyler Hill, Joshua Berk, Nicholas Long

This paper presents a mission concept for exploring the Uranian moon system utilizing two 6-U (30 cm x 20 cm x 10 cm, 8 kg) CubeSat spacecraft which will be deployed from and relay their communications to Earth through a primary spacecraft. This primary spacecraft, part of the Gap-Based Expanding Outer Region Geophysical Explorer (GEORGE) Mission, will initially work in concert with the 6-U CubeSats to create a gravity map of Uranus. Once this is completed, the spacecraft will re-dock so that the primary craft can move the two 6-U CubeSats to their orbit around Miranda to perform imaging and gravity field characterization of this moon. Umbriel and Oberon will receive a similar investigation by the two CubeSats, should resource and craft longevity allow. This paper presents a complete overview of the Miranda exploration mission (and very similar secondary Umbriel and Oberon exploration missions). A complete overview of the 6-U craft is provided including the limited onboard propulsion capabilities, the craft's attitude determination and control system, the powerful onboard computing system and a deployable solar panel configuration that will be utilized to allow the level of power generation required for operations at this distance. Mass, volume, power and communications budgets are presented and discussed. The paper pays particular attention to how, as a "hitchhiker" the spacecraft must be particularly concerned about optimizing its data transmission needs, as transmission capabilities from the primary craft to Earth must be shared between the primary and "hitchhiker" missions. The paper discusses how Model Based Transmission Reduction (MBTR) is utilized to maximize the science-value of each byte of data transmission and how MBTR is utilized to combine (on a scientific-value-maximization priority basis) data from the two spacecraft. A mission operations plan and mission data operations plan are also presented and discussed. The paper concludes by assessing the value that could prospectively be generated by this proposed mission, both in terms of the understanding gained about the Uranian moons and system and in terms of the engineering demonstration benefits of the mission.

Session: Cross cutting Technologies I and II

EntrySat : A 3U CubeSat to study the atmospheric re-entry environment

Fabien Apper, Quentin Nénon, Kevin Reilley, Laura-Sofia Urbina, David Mimoun, Alain Gaboriaud, Frédéric Sourgen, Jean-Luc Vérant, and the EntrySat team

QB50 is a constellation of 50 Cubesats planned to be launched in mid-april 2015 into a circular orbit at 320 km, inclination 98,6°. This project was created by the Von Karman Institute for Fluid Dynamics in Brussels, Belgium, to fulfill both scientific and educational purposes. ISAE's EntrySat student satellite is one of the 10 QB50 In-Orbit Demonstration (IOD) satellites. As in-orbit collisions and satellite orbital decay have demonstrated that orbital debris represents a potential threat to access to space as well as a threat to ground safety, the EntrySat experiment, with CNES and ONERA (a famous French aerospace lab) as partners, consists of inserting the EntrySat satellite into low-Earth orbit (which is similar in principle to secondary debris typically issued from launch vehicles or satellites). A science module operating during the reentry phase will be able to perform in-situ measurements of the CubeSat environment as well as integrity up to its destruction. The scientific objectives have been defined in collaboration with ONERA and are the following : to refine the atmospheric modeling between 120 and 350 km and to refine the aerodynamic coefficients modeling. Aerodynamic forces and associated kinematics (position, attitude motions) should be available and EntrySat will perform these measurements during orbit decay phase and destruction phase. As a result of the scientific goals of EntrySat, the payload includes an Inertial Measurement Unit (IMU) with magnetometers, gyroscopes and gyrometers to determine the euler angles and their derivatives. A GPS chip is integrated in the satellite to measure the position and the velocity of EntrySat. Six washer type-k thermocouples are in the center of each of the six faces of the Cubesat to give an access to the external temperature. The thermocouple position is crucial to avoid side effects. The understanding of the drag force during the mission is also a scientific goal and six piezoelectric force sensors are therefore used in addition to accelerometers. Six heat flux sensors complete the in-situ measurements. One of the biggest challenges of EntrySat is to successfully send measurement data during re-entry. The use of a UHF/VHF transmission with a ground station is not compliant with this objective as the CubeSat will be at a very low altitude during its re-entry (about ninety kilometers). Thus, we studied and chose to transmite data through the Iridium constellation. An Iridium SBD is therefore embedded in the satellite. For the ADCS, a study was conducted to determine the feasibility of a satellite attitude fine-control system using the interaction of the Earth's magnetic field with current-carrying coils to produce torque. The approximate intensity of the Earth's magnetic field was determined as a function of the satellite coordinates. Two systems for two-axis attitude control emerged from this study, one using three coils and the other using two coils. A magnetic controller was implemented in the equations of motion of the EntrySat satellite. Magnetic control laws were developed to bring the EntrySat to the desired equilibrium. This was accomplished by using a PID controller. The settling time to obtain a misalignment error of less than 3° is around 8 orbits which represents 1% of the total time of use of orbit. The satellite will be in several different modes during the whole mission, the most important ones

will be the orbit mode in which measurements will be carried out with a low frequency and only UHF/VHF transmission will be used and the entry mode in which the measurement frequency will be high and transmission will be done using the Iridium constellation. The software architecture takes into accounts those modes. It is composed by : -a main program block realizing a state machine -a communication block managing the UHF/VHF and Iridium communications -a measurements block carrying out the measurements -a housekeeping block assuring the satellite integrity -an ADCS block This architecture will be designed and implemented using TASTE. TASTE is a set of freely-available tools dedicated to the development of embedded, real-time systems. It was developed by the European Space Agency (ESA), together with a set of partners from the space industry. TASTE allows software designers to easily integrate heterogeneous pieces of code produced either manually (in C or Ada) or automatically by external modeling tools such as MATLAB Simulink, SCADE, or Real-Time Developer Studio. The deadlines for this project are the PDR on the 29th of March 2013, the CDR on the 1st of November 2013, the FRR on the 27th of February 2015 and the launch in mid-April 2015. Concerning the overall development strategy, an electrical breadboard will be realized and tested for the CDR. Afterwards a flight model will be built, tested and delivered in the beginning of January 2015.

HEART Aerothermodynamic Environment Challenges

Alireza Mazaheri, Brian Hollis, Shann Rufer, Walter Bruce, Nathaniel Mesick, Adam Wise, Stephen Hughes

Hypersonic inflatable technology is being developed at the NASA Langley Research Center for application on future hypersonic planetary entry missions. An accurate definition and prediction of the entry aerothermodynamic environment plays a significant role in success of the development of these flexible structures. The flight aeroheating environments that are predicted using the high fidelity computational reacting tools need to be properly correlated to the ground testing environments. Both arc-jet and conventional wind tunnel are needed to assess the computational predictions, and for thermal protection system development, selection and sizing. However, there are challenges in defining the flight aerothermodynamic environment on various portions of a flexible structure vehicle as a true outer mold line of the vehicle cannot be fully determined. A solid structure wind tunnel model must be appropriately scaled such that it could sufficiently represent the flight environment of the full scale flexible inflatable vehicle at different hypersonic regimes. For instance, proper geometrical scaling is needed in wind tunnel testing to study the outer mold line undulation effects on surface heating and pressure such that the flight boundary layer thicknesses are appropriately represented by the corresponding wind tunnel values. Afterbody heating is also very important as a requirement to protect the afterbody of the vehicle with a thermal protection system could further complicate the design and/or increase the overall mass of the system. Wind tunnel data are used to provide uncertainty associated with the computational data that are predicted at flight conditions. Any presence of an ill-defined uncertainty in flight environment definition will be directly transferred to ground testing that is needed for thermal protection system development. Arc-jet testing is required to develop and demonstrate the performance of new flexible thermal protection system over a range of flight relevant conditions. Arc-jet facilities are the only facilities that can provide flight relevant aerothermal loads (heat flux, pressure, enthalpy, and shear) for flexible systems for durations sufficient to test the total flight heat load. One of the major difficulties with arc-jet testing is determining the appropriate arc-jet conditions and testing configuration to sufficiently approximate the flight loads and then verifying that the arc-jet facility is providing the appropriate test conditions. In this paper, the above mentioned challenges including the current state of the analyses, modeling and testing that were performed for the support of the High Energy Atmospheric Re-entry Test (HEART) vehicle are addressed. HEART is a planned demonstration vehicle using a deployable aeroshell comprised of an inflatable with a flexible TPS designed to reenter the Earth's atmosphere from the International Space Station.

Sources of Entry, Descent, and Landing Information

Elmain Martinez, Julie Foster, Robert Powers

This poster will identify the various sources for Entry, Descent, and Landing (EDL) information and will expound on the types of material available in the NASA EDL-Repository (EDL-R). An overview of where the EDL-R is today, plans for the future, and how it fits into NASA's Knowledge Management vision will also be described. The poster will conclude with suggestions for contributing to NASA's efforts to preserve technical information.

The Legacy of EDL Information

Julie Foster, Elmain Martinez, Robert Powers

The aging of the NASA workforce and the potential loss of their implicit knowledge is well documented. In addition, spaceflight projects do not always place a priority on retaining explicit knowledge (e.g., technical memos, test reports) for reuse. Planetary entry, descent, and landing (EDL) engineering is a burgeoning discipline that is key to future human and robotic missions, but the technology advances have not been effectively documented and made accessible. This paper will discuss demographic, institutional, and other factors affecting the loss or inaccessibility of critical knowledge, and the innovative processes and tools JPL has developed to address the problem. Current JPL and NASA efforts to ensure that vital knowledge and information is preserved and retrievable include the EDL Repository (EDL-R) and a strategic plan for knowledge management. This paper is intended to spur discussion among members of the EDL community -- and among other engineers, technologists, scientists, and mission designers specializing in planetary exploration -- on how best to preserve our intellectual capital for future missions.

Entry Vehicle Technology: Aerothermal Analysis of Trajectory Environments II

Jose Aliaga-Caro, Kerry Zarchi, Steven A. Sepka

The Entry Vehicle Technology (EVT) project is developing an integrated tool to provide design parameters (such as payload mass, payload size, vehicle nose radius and TPS sizing) and aerothermal heating estimates for Earth Entry Vehicles (EEV). This poster focuses on the computational fluid dynamic (CFD) aerothermal analyses for such missions. The goal for this analysis is to provide heating estimates for the entire probe surface that could be further used for thermal soak analysis (Agrawal, et al., AIAA 2012-3010) and to establish correlation coefficients for aerodynamic heating based on the Sutton Graves equation for the vehicle that could be used for rapid design trades. A trajectory space consisting of over 2700 cases was created. From these, a representative set of trajectories for the CFD analyses was selected based on the statistical average and standard deviations of peak heating and heat load. Entry velocities for these trajectories range from 10 to 16 km/s. The range of ballistic coefficients is from 42 to 129 kg/m2 and the range of entry flight path angles is from -5 to -25 degrees. The Data Parallel Line Relaxation (DPLR) CFD code was used for the analyses (Wright, et al., AIAA Journal, Vol. 36, No. 9, 1998, pp. 1603-1609). It is a parallel, structured multi-block, finite volume code that solves the reacting Navier-Stokes equations for continuum flow, including finite rate chemistry and thermal non-equilibrium. CFD predicts a higher heat flux at the stagnation point than the Sutton Graves predictions and a correlation coefficient was established based on the higher fidelity CFD analyses using DPLR software. The effect of shoulder radius on maximum shoulder heat flux was also assessed at peak heating for one trajectory at Mach 30. A quick study varying shoulder radius while keeping nose radius and vehicle diameter constant shows a power law correlation between the maximum shoulder heat flux and non-dimensional shoulder radius, with decreasing heat flux corresponding to increasing shoulder radius.

Determination of the Deformed Structural Shape of HIADs from Photogrammetric Wind Tunnel Data

Cole D. Kazemba, Alan M. Cassell, Laura K. Kushner, Kevin Tran, Lin Li, John W. Van Norman, Justin D. Littell, Stephen J. Hughes, F. McNeil Cheatwood

The development of Hypersonic Inflatable Aerodynamic Decelerators (HIADs) is one of the primary means by which NASA is attempting to expand its science and exploration capabilities. The aerodynamic and structural characterization of these vehicles is a key aspect of their development. The first round of wind tunnel tests conducted in the National Full-Scale Aerodynamics Complex 40 x 80 foot wind tunnel at NASA Ames Research Center has provided a rich set of 3D photogrammetric data from which much of the characterization will derive. Reconstructing the structural shape from this data will correlate the deflected shape of the vehicle to angle of attack, freestream dynamic pressure, and HIAD inflation pressure. These correlations can then be fed to aerodynamic and structural computational models and lead to an improved understanding of relevant fluid-structure interactions and modeling capabilities. This work presents a methodology for data acquisition and post-processing, solid modeling of HIADs from the acquired data, decoupled fluid-structure interaction analysis and characterization of the deformed shape using 3D photogrammetric data obtained during the first series of wind tunnel tests of 3 and 6 meter test articles.

The Status of Spacecraft Bus and Platform Technology Development under the NASA ISPT Program

Lou Glaab, David Anderson, Eric Pencil, Michelle Munk, Todd Peterson, John Dankanich

The In-Space Propulsion Technology (ISPT) program is developing spacecraft bus and platform technologies that will enable or enhance NASA robotic science missions. The ISPT program is currently developing technology in four areas that include Propulsion System Technologies (electric and chemical), Entry Vehicle Technologies (aerocapture and Earth entry vehicles), Spacecraft Bus and Sample Return Propulsion Technologies (components and ascent vehicles), and Systems/Mission Analysis. Three technologies are ready for near-term flight infusion: 1) the high-temperature Advanced Material Bipropellant Rocket (AMBR) engine providing higher performance; 2) NASA's Evolutionary Xenon Thruster (NEXT) ion propulsion system, a 0.6-7 kW throttle-able gridded ion system; and 3) Aerocapture technology development with investments in a family of thermal protection system (TPS) materials and structures; guidance, navigation, and control (GN&C) models of blunt-body rigid aeroshells; and aerothermal effect models. Two component technologies being developed with flight infusion in mind are the Advanced Xenon Flow Control System and ultra-lightweight propellant tank technologies. Future directions for ISPT are technologies that relate to sample return missions and other spacecraft bus technology needs like: 1) Mars Ascent Vehicles (MAV); 2) multi-mission technologies for Earth Entry Vehicles (MMEEV); and 3) electric propulsion. These technologies are more vehicle and mission-focused, and present a different set of technology development and infusion steps beyond those previously implemented. The Systems/Mission Analysis area is focused on developing tools and assessing the application of propulsion and spacecraft bus technologies to a wide variety of mission concepts. These in-space propulsion technologies are applicable, and potentially enabling for future NASA Discovery, New Frontiers, and sample return missions currently under consideration, as well as having broad applicability to potential Flagship missions. This paper provides a brief overview of the ISPT program, describing the development status and technology infusion readiness of in-space propulsion technologies in the areas of electric propulsion, Aerocapture, Earth entry vehicles, propulsion components, Mars ascent vehicle, and mission/systems analysis.

Development Of FIAT-Based Parametric Thermal Protection System Mass Estimating Relationships For NASA's Multi-Mission Earth Entry Concept

Steven Sepka, Kerry Zarchi, Robert Madock, Jamshid Samareh

Part of NASA's In-Space Propulsion Technology (ISPT) program is the development of the tradespace to support the design of a family of multi-mission Earth Entry Vehicles (MMEEV) to meet a wide range of mission requirements. An integrated tool called the "Multi Mission System Analysis for Planetary Entry Descent and Landing" or M-SAPE tool is being developed as part of Entry Vehicle Technology project under In-Space Technology program. The analysis and design of an Earth Entry Vehicle (EEV) is multidisciplinary in nature, requiring the application many disciplines. Part of M-SAPE's application required the development of parametric mass estimating relationships (MERs) to determine the vehicle's required Thermal Protection System (TPS) for safe Earth entry. For this analysis, the heat shield was assumed to be made of a constant thickness TPS. This resulting MERs will then be used to determine the pre-flight mass of the TPS. Two MERs have been developed for the vehicle forebaody. One MER was developed for PICA and the other consisting of Carbon Phenolic atop an Advanced Carbon-Carbon composition. For the the backshell, MERs have been developed for SIRCA, Acusil II, and LI-900. How these MERs were developed, the resulting equations, model limitations, and model accuracy are discussed in this poster

Influence of Astronomical Phenomena on Entry, Descent and Landing Sequence on Mars

Dusan Marceta, Stevo Segan, Bosko Rasuo

Due to the high eccentricity of Martian orbit, which is almost six times larger than the Earth's, there are large differences in energy received from the Sun throughout a Martian year. This difference between the perihelion and aphelion on Mars orbit is more than 45%, comparing to less than 7% for the Earth. This big difference and near coincidence of southern summer solstice with the orbital perihelion are the most important causes of very dynamic behavior of the atmosphere of Mars. Beside this, the solar activity and the amount and distribution of suspended dust also have significant influence on the variability of the Martian atmosphere. In the last four decades since the Vikings' landings, significant climatic changes on Mars have been observed. These changes have caused cooling of the atmosphere which has also induced changes of vertical density profiles, especially in the region between 20 km and 50 km of height, where almost all kinetic energy of an entry vehicle is dissipated. These astronomical phenomena, separately or together, can significantly alter not only the level of some atmospheric parameter but also the characteristics of its diurnal and seasonal cycle. Considering that entry, descent and landing (EDL) sequence is mainly driven by the density profile of the atmosphere and aerodynamic characteristic of the entry vehicle, it is very important to analyze EDL scenarios under different atmospheric conditions. We performed the analysis of the influence of the mentioned phenomena on the EDL profiles by using Mars Global Reference Atmospheric Model (Mars-GRAM) and European Mars Climate Database (EMCD). The obtained results show that the influence of these phenomena can alter possible landing site elevation for several kilometers. This fact could be taken into account in planning landings on the higher elevations on Mars, mainly in the southern hemisphere.

Tension Measurements of Hypersonic Inflatable Aerodynamic Decelerator Structural Straps under Static and Dynamic Loading

Greg Swanson, Alan Cassell, Keith Johnson, Steve Hughes, Anthony Calomino, Neil Cheatwood

To support NASA's goal of landing humans on Mars, development of technologies to facilitate the landing of heavy payloads are being explored. Current entry, decent, and landing technologies are not practical when utilizing these heavy payloads due to mass and volume constraints dictated by limitations imposed by current launch vehicle fairings. Therefore past and present technologies are now being explored to provide a mass and volume efficient solution to atmospheric entry, including Hypersonic Inflatable Aerodynamic Decelerators (HIADs). In the summer of 2012, a 3 meter diameter stacked-torus HIAD was successfully tested in the National Full-Scale Aerodynamics Complex's 40ft by 80ft wind tunnel at NASA Ames Research Center. During this test series the inflatable article was run in two different configurations at various combinations of torus inflation pressures, yaw angle (representing flight angle of attack), and wind tunnel dynamic pressure. The HIAD article was constructed in a stacked toroid configuration using inflatable torus segments composed of fiber-reinforced thin films, which were joined together with high strength textile woven structural straps to distribute the aerodynamic load throughout the inflatable structure. To better characterize the HIAD article's structural performance and provide key data to validate computational models, structural strap load measurement techniques were researched, developed, and implemented during static load testing and aerodynamic wind tunnel testing. This measurement system consisted of 21 in-line strap tension loadcells that were integrated with both configurations. In this paper, both 3 meter configurations will be described along with the static load and wind tunnel test setups. Additionally, the custom structural strap measurement system will be described. Preliminary results from the static load testing and the wind tunnel testing for both configurations of the 3 meter HIAD will be presented.

Classification of observed atmospheric trajectories and prediction of impact consequences

Daria Kuznetsova, Maria Gritsevich

In this study, we analyze different observed atmospheric trajectories as well as the consequences that can accompany collisions of cosmic bodies with planetary atmosphere and planetary surface. Using the basic differential equations [1] one can introduce dimensionless parameters describing a problem. Then we distinguish three key dimensionless parameters with the following physical meaning: (1) the ballistic coefficient, which shows the ratio between the mass of the atmospheric column along the trajectory and the body's pre-entry mass; (2) the mass loss parameter, which is proportional to the ratio between the initial kinetic energy of the body and energy which is required to insure total mass loss of the body due to ablation and fragmentation; (3) the shape change coefficient, which describes possible rotation of the body along the trajectory. These parameters explicitly characterize the ability of an entering body to survive during atmospheric entry and to reach the ground. The processes accompanying atmospheric entry depend essentially on values of these parameters as well. Thus different events could be associated with different groups with similar predictable consequences. The ballistic coefficient, mass loss parameter and shape change coefficient can be derived by several existing techniques, see e.g. [2, 3]. Therefore we analyze the consequences for different types of events, for example, formation of a huge single crater (Barringer, Lonar Lake) due to meteoroid impact, dispersion of craters and meteorites over a large area (Sikhote-Alin), absence of craters and meteorites, but huge damage (Tunguska). The proposed scale helps summarizing the data on the existing impacts and to formulate the recommendations for further studies valuable for planetary defense. References 1. Stulov, V.P. Interactions of Space Bodies with Atmospheres of Planets. // Applied Mechanics Reviews, 50, pp. 671-688, 1997. 2. Gritsevich, M.I. Determination of Parameters of Meteor Bodies Based on Flight Observational Data. // Advances in Space Research, 44 (3), pp. 323–334, 2009. 3. Gritsevich, M., and Koschny, D. Constraining the Luminous Efficiency of Meteors. // Icarus, 212(2), pp. 877-884, 2011.

Post-Flight Evaluation of PICA and PICA-X - Comparisons of the Stardust SRC and the Dragon 1 Forebody Heatshield Materials

Mairead Stackpoole, David Kao, Vince Qu

Phenolic Impregnated Carbon Ablator (PICA) was developed at NASA Ames Research Center. As a thermal protection material, PICA has the advantages of being able to withstand high heat fluxes with a relatively low density. This ablative material was used as the forebody heat shield material for the Stardust sample return capsule, which re-entered the Earth's atmosphere in 2006. Based on PICA, SpaceX developed a variant, PICA-X, and used it as the heat shield material for its Dragon spacecraft, which successfully orbited the Earth and re-entered the atmosphere during the COTS Demo Flight 1 in 2010. Post-flight analysis was previously performed on the Stardust PICA heat shield material. Similarly, a near-stagnation core was obtained from the post-flight Dragon 1 heat shield, which was retrieved from the Pacific Ocean. Materials testing and analyses were performed on the core to evaluate its ablation performance and post-flight properties. Comparisons between PICA and PICA-X are made where applicable. Stardust and Dragon offer rare opportunities to evaluate materials post-flight - this data is beneficial in understanding material performance and also improves modeling capabilities.

Stagnation-Line Simulations of Planetary Reentry Probe with Surface Ablation

Alessandro Turchi, Bernd Helber, Alessandro Munafò, James B. Scoggins, Thierry Magin

During the past 40 years, space agencies have sent landers to the Moon, Mars, and Venus, as well as atmospheric probes to Jupiter, Venus, and Titan. When entering a planet atmosphere, these probes need to be protected from the extremely harsh conditions characterized by high temperature and high heat flux. Thermal protection system (TPS) materials are used to build the heat shield that protects the vital structure from the severe heating encountered during hypersonic flight through a planetary atmosphere. The TPS is a single point-of-failure subsystem. It is critical and its performance needs to be validated through ground test and analysis. For present and future space exploration missions, TPS can be mission enabling, significantly impacting the launch mass or science and instrumentation payload. Therefore, it is of fundamental importance to advance the development, design, and analysis of TPS materials. In order to better understand the behavior of these kinds of materials, a strong effort started at the von Karman Institute for Fluid Dynamics (VKI) over the past years to advance the fundamental knowledge of ablation phenomena. Both, experimental and numerical studies have been performed to investigate the flow-material interaction. Ablation tests of several materials, from pure graphite to newer low-density pyrolyzing materials, have been carried out in the VKI Plasmatron facility in air plasmas. A new thorough library for the evaluation of thermal and transport properties of gas mixtures and the resolution of chemical equilibrium by means of a new reliable procedure has been implemented. More recently, a stagnation-line code, able to reproduce the flow along the stagnation line of spherical and cylindrical bodies by solving the dimensionally reduced Navier-Stokes equations, has been developed in order to be used in the investigation of reentry-related phenomena. Aim of the present work is to reproduce Plasmatron experimental test cases using a new-developed numerical procedure that allows the simulation of the gas-surface interaction over the TPS surface. As a first step in developing an high-fidelity solver of the flow-material coupled problem, this work deals with the development of an ablation modeling procedure and its implementation in the CFD stagnation-line code. The approach relies on the surface energy and mass balances that determine a mutually dependent boundary condition for the solution of both the external flow and the TPS response to the aero-thermal loads. Since no coupling with a material flow solver is performed at this stage, the steady-state ablation approximation is used in the analysis. By this approximation the solid heat conduction term in the surface energy balance can be approximated by means of its steady-state value and the steady-state ablation rate of the TPS can be evaluated as a solution of the CFD simulation. The concentrations of all surface chemical species are determined from gas-surface chemical equilibrium calculations. This permits to bypass all of the classical approximations such as heat and mass transfer coefficients, and blowing correction approximations which needs to be used when ablating boundary conditions are not accounted for in the CFD solution. The mass and energy surface balances, combined with a suitable ablation model, provide complete

surface thermochemistry conditions for the numerical solution of the coupled CFD/ablation problem for TPS analysis. As it focuses the attention on the most solicited point of the heat shield, the stagnation-line approach look promising and particularly useful in the development phase of the heat shields. It allows for quick calculations, by putting aside the flow complexity due to the body geometries, normally computationally demanding, but retains all the advantages deriving from the accurate resolution of the flow-material thermochemical interaction. Results obtained using different air chemistry model and heating conditions will be shown in the paper. A thorough comparison with in-house experimental data and with fully-dimensional calculations available in the literature will be performed in order to asses guidelines for future developments of this simplified approach.

Modeling of gas-surface interactions and assessment of their sensitivity to flowfield composition

Abhilasha Anna, Iain D. Boyd

The high temperatures on a hypersonic vehicle surface caused by heat loads encountered during (re-)entry through a planetary atmosphere require a reliable Thermal Protection System (TPS) that makes a good understanding of the physical and chemical processes essential for its design. Aerothermal heating environment of the vehicle TPS is defined by gas-surface interaction processes (eg. surface catalysis, surface recession, etc.). Accurate modeling of these processes is necessary to predict heat flux experienced by the re-entry vehicle precisely. The effects of gas-surface interactions for graphite exposed to high-enthalpy nitrogen flow are examined in this study. The objective of this study is to investigate and implement surface chemistry models to describe dominant gas-surface interaction processes. The effect of different gas-surface interaction processes on the species concentration and temperature gradients near the material surface, and on the heat transfer to the material surface is investigated. The gas-surface interaction processes studied are the recombination of nitrogen atoms to molecules at the surface due to catalysis and the carbon nitridation reaction where nitrogen atoms react with the surface carbon to form gaseous CN. A finite-rate surface chemistry (FRSC) model that can account for different surface reactions such as particle adsorption/desorption, the recombination of an atom of the gas with an atom adsorbed on the wall [Eley-Rideal (E-R) reaction] as well as reactions leading to surface recession, is used to study these processes. The numerical simulation is performed using the Michigan Aerothermodynamics Navier-Stokes computational fluid dynamics (CFD) code LeMANS, developed at the University of Michigan. It is a general purpose, parallel, three-dimensional code that solves the laminar Navier Stokes equations including chemical and thermal nonequilibrium effects on unstructured computational grids. Assessment of the computations is performed using experimental tests that were conducted in the 30 kW Inductively Coupled Plasma (ICP) Torch Facility at the University of Vermont. It is designed to test scaled material samples in high-enthalpy gas flows for simulation of planetary entry and Earth atmosphere re-entry trajectory heating conditions. The test article is a 25 mm diameter graphite sample that is exposed to hot nitrogen plasma in the test chamber of the ICP torch facility. The free stream conditions and wall temperature used are based on the experimental set up. The free stream conditions i.e. the ICP torch exit conditions are not well defined. Therefore, a sensitivity analysis on the ICP torch exit chemical composition is performed to evaluate its effect on the flow around the graphite sample and its mass loss. The chemical composition is calculated for different values of inlet power and temperature. The comparative analysis of the computed profiles of gas translational temperature and normalized nitrogen atom density with the experimental results is performed. Results of the variation in the heat flux to the surface and surface temperature as a result of different free stream conditions are also analyzed. The carbon mass removal due to carbon nitridation reaction is also computed and compared to the measured value. All the simulations are performed for thermochemical nonequilibrium flowfield. The results from the simulations show that the temperature in the boundary

layer is not affected by different surface reactions whereas the nitrogen atom density decreased in the boundary layer and the heat flux transferred to the surface increased with increasing values of nitrogen catalytic efficiency. The results also show that the temperature and the normalized nitrogen atom density in the boundary layer are not considerably affected by different free stream conditions. The heat flux transferred to the surface, carbon mass removal rate and the surface temperature increased considerably with increasing values of input power with constant free stream temperature as compared to the analysis where increasing values of free stream temperature and constant input power are considered.

Molecular Simulation of Hypersonic Flows

Savio Poovathingal, Chongling Zhang, Paul Norman, Paolo Valentini, Ioannis Nompelis, Thomas E. Schwartzentruber

Carbon based ablative Thermal Protection System (TPS) are widely used in the aerospace industry today. The study of ablation requires a non-equilibrium finite-rate treatment of gas-surface chemistry. There have been several attempts to model finite-rate oxidation of carbon based TPS but much uncertainty remains. In order to understand current TPS and design future TPS with desired properties, it is important to understand and characterize the dominant reactions occurring on such ablating surfaces and how these influence recession rate and heat flux. The study and modeling of ablation processes can be divided into two aspects. First, the macroscopic/mesoscopic TPS morphology (polycrystalline, fibrous, resin, etc.), the resulting surface area, and the boundary layer diffusion processes that dictate the supply of reactive species to the surface area of the TPS. Second, the dominant chemical reactions occurring on an atomistic scale locally on the surface, the chemistries of the reactive sites (crystallographic orientation and defects), as well as activation energies and steric factors associated with reactions on these surface sites. Molecular dynamics is a useful computational chemistry tool to investigate dominant surface reactions and associated activation energies at the atomistic level. In this study, large-scale molecular dynamics (MD) simulations are performed to study the oxidation of Highly Oriented Pyrolytic Graphite (HOPG) by hyperthermal atomic oxygen beam (5eV). Simulations are performed using the ReaxFF classical reactive force field. HOPG is modeled as multilayer graphene and etch pit formation and evolution is directly simulated through a large number of sequential atomic oxygen collisions. The simulations predict a cylindrical etch pit clearly observed in the experiments. The simulations demonstrate that oxidation is rapid and primarily occurs around defects in the graphene sheets. It is observed that the number of carbons atoms removed scales linearly with the number of oxygen atoms striking the etch pit. Furthermore, the simulations show an increase in oxidation as the surface temperature is increased. In further quantitative agreement with the experiments, the simulations predict the most abundant product species to be O2 (via recombination reactions), followed by CO2, with CO as the least abundant product species. While recombination occurs all over the graphene sheet, the carbon-removal reactions occur only about the edges of the etch pit. Through isolated defect analysis on small graphene models as well as trajectory analysis performed directly on the predicted etch pit, the activation energies for the dominant reaction mechanisms leading to O2, CO2, and CO product species are determined to be 0.3 eV, 0.52 eV, and 0.67 eV respectively. Overall, the qualitative and quantitative agreement between MD simulation and experiment is very promising. The MD simulation approach can be extended to study and simulate high-temperature gas interactions with graphitic materials which are relevant to the ablation of carbon based TPS.

Micro-scale DSMC Simulations of Porous TPS Materials Using High-Resolution Tomography

Eric C. Stern, Savio J. Poovathingal, Graham V. Candler, Tom E. Schwartzentruber, Nagi N. Mansour, Francesco Panerai

This poster presents progress on efforts underway at the University of Minnesota to perform micro-scale simulations of the ablation of porous carbon TPS materials. This effort attempts to build on previous work on multiscale modeling of ablation done by Lachaud et. al. where a Brownian motion technique was used to model the ablation of fibrous materials in the diffusion regime. The first section of this work will detail the experimental effort currently being undertaken at the Advanced Light Source (ALS) at Lawrence Berkeley National Laboratory (LBNL) to obtain high resolution tomographic data of various porous thermal protection system (TPS) materials. It will also outline the process by which the tomographic data is processed, and used to create triangulated surface meshes for the numerical simulations. The second section will outline the procedure with which we will model the gas-phase physics using the Direct Simulation Monte Carlo (DSMC) method. Using DSMC will enable us to capture more physical phenomena that may be important in future applications, such as convective mass-transfer, gas-phase kinetics, and higher-fidelity gas surface interaction (GSI) models. In this work, we will present results from DSMC simulations of flow through PICA preform. These simulations will use tomographic data to generate the computational surfaces. We will use the data from the simulations to determine Darcy coefficients for the flow through the porous media. Finally, we will present preliminary results from simulations with multiple gas species, as well present up-to-date progress on performing simulations with a moving (ablating) surface.

A Continuation-Krylov Method for the Computation of Multiphase Equilibria

James B. Scoggins, Benjamin Graille, Thierry Magin

A new algorithm for the robust computation of multiphase equilibria under general linear constraints is presented which is based on the single-phase Gibbs function continuation method. System solutions are obtained through a simple Krylov iterative scheme, which avoids convergence stalling due to singular systems. The method is guaranteed to converge to the correct solution even when one or more phases are depleted for any well posed problem. The merits of the algorithm are presented, including performance and accuracy, and compared with current state-of-the-art equilibrium codes such as STANJAN or CEA, which have been shown to fail to converge under certain conditions. An application to space missions is also envisaged through the design of a robust mass balance solution methodology for the coupling of ablative thermal protection system response models with flow solvers. A robust methodology for the calculation of the so-called B-prime tables is especially important for pyrolysis gas mixtures that have very stiff chemistry or for temperature and pressure ranges very different from a priori conditions.

Comparison of Failure Modes in 2-D and 3-D Woven Carbon Phenolic Systems

Grant A. Rossman, Mairead Stackpoole, Jay Feldman, Robert D. Braun

Woven Thermal Protection System (WTPS) materials are a new class of heatshields being developed for entry vehicles. Currently, there are few options for ablative entry heatshield materials, none of which is ideally suited to the planetary probe missions currently of interest to NASA. While carbon phenolic was successfully used for missions such as Pioneer Venus and Galileo (to Jupiter), the heritage constituents are no longer available. An alternate carbon phenolic would need to be qualified for probe missions, which is most efficient at heat fluxes greater than those currently of interest. Additional TPS materials such as Avcoat and PICA are not sufficiently robust for the heat fluxes required. As a result, there is a large “TPS gap” between the materials efficient at very high/extreme conditions (carbon phenolic) and those that are effective at low/moderate conditions (all others). Development of 3D Woven TPS is intended to fill this gap, targeting mid-density weaves that could with withstand mid-range heat fluxes between 1100 W/cm2 and 8000 W/cm2. Preliminary experimental studies have been performed to show the feasibility of WTPS as a future mid-range TPS material. One study performed in the mini-ARC (mARC) Jet Facility at NASA Ames Research Center characterized the performance of a 3D Woven TPS sample and compared it to 2D carbon phenolic samples at ply angles of 0°, 23.5°, and 90°. Each sample contained similar compositions of phenolic and carbon fiber volume fractions for experimental consistency. The goal of this study was to compare the performance of the TPS materials by evaluating resulting recession and failure modes. After exposing both samples to similar heat flux and pressure conditions, the 2D carbon phenolic laminate was shown to experience significant delamination between layers and further pocketing underneath separated layers. The 3D Woven TPS sample did not experience the delamination or pocketing failure modes because z-fibers in the through-thickness direction provided extra reinforcement to hold material layers together. Therefore, the benefit of using a 3D weave architecture was shown to alleviate failure modes experienced by a 2D laminate sample of similar material composition. In summary this poster reviews the thermal response performance comparisons drawn between a 3D Woven TPS sample and 2D Carbon Phenolic samples after performing rigorous heating experiments. Although the mARC facility is still in its developmental stages, researchers expect similar trends in failure modes observed from large scale arc jet facilities. This work helps demonstrate the viability of 3D Woven TPS’s as a new TPS option for future atmospheric entry missions.

Overview of the QB50 mission

Jan Thoemel, Cem Asma, Fiona Singarayar, Ruedeger Reinhard, Isil Sakraker, Thorsten Scholz, Gilles Bailet, Jean-Marie Muylaert

The QB50 mission is presented. It consists of in total 50 cubesats, which are deployed from their mothership in Low Earth Orbit at an altitude of approximately 320 km. The expected mission duration is limited by the natural decay of the nanosatellites and is predicted to be approximately to be 3-4 months. 40 of the QB50 cubesats will be Science Double Cubesats of a 10x10x20 cm size. They will be equipped with common sensors allowing a multipoint long duration measurement of key quantities in the Thermosphere. Three sets have been identified: An Ion and Neutral Mass Spectrometer (INMS), an Oxygen Probe (FIPEX) and a Multi Needle Langmuir Probe (MNLM). The obtained and correlated data is expected to improve high altitude atmosphere models. In addition, around 10 triple cubesats will be embarked aiming at the demonstration of key technologies such as the deorbitation of structures using atmospheric drag sails, formation flying and reentry. The project will also show the potential of hardware miniaturization and the cubesat standard. A consortium led by the von Karman Institute, Belgium, is carrying out scientific, engineering and coordination work aiming at a launch of the constellation in 2015. The deployment system, the common sensors, mission and engineering analyses are provided by the consortium whereas the cubesats are a contribution from different universities of the cubesat community. The elements of the project are currently undergoing their Preliminary Design Review showing the compliancy to initially defined mission and design requirements. The support of the QB50 consortium partners, the cubesat providers, the Research Executive Agency (REA) and the European Space Agency (ESA) is gladly acknowledged. The QB50 Project all related activities are supported by the European Community Framework Programme 7, Grant Agreement no. 284427

Low Pressure, High Temperature Flow-Tube Experiments on a Carbon Preform Material

Francesco Panerai, Alexandre Martin, Steven A. Sepka, Jean Lachaud, Nagi N. Mansour

The entry of a spacecraft into a planetary atmosphere presents a challenging thermo-chemical environment, requiring a suitable heat shield to protect the spacecraft and its payload from the surrounding gas. During moderate speed entry, typically below 7.5 km/s, and mild heat fluxes, up to 1 MW/m2, reusable materials are a suitable solution. A famed example are the ceramic tiles mounted on the Space Shuttle orbiter. Higher entry speeds (that can reach 12 km/s), heat fluxes exceeding 1.5 MW/m2, and high density atmospheres impose the use of ablative thermal protection systems, that handle the incoming heating by phase changes, chemical reactions and material removal. The last decade has seen a tremendous e ort by scientists and engineers towards the development of a new class of low density carbon/resin ablators, made of a carbon fiber preform impregnated in phenolic resin. A successful example is the phenolic-impregnated carbon ablator (PICA) developed at NASA Ames Research Center and flight qualified during the recent reentry missions. The underlying idea of carbon/phenolic ablators is to allow the use of different carbon preforms and to adapt the properties of the impregnating matrix to the characteristics of the flight trajectory. Despite their high technology readiness level, the TPS design process is still full of conservatisms due to assumptions that need to be adopted about material properties and physical phenomena. To reduce design uncertainties, existing pyrolysis-ablation models are being revisited and improved. A multi-scale approach has been recently proposed by Lachaud et al. from NASA in order to model the behavior of carbon/phenolic composites. It is based on the volume-averaging of a microscopic scale model for the oxidation of the fibers and the impregnating phenol. The diffusion of oxygen is treated with a porous-medium formulation for the mass transport inside the material. Despite numerous wind tunnel tests and flight data available in the literature, dedicated testing and quantification of the different parameters is needed to validate this model. The aim is to implement different experiments able to decouple and address separately the various effects occurring during the ablation process of a carbon/phenolic composite. Studying the oxidation of carbon preform is a first step towards this goal. Here we document oxidation experiments on the carbon preform of PICA in the NASA Ames flow-tube reactor. The material is tested under low pressure dry air and high surface temperatures, promoting the recession of the carbon fibers by oxidation, according to the leading reaction C(s)+O2?CO2. We present the results obtained at temperatures between 700 and 1300 K and dry air at gas pressures of 1.6, 10 and 60 kPa. Mass losses and recession rates were measured and oxidation features of the specimens were characterized. Scanning electron microscopy analyses allowed comparison of the degradation of the carbon fibers to the virgin material.

University of Idaho RLEP Ball Bot

George Korbel, Sophie Milam, Joe Hepner, Nathan Clark, Mogley Samter, Karen Jolly, Sarah Lynn, Kyle Morse, David Atkinson, Vytas Sunspiral

Tensegrity describes a structure comprised of compression members connected by a network of tension members. These structures have a high strength to weight ratio making them ideal for space applications. An interdisciplinary senior design team and two graduate student mentors at the university of Idaho designed and built a tensegrity robot as the first step toward replacing current, more complex EDL systems. It will deploy from a stored compact state and land using the structure itself to absorb the impact instead of a powered landing system, parachute or airbag. The robot the UI design team is designing must be capable of deploying and compacting by actuating its tensile members and must survive a thirty foot drop test without damaging its actuators.

Time-varying entry heating profile replication with a rotating arc jet test article

Jay Grinstead, Raj Venkatapathy, Eric Noyes, Jeff Mach, Dan Empey

Arc jets are the only means to validate the most critical performance parameters of a spacecraft heat shield's thermal protection system (TPS) at conditions simulating atmospheric entry. Arc jet testing of thermal protection materials is conducted by exposing a small sample to the arc jet stream for a prescribed duration. Facility operating conditions, test model size and shape, and distance from nozzle exit are specified to apply a prescribed constant heat flux and surface pressure to an instrumented test model. The data from several tests at a variety of (constant) flow conditions are used to develop and validate material response simulations employed for the design of a vehicle's thermal protection system. In flight, however, a spacecraft's heat shield encounters time-varying conditions where the heat flux increases, peaks, and then recedes as the vehicle descends through the atmosphere. A material's response to time-varying conditions may be different than the response to the constant condition of an arc jet test, however – for example, coking observed in Apollo-era flight tests of Avcoat has not been seen in prior or subsequent arc jet tests. Simulations developed from constant-condition arc jet testing may not capture critical thermophysical phenomena that, if known, could ultimately influence design decisions. Attempts to replicate the time-dependent aerothermal environment of an entry event by varying the arc jet facility operating conditions during a test have proven to be difficult, expensive, and only partially successful. An alternative approach is to operate the facility at a constant condition but takes advantage of the varying heat flux and pressure distribution over the surface of a curved test model. The concept utilizes a cylindrical arc jet test model that rotates on its axis (perpendicular to the flow direction) during a test run. The heat flux and pressure at a single point on the model will increase to their peak values then decrease as a function of time as the point rotates towards and away from the stagnation point. The model size and (constant) arc jet operating condition are chosen such that the stagnation point heat flux matches the anticipated peak heat flux at a targeted location on an entry vehicle's heat shield. Since the flight profile spans from zero to its maximum and back to zero, the angular direction and instantaneous rate at which the model is rotated will be programmed to realize a time-accurate heat flux profile that maps, as closely as possible, to the profile of the targeted location on the vehicle. This concept, informally known as ROAST, was carried forward to development to meet a need of the MSL Entry, Descent, and Landing Instrumentation (MEDLI) project for NASA's Mars Science Laboratory mission. The forebody heat shield of the MSL entry vehicle was embedded with MISPs (MEDLI Integrated Sensor Plugs) at several locations to measure surface pressure, in-depth temperature, and isotherm depths within the PICA (Phenol Impregnated Carbon Ablator) heat shield material during hypersonic entry. Once processed, the MEDLI data will be correlated with the results of high fidelity simulations of the applied aerothermal environment and PICA material response. This post-flight reconstruction phase will rely on analysis tools developed specifically for interpreting MEDLI's sensor data. ROAST will aid the validation of MEDLI's data reconstruction methodology with a means to obtain ground test data at

authentic time-varying conditions. A series of demonstration tests were conducted to prove the viability of the rotating test model concept. The ROAST arc jet test article consists of an instrumented cylindrical PICA model attached to an electromechanical mechanism that rotates the model. The mechanism is contained within a water-cooled housing attached to a facility sting arm. A controller located remotely from the mechanism drives the motor. A shaft encoder provides positional feedback to the controller. The cylindrical PICA model contains one of the MISP sensors mid-way along the model's length. The ROAST assembly is positioned in the facility such that the MISP sensor plug lies at the stagnation point of the flow when the mechanism is rotated to achieve peak heat flux. The tests were conducted in the Ames Interaction Heating Facility (IHF) arc jet. Six ROAST models were tested. A cylindrical copper calorimeter of the same dimensions as the PICA models was also used to measure heat flux and pressure at 0?, 45?, and 90?. The calorimeter data will be used to validate computational fluid dynamics (CFD) simulations that will define the aerothermal environment for the PICA response analyses. The results of the demonstration tests and possible future applications of the ROAST concept will be presented.

The TechEdSAT-3P: The First Passive De-orbit Flight Test From The ISS

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The TechEdSAT-3p will be the first flight test of an Exo-Brake passive de-orbit system. Currently scheduled for an August launch on the HTV-4, it is second of a series of TechEdSATs to be jettisoned from the International Space Station (ISS). The first TechEdSAT-1, jettisoned on October 4, 2012, was a single unit cubesat (1U) that tested the jettison system as well as the core avionics package and power subsystem. Building on the predecessor, TechEdSAT-3p is a triple cubesat (3U) with a set of dual advanced avionics systems powered by the previous core system. In 2U of the packaging (roughly 2/3 of the volume), there is stowed an erectable deployment system that would place the Exo-Brake in the desired orientation. Scaled as a 10 day experiment, the deployed system has the ballistic coefficient of 5 kg/m2 - and is constructed as a tension-based structure that would not entangle itself. During the de-orbit process, GPS data from the dual avionics as well as collected TLE (Two Line Elements) provided by the USSTRATCOM will be used to analyze the flight dynamics and compared to computational models. This experiment will be the antecedent of related experiments at larger scale (e.g., for a 1.5 day de-orbit) and controllability. The effort was built by teams from the San Jose State University and the University of Idaho as a collaborative engineering project, with oversight from the NASA Ames Research Center.

Session: Future Mars

30kWatt Solar Electric Propulsion transfer stage for Phobos Sample Return Mission

Benjamin Donahue

This investigation presents an evaluation of 30kW to 100kW Solar Electric Propulsion (SEP) in-space transfer stages for Phobos and Demos Exploration and Sample Return missions. Because of SEP's high specific impulse (Isp) thrusters and low specific mass Solar arrays, SEP transfer stages could provide robust and affordable payload delivery capability. All electric satellite buses are now in production, and derivatives of these might form the basis for reliable, affordable, and reusable SEP in-space transfer systems. In this paper, Solar array and electric propulsion (EP) elements, suitable to Mars missions are discussed, spacecraft and mission parameters are described, and the benefits of 30kW to 100kW systems for Mars/Phobos missions are presented. Samples collected from Phobos and/or Demos, might form a first step in a series of SEP based missions that grow in power level and capability, culminating with a SEP stage that supports a Mars Sample Return (MSR) lander and ascent stage. Boeing's all electric 702SP satellite bus represents one possible starting point for SEP stage development. A summary of Boeing's recent DARPA sponsored FAST array technology development, conducted in 2009-2012, will be included. The advantage that SEP stages may have for future Mars science missions stem from the fact that they can realistically evolve from 30kW to higher power levels, enabling more missions. This evolution to higher power can be accomplished via a cost effective, modular approach that builds on multiples of proven designs and earlier successess. Trajectory simulations of EP flight to Phobos will be shown. With several SEP based NASA and ESA science missions successfully accomplished, SEP based in-space stages may be poised to surpass chemical propulsion based transfer stages for future, high energy science missions.

Evolution of the Multi Mission Earth Entry Vehicle Impact Landing Protective System

Scott Perino, Javid Bayandor

This paper discusses the development of an earth entry vehicle (EEV) shown in Fig. 1, and the methodology to evaluate its landing response when returning to earth. The upcoming NASA Mars Sample Return Mission (MSR) currently planned for launch in 2020 requires a simple and reliable method to return Martian samples back to earth for analysis [1]. The EEV is one of the proposed solutions to this MSR requirement. During reentry the EEV is designed to slow within the atmosphere and then impact the earth at terminal velocity thereby completing its mission. To protect and contain the Martian samples in the EEV during this severe collision a high performance impact sphere (IS) is being designed [2]. The IS is a low weight high energy absorption structural system comprised of radially oriented interlocking hybrid fiber composite panels and crushable foam filled chambers. The features of the IS allow the structure to be subjected to progressive crush failure rather than less efficient buckling failure. Numerical representation of the interactions between stiff and soft components under high-rate compressive loading is difficult and often fails to reproduce the proper mechanics. To incorporate such intricacies, in this methodology complex structural interactions and a wide range of failure physics were uniquely addressed. Numerical models using the developed methodology were created and benchmarked against experimental tests conducted at NASA Langley Research Center (LaRC). The post impact failure analysis provided close agreement between the numerical data and test results. It was shown that a wide range of parameters and properties including acceleration, velocity, displacement, damage modes, and failure mechanisms, were closely matched. Through these investigations it was shown that the advanced vehicle developed has great potential in facilitating a myriad of future cost-effective space and planetary exploration missions. Figure 1: The EEV structure is comprised of structural components and the integrated impact landing protection system [1] References 1. Mattingly, R., May, L., “Mars Sample Returns as a Campaign,” IEEE Aerospace Conference, Big Sky, MT 2011. 2. Perino, Scott, Javid Bayandor, Aaron Siddens. “A Comprehensive Structural Dynamic Analysis Approach for Multi Mission Earth Entry Vehicle (MMEEV) Development”, NASA Tech. Report, CR-2012-TBD., Hampton, VA: NASA STI, 2012.

COMPARATIVE ANALYZIS OF RADIATIVE AEROTHEMODYNAMICS OF MARTIAN ENTRY PROBES

Surzhikov S.T.

Three dimensional numerical simulation data on radiative aerothermodynamics of Martian entry probes Pathfinder, Exomars and Mars Science Laboratory (MSL) are presented and analyzed. It is shown that integral radiative heating of leeward surface of the entry probes exceeds corresponding convective heating and should be taken into account for probes of large sizes. Modern models of computational aerophysics of entry probes, entering into Martian atmosphere are based on gasdynamic models of thermally and chemically non-equilibrium gases. These models use Navier ? Stocks (NS) or Reynolds averaged Navier ? Stocks (RANS) equations in one-, two-, and three dimensions. Also they take into account radiative and non-equilibrium processes of chemical and physical kinetics, as well as different models of turbulent mixing and radiation transfer. Such models are realized, as a rule, on multi-block inhomogeneous grids, or on unstructured grids. Used models of chemical kinetics are conjugated often with heterogeneous chemical processes on solid surfaces, or with kinetics of ablative processes. Electronic data bases on chemical-physical kinetics, thermodynamic and transport properties, as well as the spectral optical properties are the important components of these computational models. Aerophysics of the Martian entry probes has some characteristic properties in comparison with the Earth entry. These are: relatively low velocity of atmospheric entry (~6-7 km/s), relatively rare atmosphere, and significant role of radiative processes in ultraviolet (UV) spectral region, connected with intensive electronic band CO (4+), and in infrared (IR) spectral region, connected with vibrational bands of CO2 and CO. Development of the Martian radiative gasdynamic models was also the subject of some recently published papers. These papers contain calculated data obtained with two- and three-dimensional codes. Calculation data for convective heating of front shield and leeward surface of probes were used for verification of numerical simulation results of the given paper. General goal of given aerothermodynamic calculations is the prediction of conductive and radiative heat fluxes to surface of space vehicles Pathfinder, Exomars and MSL along their entry trajectories. These calculations are complicated by the nonequilibrium character of radiation heat transfer. Attempts to predict radiative heating along the whole surface of space vehicle pose a number of additional problems of radiation heat transfer in shock layer and wake regions. General ways for verification and validation all of such models and codes are the comparison with available experimental and flight data. Unfortunately, the list of the data is utterly small. Therefore comparative investigations of aerothermodynamic data obtained with the use of different computational models and codes are also of much current interest. Given paper presents three dimensional (3D) multi-physics radiative-gasdynamic model of Martian descent space vehicles, which is realized in NERAT-ASTEROID computer platform. Gasdynamic code NERAT-3D included unto the computing platform uses the time-relaxation method on structured multi-blocks curvilinear calculation grids. On the each time step the following groups of governing equations are integrated successively: the Navier ? Stokes (or Reynolds averaged N-S equations, with several

turbulence models) and continuity equations, the equations of mass conservation of chemical species, the equation of energy conservation together with equations for vibrational energy conservation for separate vibrational modes, the electron energy conservation, and the radiation heat transfer equation (in multi-group approximation). The energy conservation equation is integrated in the form of the Fourier-Kirchhoff equation, which is formulated relative to translational temperature. Numerical simulation results on radiative aerothermodynamics of three Martian entry probes, namely for Pathfinder, Exomars and MSL, correspond to non-equilibrium trajectory points, obtained with the use of NERAT-ASTEROID computer platform are presented and analyzed at the final part of the paper. Three-dimensional laminar convective and radiative heating predictions have been performed for three Martian entry probes. These are: Pathfinder, Exomars and Mars Science Laboratory. Presented numerical simulation results for convective heating were successfully compared with available calculation data for Pathfinder and Mars Science Laboratory. Presented calculation data on radiative heating predict significant contribution of the heat radiation into total heating of leeward surface of the space vehicles. Afterbody radiative heating rates are predicted to be generally one order higher than the laminar convective heating. Finally, some attempts of numerical rebuilding of the first experimental data from MSL are discussed.

Photosynthetic life on mars

Divya Krishnamoorthy

The space science and engineering community has made tremendous strides in scientific research and technology development since the Viking era, culminating in the launch of MSL, the first roving analytical laboratory and first astrobiology mission to Mars since Viking.Three primary requirements for life—water, essential elements and energy—are found within the near-surface of Mars .Shielded from the various stresses imposed by the Martian surface environment, it is believed that life could find the perfect niche within meters of the surface Simple photosynthetic organisms could theoretically receive enough energy from the sun at certain periods whether once every year or once every ten thousand years to thaw out from a frozen, dormant state and reproduce until they refreeze. My Paper is based upon examples set by extremophiles—many of them photosynthetic cyanobacteria—currently living within the Antarctic and Arctic ice caps on Earth that hibernate through frozen conditions and remain viable upon thawing . Potential Martian habitats sufficient to protect and sustain such cyanobacteria for long periods include the north and south polar ice caps as well as various sedimentary rock formations . Hence, there are so many possibilities for Photosynthetic life on mars...

Wind Measurements on the Martian Surface - A Review

Ralph Lorenz

The Viking landers provided our first, and so far best, record of in-situ wind measurements at Mars. I review these data and summarize them with some statistical models. Thermal anemometer measurements were acquired by the Pathfinder lander, but have not been reduced to windspeeds as the superheat of the sensor was sometimes swamped by turbulent fluctuations in the atmosphere. Unfortunately Spirit and Opportunity were not equipped with any meteorological instrumentation. While the Phoenix lander had only a 'telltale' wind indicator, it provided a record of many thousands of measurements over 150 sols. I compare these measurements with the Viking record. I also point out another source of wind information : the seismometer on Viking 2. The Viking seismometer, mounted on the lander deck prove sensitive to wind (as expected) raising the noise floor for geophysical seismic detections. However, from a meteorological perspective this affords a new window into wind at the surface of Mars, as the seismometer was more frequently-sampled than the wind sensors. I review this 'new' dataset. These data set the stage for wind measurements expected to be forthcoming from the Curiosity rover.

Mars-X: Human Exploration of Mars From Martian Orbit

Alessandro Grasso, Adrianos Golemis, Udrivolf Pica

It is important for the advancement of the human species to leave the cradle of Earth and extend human presence to the rest of the solar system and beyond. Twenty seven graduate students have come together at the International Space University to research manned exploration of Mars from Martian orbit. The team is highly diverse with students from sixteen countries collaborating in an international, interdisciplinary and intercultural (3i's) way. The Mars-X project identifies the problems associated with sending humans to a Martian moon and returning them safely to Earth. A framework is suggested for addressing these challenges by proposing an alternative to the International Space Exploration Coordination Group's Global Exploration Roadmap. This alternative advises going to Phobos as a preliminary step to explore Mars, instead of missions to the Moon and Near-Earth Asteroids. The Red Planet is located at an enticing distance from the Earth and possesses characteristics which make it an appealing destination for the advancement of human space exploration. Mars-X focuses on investigating whether all the technologies required for a manned mission to Mars can be tested on a single mission to Phobos. For the mission to achieve 439 days in orbit around Mars, a three-phase launch architecture is proposed, with the first phase launching in 2020. Mars-X outlines how technologies such as nuclear thermal propulsion, advanced radiation shielding, teleoperation of robotics, and artificial gravity will help achieve the goals of this mission. Political, legal, ethical, and economic aspects such as fundraising through Public Private Partnerships are addressed. The potential for involvement of spacefaring nations in this international venture is also discussed. Investigating the Mars-X mission from the 3i's approach is vital for its success. Therefore, Mars-X brings a unique perspective to the challenge of advancing human exploration throughout space.

System-level impacts of the InSight SEIS operations concept

Cecily Sunday, David Mimoun, Naomi Murdoch

The upcoming InSight mission sets out to study the composition and structure of Mars in order to help scientists first, to understand the formation and evolution of terrestrial planets and second, to determine the current level of tectonic activity and impact flux on Mars. To achieve these baseline goals, InSight's payload includes a number of instruments that will collectively measure the planet's interior seismic, rotational, and thermal activity. In particular, SEIS, the Seismic Experiment for Internal Structures, will be used to fulfill a number of science objectives that have been derived from the baseline mission goals. These objectives include, 1) to measure the rate and distribution of internal seismic activity on Mars, and 2) to measure the rate of meteorite impacts. SEIS consists of two independent, 3-axis seismometers, which will provide the desired frequency measurements and redundancy required for mission success. The support and monitoring of SEIS falls under the care of the Instrument Operations Team (IOT) at JPL and CNES, the Centre National d'Études Spatiales (France). Later in the mission, the IOT will also be responsible for command generation and lower level-science data processing, but first, they must develop the sequence planning and operations concept for the instrument. The operations concept describes all the processes for information uplink, downlink and functional activities as they relate to SEIS. The SEIS operations concept begins with the functional tests and sensor calibrations that take place during the early Cruise phase of the InSight mission. Next, during the Commissioning phase of the mission, the post-landing evaluation, deployment, and calibration operations of SEIS are outlined. Lastly, the operations concept describes the planning and execution of all science monitoring procedures occurring from deployment until the conclusion of the InSight mission, which has an estimated length of one Mars year and forty sols (beginning after Entry, Descent and Landing). Regardless of mission phase or activity, instrument operations raise a number of functional challenges, ranging from potential interactivity with other instruments to required compliance with data storage and transmission constraints. This presentation outlines the SEIS functional conditions described by the operations concept within the scope of constraints such as those mentioned above. In particular, the poster will focus on the system-level impacts of the seismometer's operations on the lander's power and data budgets as a function of time during the cruise, commissioning, and science monitoring mission phases. This poster presentation will also introduce an Excel Macros tool developed to support the SEIS operations concept. This tool automatically updates relevant parameters within an Excel spreadsheet as changes are made to the operations concept and re-calculates and evaluates SEIS data storage and energy consumption during any given operation. The purpose of quantifying metrics such as operation duration, power demand and data rate is to verify that the concept fits within the schedule and requirements of the overall Mission Operation System. This tool provides a quick and thorough means to perform this evaluation.

Honeybee Robotics Planetary Excavation and Drilling Technologies

Kris Zacny

Since 1990s, Honeybee Robotics has been developing planetary sample acquisition, excavation, drilling and grinding systems. These systems were built to meet specific requirements such as reaching certain depth, exhibiting certain level of autonomy, acquiring samples of certain sizes and integrity (e.g. core vs. powder). They were built to operate at restricted energy, power and Weight on Bit, and have reached various Technology Readiness Levels ranging from TRL of 4 to 9. Most of these systems were tested either in the planetary analogs such as the Antarctic, Arctic, Greenland, Atacama and in large environmental chambers. Three systems have been deployed on Mars: Rock Abrasion Tool of Mars Exploration Rovers, Icy Soil Acquisition Device on Mars Phoenix lander and Dust Removal Tool on the MSL Curiosity rover. The poster will present all the systems that have been developed to date and group them into the following categories: surface, near surface, 1 meter, 10 meter, > 10 meter.

A High-Altitude Divert Architecture for Low Ballistic Coefficient Vehicles at Mars

Amit Mandalia, Prof. Robert D. Braun

For future Mars entry, descent, and landing missions, it is important to increase the landed mass while increasing landing precision. These objectives force a trade-off for propellant mass, as less propellant mass will increase the total payload mass, but decrease the overall divert capability for a lander. State-of-the-art propulsive descent maneuvers are capable of performing both a deceleration task to slow the vehicle down to a safe touchdown speed, and a divert task to target a landing position optimally. Current architectures plan propulsive descent initiation in the terminal phase of flight when the altitude is very low. It has been observed that when large diverts need to be performed at low altitudes, the increased glideslope leads to increased propellant mass fractions. By decoupling the deceleration and divert tasks and by moving the divert task to earlier in the trajectory, the propellant mass fraction of a lander can be decreased. Low ballistic coefficient systems at Mars have been shown to complete hypersonic deceleration at much higher altitudes than traditional Mars entry systems. This leads to an increased flight time between the end of hypersonic deceleration and the deployment of a decelerator like a parachute. It is proposed to leverage this added timeline by performing a divert maneuver during this time. The performance of this entry, descent, and landing architecture is analyzed using a modified low ballistic coefficient version of the Mars Science Laboratory entry system which has the following phases: hypersonic entry, propulsive divert maneuver, parachute deploy, and then a non-precision propulsive deceleration maneuver to the ground. The propulsive divert maneuver uses the application of thrusts (from ADCS thrusters) orthogonal to the velocity vector to target a landing site. The achievable divert distances and propellant mass fractions are determined as a function of vehicle ballistic coefficient, thruster size, and maneuver initiation speed and flight-path angle. The statistical significance of this maneuver is evaluated through the use of Monte Carlo simulations. Finally, this architecture is compared to contemporary architectures that utilize optimal propulsive descent maneuvers.

EXOMARS 2016 ENTRY PROBE TRAJECTORY SIMULATION

Aboudan Alessio, Giacomo Colombatti, Francesca Ferri

In the framework of the ESA EXOMARS project and the AMELIA experiment for entry and descent science, an accurate physical and dynamical model of EXOMARS 2016 entry probe has been developed and simulation of the trajectory dynamics has been investigated. The Entry and Descent Module (EDM) of the ExoMars 2016 mission will land during one of the dustier season on Mars, around Ls=243.7°. To prepare the landing and provide hints for the exploitation of the housekeeping sensor data in the probe reconstruction trajectory process we perform simulation studies in order to characterise the sensor outputs during the different Entry Descent and Landing (EDL) phases. During the entry phase the EDM, protected by the front heat shield, will be dragged by the Martian atmosphere. An ad hoc developed 6DoF model allows simulating the aero-dynamical forces that act on the probe and the computation of probe position, velocity, angular rate and attitude. These parameters are used to compute the expected measurements of the on-board sensors. The previous experience and lesson learned from Huygens mission provided a solid base for the development of the probe and sensors models [1]. This activity will therefore be preparatory for the analysis of the real 2016 Exomars housekeeping data and the simulation techniques here developed will contribute not only to the trajectory reconstruction but also to the retrieval of the vertical profiles of significant atmospheric parameters such as density, pressure and temperature [2]. The EDM trajectory starting from 120km above surface (@ EIP - Entry Interface Point) has been simulated; the atmospheric parameters from the European Mars Climate Model [3] have been considered and used as inputs for the simulation; the EDM mechanical and aerodynamical coefficients have been derived from the ESA ExoMars 2016 databases [4]. Outputs of the EDLS sensors are simulated considering their location onboard the EDM and their sensitivities: front shield and backshell sensors and EDM inertial unit. These simulations will allow us to define the possible accuracy we could achieve with the actual EDLS housekeeping sensors on the trajectory and attitude and therefore on their inversion for the reconstruction of the atmospheric vertical profile and definition of atmospheric structure and dynamics. The analysis of the sensor outputs sensitivities will also contribute on the possible selection of the sensor acquisition timeline in order to increase the scientific return of the EXOMARS 2016 mission (i.e. acquire data at altitudes higher than the 120km up-to-date baseline value). [1] Colombatti, G., A. Aboudan, F. Ferri, F. Angrilli Huygens probe entry dynamic model and accelerometer data analysis Planet. Space Scie., 56, 601–612, 2008. [2] Aboudan, A., G. Colombatti, F. Ferri, F. Angrilli, Huygens probe entry trajectory and attitude estimated simultaneously with Titan atmospheric structure by Kalman filtering Planet. Space Scie., 56, 573–585, 2008. [3] http://www-mars.lmd.jussieu.fr/ [4] Calantropio, F., ExoMars EDM Aerodynamic Database. EXM-DM-TNO-AI-0058, 10 Feb. 2012

A Mars Sample Return Sample Handling System

David Willson, Carol Stoker

We present a sample handling system, a subsystem of the proposed Dragon landed Mars Sample Return (MSR) mission [1], that can return to Earth orbit a significant mass of frozen Mars samples potentially consisting of: rock cores, subsurface drilled rock and ice cuttings, pebble sized rocks, and soil scoops. The sample collection, storage, retrieval and packaging assumptions and concepts in this study are applicable for the NASA's MPPG MSR mission architecture options [2]. Our study assumes a predecessor rover mission collects samples for return to Earth to address questions on: past life, climate change, water history, age dating, understanding Mars interior evolution [3], and, human safety and in-situ resource utilization. Hence the rover will have "integrated priorities for rock sampling" [3] that cover collection of subaqueous or hydrothermal sediments, low-temperature fluid-altered rocks, unaltered igneous rocks, regolith and atmosphere samples. Samples could include: drilled rock cores, alluvial and fluvial deposits, subsurface ice and soils, clays, sulfates, salts including perchlorates, aeolian deposits, and concretions. Thus samples will have a broad range of bulk densities, and require for Earth based analysis where practical: in-situ characterization, management of degradation such as perchlorate deliquescence and volatile release, and contamination management. We propose to adopt a sample container with a set of cups each with a sample from a specific location. We considered two sample cups sizes: (1) a small cup sized for samples matching those submitted to in-situ characterization instruments, and, (2) a larger cup for 100 mm rock cores [4] and pebble sized rocks, thus providing diverse samples and optimizing the MSR sample mass payload fraction for a given payload volume. We minimize sample degradation by keeping them frozen in the MSR payload sample canister using Peltier chip cooling. The cups are sealed by interference fitted heat activated memory alloy caps [5] if the heating does not affect the sample, or by crimping caps similar to bottle capping. We prefer cap sealing surfaces be external to the cup rim to prevent sample dust inside the cups interfering with sealing, or, contamination of the sample by Teflon seal elements (if adopted). Finally the sample collection rover, or a Fetch rover, selects cups with best choice samples and loads them into a sample tray, before delivering it to the Earth Return Vehicle (ERV) in the MSR Dragon capsule as described in [1]. This ensures best use of the MSR payload mass allowance. A 3 meter long jointed robot arm is extended from the Dragon capsule's crew hatch, retrieves the sample tray and inserts it into the sample canister payload located on the ERV stage. The robot arm has capacity to obtain grab samples in the event of a rover failure. The sample canister has a robot arm capture casting to enable capture by crewed or robot spacecraft when it returns to Earth orbit. [1] Lemke et al. 2013, IPP workshop 10, San Jose CA June 17-21, 2013 [2] MPPG Summary of final report, 2012 NASA at http://www.nasa.gov/offices/marsplanning/home/, [3] Planning for Mars Returned Sample Science: Final report of the MSR End- to-End International Science Analysis Group (E2E-iSAG), 101 pp., posted December, 2011, by the Mars Exploration Program Analysis Group (MEPAG) at http://mepag.jpl.nasa.gov/reports/. [4] Zacny et al. 2013, LPSC 1331, [5] Younse et al. 2013, LPSC 1198.

Design development and testing of throttable hybrid rocket engine for soft landing on planetary surface

D.Pavarin, D. Pavarin, J.E. Ronningen, P. Van Put, R. Popela, L. Mollinedo Herrera, V. Stasse, L.T. De Luca

The SPARTAN (SPAce exploration Research for Throttleable Advanced eNgine) research program aims at developing a throttleable propulsion technology, which is mandatorily needed for any planetary soft and precision landing. It relies on the hybrid engine technology, exploiting its capability of being throttled and its proper performance. The research focuses on three major objectives, needed to achieve the soft and precision landing capabilities: * The engine design, specific for throttling functionality * The oxidizer throttleable device development * The design of the landing case: test bench and testing procedures The development is supported by establishing an advanced coding, enabling the definition of the fuel and the throttling behavior of the hybrid engine. The design is supported by development tests: cold injection case, dedicated to the throttling device, and hot firing on subscale model, merging the throttling device and a subscale engine. In parallel it is developed a landing test and the associated landing model (flying test bed), providing the availability of proven landing model and landing test capabilities. These capabilities will allow demonstrating the soft and precision landing features of a throttleable hybrid propulsion technology. This research program implements and strengthens the technological base in view of the future robotics and manned space exploration missions. The outcomes from the SPARTAN development can be reflected also in many Earth/Space civilian and military applications, exploiting both the throttling capability of the propulsion system and the peculiar characteristics of the hybrid engine technology, like: safety, minimum environmental impact (green propellants), lower life cycle costs, responsiveness, competitive performance, increased reliability, soft ignition, and shutdown. The SPARTAN program has been approved for support in the frame of the European FP7, COOPERATION THEME 9 - SPACE, REA Grant Agreement n. 262837.

High Mass EDL Architecture Performance Analysis and Engineering Challenges for Crewed MARS Entry Vehicle

Prabhakar Subrahmanyam

Mars has been explored historically only by robotic crafts, but a crewed mission encompasses several new engineering challenges primarily landing mass being a critical factor for evaluation. The Viking heritage class of Landers has been the largest Landers insofar that had delivered to the Martian surface a landed mass of 600 kg. The MSL Lander of about 2600 kg used propulsive descent and sky crane technology during the final landing of the EDL sequence after the heat shield and back shield were jettisoned pushing the envelope of landed mass performance to newer levels. MSL in August 2012 delivered the largest and heaviest payload (~900 kg payload mass) to the Martian surface ever making history in payload delivery for robotic exploration. Robotic landers have delivered 1-2 ton payloads within a range of about 10 kilometers. Albeit, this may seem challenging by itself, these requirements are extremely moderate in comparison to space-qualifying a human rated EDL for Mars mission where the requirements are much more stringent (i) to safely land payloads anywhere between 40 to 80 t at scientifically interesting locations for exploration and (ii) within a considerable and acceptable precision range of tens of meters. A sensitivity analysis tool should be able to support variation of several different parameters and generate the performance data for the designer. A comprehensive Entry Descent Landing (EDL) parametric analysis has been conducted in support of a high mass landing architecture. A study on aerocapture corridor over a range of entry velocities (6 to 9 km/s) suggests that a hypersonic L/D of 0.3 is sufficient for a Martian aerocapture. Parametric studies conducted by varying aeroshell diameters from 10 m to 15 m for a hypersonic lift to drag ratio (L/D) of 0.3 for several entry masses up to 100 t are summarized. Entry-from-orbit and aerocapture scenarios were investigated at velocities of 4 km/s and 6.75 km/s respectively. Sensitivity analysis results shows that vehicles with entry masses in the range of about 40-80 t are capable of delivering cargo with a mass on the order of 5-20 t. Strategies and technologically challenging issues that we currently encounter for a human rated EDL to the Martian surface are evaluated and presented in this Poster/study.

ExoMars Entry & Descent System ready for Flight model manufacturing

Denis LEBLEU, Jean MONIER, Baptiste MARCHAND, Jean-Regis SQUILLACI, Gilles LUBRANO, Patrice LAURENTI, Philippe CAPUS

ExoMars is the second planetary entry probe vehicle directly developed by ESA, after HUYGENS. Some technologies used on HUYGENS are reused for ExoMars, but specific development and extension of qualification range were necessary to fulfil the objectives of this mission. This paper will present the technologies selected for ExoMars Entry & Descent System, and their development status, leading to Flight model manufacturing in 2013. During Entry and Descent, mechanical and thermal environment history leads to specific constraints for EDM aeroshell structure. Both Frontshield and Back Cover are composite sandwich, with carbon fibers embedded in epoxy resin. The operating temperature induced by Entry aerothermal environment combined with the significant mechanical loads generated by aerodynamic deceleration and parachute deployment have led to qualify the classical structure in an unusual operational range. Manufacturing of the Aeroshell structural model has started in this mid 2012, and it was tested in specific static test configurations, in order to stress the structures in a representative way. Fligh model is now almost completed. Light ablative TPS material selected to protect ExoMars Descent Module from aerothermal flux is derived from launcher application. The material TRL was increased for Entry vehicle application via ARD, BEAGLE2 and NetLander project, but a significant step was necessary for ExoMars. Impact of sterilization process, convective heat flux up to 2 MW/m², new inter-tile joint material, all specific features which have led to follow a specific development plan to improve TPS qualification range. Parachute subsystem is based on a classical Disc Gap Band single stage parachute, using the successful HUYGENS specific shape. However, the very significant deployment load, combined with an extension of operating Mach range has induced both to extend this DGB performance at higher Mach and to qualify all parachute materials for ExoMars. Scaled parachute was also tested during a high altitude drop test. The Parachute Deployment Device takes advantage of NASA Mars mission developments, the parachute mass to be ejected being significantly higher than HUYGENS one. Final Parachute System qualification will be reached during a High Altitude Drop Test on a fully representative Qualification Model in fall 2013. ExoMars terminal deceleration is performed via the Pulse Modulation of three clusters of three 400N engines. During the 30s continuous operation of the Reaction Control Subsystem, about 39kg of hydrazine will feed the nine engines in their proper inlet pressure range. The physical phenomena acting in the RCS are unusual compared to both Satellite or Launcher design. Specific development models are under test to verify architecture in a comprehensive way. A first development model was successfully used in 2011 to qualify fluidic simulator which allows to predict RCS performances. Manufacturing of ExoMars Entry and Descent Subsystems flight models are in progress, and all will be delivered to Instrustrial Prime level end 2013 – Beginning 2014 for integration, and for the Launch in January 2016.

Characterizing An Experimental Decelerator For Delivering Nano-Sat Payloads To Planetry Surfaces

Mitch Bodmer, Marcus Murbach, Kevin Ramus, Nicholas Rodriguez, Joshua Benton, David H. Atkinson

NASA Ames Research Center is developing the Small Payload Quick Return (SPQR) system, designed to quickly deliver a small payload from the International Space Station (ISS). The SPQR system uses a unique atmospheric drag device (Exo-Brake) which makes it compatible with crew operations. A crew member would release the system from the ISS. The atmospheric drag device would deploy, and de-orbit the system in roughly 40 orbits. A tube deployed re-entry vehicle would then allow the payload to enter the atmosphere. At an altitude near 30,000 meters (95,000 feet), a parafoil system would deliver the payload to the surface of Earth. In order to improve the accuracy of the targeting system over the gradual de-orbit process, it was necessary to develop an on-orbit position determination system. A Short Burst Data (SBD) modem offered by satellite phone vendors is to be tested to determine the viability of the uplink/downlink capability. This modem will include a GPS receiver and will transmit its location multiple times per orbit. The position resolution will help accurately characterize the aerodynamics of the atmospheric drag device throughout entry and descent. The University of Idaho's Near Space Engineering program has worked with NASA Ames to test various subsystems for the SPQR system. The balloon program had been able to serve as a test bed for iterative tests of the telecom system and GPS guided parafoil. The first experimental atmospheric drag device (Exo-Brake) test is scheduled for April 2013. This will consist of a scale model being dropped at 50,000 feet (15000 m). A 3U cube satellite will be delivered to the ISS in June 2013 with Exobrake and telemetry systems tested by the Balloon program. Once the atmospheric drag device is properly characterized and tested, the SPQR system could be used to deliver Nano-sat scale payloads to the surface of Mars.

The Exo-Brake: An Exo-Atmosphere Drag Device for Earth De-Orbit and Aero-Assist

M.Murbach, P. Papadopoulos, D. Atkinson

The current and future development of the Exo-Brake concept is described. The device is an erectable drag device for use in free molecular flow regime in the exo-atmosphere of the earth or potentially other planetary bodies. The concept is compared to other concepts, including solar sails which have been recently attempted, as well as various deployable/inflatable concepts. The design is essentially a tension-based structure - and is not susceptible to the beam buckling that would be expected of a typical solar sail construction. Also, the design may be more robust than some of the inflation-based aeroassist concepts which have been proposed - in that the leakage of a pressurant gas would not lead to the catastrophic failure during the critical orbit insertion maneuver. The Exo-Brake also has a capability of modulating the drag, which would be critical in overcoming atmospheric uncertainties (which could result in reduced navigation/targeting capability). Current laboratory and high altitude tests are described, as well as the first planned de-orbit demonstration. Currently scheduled as a jettison experiment in September, 2013, the first Exo-Brake flight test would perform a de-orbit maneuver from the International Space Station (ISS). This effort is also coupled to the improvement in navigational system inputs, that would lead to a capability of being able to return small payloads from the ISS on a regular basis.

Session: Giant Planets

Science at Saturn from Shallow Entry Probes

David H. Atkinson, J.I. Lunine, A.A. Simon-Miller, S.K. Atreya, W. Brinckerhoff, A. Colaprete, A. Coustenis, T. Guillot, P. Mahaffy, K. Reh, L.J.Spilker, T.R. Spilker, C. Webster

Data from atmospheric entry probe missions at the giant planets could uniquely discriminate between competing theories of solar system formation and the origin and evolution of the giant planets and their atmospheres, providing for valuable comparative studies of giant planets as well as providing a laboratory for studying the atmospheric chemistries, dynamics, and interiors of all the planets including Earth. The giant planets also represent a valuable link to extrasolar planetary systems. For these reasons, a Saturn Probe mission with a shallow probe is ranked by the recent U.S. Planetary Science Decadal Survey as a high priority for a New Frontiers class mission. Atmospheric constituents needed to constrain theories of solar system formation and the origin and evolution of the giant planets could be accessed and sampled by shallow entry probes. Many important constituents are either spectrally inactive or are beneath an atmospheric overburden that is optically thick at useful wavelengths and are therefore not remotely accessible by flyby or orbiting spacecraft. A small, scientifically focused shallow entry probe mission could make critical abundance measurements of key constituents, and could measure profiles of atmospheric structure and dynamics at a vertical resolution that is significantly higher than could be achieved by remote sensing techniques. The Galileo mission began the detailed study of the solar system's two gas giants by dropping an entry probe into the atmosphere of Jupiter and deploying an orbiter around Jupiter. In 2016-2017 the Juno mission will make meas-urements of Jupiter's deep oxygen abundance, and gravitational and magnetic fields. In the same epoch, the Cassini orbiter is planned to pursue a set of Juno-like orbits to make comparable gravitational and magnetic field measurements of Saturn. A Saturn atmospheric entry probe would complete the quartet of missions needed for a comparative study of the two gas giants, leading to improved models of solar system formation. A highly focused entry probe mission at Saturn carrying a minimal science payload could address unique and critical science while fitting within existing program budget caps. Fundamental measurements include abundances of the noble gases He, Ne, Ar, Kr, and Xe and, abundances of key isotopic ratios 4He/3He, D/H, 15N/14N, 18O/16O, and 13C/12C. Detection of disequilibrium species CO, PH3, AsH3, and GeH4 is diagnostic of deeper internal processes and dynamics of the atmosphere along the probe descent. Abundances of these key constituents, as well as carbon which does not condense at Saturn, sulfur which is expected to be well-mixed below the 4 to 5-bar ammonium hydrosulfide (NH4SH) cloud, and gradients of nitrogen below the NH4SH cloud and oxygen in the upper layers of the H2O and H2O-NH4 solution cloud, could be measured by an entry probe descending through 10 bars. In concert with the results from Galileo, Cassini, and Juno, a shallow Saturn probe capable of measuring abundances of key constituents not accessible by a remote sensing mission would provide critical measurements enabling a comparison of composition and dynamical processes on the giant planets while also providing an improved context for understanding exoplanets.

Design development and testing of throttable hybrid rocket engine for soft landing on planetary surface

D.Pavarin, D. Pavarin, J.E. Ronningen, M. Van der List, R. Popela, L. Mollinedo Herrera, V. Stasse, L.T. De Luca

The SPARTAN (SPAce exploration Research for Throttleable Advanced eNgine) research program aims at developing a throttleable propulsion technology, which is mandatorily needed for any planetary soft and precision landing. It relies on the hybrid engine technology, exploiting its capability of being throttled and its proper performance. The research focuses on three major objectives, needed to achieve the soft and precision landing capabilities: * The engine design, specific for throttling functionality * The oxidizer throttleable device development * The design of the landing case: test bench and testing procedures The development is supported by establishing an advanced coding, enabling the definition of the fuel and the throttling behavior of the hybrid engine. The design is supported by development tests: cold injection case, dedicated to the throttling device, and hot firing on subscale model, merging the throttling device and a subscale engine. In parallel it is developed a landing test and the associated landing model (flying test bed), providing the availability of proven landing model and landing test capabilities. These capabilities will allow demonstrating the soft and precision landing features of a throttleable hybrid propulsion technology. This research program implements and strengthens the technological base in view of the future robotics and manned space exploration missions. The outcomes from the SPARTAN development can be reflected also in many Earth/Space civilian and military applications, exploiting both the throttling capability of the propulsion system and the peculiar characteristics of the hybrid engine technology, like: safety, minimum environmental impact (green propellants), lower life cycle costs, responsiveness, competitive performance, increased reliability, soft ignition, and shutdown. The SPARTAN program has been approved for support in the frame of the European FP7, COOPERATION THEME 9 - SPACE, REA Grant Agreement n. 262837.

Investigating Uranus with GEORGE: The Gap-Based Expanding Outer Region Geophysical Explorer Mission

Jeremy Straub, Joshua Berk, Nicholas Long, Tyler Hill

This paper presents an overview of a two-part mission to Uranus and its moons. Key questions about the Uranian system remain unanswered which can and should be resolved by this mission. These include gaining an understanding about the offset of Uranus's magnetic field and its axial tilt. Additional details about the composition and formation of Uranus's moons are also desired. The mission will be comprised of a primary craft and three 6-U CubeSats. The CubeSats will initially be deployed in Uranian orbit to perform a characterization of the gravity field of the planet. They will then be moved by the primary spacecraft to the moon Miranda where they will collect imagery and produce a gravity map. The primary spacecraft will also perform magnetospheric and electric dynamo measurements of Uranus. Umbriel and Oberon have been identified as secondary mission targets, should resources (and craft longevity) allow. This paper presents a detailed overview of the GEORGE mission, largely focusing on the primary spacecraft. The paper begins by making the case for Uranian exploration. It then focuses on the scientific and engineering benefits prospectively offered by such a mission. It discusses several prospective architectures for this mission and explains why the primary and three-subsatellite approach was utilized and the benefits thereof. The enabling technologies for this mission are discussed and an evaluation on the current status of each requisite technology is presented. In the case of technologies requiring further development, a brief overview of what is required and an estimated timeline are presented. A mission operations and mission data operations plan are presented. The data operations is a critical portion of the mission as the spacecraft's cost level is reduced significantly through the use of model-based transmission reduction (MBTR) which allows a lower-bandwidth communications link and a reduction in spacecraft size due to lower power generation and gain (and thus antenna size) requirements. The paper concludes by placing the mission in the broader context of the United States and the world's space exploration plans. The 'stepping stone' value of this mission is discussed and prospective follow-on mission concepts are considered.

Europa Exploration Challenges

Satish Chetty, Bobby Machinski, Vangelis Christodoulou, Bob Higgis, Simon Kanis, Periklis Papadopoulos

Jupiter's moon Europa, appears to have an ocean of liquid water under its hard icy shell with hydro thermal vents. With liquid water, the ocean floor may be capable of supporting life. Robotic exploration of Europa's ocean floor will be one of the greatest space engineering challenges. Similar manual ice drilling efforts are underway to reach sub glacial lakes in Antarctica. An equipment and fuel tracker convoy took several days to reach the drilling site in Antarctica. A custom built hot water drill with filtration system was used to drill Lake Whillans. Such drills or drilling system will not be suitable for autonomous drilling in Europa. Instead we suggest the use of MMRTG based heat/drill based drilling cyrobot to reach Europa's ocean. This cyrobot will also serve as hydrobot when it reaches liquid water. This poster presentation will explore some of these challenges faced in performing such an exploration i.e. from landing on an airless, ice surface, to the use of cyrobots/hydrobots for drilling, and under water exploration.

Ongoing TPS development at NASA Ames Research Center

Mairead Stackpool

Ongoing TPS development at NASA Ames Research Center Abstract NASA Ames has a rich history in TPS development ranging from shuttle tiles to Phenolic Impregnated Carbon Ablator (PICA). This poster will outline past TPS development efforts at Ames as well as new TPS development in ablative polymer composites. Phenolic resin has been used extensively as a matrix for TPS applications. However, phenolic resin has limitations in both properties and processing, therefore other high performance polymers are being considered.

Session: Mars Science Laboratory

Mars Science Laboratory Trajectory, Atmosphere, and Aerodynamic Reconstruction Using Statistical Estimation Techniques

Soumyo Dutta, Robert D. Braun

On August 5, 2012, Mars Science Laboratory became the seventh successful U.S. mission to complete entry, descent, and landing on Mars. The vehicle charted a lot of new ground by flying the largest aeroshell and the largest supersonic parachute for any planetary entry mission, while also becoming the first spacecraft on Mars to use hypersonic guidance. Also among the firsts was the large amount of data that Mars Science Laboratory collected during its descent through the Martian atmosphere. The vehicle not only collected inertial measurement unit data (3-axis accelerometer and 3-axis gyroscopes) and radar altimeter data (range and range rate measurements), but it also gathered pressure measurements on the aeroshell at seven different locations during the hypersonic and supersonic phases of the flight. The large variety of on-board sensors allow for the reconstruction of many parameters of interest, such as the flight's trajectory, Martian atmospheric parameters describing the atmosphere experienced by the flight, and the vehicle's aerodynamic characteristics. Additionally, due to the existence of a variety of independent data sources, atmospheric and aerodynamic uncertainties can be separated, unlike for the limited datasets of past Mars missions. This paper demonstrates a statistical estimation methodology for the reconstruction in which measurement calibration data, vehicle's initial state uncertainty, and other statistical information are utilized to estimate the parameters of interest and their uncertainties. Three different statistical methods – extended Kalman filter, unscented Kalman filter, and covariance-matching adaptive filter – are used within the estimation methodology to determine the best estimate of Mars Science Laboratory's trajectory, atmosphere, and aerodynamics and their associated uncertainties.

Inverse Estimation of Mars Science Laboratory's Entry Aeroheating and Thermal Protection System Response

Milad Mahzari, Robert D. Braun, Todd R. White, Deepak Bose

There are substantial uncertainties in the computational models currently used to predict the heating environment of a spacecraft and the Thermal Protection System (TPS) material response during Mars entry. Flight data will provide a better quantification and possible reduction of such uncertainties as well as improvement of the current computational tools. The Mars Science Laboratory (MSL) entry vehicle landed on Mars in August 2012 and was instrumented with a comprehensive set of sensors in its heatshield. The aeroheating subsystem of MSL Entry, Descent and Landing Instrumentation (MEDLI) called MEDLI Integrated Sensor Plug (MISP) provided subsurface temperature measurements of MSL's PICA heatshield at different locations. The objective of this paper is to perform an inverse analysis based on the MISP data to reconstruct the entry vehicle's aerothermal environment and TPS material response. Prior to the inverse analysis, a direct analysis is performed to calculate the surface heating and TPS response predicted by current models. The predicted heating and TPS response will be used as an initial guess for the inverse methods and will provide a baseline for comparison with reconstruction results. In the direct approach, Computational Fluid Dynamics (CFD) simulations are performed using the Data Parallel Line Relaxation (DPLR) code to calculate the expected MSL's surface heating environment based on the vehicle's reconstructed trajectory. Next, the in-depth material temperature response is calculated using the Fully Implicit Ablation and Thermal response (FIAT) program. An inverse analysis is performed where PICA material properties and MSL's surface heating are estimated using inverse methods in order to achieve a reasonably close match with the data. A thermocouple driver (TC driver) approach is used in this work to decouple the surface heating form the in-depth heat transfer problems. The top thermocouple data is used as a boundary condition and then the simple in-depth heat conduction problem is solved in conjunction with inverse parameter estimation techniques to estimate PICA material properties from the deeper thermocouple data. Uncertainty and sensitivity analyses are used to provide information required for the material property estimation. Once the material properties are refined, the top thermocouple data are used to reconstruct the time-dependent surface heating of MSL vehicle. Two different approaches will be used. In the first approach, we attempt to estimate the surface heat transfer coefficient from the thermocouple data using the equilibrium surface energy balance formulation in FIAT. This approach will provide an estimate of surface heating based on equilibrium recession calculations. In the second approach, we first estimate the effective surface heating from the thermocouple data assuming no surface recession. Then, a recession profile is assumed and the surface energy balance at the wall is re-solved to provide an estimate of the surface heating for the pre-defined recession profile. This approach will allow us to provide surface heating estimates for a surface recession smaller than the one predicted by equilibrium models. Additionally, the effect of material property uncertainty and TC location uncertainty on the estimated surface heating will be explored. The authors have extensive experience with trajectory reconstruction, Mars

CFD modeling, TPS material response modeling and inverse data analysis topics. The authors have previously investigated using inverse methods for the estimation of TPS material properties from arcjet data and studied the reconstruction of MSL surface heating from simulated MISP data. Similar methods have also been used by the authors to estimate Mars Pathfinder surface heating from its heatshield subsurface temperature data. The authors performed a preliminary analysis of MSL aerothermal and TPS performance focusing on direct comparison of data and model predictions. The work presented in this paper will focus on a detailed inverse analysis.

Verification and Validation of the MSL/Curiosity Rover Entry Descent and Landing System

Richard Kornfeld, Ravi Prakash, Allen Chen, Ann S. Devereaux, MarHn E. Greco, Corey C. Harmon, Devin M. Kipp, A. Miguel San Martin, Steve W. Sell, Adam D. Steltzner

On August 5/6, 2012, the Curiosity rover successfully touched down on the Martian surface setting off the most ambitious surface exploration of this planetary body. Preceding this significant step were years of design, development and testing of the Curiosity Entry, Descent and Landing (EDL) system to prepare for the most complex landing endeavor ever attempted at Mars. This paper discusses the approach and implementation of the overall EDL verification and validation (V&V) program. The EDL V&V program had to face a number of major challenges. First, Curiosity's EDL system entailed a number of new and challenging technologies, including guided entry, a larger parachute, a new radar system, and most notably, the new and innovative sky crane approach during powered descent. Moreover, the presence of redundant avionics strings during EDL introduced additional complexity in both system design and V&V. Finally, a full-up end-to-end test of Curiosity's EDL systems in a terrestrial environment was considered prohibitive and beyond the means of the Curiosity project. Instead, the EDL V&V program was decomposed into three main domains. First, the EDL Flight Dynamics domain encompassed the external behavior of the flight vehicle, specifically the interactions of the vehicle with its environment during EDL. Key testing tools for this domain included an end-to-end EDL simulation using validated models. Secondly, the Command and Response domain addressed the internal behavior of the flight vehicle during EDL, in particular the hardware/software interactions. The latter was tested on a number of platforms with increased fidelities, starting with a software-only based softsim at one end of the spectrum and running full-up system tests on the actual flight vehicle at the other end. Finally, the EDL Subsystem domain dealt with hardware and software unit-level functional verification. Examples of this domain included the certification of Curiosity's parachute, field tests of the new landing radar, and sky crane drop tests. The approach and implementation of Curiosity's EDL V&V was thus tailored to identify and address the challenges of each of these three domains, while at the same time each domain fed into the other two. In addition to the nominal testing outlined above, a comprehensive stress and robustness test program was conducted to characterize both the flight dynamics and flight system performance of the vehicle outside the nominal envelope. This allowed the testing team to gain valuable insights into the robustness of the design and, where necessary, to change the software design to increase robustness and survivability. A final corner stone of the verification and validation of the MSL/Curiosity Rover Entry Descent and Landing System was the preparation and test of the processes, tools and teams that operated and ultimately landed the spacecraft on the surface of Mars. To this end, a number of Operational Readiness Tests were conducted. This paper will start out with an overview of Curiosity's EDL system design, its new technologies and challenges. It then discusses the three EDL V&V domains in greater detail, including the methodology to identify a complete set of verification and

validation items, and the various testing and simulation venues available for testing. Finally, the stress and robustness testing and the Operational Readiness testing will be described. The overall lessons learned and conclusions described herein can serve as a pathfinder for the EDL system testing approach and implementation of future Mars landed missions.

Trajectory Reconstruction and Uncertainty Analysis using Mars Science Laboratory Pre-Flight Scale Model Aeroballistic Testing

Rafael Lugo, Robert Tolson, Mark Schoenenberger

As part of the Mars Science Laboratory (MSL) trajectory reconstruction effort at NASA Langley Research Center, free-flight aeroballistic experiments of instrumented MSL scale models were conducted at Aberdeen Proving Ground in Maryland. The models carried an inertial measurement unit (IMU) and a flush air data system (FADS) similar to the MSL Entry Atmospheric Data System (MEADS) that provided data types similar to those from the MSL entry. Multiple sources of redundant data were available, including tracking radar and on-board magnetometers. An inertial navigation trajectory was obtained using accelerometers and rate gyros, but was inaccurate due primarily to poorly-known initial conditions. The aerodynamic parameters Mach number, angle of attack, and sideslip angle were estimated using minimum variance with a priori to combine the pressure data and pre-flight CFD aerodynamics data. Both linear and non-linear pressure model terms were also estimated for each pressure transducer as a measure of the errors introduced by differences in transducer behavior between calibration and flight. Parameter uncertainties were estimated using a “consider parameters” approach, which provided a more realistic measure of the uncertainties than those associated with noise. The FADS-based aerodynamic parameter estimates were validated independently of pressure and CFD data using IMU sensors and inertial position and velocity from radar data. This solution validation analysis showed that the differences between FADS- and radar-based aerodynamic parameter solutions exceed the 1-sigma uncertainty estimates. Potential error sources include transducer biases, imperfections in the hardware, and modeling errors. The loss of two pressure ports along the vertical axis on MSL-PTM-01, which are primarily used to determine angle of attack, also contributed to differences that exceeded the 1-sigma uncertainties by biasing the solutions. For the lifting flights where FADS estimation methods were unable to be applied, validation techniques were still able to provide reasonable estimates of the aerodynamic parameters. Thus, even in a close to “worst-case” testing scenario where IMU and pressure data behavior render inertial navigation and FADS solutions impossible, redundant data sources from radar and magnetometers can still provide useful aerodynamic parameter estimates. The vehicle/sabot interactions in the lifting flights that caused unexpectedly high oscillation rates did not appear in the 120 mm scale model radar calibration firings and may have been exacerbated by larger scales. The limited number of 7-inch models and accompanying sabots heavily restricted the qualification of mechanical solutions. In the case of MSL-PTM-02, the unstable flight was caused by high roll angle at muzzle exit coupled with abnormally high pitch rate amplitudes due to unexpected sabot/barrel and sabot/vehicle interactions, which indicate that sabot issues may cause experiment repeatability problems. Future free-flight aeroballistic experiments would benefit from more extensive sabot qualification tests that simulate actual test conditions, e.g. similar scales and CG offsets. More economical alternatives may include using models instrumented only with accelerometers and magnetometers and tracking them with radar, and utilizing the validation techniques described in the present paper to recover

aerodynamic parameters and qualify the sabot design and in-barrel behavior. Calibrating the pressure transducers over the entire pressure range that is expected to be observed would also aid in FADS solution accuracy, as would re-calibrating the transducers post-flight. The latter option would require the recovery of the model, which in the past has been prohibitive due to cost. The results of the aeroballistic experiments show that useful pre-flight data can be acquired from aerodynamically scaled instrumented models in free flight. Combined with the “truth” trajectories provided by tracking radar and other redundant data sources, these data may be used to validate the tools and methodologies designed for trajectory reconstruction.

Mars Science Laboratory Entry, Descent, and Landing Trajectory Reconstruction Uncertainty Assessment

Prasad Kutty, Christopher Karlgaard

The Mars Science Laboratory (MSL) entry vehicle successfully entered the Mars atmosphere and delivered the Curiosity rover to the Mars surface on August 5th, 2012. In addition to safely landing the rover, the entry vehicle returned valuable Entry Descent and Landing (EDL) data to earth for a variety of purposes, including trajectory reconstruction. Three semi-independent reconstruction methods were utilized for the MSL EDL – an inertial reconstruction, an aerodatabase reconstruction, and a Mars Entry Atmospheric Data System (MEADS) based reconstruction. Each method relied on distinct measurements taken by the MSL entry vehicle during EDL to compute time histories of common trajectory parameters, such as position, velocity, attitude, dynamic pressure, angle of attack, and angle of sideslip. In addition to performing the reconstruction of the entry trajectory, an uncertainty assessment was performed in order to identify the accuracy of each analysis technique. The uncertainty or error assessment captures the statistical variances of the reconstructed outputs based on the known uncertainties of the measurements used to generate those reconstructions. This paper is concerned specifically with the uncertainty assessments of the three reconstruction analyses. The inertial reconstruction was performed by integration of the non-linear dynamic equations of motion of the vehicle state. This deterministic approach relied only on the measured accelerations and angular rates from the inertial measurement unit (IMU), as well as the initial condition of the entry vehicle. Similarly, the input uncertainties and linear system dynamic matrices were used to integrate the covariance equations of motion and determine the uncertainties associated with the inertial reconstruction outputs. This approach, sometimes called a deterministic or navigated solution, is a very common reconstruction technique. It was used to provide an initial uncertainty assessment as an input to other analysis methods. The aerodatabase reconstruction was applied to extract angle of attack and angle of sideslip by solving a non-linear system of equations relating accelerations from the IMU to force coefficients from the MSL aerodynamic database. Additionally, estimates of density, pressure, and Mach number were recovered through computations involving the aerodatabase. Linear sensitivity matrices for each non-linear reconstruction equation were computed so that a covariance mapping computation could be performed to transform the input uncertainties to the reconstructed output uncertainties. The input uncertainties were known beforehand, either through measurement specifications or previous analysis, such as the inertial reconstruction. While this method of reconstruction has been previously used on several planetary missions (such as Pathfinder, MER, and Phoenix), the covariance mapping technique to determine output uncertainties is a technique not previously applied to this problem and is a new result given in this paper. The MEADS-based pressure reconstruction makes use of seven forebody pressure measurements as a Flush Air Data System (FADS) to produce estimates of the angle of attack, angle of sideslip, dynamic pressure, static pressure, and Mach number. The estimates are obtained from a nonlinear weighted least-squares algorithm at each measurement time. A novel IMU-aiding approach

has been implemented in which the IMU velocity estimate is combined with MEADS-derived speed of sound to produce an initial estimate of the Mach number. This initial estimate is then refined in the nonlinear weighted least-squares algorithm. The algorithm is iterated globally over the entire data until the estimated speed of sound profile (and, consequently, Mach number) converges. The uncertainties in the computed quantities arise naturally as a byproduct of the weighted least-squares algorithm, corresponding to the covariance matrix of the converged solution. This paper will provide an overview of the reconstruction algorithms and the mathematical details of the uncertainty analysis procedures and results of the MSL EDL reconstruction.

CHEMCAM Instrument on MSL : performance measured at Gale crater

Rene Perez, B.L. Barraclough, S. Bender, A. Cousin, L. DeFlores, Ch. Yana, O. Gasnault, S. Maurice, R. Wiens

The ChemCam experiment is one of ten science investigations onboard the NASA Mars Science Laboratory (MSL) rover “Curiosity”, landed at Gale crater on August 2012. The overall scientific goal of the mission is to explore and quantitatively assess a local region on Mars’ surface as a potential habitat for life, past or present. The Curiosity rover, built by Jet Propulsion Laboratory, carries ten scientific instruments and a sample acquisition, processing, and distribution system. The various payload elements work together to detect and study targets of interest with remote and in situ measurements, to acquire samples of rock, soil, and atmosphere and analyze them in on-board analytical instruments, and to measure the local environment of the rover. The ChemCam experiment is a package of a Laser-Induced Breakdown Spectrometer (LIBS) coupled to a Remote Micro-Imager (RMI) camera. Its main objective is to remotely determine the elemental composition of soils and rocks. The LIBS subsystem provides remote sensing data via active interrogation by a high-power laser, and passive spectra of targets using natural illumination. The RMI subsystem provides high-resolution images of the target regions interrogated by the LIBS laser, and is used to provide geologic context for the LIBS data. The LIBS instrument is probing samples up to 7m from the mast of the rover. The RMI instrument could provide images from 1.1m up to infinity. This is the first use of a LIBS system in space. Extensive testing before launch had showed that performance requirements shall be met [IPPW-8], and that ChemCam was capable of achieving its science goals. Additional tests have been performed on the Engineering Model Rover (JPL Testbed) and the Flight Software simulator prior to landing in order to assess that the ChemCam instrument successfully interacts with the different subsystems onboard the Curiosity rover. A special emphasis has been put on the pointing capability of the Rover. Another goal of these tests was to prove that the Rover Flight Software successfully protected the instrument and the rover against possible hazards during operation: laser beam collisions, sun safety, and fault protection processes. We report, here, on these additional tests performed during cruise, and on the verifications performed in-situ on Mars during the commissioning phase, confirming the excellent performance of the ChemCam measurements.

Session: Venus and Titan

Candidate regions on Titan as promising landing sites for future in situ missions

Anezina Solomonidou, Athena Coustenis, Pierre Drossart, Ralf Jaumann, Katrin Stephan, Frank Sohl, Hauke Hussmann, Mathieu Hirtzig, Georgios Bampasidis, Emmanuel Bratsolis, Xenophon Moussas, Konstantinos Kyriakopoulos

The highly successful and still on-going Cassini-Huygens mission to the Saturnian system provides us with valuable scientific results and discoveries of many complex phenomena while it significantly advanced our understanding of Titan, Saturn's largest satellite. Furthermore, it points to the need for a return mission, with both remote and in situ instrumentation, as a high priority in space exploration. Titan's surface has proven to be a complex and multivariable environment with many processes acting on it and changing its landforms. Additionally, many geological features seem to resemble a variety of terrestrial geological features enhancing the 'Earth-like' view of Titan. The observed morphotectonic features [1], the cryovolcanic candidate regions [2] and the correlation of the interior-surface-atmosphere system are some of the many issues that demand further extensive exploration by a return mission that will focus on these aspects with developed instruments affording higher spectral and spatial resolution, dedicated and in situ capabilities. Several such missions were proposed in the past and have been studied by the space agencies. Some of them included a lander, like the ambitious Titan Saturn System Mission (TSSM) [3] or the Titan Mare Explorer (TiMe) [4]. One of the most important goals of a future mission on Titan would be the achievement of a safe and successful landing over a location that provide as many as possible scientific interesting aspects. Such location should combine a significant variability in geological features attached to atmospheric processes as well as a safe and 'friendly' geological terrain for landing. The candidate cryovolcanic regions seem to be related with internal processes and are also possible openings for the release of methane in the atmosphere [2;5]. We suggest the possible cryovolcanic and morphotectonic regions on Titan as promising candidate locations for future landing on Titan and we present the case as to why they could provide a wealth of scientific data. References: [1] Solomonidou, A., et al. Planetary Space Science doi: 10.1016/j.pss.2012.05.003; [2] Solomonidou, A., et al. in prep.; [3] TSSM, NASA/ESA Final Report, 30 January 2009. [4] Stofan, E., et al. 2011. EPSC-DPS Joint meeting #909. [5] Solomonidou, A., et al. 2010. Hellenic Journal of Geosciences 45, 257-268.

MECHANICAL TESTING OF CARBON BASED WOVEN THERMAL PROTECTION MATERIALS

Parul Agrawal, John T. Pham, James O. Arnold, Keith H. Peterson, Ethiraj Venkatapathy

MECHANICAL TESTING OF CARBON BASED WOVEN THERMAL PROTECTION MATERIALS Parul Agrawal(1), John T. Pham(2), James O. Arnold(3), Keith H. Peterson(1), Ethiraj Venkatapathy(3) (1) ERC Inc., NASA Ames Research Center, Moffett Field, CA, 94035. (2)University Space Research Associates, NASA Ames Research Center, Moffett Field, CA, 94035. (3) NASA Ames Research Center, Moffett Field, CA, 94035. ABSTRACT Three Dimensional Woven thermal protection system (TPS) materials are one of the enabling technologies for mechanically deployable hypersonic decelerator systems. These materials have been shown capable of serving a dual purpose as TPS and as structural load bearing members during entry and descent operations. In order to ensure successful structural performance, it is important to characterize the mechanical properties of these materials prior to and post exposure to entry-like heating conditions. This research focuses on the changes in load bearing capacity of woven TPS materials after being subjected to arcjet simulations of entry heating. Preliminary testing of arcjet tested materials [1] has shown a mechanical degradation. However, their residual strength is significantly more than the requirements for a mission to Venus [2]. A systematic investigation at the macro and microstructural scales is reported here to explore the potential causes of this degradation. The effects of heating on the sizing (an epoxy resin coating used to reduce friction and wear during fiber handling) are discussed as one of the possible causes for the decrease in mechanical properties. This investigation also provides valuable guidelines for margin policies for future mechanically deployable entry systems. Figure 1. Pre (left) versus post (right) exposure to entry-like conditions imaged at 800x magnification. [1] Agrawal, P, Pham, J. T., Arnold, J. O., Peterson, K. H. and Venkatapahty, E., “Micromechanical Characterization and testing of Carbon Based Woven Thermal Protection Materials,” 37th Composites, Materials, and Structures Conference, Cocoa Beach, FL. Jan 21-24 2013. [2] Arnold, J. O., Peterson, K. H., Yount, B. C., Schneider, N. and Chavez-Garcia, J., “Thermal and Structural Performance of Woven Cloth for Adaptive Deployable Entry and Placement Technology” 22nd AIAA Aerodynamic Decelerator Conference, Daytona Beach, FL, March 25-29, 2013.

Balloon Options for Exploring the Upper Atmosphere of Venus

Jeffery L. Hall

The upper atmosphere of Venus spanning approximately from 57 to 72 km in altitude contains an unknown material that absorbs significant amounts of ultraviolet radiation from the sun. Understanding the composition and dynamics of this unknown UV absorber is an important science goal for Venus. However, this region of the atmosphere is difficult to access with standard entry probes because of the problem of decelerating at a sufficiently high altitude to allow for aeroshell separation and instrument activation and data collection. This is the reason why none of the previous Venus entry missions have been able to make the measurements needed to discover the nature of the UV absorber. This paper addresses the approach of exploring this region of the Venus upper atmosphere with different kinds of balloons. The options consist of helium-filled zero pressure and superpressure balloons, and altitude-cycling balloons that utilize a phase-change buoyancy fluid. All of these balloons would notionally be deployed at a lower altitude such as 55 km, much like the Soviet VEGA balloons, and then ascend to cover the desired 57-72 km range. The paper discusses the advantages and disadvantages of the different options and presents parametric design data that quantifies payload mass as a function of balloon size and flight altitude. The paper concludes with a brief discussion of the technological maturity of all three balloon options.

Possible life detected on the planet Venus' surface

Leonid V. Ksanfomality

The position of the hypothetical habitability zone in extrasolar planetary system was considered by many authors. Approximately 1/4 of exoplanets orbit their stars at very low orbits, which leads to high temperatures of their surface (if any), up to 800 K or more. Some of them should have the physical conditions close to those of Venus. Is there any possibility that the life forms can exist at quite different environment than "normal", Earth-like physical conditions? Namely the planet Venus could be the natural laboratory for studies of this type, having the dense, hot (735 K) oxygenless CO2 - atmosphere and high, 9.2 MPa, pressure at the surface. It should be recalled that the only existing data of actual close in TV-observations of Venus' surface are the results of a series of missions of the Russian VENERA landers which took place the 1970s and 80s, working in the atmosphere and on the surface of Venus. No other results of this kind were obtained since. A re-exemination of images of venusian surface obtained from the VENERA landers has been undertaken using a modern processing technique, with a view to detect any possible signs of life under the specific conditions on Venus. This speculative identification rests on two characteristics of these features: (a) their somewhat suggestive morphology and (b) their temporal appearance and behavior (present, than absent on subsequent images of the same area; or changing appearances). The re-exemination has identified previously unreported features that may correspond to hypothetical life forms on Venus' surface. Two of them, 'mushroom" (1) and "hespera" (2) are shown here. Analysis and comparison of the contents the sequence of panoramas of the venusian surface obtained in the course of the TV-experiments on the VENERA landers (1975-82), allowed the author to detect some interesting objects displayed on the panoramas. Following the change in their appearance on the sequence of images allowed a suggestion that such changes may be related to the possible habitability of the planet. Some of the objects found were described in few papers of L.Ksanfomality (2012). There are also found and listed in the report images of objects with special morphology resembling the shape of some terrestrial fauna. In the absence of new landing missions to Venus, the same study was carried out on the other remaining panoramas. There is a reason to believe that in the panoramas few class of unusual objects has been found, which will be shown in the report. Published in 2012: Ksanfomality L.V. Possible Detection of Life on the Planet Venus // Transactions of the Academy of Sciences, 2012, V. 446, No. 1, pp. 42–47. DOI: 10.1134/S1028335812090. Ksanfomality L.V. Hespers: Unusual Objects on Venus According to the Data of the VENERA Missions of 1975-1982 // Transactions of the Academy of Sciences, 2012, V. 447, No. 1, pp. 1–5. See also: 2012. 'Online First' on SpringerLink: (DOI)10.1134/S102833581211002X. Ksanfomality L. V. Venus as a Natural Laboratory for Search of Life in High Temperature Conditions: Events on the Planet Venus on March 1, 1982. // Solar System Research, 2012, Vol. 46, No. 1, pp. 41–53. Ksanfomality L.V. Results of the New Processing of Images Obtained from the Surface of Venus in a TV Experiment Onboard the Venera-9 Lander (1975) // Solar Syst. Res., 2012. V. 46 (5) pp.393-403. Ksanfomality

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Huygens Probe Heat Shield : Performance Constraints at Titan from Flight Data

Ralph Lorenz, The Huygens probe heat shield was not instrumented, so no direct information on its performance margin is available, beyond the fact that the probe was deployed successfully. Although it is generally assumed that no indirect information is available either, I show that a combination of housekeeping temperature sensors before, during and after entry, and Huygens Atmospheric Structure Instrument (HASI) temperature measurements at the beginning of descent, show some indication of the entry heat pulse. This allows at least crude constraints on the heat shield performance to be inferred. A lesson for future missions is that even when no heat shield instrumentation is available, recording of housekeeping temperatures during entry (before one might nominally operate most science payloads) can provide useful diagnostic constraints.

Venus Lifting Trajectory Studies of Low and Mid L/D Aeroshells

Joseph A. Garcia, M. Kathleen McGuire, Jeffrey V. Bowles, Loc C. Huynh

In NASA's latest planetary science decadal survey, entitled "Visions and Voyages for Planetary Science in the Decade 2013-2022," the National Research Council recommended several atmospheric entry missions. The mission of interest to this present study is the "Venus In Situ Explorer" (VISE), which is called out in the survey to be a New Frontiers class mission. An objective of the VISE mission is to obtain in situ measurements of Venus' atmosphere in order to understand its physics and chemistry. This objective involves atmospheric entry and hence requires protection of the science payload against aerodynamic heating. Protecting the payload against atmospheric heating on entry or aerocapture at the planet will depend strongly on the entry system architecture, and the entry state (velocity, ballistic coefficient, heading angle, and entry flight path angle). Traditionally, the architecture for this type of mission has been a rigid axisymmetric aeroshell based on a spherically-blunted cone geometry. This architecture requires ballast mass to offset its center of gravity (CG) to produce lift. In this study two alternate vehicle shapes, a slender mid L/D vehicle and an asymmetric capsule vehicle are also explored which enable lifting trajectories with little to no ballast mass required to offset the center of gravity (CG). Furthermore, the mid L/D vehicle has the additional advantage of providing over twice the L/D as compared to the capsule vehicles which could enable even larger mission payloads. The objective of the present study is to explore the potential benefits of employing lifting trajectories for a conceptual Venus mission by utilizing three classes of entry vehicles – (1) a low L/D axisymmetric capsule vehicle, (2) a mid-L/D slender blunt-body vehicle, and (3) a low L/D asymmetric capsule vehicle. For each configuration, two scenarios were analyzed – (i) direct entry, and (ii) aerocapture into orbit followed by entry. The VISE mission study, performed in support of the latest Planetary Science Decadal Survey, was used as the reference for entry conditions and masses. Additionally, the present study assessed the use of lower density Thermal Protection System (TPS) alternatives to high density Carbon Phenolic, which was used in the Pioneer Venus Multiprobe Mission. The results of the present study have shown that the use of lifting trajectories as compared to non-lifting (ballistic) trajectories reduces the overall required TPS mass of the vehicle. Furthermore, these results show that a direct lifting entry has an overall lower integrated heat load and hence a lower TPS mass as compared to an aerocapture followed by entry scenario.

CubeSat Technology on Titan Exploration

Demet Çilden, Cem Oran

Our study introduces a comprehensive analysis of Titan findings obtained by past missions, also an offer for interplanetary mission via a group of nano-satellites for exploring to celestial bodies. The study especially focuses on bodies which their dense atmospheres are posing an obstacle for observation from in-orbit spacecraft, such as Titan. In our solar system, Titan is the only celestial body on which findings indicates evidence for liquid except Earth. Moreover, the findings of Cassini-Huygens Mission show that the atmospheric conditions and hydrological cycle form Titan's surface features similar to those on Earth, such as dunes, lakes, seas with periodically liquid methane rains. Such balance between geological and atmospheric processes may be indicating the same predicted conditions of the Earth during its primordial age. Even complex self-organizing biological systems may be formed on Titan. These specialties make Titan a unique exploration area for planetary sciences. A large altitude range in the atmosphere remained poorly explored even after Cassini-Huygens mission, launched on 15 October 1997. After a seven year journey from launch to Saturnian system, Huygens probe, separated and successfully landed on Titan's surface on 14 January 2005. During its mission, Huygens probe took images, made spectral measurements, deduced atmospheric properties, measured physical and magnetic properties of Titan's atmosphere and surface, etc. In addition to the actual data obtained by Cassini-Huygens mission, the data from the future Titan missions may help scientists to understand how the Earth look like in its early ages and how life began. Much remains to be understood such as seasonal changes of the geology, atmosphere, climate, atmosphere-surface interaction and probability of existence of life on Titan. Nano-satellites can be thought as an effective choice for the interplanetary missions because of their advantages such as low cost, low weight, simple and task oriented design. Especially, CubeSats which are kind of standardized nano-satellites can be developed by utilizing commercial off-the-shelf technologies. A comprehensive research on celestial bodies, such as Titan, can be accomplished via numerous numbers of CubeSats with various scientific instruments. Titan's cloudy weather requires that CubeSats must dive into atmosphere in order to analyze low atmosphere segments and surfaces. Depending on their instruments, CubeSats can be grouped by assigning each group to a different specific task to increase the redundancy. For example, one group collects the temperature and pressure data from different altitudes, while another group exploring bio-signature or complex chemical compounds on Titan's atmosphere and surface. The main advantage of this idea is to allow Titan to be explored more widely compared to other options, such as lander, rover etc. CubeSats can be carried on a mother spacecraft (the station) during their journey to Titan. At the end of the journey, utilizing an aero capture maneuver will provide the mother spacecraft to pass through different layers of atmosphere and lets the CubeSats to be deployed at the desired altitudes suitable to their assigned tasks. After deploying all the CubeSats, the station will orbit Titan at suitable altitude to serve as a relay communication satellite between the CubeSats and the Earth. Among the satellites, a network using an internet like communication protocol should be

established to transfer data from each satellite to the station. Atmospheric drag will circularize the orbits of the CubeSats at the deployment altitude in time. Some of the CubeSats may be designed to survive the re-entry conditions and land on the surface using parachutes and/or airbags. All of the satellites will be equipped with radioisotope power supply subsystems appropriate for CubeSats and an integrated attitude control system comprising gravity gradient boom and reaction wheels. In conclusion, the advantages listed above provide reduction on the mission cost and effort significantly. Our poster includes detailed portrait of ideas for interplanetary nano-satellite missions.

Solar Airplanes for Venus Exploration

Geoffrey A. Landis, Anthony Colozza, Tibor Kremic

The atmosphere of Venus above the cloud layer is a region that has been identified to be of significant scientific interest, and which is, to date, been unexplored by spacecraft. Use of a remotely-piloted vehicle for exploration of this region is proposed. Such an airplane as a probe for atmospheric exploration has several advantages over other concepts, including the ability to achieve an extended mission duration, to visit different locations and altitudes in the atmosphere, and the ability to fly faster than the winds, allowing the vehicle to be stationary either with respect to the surface, or at the subsolar point. Design constraints for a Venus airplane will be discussed, and results of conceptual design studies presented. A Venus airplane is shown to be a feasible approach to Venus exploration, and such a mission may be possible within the constraints of a low cost mission.

Venus In Situ Explorer Mission Design using a Mechanically Deployed Aerodynamic Decelerator

Brandon Smith, Ethiraj Venkatapathy, Paul Wercinski, Bryan Yount, Dinesh Prabhu, Peter Gage, Lori Glaze, Charles Baker

The Venus In Situ Explorer (VISE) Mission addresses the highest priority science questions within the Venus community outlined in the National Research Council's Decadal Survey. The heritage Venus atmospheric entry system architecture, a 45º sphere-cone rigid aeroshell with a carbon phenolic thermal protection system, may no longer be the preferred entry system architecture compared to other viable alternatives being explored at NASA. A mechanically-deployed aerodynamic decelerator, known as the Adaptive Deployable Entry and Placement Technology (ADEPT), is an entry system alternative that can provide key operational benefits and risk reduction compared to a rigid aeroshell. This presentation describes a mission feasibility study performed with the objectives of identifying potential adverse interactions with other mission elements and establishing requirements on decelerator performance. Feasibility is assessed through a launch-to-landing mission design study where the Venus Intrepid Tessera Lander (VITaL), a VISE science payload designed to inform the Decadal Survey results, is repackaged from a rigid aeroshell into the ADEPT decelerator. It is shown that ADEPT reduces the deceleration load on VITaL by an order of magnitude relative to a rigid aeroshell. The more benign entry environment opens up the VISE mission design environment for increased science return, reduced risk, and reduced cost. The ADEPT-VITAL mission concept of operations is presented and details of the entry vehicle structures and mechanisms are given. Finally, entry aerothermal analysis is presented that defines the operational requirements for a revolutionary structural-TPS material employed by ADEPT: three-dimensionally woven carbon cloth. Ongoing work to mitigate key risks identified in this feasibility study is presented.

Session: Other

Systems Engineering of a Mission to Develop a permanently Crewed Lunar Base

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This paper summarizes a proposal for a Systems Engineering portion of a project to develop a Lunar Base necessary to sustain up to 6 astronauts/cosmonauts on an ongoing basis, to perform scientific research extending the National Laboratory concept of operation from the International Space Station (ISS) to the Lunar surface. Starting with this as a mission statement, a mission operational scenario was developed centered around multiple launches to launch and land on the Moon living and laboratory modules, life support, power and communication infrastructure and other necessary hardware required to support ongoing habitation and scientific exploration within the base and on the Lunar surface. The technology readiness level of the base infrastructure, anticipated level and type of scientific research and other aspects of the base operation will be a direct extrapolation of the functional and physical architecture of the ISS. In other words, in order to minimize cost and complexity and minimize risk, only processes, hardware and software with an ISS pedigree will be considered. This will allow for a start of the Lunar Base development no later than 10 years after program inception, planned for 2015.

The Systems Engineering effort proposed here as a part of a broader Lunar base development project follows an industry standard Systems Engineering process. Mission functional and operational requirements are developed, followed by a set of System Level requirements. The system requirements are generated by a process of functional requirements decomposition, taking into account candidate concepts for each major mission segment and subsystem. Where applicable, trade studies are conducted to pick the best baseline configuration, using a set of pre-established criteria. Finally, physical and functional architecture of the base is presented, along with the overall mission operational concept. The study also provides an analysis of major risks, a tier I plan and schedule, as well as a top level resource allocation (e.g. organizational structure, cost allocation, international partnership breakdown, work breakdown structure, etc.)

Systems Engineering of a Europa Life Exploration Mission

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This paper summarizes a proposal for a Systems Engineering portion of a project to develop and launch a mission to Europa with the goal of gathering scientific data that will help determine the existence of necessary conditions to sustain life, and the possible existence of life there. Starting with this as a mission statement, the operational scenario is developed, centered around two unmanned spacecraft – an orbiter and a lander. The first is to be sent 5 years before the lander and will act as an initial scientific platform that will analyze the surface of Europa, find a landing site for the lander and act as a communications relay between the lander and Earth. The lander will be launched using the Falcon Heavy type launch vehicle into an Earth orbit from which it will be sent on a trajectory to Europa. Once the lander arrives at Europa a soft entry landing similar to the Mars Science Laboratory (MSL) is utilized to land it on a predetermined location. The primary mission of the lander is to conduct robotic studies to detect the presence of a liquid-ocean underneath the icy crust, and to determine the surface composition and sub-surface structure of the icy crust. Also, the lander will provide basic scientific data about the atmosphere of Europa, that can be used to determine the presence of life there.

Following an industry standard Systems Engineering process, functional and operational requirements for the entire mission and the two spacecraft are developed, followed by a set of System Level requirements – one set each for the orbiter and the lander. The system requirements are generated by a process of functional requirements decomposition, taking into account candidate concepts for each major mission segment and subsystem. Where applicable, trade studies are conducted to pick the best baseline configuration, using a set of pre-established criteria. Finally, physical and functional architecture of the two spacecraft are presented, along with the overall mission operational concept. The study also provides an analysis of major risks, a tier I plan and schedule, as well as a top level resource allocation (e.g. organizational structure, cost allocation, international partnership breakdown, work breakdown structure, etc.)

Prototyping and improving the design of planetary deep atmosphere probe decent modules for testing

Amato, M., Joe Generie, Chris Ross, Drew Jones, et Al, NASA GSFC

Our team has created deep and shallow atmosphere probe prototypes for Venus, Saturn and Uranus. Our prototype design, fabrication, assembly and test has resulted in testing refinement and altered approaches. Our goal is to refine our preliminary flight designs to create a lower cost, lower risk, design which is well integrated with the instruments and built for testing and assembly approaches. The challenges for extreme environments or very deep probes or descent spheres are very different than shallow probes.