Phoenix – The First Mars Scout Mission

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Abstract—[1] [2]As the first of the new Mars Scouts missions, the Phoenix project was selected by NASA in August of 2003. Four years later, almost to the day, Phoenix was launched from Cape Canaveral Air Station and successfully injected into an interplanetary trajectory on its way to Mars. This paper will highlight some of the key changes since the 2006 IEEE paper of the same name, as well as activities, challenges and problems encountered on the way to the launch pad.

Phoenix "Follows the water" responding directly to the recently published data from Dr. William Boynton, PI (and Phoenix co-I) of the Mars Odyssey Gamma Ray Spectrometer (GRS). GRS data indicate extremely large quantities of water ice (up to 50% by mass) within the upper 50 cm of the northern polar regolith. Phoenix will land within the north polar region at 68.2°N, 233.4°W identified by GRS to harbor near surface water ice and provide in-situ confirmation of this extraordinary find. Our mission will investigate water in all its phases, and will investigate the history of water as evidenced in the soil characteristics that will be carefully examined by the powerful suite of onboard instrumentation. Access to the critical subsurface region expected to contain this information is made possible by a third generation robotic arm capable of excavating the expected Martian regolith to a depth of 1m.

Phoenix has four primary science objectives:

1) Determine the polar climate and weather, interaction with the surface, and composition of the lower atmosphere around 70° N for at least 90 sols focusing on water, ice, dust, noble gases, and CO2. Determine the atmospheric characteristics during descent through the atmosphere.

2) Characterize the geomorphology and active processes shaping the northern plains and the physical properties of the near surface regolith focusing on the role of water.

3) Determine the aqueous mineralogy and chemistry as well as the adsorbed gases and organic content of the regolith. Verify the Odyssey discovery of near-surface ice.

4) Characterize the history of water, ice, and the polar climate. Determine the past and present biological potential of the surface and subsurface environments.

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1. INTRODUCTION

The first of a new series of highly ambitious missions to explore Mars, Phoenix was selected in August 2003 to demonstrate the NASA Mars Program's effort at responsive missions to supplement the Program's systematic, long term planned exploration of Mars. These competed, PI-led missions are intended to be lower cost missions that are responsive to discoveries made through this systematic program of exploration. Mr. Peter Smith from the University of Arizona is the Principle Investigator for Phoenix. Peter Smith has a long history of Mars science and has been actively involved in the exploration of Mars from the Mars Global Surveyor through the development of the HiRISE telescope being flown on the Mars Reconnaissance Orbiter.

Phoenix "Follows the water" responding directly to the recently published data from Dr. William Boynton, PI [1,2,3,4] (and Phoenix co-I) of the Mars Odyssey Gamma Ray Spectrometer (GRS). GRS data indicate extremely large quantities of water ice (up to 50% by mass, Fig 1) within the upper 50 cm of the northern polar regolith. Phoenix, a re-flight if the inherited Mars Surveyor program 2001 lander, will land within this north polar region (65N – 72N) identified by GRS and provide in-situ confirmation of this extraordinary find. Phoenix will investigate water in all its phases, and will investigate the history of water as evidenced in the soil characteristics that will be carefully examined by the powerful suite of onboard instrumentation. Access to the critical subsurface region expected to contain this information is made possible by a third generation robotic arm capable of excavating the expected Martian regolith to a depth of 1m.

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Additionally, Phoenix will address several key areas in the preparation for human exploration of Mars (MEPAG section IV) [5].

This rich set of investigations is made possible through a selected set of instrumentation previously selected for the Mars Polar Lander and MSP 2001 missions and augmented by a Canadian Space Agency (CSA) provided Meteorological Station including a Lidar system. The mission timeline for the Phoenix investigation is shown in figure 2.

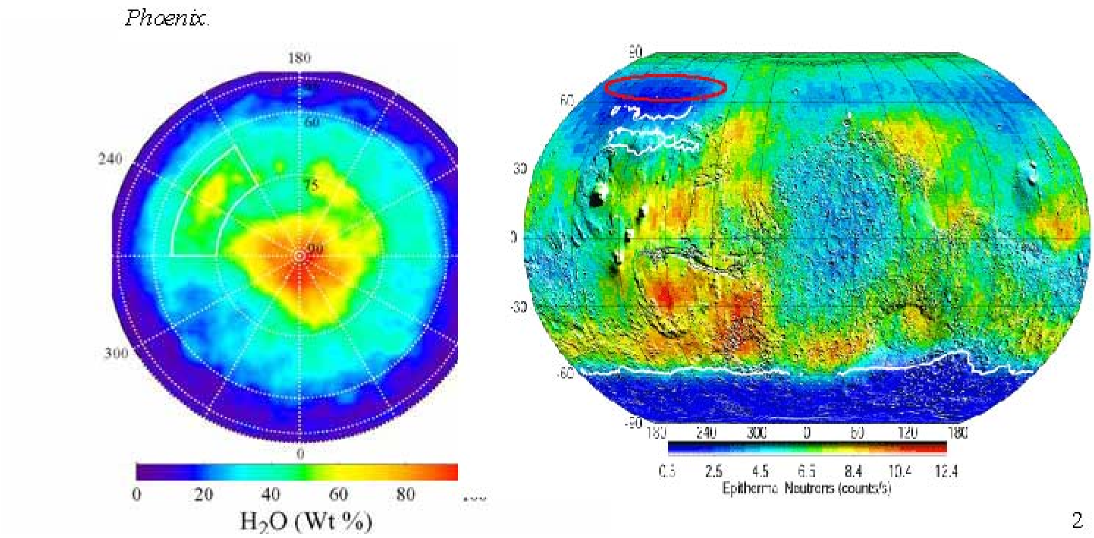

Figure 1 - Below left, recent GRS data identifying large quantities of near subsurface ice in the 70 North region of Phoenix interest. To the right, a global distribution of epithermal neutrons indicating water rich sites. The zone identified is a prime candidate landing site for Phoenix.

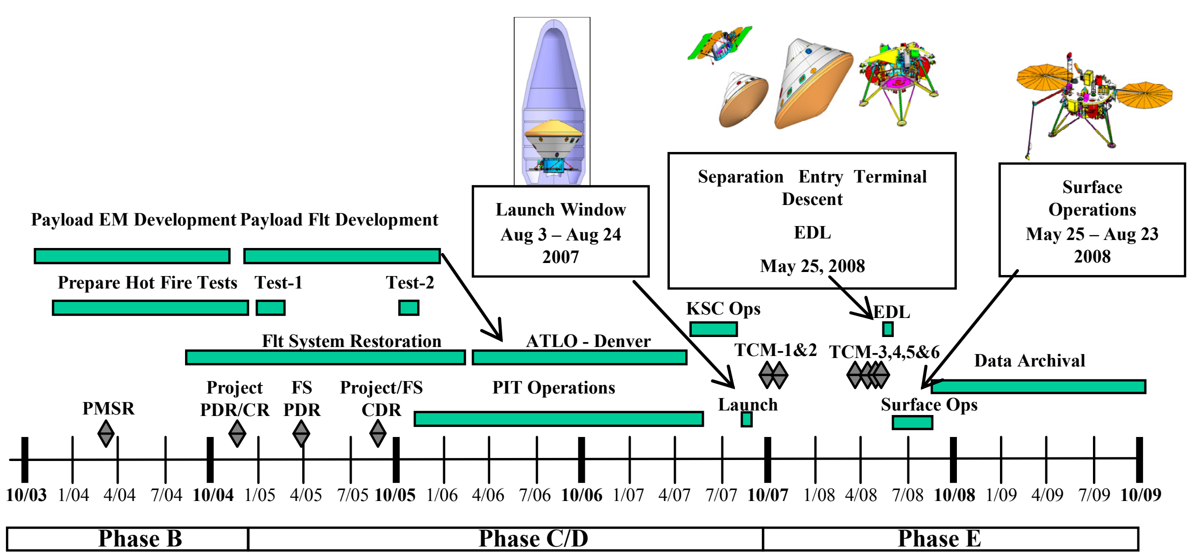

Figure 2 – Phoenix Mission Timeline

2. DEVELOPMENT PHASE ACTIVITIES

The development and test phase of the Phoenix mission lifecycle began with the Preliminary Design Review in February of 2005, followed by the Critical Design Review in November of 2005. Between these two events, several open design issues were resolved, including the design of the Entry, Descent and Landing (EDL) communications antenna and strategy, addressing the late cruise power margin (and associated need for an all stellar navigation mode or other power savings strategies), as well as some key EDL design features (see later section). The Assembly, Test and Launch Operations (ATLO) phase began in April 2006. Phoenix was successfully launched on Aug 4, 2007 from Cape Canaveral Air Station concluding the development phase of the mission. This section will cover each of these aspects of the development phase in more detail.

Phase C, Design Reviews

In the JPL/ NASA lifecycle, phase C covers the period wherein detailed design reviews must be successfully completed prior to initiating the assembly and test phase. Phoenix is somewhat unique in that the mission on which it is based was already past this phase when it was terminated in 2001 (MSP 01). As such, Phoenix is probably one of the most reviewed projects in JPL history. Due to the failure of the Mars 98 spacecraft, in addition to the normal PDR and CDR reviews, Phoenix was also subject to additional Return to Flight reviews chartered internally as well as directed by NASA Headquarters.

Within the project, PDR's and CDR's were held for each of the payloads and each spacecraft subsystems prior to the commensurate Project level review. Due to the degree of inheritance, each of the payload and subsystems had engineering model (EM) hardware at the time of the Project PDR and had completed substantial testing by the time of the Project CDR. This was a major risk reduction for the Phoenix implementation effort.

Design Changes

Several key hardware design changes were realized during the Phoenix development. As mentioned in the previous paper (2006 IEEE) both the hazard avoidance system during EDL and the X-band system for the lander were descoped. As shown in figure 3, the design of a wrap around antenna was initiated to fit directly onto the aft region of the backshell instead of the 3 switched patch antennas which was the baseline. This allows for continuous transmission of data from the entry vehicle even in the event that the lander were to lose attitude knowledge. This also eliminated the need for additional UHF switches and active control during EDL. The design is based on similar antennas used for ballistic missiles for a similar purpose.

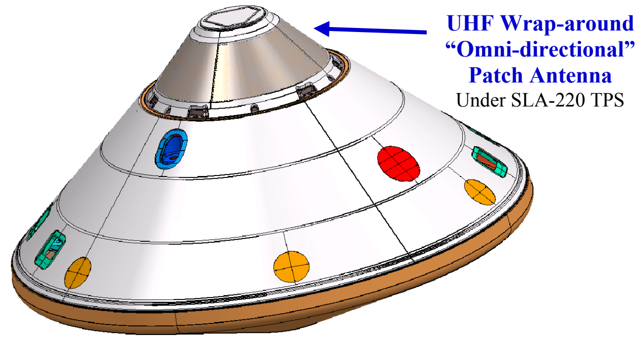

Figure 3 – *UHF Wrap-around Antenna*

The EDL communications design has been defined and tested end to end with the Mars Reconnaissance Orbiter (MRO) and Mars Odyssey test sets. Analysis tools have been developed to perform Monte Carlo assessment of link performance between Phoenix and the two orbiters. Link performance demonstrates 3dB plus 3 sigma margin across the entire EDL phase (Figures 4 & 5). Due to the loss of the Mars Global Surveyor mission during the development phase of Phoenix, tests were successfully performed with the European Mars Express (MEX) spacecraft, also in orbit at Mars, which will provide an independent acquisition of signal. Both MEX and MRO will perform Open Loop signal

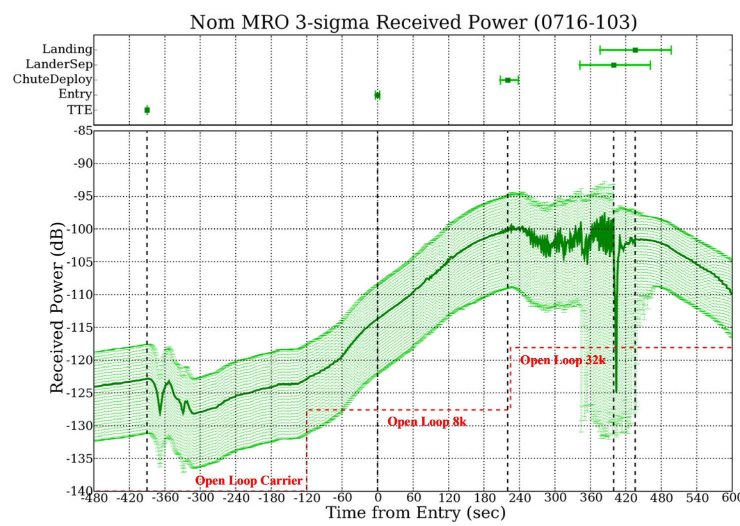

Figure 4 – *UHF link margin assessment for MRO during EDL*

acquisition which requires post processing on the ground to extract signal information. This process has also been developed and tested as part of the Phoenix design and test phase. Mars Odyssey (ODY) will switch from an Open Loop record mode from Cruise Stage Separation (CSS) through parachute deploy, at which time it will switch in synch with Phoenix to 32 kbps unreliable mode and will provide real time telemetry via a bent pipe operation of the remainder of EDL through post touchdown.

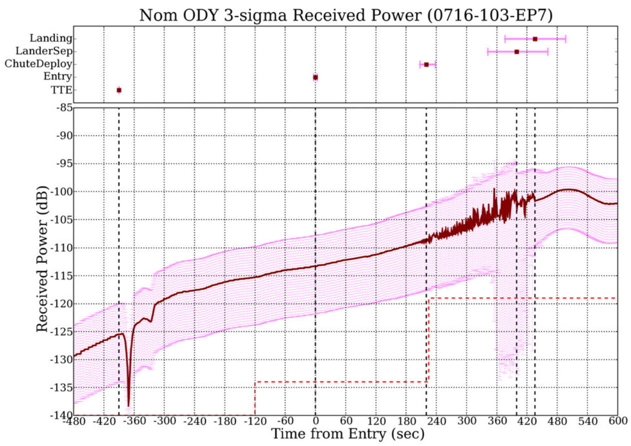

Figure 5 – *UHF link margin assessment for Odyssey during EDL*

While testing the landed solar array (Figure 6), it was discovered that at low temperatures, some of the gores, which are the stacks of the array, were sticking together and were unable to properly deploy. The solar arrays were modified to include Tedlar, film over the exposed adhesives surrounding the cells. Subsequent to the modifications, several test coupons were put into long term cold storage under a flight preload and tested to validate the modification. All coupons showed minimal stiction with the Tedlar addition and leave the array with 100% deployment margin.

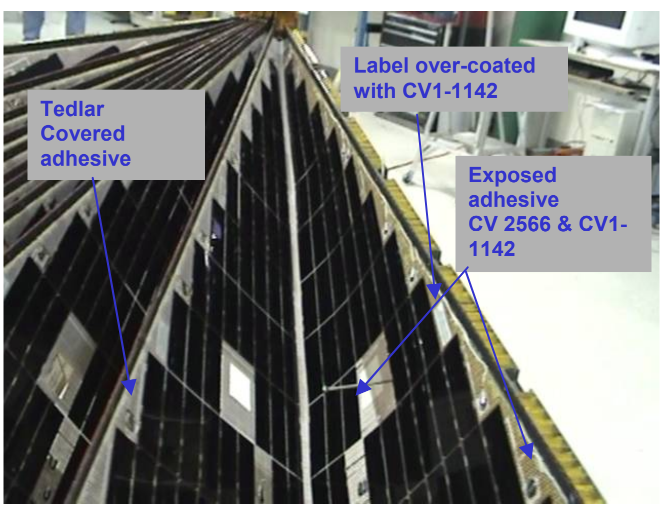

Figure 6 –Landed solar arrays with material modifications identified.

Another problem discovered with the landed solar array during deployment tests related to the tape used to pull the arrays out of the stowed position and into the fully deployed state. Several "keepers" are employed to maintain the tape in the desired plane during deployment and to keep the tape from coming off the deployment spool. During cold tests, a pre-existing flaw in one of the tapes caught on the keeper and failed catastrophically. As a result of the failure investigation, it was determined that the current keeper design imparted too much bending stress on the tape, and that the tape material was not adequately ductile. The keeper was redesigned and the tape was replaced with a more ductile material. All subsequent tests were successful.

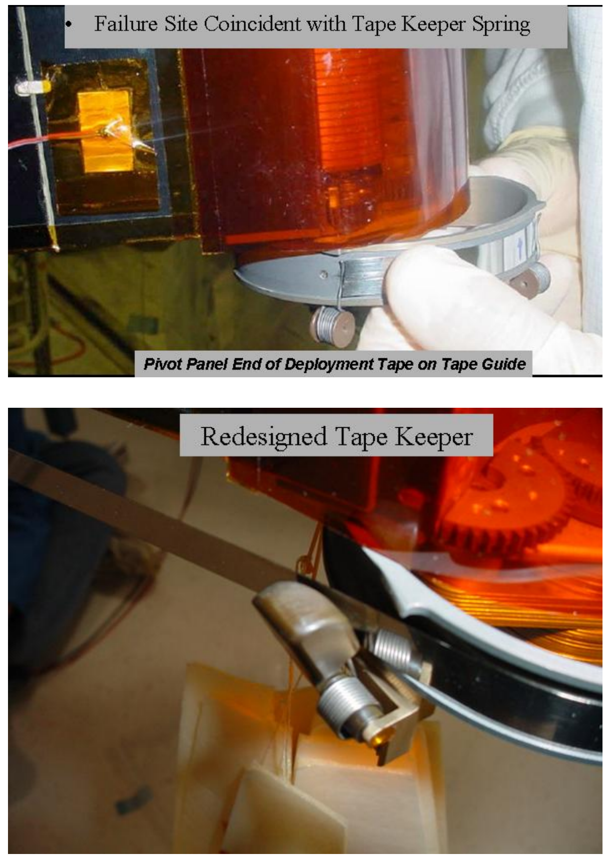

Figure 7 – *Landed solar array keeper mechanisms*

Within the payload domain, several payload features were added. On the tip of the deployable MET mast a lightweight "telltale", figure 8, was added that can be imaged by the SSI camera and provide some visibility into the wind velocity and direction. Provided by the Institute of Astronomy and Physics in Denmark, this late addition was added to the tip of the mast on the upper end cap. Calibrated at the University of Copenhagen's environmental chamber, it is believed that it will be sensitive enough to detect winds from 1-5 m/s with a resolution of 0.3 m/s.

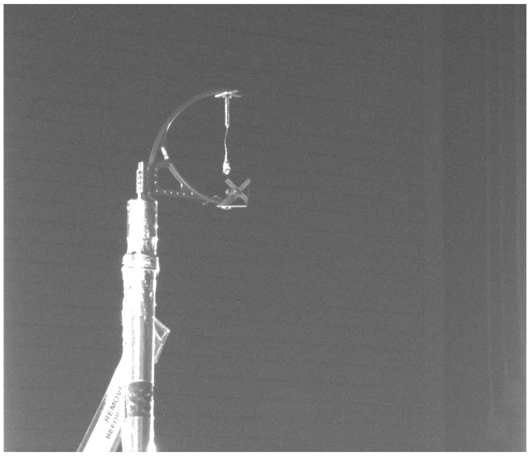

Figure 8 – *MET Mast wind sensor "Telltale"*

The Icy Soil Acquisition Device (ISAD) was the most significant payload design addition. Due to limitations on the robotic arm (RA) actuators and the potential for extremely hard compressive strength ice-soil compositions, the RA scoop was redesigned to accommodate a motor driven rasp that can bore into extremely high compressive strength materials and pass the cuttings into the scoop for delivery to the on board instruments for analysis. Designed, fabricated and tested by Honeybee Robotics of New York, the ISAD is comprised of a single motor and drive train that actuates and spins a tungsten carbide rasp bit. It is operated after the scoop has been firmly seated onto the material to be sampled. Several boring operations are required to obtain a sufficient sample size for instrument delivery. The ISAD can bore into >30 MPa material in less than 60 seconds. It is qualified for materials with compressive strength in excess of 45 MPa. The ISAD position on the back side of the RA scoop is highlighted in figure 9.

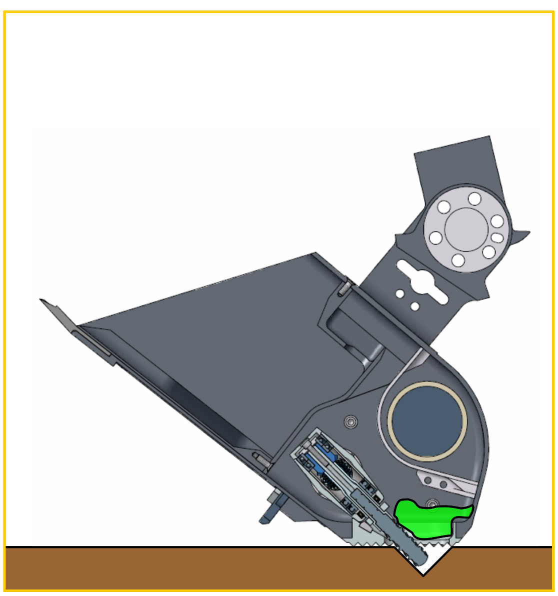

Figure 9 – *Robotic Arm Scoop with ISAD addition*

Due to the concern of molecular contamination during processing and flight, it was desired to add a reference source for calibration purposes in situ. The TEGA instrument with its mass spectrometer has the ability to detect organic molecules in the part per million range. If organic signatures are detected in acquired samples after landing, a sample from this reference will be acquired to determine if the measurement system or chain is contaminated and providing the positive indication, or if in fact the organic molecules observed may in fact be of martian origin. To help prevent confusion, another payload addition was the organic free blank (OFB), added to the robotic arm base-plate. The OFB is made of a specially formulated ceramic material that is void of organic materials and baked at >1000 degrees centigrade after forming to ensure no potential embedded organic molecules suvive. When sampled by the (ISAD), this should provide a minimum detectable threshold within the TEGA instrument for the purpose of organic detection.

Phase D, Test Program

The Assembly, Test and Launch Operations (ATLO) program for Phoenix began in April 2006 following the Project level ATLO Readiness Review in March. This program began with disassembly of the mostly fabricated lander structure while the electronics were still in assembly level testing. Disassembly of the core structure was necessary to accommodate minor modifications, updated testing and qualification, and to accommodate a different payload suite than was originally planned for MSP'01. In some cases, components were replaced completely (for example, the LiIon batteries and the descent thrusters were replaced with newer versions. The original star trackers were replaced with a more advanced version developed by Galileo of Italy for the MRO mission. In other cases, components were modified, such as the science deck adaptation with new inserts to accept a different payload and a different payload configuration, the component deck was modified to accept a redundant UHF radio and remove the X-band radio, additional heat pipes were added to the cruise stage to better regulate the temperatures of the cruise stage components.

The Phoenix spacecraft must undergo several configuration transitions during the mission. The first hard deadline faced by the ATLO team was the Cruise Thermal Vacuum Test in the Fall of 2006. To do this, the lander had to be modified and assembled sufficiently in advance of the test date to provide test margin. In addition, all spacecraft deployments were required to execute both prior to and after environmental testing to ensure exposure to the environments had no detrimental affects. This initial schedule for accomplishing this is shown in figure 10.

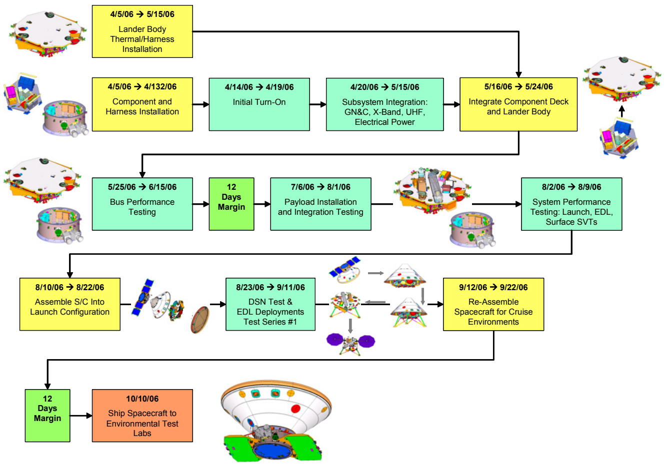

Figure 10 – *First phases of Phoenix ATLO program*

Dynamics Testing

The separation tests were performed as flight like as possible. New fixtures were constructed to allow the deployments to occur dynamically where possible. In the fully stacked configuration, the first deployment encountered is the Cruise Stage Separation (CSS). With the test fixture developed, the lander is held by support structure from below and the cruise stage is affixed to a spring loaded retraction mechanism. The spacecraft is instrumented with sensors to measure the forces and accelerations experienced by the various components. The normal on board sequence is executed with the inertial measurement units (IMU) operating in high rate acquisition mode and the pyrotechnic devices are fired that separate the cruise stage from the entry vehicle. The range of motion is sufficient to completely separate the electrical separation connectors. Both the pre and post environments CSS was successful.

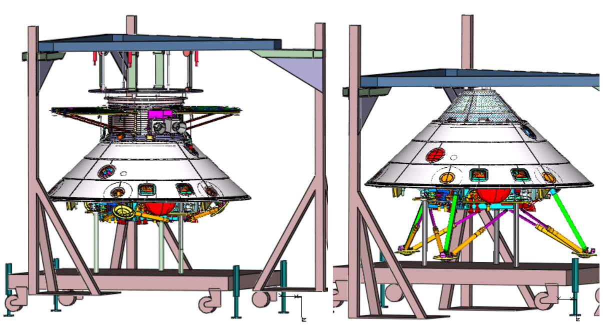

Figure 11 – *Cruise stage and backshell Separation Test Fixture*

In the nominal timeline, the heat-shield is deployed next, however due to the configuration limitations and holding points on the spacecraft, this test was performed independently. The test flow proceeded with the next deployment which was landing leg deployment. This was performed again using the flight sequence and with the IMU's and instrumentation collecting information. Each leg is deployed approximately 0.5 seconds apart. Leg deployment tests, both pre and post environments were successful.

The same test configuration is used for the last separation event, lander separation. Lander separation cannot be safely performed dynamically. Instead, the backshell is held in place by the overhead fixture and the lander is supported from underneath at the pickup points. The flight sequence is executed and the pyrotechnics are fired to execute the separation. The actual separation is effected by lowering the jack screws on the lower lander platform. This drops the lander away from the backshell slowly and approximately linearly.

The first lander separation test was successful; however the second test was not. During the lowering of the lander out of the backshell, the reaction control (RCS) thrusters which are scarfed through the backshell for use during cruise caught on the flexible quartz cloth that provides a plasma barrier between the exterior and the interior of the lander.

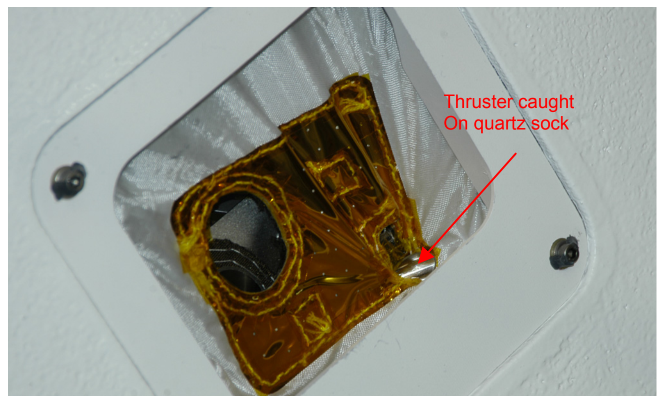

Figure 12 – *RCS & TCM thrusters scarfed through backshell*

Upon investigation, it was found that insufficient margin existed in the baseline design and the flexible sock that affixed to a plate around the nozzles allowed too much play and was non-deterministic. All 4 of the RCS thruster standoffs were bent during the test. These thrusters were removed from the lander at the valve interface and returned to Aerojet for refurbishment. Aerojet turned around completely rebuilt thrusters in 6 weeks and they were re-installed on the lander. In the meantime, a new Rocket Engine Module (REM) seal design was developed and tested in the AMES arc jet facility. This new design used a titanium finger tab approach to provide adequate plasma rejection, and flexibility between the lander and backshell to absorb flexing during dynamics, while also being significantly more deterministic. More dynamic and static clearance was also obtained in the updated design. The test was repeated successfully.

In parallel with the RCS thruster repair and REM seal redesign effort, additional problems with the Thermal

Protection System (TPS) material on the backshell triggered a full replacement. The material is a proprietary mix of cork, binder and other materials and comes in various densities and thicknesses. The material installed on the backshell at the time was not meeting the density requirements and the formulation as applied for the MSP'01 program could not be verified. The backshell was shipped to the Michoud facility (the same that applies TPS material to the space shuttle exterior tank) where the old TPS material was scraped off, and new material with the correct properties was applied. This rework effort was pulled off on an extremely short timescale and both the LM Denver and Michoud personnel performed heroically getting the backshell completely reworked, repainted and baked out in time to meet the reworked thrusters and REM seals in early April for a verification Lander sep test, just prior to shipment to the Cape, a few weeks later.

Finally, the heatshield separation test was performed two times as well. This first test was conducted prior to the systems environmental tests, but the last had to await final completion of the backshell rework necessary for the REM seal brackets and thruster replacements. In addition, minor rework was required on two of the 6 restraint mechanisms, as well as the o-ring seal and the flexible quartz cloth interface that was starting to show signs of wear after so many installations and removals over the previous 10 years. The test fixture allows spring preload to remove the weight of the heatshield and provides for sufficient travel to fully demonstrate dynamic performance and clearance. The final heatshield separation test was performed in the Payload Hazardous Servicing Facility (PHSF) at the Kennedy Space Center in Florida. The test fixture for this test was shipped to the clean room there, and the test was repeated successfully.

Environmental Tests

In order to ensure that the spacecraft will survive and operate as expected after exposure to the conditions of launch, deep space, Entry Descent and Landing, and on the Surface of Mars, a series of tests are performed on the spacecraft to simulate these environments. These were performed at the Lockheed Martin Environmental Test Facilty (ETF) in Denver, Colorado.

Starting in the fully stacked condition, including attachment to a test Payload Adapter Fitting (PAF) used to connect the spacecraft to the launch vehicle, the first tests performed included modal and electro-magnetic interference (EMI) testing. Modal testing is performed by instrumenting the entire spacecraft with load and accelerometer sensors, and then applying loads at various locations to the spacecraft dynamically and looking at the overall dynamic response. This allows the complex finite element model of the whole configuration to be checked against actual behavior, thereby validating the model and ensuring its ability to predict behavior during other stimuli.

While fully instrumented, the spacecraft is then subject to the acoustic environment it will see during the launch induced by the rocket engines on the launch vehicle, and the aerodynamic noise experienced in flight. This data is also similarly correlated to the finite element model.

Further simulating the launch effects, the pyrotechnic devices attaching the spacecraft to the PAF are fired twice and the shock response is measured each time.

The last set of tests prior to thermal vacuum include the launch-cruise EMI/EMF tests. This is a series of tests where the spacecraft is operated as expected during launch and shortly thereafter to assess whether there are any electromagnetic interference issues associated with the configuration. Within the ETL, radio frequency absorber panels were placed around the spacecraft in order to insulate the spacecraft from unwanted external sources. Special antennas are placed around the spacecraft to radiate the spacecraft with signals that it might see while sitting on the launch pad or in flight (radar installations, launch vehicle transmitters) and ensure that these signals do not affect the performance of the spacecraft.

Following this series of tests and in preparing for thermal vacuum testing, the spacecraft runs a set of System Verification Tests (SVTs). SVT's are tests that simulate key periods or events in the mission. In the fully stacked configuration, the launch, cruise and EDL SVT's were executed. These use the actual flight sequences as designed to perform these events, with hazardous activities prohibited either by software, by mechanical inhibits or by disconnecting the flight articles (like pyrotechnics) before

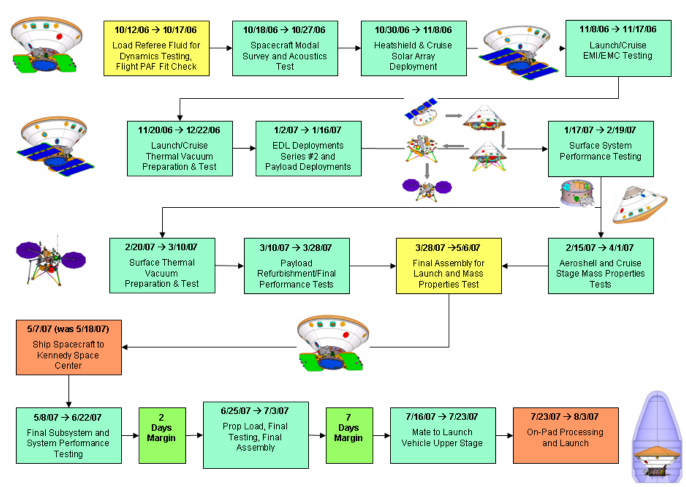

Figure 13 – *Final phases of Phoenix ATLO program*

execution. These tests help to assess whether exposure to the environmental tests had any unexpected detrimental effects.

Thermal Vacuum

Launch-Cruise thermal vacuum is the first of two planned tests of this type and are the most complex and expensive series of tests normally executed.. Testing is performed in the large (29' x 65') thermal vacuum chamber within the ETL. The solar simulator was not used for these tests as it can only be used from the top of the chamber facing downward (counter to the attitude in flight) and the gravity effects on the heat pipes required maintaining the spacecraft in the vertical position. Instead, infrared lamps and heaters were used to simulate the energy flux that would be incident on the spacecraft during these conditions. The spacecraft was heavily instrumented with thermocouples as a ground truth check of the spacecrafts own internal sensors, as well as to provide additional data not obtained normally.

The test starts by pumping down the chamber and exposing the spacecraft to high vacuum conditions. It is then subject to a series of hot and cold tests which simulate the maximum and minimums that might be seen in flight. Figure 17 identifies the thermal profile executed in the Phoenix launch cruise thermal vacuum test. The conditions are not exactly identical and do purposefully are not intended to achieve the worst case temperatures potentially posing a hazard to the spacecraft or its components, but

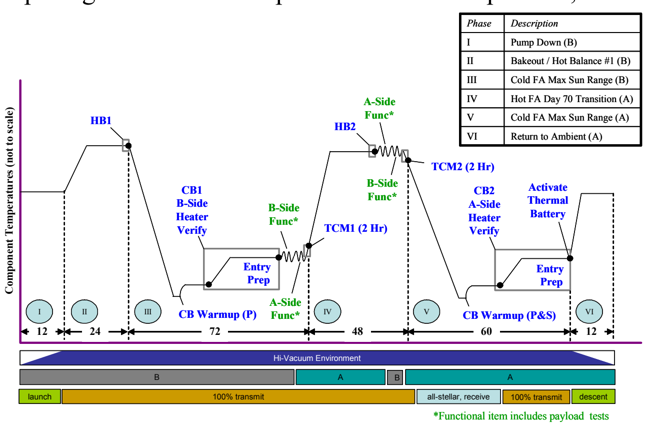

Figure 14 – *Launch / Cruise Thermal Vacuum test profile*

instead validate the spacecraft thermal model which can then be applied analytically to the worst case possible environments. In some cases, special "guard" heaters are applied to prevent items from getting too cold due to non-flight like conditions. Taking the spacecraft environments cold also allow for testing of the thermal controllers and heaters which otherwise would never actuate at ambient conditions. After going through the launch and cruise cases, the activities leading up to EDL are performed and the EDL sequence is executed. One of the findings during this phase of TVAC testing was some minor control issues with the propulsion lines and tank heater settings and these were later corrected (and re-verified in the subsequent landed test) A few problems occurred during the L-C TVAC tests, including but not limited to a blizzard at the Denver plant, but the test was completed in time for plant shutdown during the Christmas holidays. The spacecraft was left in the TVAC chamber for safe storage.

After completion of the launch-cruise TVAC tests, the spacecraft had to be reconfigured for Mars surface testing. This required a full suite of separations or deployments to go from the Launch configuration back down to the Surface configuration (see previous section on deployments).

Once down to the surface configuration with the lander legs deployed, the spacecraft was reinstalled back into the large chamber. Due to size restrictions, both landed arrays cannot be fully deployed in the chamber. For this test, the Plus (+) Y wing was removed and the minus (-) Y wing was deployed during the test. All payloads were similarly deployed within the chamber as part of the test, except for the Surface Stereo Imager (SSI) which requires an offload fixture to account for Earth versus Mars gravity and could not be made to reliably work unattended within the TVAC chamber.

The same philosophy was applied in the launch cruise TVAC tests as employed by the surface TVAC tests. These tests are intended primarily to exercise the spacecraft thermal control hardware, and to validate the spacecraft thermal model for this configuration. Unlike the launch cruise TVAC tests, the lander is not exposed to hard vacuum, but instead the chamber is maintained at 8 Torr of GN_2 (note that Mars conditions are 6-10 Torr of CO_2, so the difference in thermal constant properties had to be accounted for). Convective cooling adds another level of complexity in the landed conditions not present during cruise. In addition, the spacecraft will encounter diurnal cycles with extreme temperature swings each day from a low of -80 deg C to a possible high of +50 deg C (in the sun).

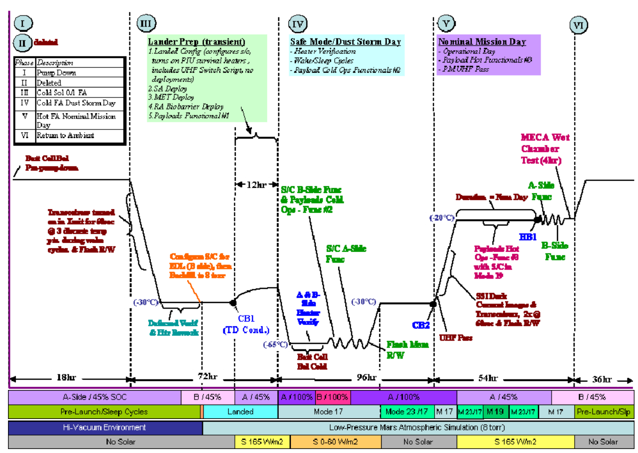

Figure 15 – *Surface Thermal Vacuum test profile*

The Surface TVAC tests were not as uneventful as the launch cruise tests. The initial activities all completed successfully, including deployment of the landed solar array, deployment of the MET mast, deployment of the bio-barrier and the robotic arm. During the payload checkout portion of the test, the TEGA instrument experienced anomalous behavior and the spacecraft recorded current spikes in excessive of the design value during the high temperature portion of the TEGA operation. This problem was later alleviated by providing a 3rd electrical switch to the TEGA instrument to provide higher instantaneous current capacity during operation.

An additional surprise was observed through the facility cameras within the chamber when the MECA instrument was running through its full test sequence on a wet chemistry cell. Due to an error in the sequence, the liquid water that normally gets injected into the wet chemistry cells along with the reagents to form solution was released prior to the cell being fully closed. The result was a visual flash observed as the water instantly flashed to vapor form as it would at that temperature and pressure. Due to the very small quantity and conditions, it was determined that this posed not threat to any of the spacecraft components, and the sequence was later fixed.

Thermal vacuum testing occurs around the clock, and can run for multiple weeks. The Phoenix team went the extra mile attempting to address and correct the various problems encountered during TVAC testing to keep the duration of the test as short as possible and to save testing costs. It was not unusual for people to repeatedly put in over 16 hours a day supporting the test, and their contribution should be duly recognized.

Cape Operations

The original plan at the start of ATLO had the spacecraft being shipped to the Cape on the 15th of May, 2007. Due to issues with the radar delivery and with the availability of the flight batteries for installment, as well as the need to repeat the heatshield separation test and the addition of several other tests not originally planned for execution there, the spacecraft ship date was moved forward 8 days, and was shipped on the 7th of May. The spacecraft was built back up into the fully stacked configuration and instrumented, then boxed into the MRO transport container. It was shipped to the Cape via C-17 transport from Buckley Air Force base (figure 16) directly to the space shuttle landing facility at Kennedy Space Center, and was then trucked from the landing strip to PHSF, all in a single though long day.

Figure 16 – *Phoenix Spacecraft being loaded to C-17 for transport to Kennedy Space Center*

At KSC, the spacecraft and all test equipment were verified clean and deployed in PHSF until installed on the launch vehicle. All electronic control systems and data acquisition systems were installed in the control room in the Multi-Mission Operations Support Building (MOSB). MOSB was also the location for all deployed personnel from LM and JPL providing offices and conference rooms over the 3+ months they were deployed there.

Processing at KSC took three main flavors:
1) Completing testing that would have otherwise been completed in Denver
2) Performing tests and installations that could only be performed at KSC, and
3) Activities and testing associated with the launch vehicle.

The first series of activities included post ship testing to ensure nothing was damaged during transportation. This was followed by heatshield separation testing, and final phasing and functional tests. In particular, the propulsion phasing and functional tests repeated tests that were performed prior to the thruster rework, thereby ensuring no errors were introduced during the rework, and preparing the system for propellant loading. A set of SVT's was performed on the spacecraft as well as an abbreviated set of tests called Baseline System Tests used to test the system after any hardware change or event such as transportation. The flight radar was installed at the Cape since it was not ready prior to shipping. This was performed and the system checked out successfully. The flight load of FSW (version 6.4) was also loaded at KSC.

The largest set of activities performed at the Cape included those items that could only be performed there. The first of which was the installation and first testing / conditioning of the flight batteries. These are considered hazardous currently by Cape and Range safety because they are LiIons and the industry is still learning their specific hazards and idiosyncrasies. This is followed by dry spin balancing which is performed to get the dry mass properties of the vehicle. Following that, end to end testing was performed with the Deep Space Network simulator located at the cape known as MIL-71. Lastly, a repeat of the EDL phase EMI/EMC testing was performed in the final flight configuration.

The flight ordnance was then installed (explosive devices, pyrotechnic actuators, etc). At this time the flight parachute and deployment mortar were installed. This again is a hazardous operation and restricted access was employed during this.

The spacecraft is then ready for propellant loading. The spacecraft was loaded to the maximum propellant allowed in the diaphragm tanks (85% by volume). This was done to less than 1% accuracy and after loading a concern was raised regarding the potential for expansion due to thermal increases around or during launch and the possibility of exceeding the specification on the tank bladders. Subsequent analysis and thermal controls indicated that this was not going to be a problem, but the margin was slim.

Propellant loading was followed by wet spin balancing of the spacecraft. The propellant tanks are about 75 cm off the spin axis and the propellant lines are not deployed perfectly symmetrically around the vehicle. The final wet spin balance allows for additional changes to be made to the CG and the principle axis of misalignment to meet the launch vehicle and EDL entry conditions.

At this point, the spacecraft is ready to fly. The following activities fall into the last category of Cape processing.

In parallel with the final spacecraft activities, testing and preparation, the launch vehicle 3rd stage, a Star 48 solid propellant motor from Thiokol is prepared and balanced as well. About 2 weeks prior to launch, the 3rd stage is transported to the PHSF airlock and prepared for mating to the spacecraft (figure 17). The spacecraft, without the heatshield installed (spacecraft lifts can only be performed with the heatshield removed in order for the crane to attach to the certified pick up points) was lifted over the 3rd stage and lowered into place on the flight PAF. The V-band that mates the two together for flight was then installed. Electrical connections between the spacecraft and the LV harness were performed to verify the integrity of the 3rd stage separation connectors. The entire assembly was then installed onto the Delta transport canister on a flatbed trailer. The assembly was covered with plastic and the canister sealed and a purge was applied.

Figure 17 – *Phoenix Spacecraft integration onto launch vehicle 3rd stage*

Transportation to the launch pad and lifting into the white-room for assembly onto the launch vehicle takes place on the same day. At 3:30 am on July 23rd, the assembly left PHSF on route to the launch complex 17-A. The assembly arrived at the launch complex at 5:15 am. The canister was then affixed to the mobile service tower crane and began the lift that would take it to the white room 9 floors up. At approximately7:45 am, about 15 minutes into the lift, rains and a stage II lightning warning suspended operations. The canister and enclosed spacecraft were left hanging on the side of the mobile servicing tower until the rains and lightning warnings finished at 9:30 am. The lift completed and the spacecraft and 3rd stage were mated to the 2nd stage. During the mating procedure, water was noted within the transport canister and on the exposed parts of the 3rd stage. Fortunately, the spacecraft was protected by the plastic sheeting placed over it prior to sealing of the canister.

Final end to end electrical checkouts were performed and the spacecraft BST was successfully executed. The heatshield was installed and the final closeouts were performed on the backshell access ports. All remaining Remove before Flight items were removed. The spacecraft was ready to fly. Launch took place at 5:26 am on August 4th, 2007, the first available day of the launch period, and on the very first attempt. The Delta II rocket lifted off on a clear morning for a pre-dawn launch and successfully injected the spacecraft onto its interplanetary trajectory to Mars.

Figure 18 – *Phoenix Spacecraft on top of Delta-2 rocket at KSC*

Figure 18 – *Phoenix launch, August 4, 2007*

3. ENTRY DESCENT & LANDING MATURITY

When the Phoenix Project was selected as the first Mars Scout, the overarching concept was to utilize existing hardware from the MSP'01 lander and use resources other projects might have for hardware development to facilitate an extensive test program. Obviously mission success is a measure of success of this effort. During the development phase another measure of success is the uncovering of and eventual mitigation of potential failure modes in the Entry Descent and Landing (EDL) system for Phoenix. As the Phoenix EDL system is architecturally similar to the Mars Polar Lander (MPL), uncovering these potential problems also identify failure modes from this mission. Figure (19) delineates the Phoenix EDL timeline highlighting the sub-phases of EDL, and an issue within each sub-phase which was addressed during the projects development.

Cruise Stage Separation Connectors

Prior to entering the Martian atmosphere, the Phoenix cruise stage which supports the cruise solar arrays, X-band communication hardware, and celestial sensors separates from the entry vehicle. The cabling between the entry vehicle and the cruise stage run through several pyro-initiated separation connectors. As part of the Projects separations test program, these connectors separation force were verified. During these tests, it was uncovered that under the cold temperatures that our thermal models predict for the separation connectors, the spring force margin for these connectors was inadequate. The solution to this problem was to add both shave some of the material between the two portions of the connectors, and to add heaters to these connectors to assure the temperatures at the time of separation is above the point where margin exists.

Cruise Stage Component Re-contact

After separation from the cruise stage the entry vehicle must assure adequate separation from the cruise stage as it continues down its trajectory thought the Martian atmosphere. This verification is accomplished via a re-contact analysis. This analysis utilizes models of the atmosphere, entry velocities (including dispersions), and the aerodynamic parameters of the two bodies' through the

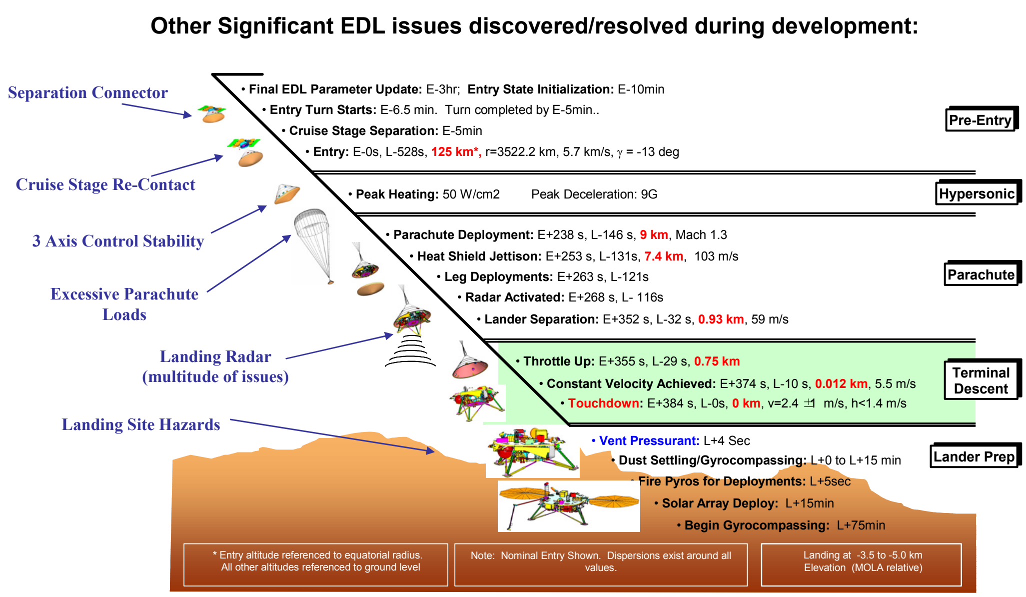

Figure 19 – *Phoenix EDL sub-phases*

entry profile. Initial looks at this analysis, which was conducted in a similar manner as previous Mars landing missions looked to have adequate margins. An additional concern for all Mars landers is Planetary Protection. This is a requirement which limits the number of potential biological spores on the vehicle which either land, or in other ways impact the planet. To certify our assessment of our bio-burden, and "burn-up / break-up" analysis of the cruise stage is conducted. In conducting this analysis the assessment showed that while the cruise stage structure itself would destruct during entry, many of the telecom components would not. While this turned out not to be an issue with regard to the bio-burden, it did reveal to the analysts relatively small pieces of the cruise stage would survive. Because of this finding, the re-contact analysis was repeated, and it was found that a "cloud" of pieces from the cruise stage, with significantly high ballistic coefficients, could actually catch-up to the entry vehicle, and thus present a danger during entry. To mitigate this concern, the baseline of the mission was changed, such that after cruise stage separation, which was advanced by two minutes for more separation, the entry vehicle conducts a deflection maneuver to move away from the cruise stage to alter its trajectory before entry.

Thruster Efficacy

One of the significant differences in the EDL design for Phoenix (and MPL) from the Mars Pathfinder, and Mars Exploration Rover missions, is that the entry vehicle is three axis stabilized during entry as opposed to spin stabilized. As such reaction control thrusters are used to maintain attitude. Aerodynamics engineers at Langley Research Center (LaRC) analyzed all of the effects of these thrusts on the entry vehicle. In assessing the control effects between mach 1 and 1.5, the assumptions used by all of the system models were shown to be invalid. The efficacy of the thrusters in this regime in some cases was shown to be in the opposite direction for the pitch and roll control axis. The effects in this low mach zone were correlated with both Apollo and Shuttle data. The assumptions used by the three axis control algorithms for MPL were called into question. The Project investigated potential solutions for this control mode. The first would have been to initiate a slow roll, thus effectively making the entry vehicle spin stabilized. Detailed analysis however showed that the stability of the entry vehicle in this regime was sufficient, even with worse case conditions, to turn off the control in pitch and roll. The implementation of this "use as is" fix was to increase the control dead-bands in these two axis to levels such that thrust would in effect be turned off unless threatening stability of the vehicle.

Parachute / Structural Loads

Early in the development of the Mars Exploration Rover, work with experts at LaRC revealed that inflation loads assumed during parachute deployment were significantly understated in models developed on the Mars Pathfinder Project. Since material and construction techniques has addressed the stress issues for the parachute itself on during the MER development, the concern for Phoenix focused on the lander structure which is exposed to these loads as well. As it turned out, with the new inflation assumptions, there was less than 10% load margin on the parachute cone which attaches the risers to the entry vehicle. The Project took two courses of action. First, do to the lower landing altitude for Phoenix, relative to MSP'01, we can increase our time on the parachute, and likewise reduce the parachute diameter by approximately 10%. This decrease in surface area translated into a liner decrease in the applied loads. In addition, the structure of the parachute cone was strengthened to provide significantly more margin. Several drop tests were conducted in Idaho, which generated significant inflation load margins, and demonstrated the Phoenix parachute has the capability to sustain the expected loads.

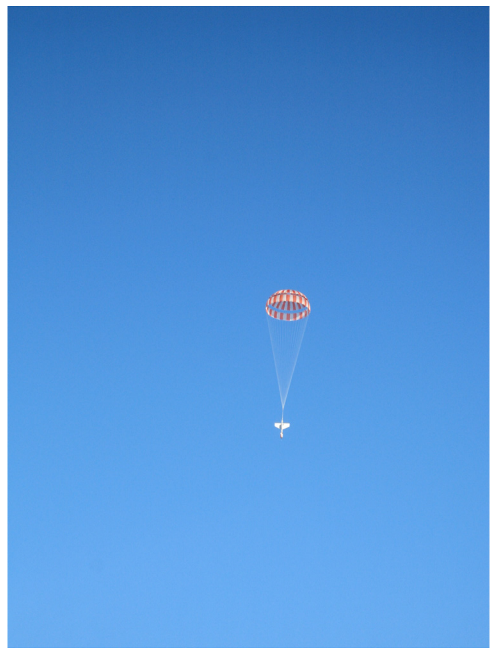

Figure 20 – *Parachute drop tests in Idaho*

Landing Radar Idiosyncrasies

The Phoenix EDL system utilizes landing radar to establish and maintain vehicle altitude, as well as synthesizing horizontal velocity. The radar used for this purpose is a derivative of military aircraft radar built by Honeywell. This same radar was used on the MPL mission, which has its history dating back to the early 1990's. Phoenix had developed an ambitious field test plan for the landing radar to assure its performance within our system. These tests included both captive carry and drop tests. Captive carry tests are conducted with the test radar hard strapped to a helicopter, and then a flight profile is performed over varying terrain. In these tests, the horizontal velocity is significantly larger than the vertical velocity. The Phoenix Drop test program required the generation of a pneumatically controlled drop vehicle, suspended from a helicopter. The control system developed for these drop tests were programmed to simulate the descent profile of the lander at Mars. These tests, which were conducted at the Dryden Research Center in California, are the most realistic tests of the radar as the vertical velocity more closely matches the expected ratio to horizontal.

The first of the captive carry tests was conducted in March of 2006. There were many idiosyncrasies noticed, so much that a special inter-organizational team was formed to assure the Project understood the phenomena we were seeing. This team, lead by JPL included team members from Lockheed Martin and Honeywell, the developers of the radar. The first set of anomalies, as it turned out were dominated by the fact that the firmware of the variant of this radar was "frozen" in the version built for MPL. Since that time, Honeywell had made several improvements, which had to be incorporated in our flight system. In utilizing a radar that was designed as an altimeter for an aircraft ($V_h >> V_v$) for a spacecrafts terminal descent sensor, it quickly became apparent to the team a more concerted system assessment effort was needed. To facilitate this effort it was important to augment the planned test program with two significant verification platforms; a detailed software simulation of the inner workings of the radar and a programmable set of test equipment so that we can simulate, with the hardware in the loop, multiple drops and get the response of the radar.

Figure 21 –*Radar drop tests at Dryden Research Center*

Honeywell was very accommodating in allowing the team the inner workings of their proprietary firmware, such that the software model was extremely accurate. This model was inserted into the EDL team's supercomputer simulation environment to run multiple 2,000 case Monte-Carlo simulations on end to end EDL. One of the system level idiosyncrasies, a radar ambiguity generated by locking on the entry vehicles heat shield after separation, and then falsely believing the ground was a lot closer by maintaining lock on the incorrect radar pulse, was initially discovered with this simulator. This anomalous condition was later verified utilizing the actual hardware, and delay lines specifically set to simulate this condition.

The programmable radar EGSE was created to allow multiple drop tests to be simulated with the hardware in the loop. In addition, it allowed us to 'run' drop tests while the hardware was inside an environmental chamber, thus allowing us to understand performance and margins around our expected conditions. Since the radar algorithms, incorporates temperature based lookup tables for calibration parameters, this latter step proved to be invaluable.

Overall, there were over 30 anomalous conditions identified in the use of the landing radar in our system configuration. Only one of these necessitated changing the hardware (the radar ambiguity resulted in the team doubling the pulse repetition frequency), however many flight software changes, and system timing changes were incorporated which eliminated many potential failure modes which may have been the cause of the MPL loss.

Landing Site Assessment

Landing site selection is a joint effort of the engineering and science teams. The concern of science is to assure that the site chosen maximizes the potential for meeting all of the level-1 science requirements of the mission. The concern of all is that the environmental conditions are survivable for the lander given the system design envelope. The process involves a series of down selection of regions on the planet, within the latitude band identified by the Odyssey spacecrafts Gamma Ray spectrometer instrument. In March of 2006, the Mars Reconnaissance Orbiter arrived at the red planet equipped with the HiRise camera. This camera brought groundbreaking resolution to remote imaging of the surface resolving 1 meter objects on the surface with clarity. As the Phoenix landing site team had progressed in narrowing down the region of choice, they had concentrated on region B (figure 22). Then, in October of 2006

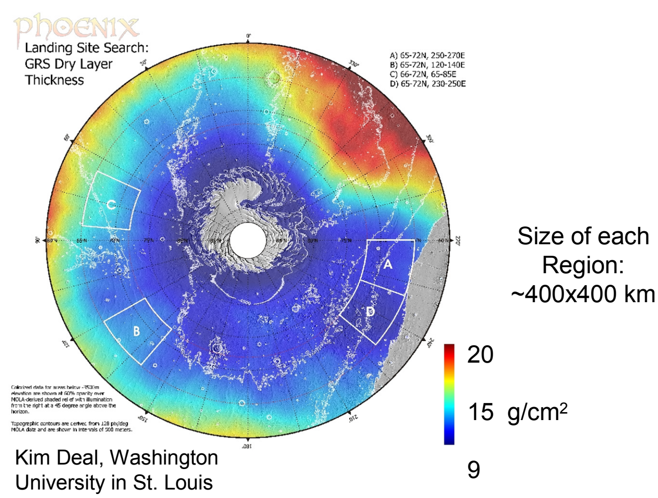

Figure 22 –*Mars Polar Map with Phoenix candidate landing regions*

HiRise took the first images of region B. Figure 23, along with its humorous quote from the HiRise team revealed boulder fields which were a significant cause for concern. The landing site team, along with the support of the HiRise imaging team, then embarked on an extensive search in the other targets for more benign landing sites for Phoenix. With global infrared imaging available from the Thermal Emission Spectrometer (TES), the team was able to correlate the HiRise data with TES data to extrapolate where would be potential boulder free zones. These zones were identified, and then a mapping within those zones with HiRise helped confirm the zones were indeed clear of boulder fields at the TES data might imply. By years end, the team had identified three potential landing boxes within the boarder of regions A and D, and safe havens. Figure 24 delineates the current landing site for Phoenix, with the highlighted rectangles identifying those areas mapped by the HiRise camera. The detail shown in Figure 25 shows the major difference from our original concerns in region B. The so called "green-valley" region (Figure 26) is the teams current target, so named by the color coding used to map the site, where green was the lowest density of rocks and boulders. As Mars entered its northern winter darkness covered our landing site in March of 2007, however starting in January of 2008 we will be able to continue imaging of the landing regions. The strong correlation between TES data and HiRise imaging has given the team confidence that we have found the proper landing site for our vehicle. Formal certification of the landing site will be confirmed at a Project review in April 2008.

Figure 23 –*First images of landing region B unveil significant boulder fields*

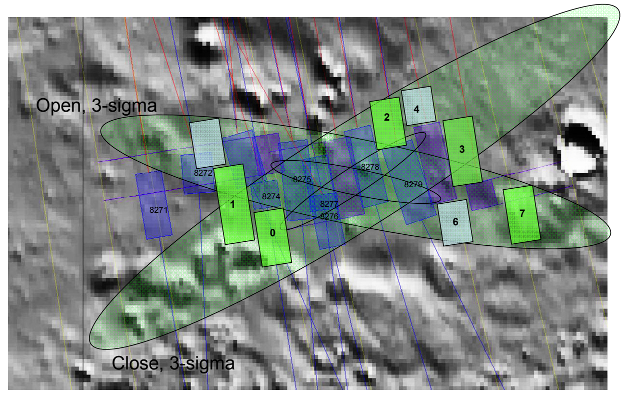

Figure 24 –*Current selected landing site with HiRise imaging coverage*

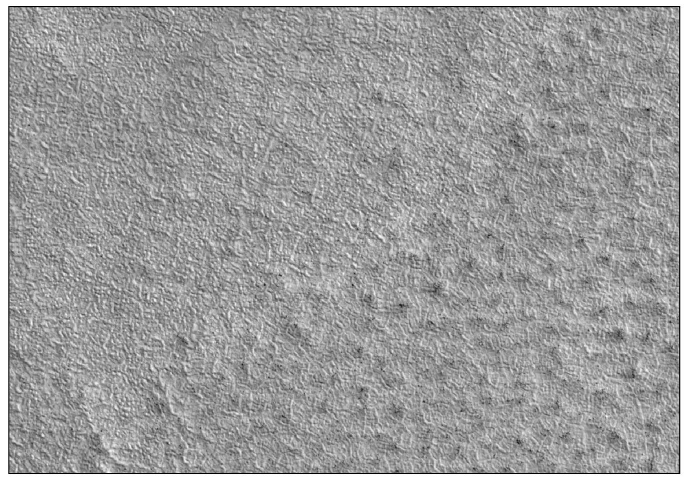

Figure 25 –*Lowland bright / lowland dark regions of selected landing box*

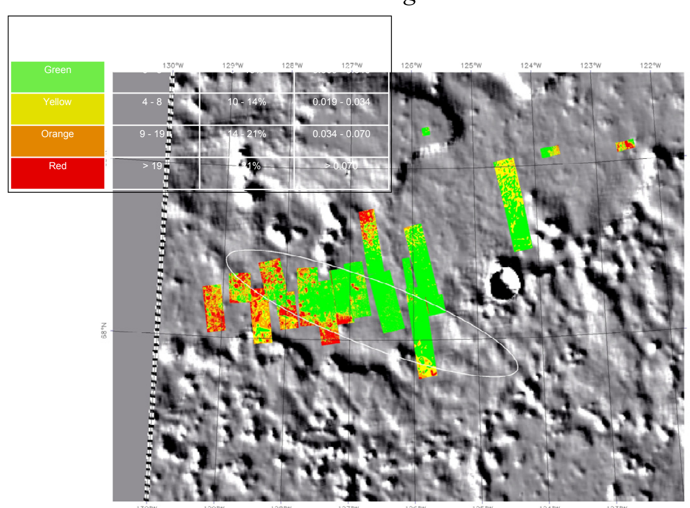

Figure 26 – *"Green valley" with landing ellipse overlay*

4. Conclusion

Phoenix is a science rich, relatively low risk kickoff to an exciting series of new NASA missions to Mars. Phoenix has the capability of obtaining key critical science information that could write whole new chapters in our current understanding of Mars. As with all planetary missions, our implementation phase was fraught with challenges which were overcome by the hard work and dedication of a very talented team. After launch on August 4, 2007 this team's focus shifted to the operations of the spacecraft on its way to Mars, as well as continued robustness testing and training of the flight team through a series of Operational Readiness Tests. On May 25, 2008, Phoenix will land on the northern polar plains, and the potential for ground breaking, figuratively and literally, discoveries have the potential to be in our future. The success of EDL is a direct reflection of the attention the team place on our verification and validation program. While EDL is still one of the most challenging activities on the planetary exploration business, our team has worked diligently to leave no stone unturned in an attempt to maximize the probability of success.

The research described in this (publication or paper) was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

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BIOGRAPHIES

Barry Goldstein is the Project Manager for Phoenix. Barry graduated from the University of Colorado in 1981, with a degree in Mathematics with a minor in Physics. He also has received and Executive Masters of Business Administration from the Peter Drucker management center at Claremont Graduate School.

Barry started his career at the Jet Propulsion Laboratory in 1982, and has worked on various deep space projects for 22 years. He started his career as a flight designer working in the Attitude and Articulation Control subsystem for the Galileo, Jupiter orbiter spacecraft. For the past nine years, Barry has been focusing on Mars missions. He was the lead system engineer for the Mars Volatiles and Climate Surveyor payload for the Mars Polar Lander. After this devolvement, he was appointed manager for the Athena Payload, which was to be core science for a rover in 2001. Barry also led the initial Mars Scouts study team, which proposed a series of small landers to investigate multiple sites of the red planet. Prior to his assignment on Phoenix, Barry served at the Deputy Flight System Manager for the hugely successful Mars Exploration Rover project.

Robert Shotwell is the Project Systems Engineer for Phoenix. Robert graduated Magna Cum Laude from Texas A&M University in 1995 with a BS in Aerospace Engineering and minor in Math and Physics. He received his MS in Astronautics from USC in 2003.

Robert has worked at the Jet Propulsion Laboratory since 1993. He began as a cooperative education student working Propulsion Operations for Galileo, then transferred to Mars Pathfinder where he was responsible for detailed design, assembly and test of the propulsion system. He then became the lead for the Xenon Feed System for Deep Space 1, and supported mechanical integration and test of DS-1. Robert then spent two years in Advanced Propulsion R&D working on Arcjets, Ion Engines, Hall Thrusters, MPD thrusters and the like, before transferring into Systems Engineering. He was the lead Systems Engineer for Outer Planets studies during the Decadal Survey, while working to develop and promote microsatellite capabilities at JPL. Robert was the Project Systems Engineer for several DARPA proposals, then became the Proposal Manger for a Mars Scout Step 1 proposal after the Scout AO was released. Robert then joined Barry Goldstein during the Scout Step II proposal phase, and later became the Project Systems Engineer for Phoenix after Phoenix selection.