THE MARS PATHFINDER SYSTEM LEVEL SOLAR THERMAL VACUUM TEST

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ABSTRACT

The Mars Pathfinder (MPF) spacecraft, launched in December 1996, is the second launch in NASA's Discovery Program. The MPF mission is primarily an engineering demonstration of key technologies and concepts for eventual use in future missions to Mars. As part of the Discovery Program, this mission was the first to be implemented with a cost cap of $ 150M (FY 92) and with a maximum three-year development cycle.

The Mars Pathfinder flight system consists of three essential elements: a Cruise Stage, that supports the solar panels the Propulsion subsystem and the cruise stage electronics; an Aeroshell and Deceleration Module; and a Lander that carries the Sojourner microrover.

A restricted budget and limited schedule affected the scope of the thermal verification test program. The system level test was the first and only verification of the thermal design of the spacecraft cruise stage and cruise stage components.

One of the main challenges of the thermal environmental test program was the need to simulate all three mission phases: (a) cruise, (b) entry, descent and landing and (c) Martian surface landed operations, each with different thermal environments.

Another test program challenge involved solar load simulation of a spinning spacecraft at off-Sun angles ranging from 0° to 60°. A novel approach of using a combination of infrared lamps and a solar simulator was implemented for that purpose.

This paper concentrates on the first phase of the system level solar thermal vacuum (STV-1) test for the Mars Pathfinder Flight System, which covered the cruise and EDL phases of the mission. It will also discuss the pre-test design and spatial temperature mapping of the infrared lamp system, a proof-of-concept test for an accelerated spacecraft cooldown, and STV-1 test results.

INTRODUCTION

Spacecraft Mission

The Mars Pathfinder spacecraft, launched December 4, 1996 by a Delta H launch vehicle, is the second launch in NASA's Discovery Program. It was designed to demonstrate technology for inexpensive entry and landing on Mars and became a technology demonstration for landers in the Mars Surveyor program. Most of the entry and descent technology demonstrated by Pathfinder will be used on Mars Surveyor 98.

Mars Pathfinder was developed under the "Faster, Better, and Cheaper" way of doing business with 3 years for development and at a cost of under $150 million dollars, in FY 92 dollars.

The mission consists of four essential phases: the launch phase; the cruise phase; EDL phase; and the surface operation phase. After liftoff the spacecraft is to be directly injected into a ballistic trajectory for cruise from Earth to Mars (Figure 1).

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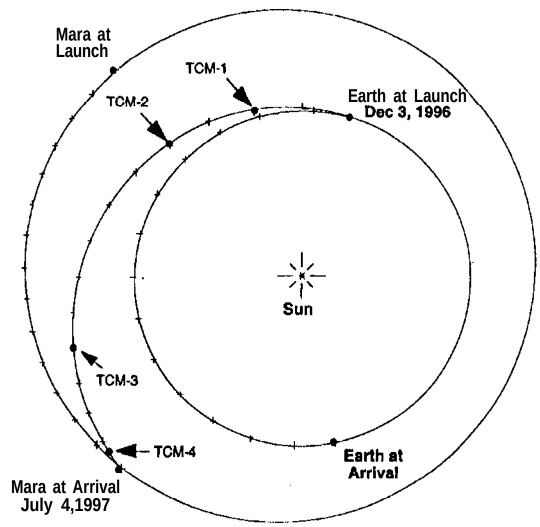

Figure 1. Mars Pathfinder Cruise Trajectory

During its cruise from Earth to Mars the spacecraft spins at a rate of two revolution per minute and points within 5° of Earth to facilitate effective communication with ground stations. The spacecraft Sun angle during cruise increased from its initial injected Sun angle, up to 60°, and then gradual] y decreased to 0° at the time of conjunction in early February, 1997. After conjunction, the Sun angle gradually increased from 0° to410 at Mars arrival.

Mars Pathfinder will arrive at Mars on July 4, 1997. At the end of the Mars cruise, the spacecraft will be maneuvered to a proper entry angle and the cruise stage will be jettisoned. It will then enter the Martian atmosphere direct] y from its approach hyperbola at about 7600 m/s without going into orbit around the planet. During the ninety-five minute EDL phase, the entry module will first go through about one minute of entry aerodynamic heating, followed by the deployment of the parachute deceleration subsystem (Figure 2), jettisoning of the heatshield, lowering of the Lander on a 20 meter bridle, firing of the RAD motors, inflation of the airbags, cutting of the bridle, touching down of the Lander on the Martian surface and deflation of the airbags. After landing, the airbags will be retracted and the petals and the Rover deployed. The Mars Surface operations are planned for one month with an extended mission of one year.

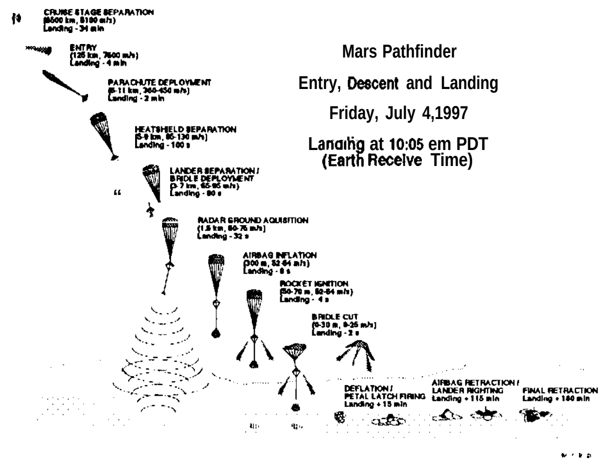

Figure 2. Entry, Descent and Landing

The thermal environments during EDL are dominated by the entry aerodynamic heating and the Mars upper and lower atmospheric conditions. On the surface the Martian diurnal ambient conditions are the thermal environments for the Lander and Rover.

Spacecraft Description

The Mars Pathfinder Flight System consists of three essential elements: the Cruise Stage, the Deceleration Module and the Lander with the Rover. A photograph of the spacecraft in its full up cruise configuration is shown in Figure 3.

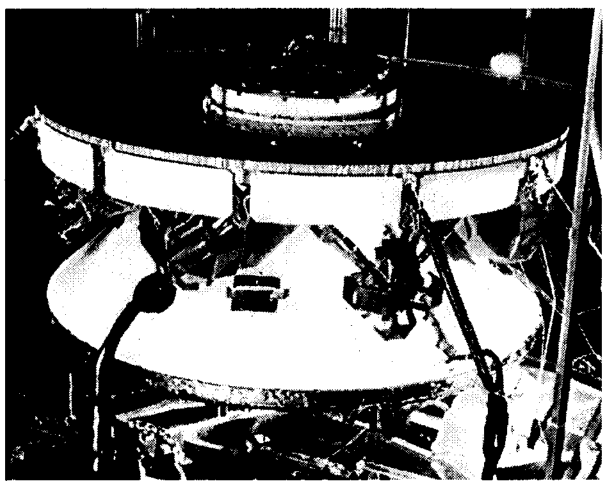

Figure 3. The Mars Pathfinder Flight S ystem

The Cruise Stage houses the main spacecraft structures that are specific to the cruise phase. Mainly it houses the cruise power, propulsion and attitude control subsystems as shown below:

- . Power subsystem: cruise solar array, shunt radiator and shunt regulator
- Propulsion subsystem: hydrazine tanks, thruster clusters, propellant lines and propulsion distribution module
- Attitude Control subsystem: Sun Sensor heads and electronics and Star Scanner,

Additionally, the Cruise Stage supports the Heat Rejection System (HRS) radiator and its Integrated Pump Assembly. In an effort to avoid hardware duplication and save mass, most of the electronics, used during cruise and landed operations, are contained inside the Lander. A small portion of the electronics, that are dedicated to the cruise phase, are kept in the Cruise Electronics Module. A Medium Gain Antenna located on the top of the Cruise Stage is used for spacecraft communication with Earth.

The entry module consists mainly of the Aeroshell (the heatshield and the backshell), the RAD Motors and parachutes, the Thermal Batteries, the Pyro Switching Unit and the entry science instrument, the Atmospheric Instrument Package. The exterior of the backshell and heatshield are also shown in Figure 3.

The main structure of the Lander consists of a base petal and three side petals covered by the Lander solar arrays. The main spacecraft equipment are mounted on an equipment shelf which is structurally tied to and thermally isolated from the base petal. The main equipment and the equipment shelf are housed underneath a insulated thermal enclosure, the Insulated Support Assembly (ISA). Items on the equipment shelf include a secondary battery, the Integrated Electronic Module which houses the Attitude and Information Management and other electronics, the telecommunication subsystem, mainly, with a Deep Space Transponder, a Solid State Power Amplifier, a Auxiliary Transmitter, a Command Detect Unit, and a Telemetry Modulation Units, and the Lander mounted rover equipment, The deployed Lander is shown in Figure 4.

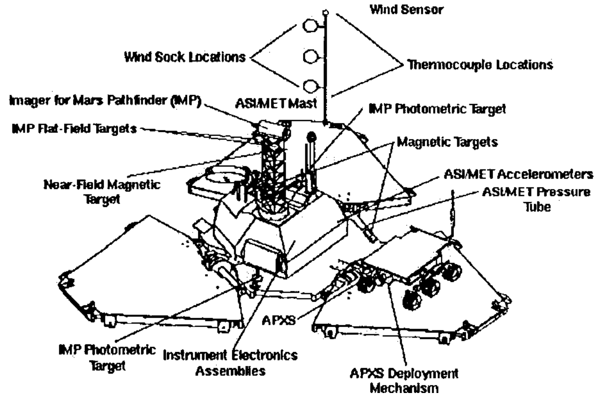

Figure 4. MPF Deployed Lander

The Lander also carries the majority of the EDL equipment: the airbags, the Gas Generators, the Lander Petal Actuators and their electronics, the Airbag Retraction Actuators, the Radar Altimeter and antennas, the Descent antenna, the Lander Thermal Batteries, the Lander Pyro Switching units the science instruments: Mars Pathfinder Imager, the atmospheric structure instrument meteorology package and the Rover which carries an alpha proton x-ray spectrometer and CCD cameras. The other equipment on the Lander, outside of the ISA, includes the High Gain Antenna assembly and the Low Gain Antenna.

<u>Thermal Desire Drivers and Description</u>

The basic philosophy of the Mars Pathfinder thermal design followed a low cost approach in which the thermal requirements was derived more from the equipment capability than from the pre-defined desirable performance needs.

In addition to the thermal environments in which the spacecraft must operate there were several unique Mars Pathfinder conditions and mechanical configurations that eventually drove the spacecraft thermal design:

- Most of common critical equipment on the Lander (electronics and battery) was shared between cruise and surface operations.

- The shared equipment had to be well insulated to survive the frigid Martian ambient conditions and, during cruise, it is required to operate constantl y in vacuum.

. The Lander secondary battery needs to be kept cold during the cruise period, to promote life-time reliability, but must be kept warm on the cold Martian environment for a more effective charging and discharging operations.

. During EDL, the entry module is subjected to extreme aerodynamic heating.

For the entire mission duration there are four key temperature control elements that maintain the spacecraft within the required temperature limits:

- The Heat Rejection System (HRS) - The HRS is an active, single-phase liquid freon fluid loop, for the rejection of excessive heat dissipated by equipment inside of ISA to a radiator on the cruise. The HRS also provides thermal control for the shunt limiter controller and the Rover during cruise. The HRS consists of four main elements: (1) The Integrated Pump Assembly which includes redundant motor-pump units, control valves and motor control electronics; (2) the radiator on the cruise stage; (3) the freon transport tubes and the associated plumbing; and (4) the heat exchanger tubes on the equipment shelf. An HRS of this type is being flown for the first time on the Mars Pathfinder Flight System (1, 2). The freon will be vented from the system just prior to EDL and the HRS tubes will be cut before cruise stage separation.

. The Aeroshell Thermal Protection System - The aeroshell consists of a heatshield and a backshell. Its main purposes are: to maintain aerodynamic stability during entry, to provide mechanical and thermal protection for the Lander and its equipment during entry and to provide an outer thermal enclosure for the Lander and EDL equipment during cruise. The aeroshell is made of an inner honeycomb structural layer and an outer layer of ablative material, SLA561V.

- The Insulated Support Assembly (ISA) - The ISA houses the major electronics, the battery and telecommunication equipment and helps in controlling their temperatures within the general range of -40°C to +40°C during Martian surface operations. It is a thermally insulated enclosure made of an Eccofoam layer and a Nomex honeycomb structural layer filled with crushed Eccofoam material.

- Other standard thermal control tools (paints, multi-layer insulation blankets, heaters and thermostats) were employed to maintain the spacecraft temperatures within allowable flight temperature (AFT) limits.

TEST OBJECTIVES

The primary STV- 1 test objectives associated with the system thermal design verification were:

. To demonstrate that the spacecraft thermal design meets the temperature requirements over expected mission- extreme environments.

. To obtain thermal test data to be used for determining design corrections.

- To obtain thermal data for using in correlating the analytical models.

Other objectives were:

. To verify that the spacecraft operates within specified performance requirements in the simulated space environment and within Flight Acceptance Limits (Functional and Margin Tests)

. To verify flight temperature sensor readings. Verification was done by comparing flight temperature sensor readings with thermocouple readings at the end of the thermal balance test cases.

SYSTEM LEVEL SOLAR THERMAL VACUUM TEST CHALLENGES

Restricted budget and schedule limited the thermal verification test program to only two thermal tests: (a) a thermal characterization test of a development Lander and the Heat Rejection System, and (b) a full-up, flight system solar thermal vacuum (STV) test. Because the first test covered only the lander thermal design verification, the STV test was the first and only verification of the thermal design of the spacecraft cruise stage and cruise stage components prior to launch.

The STV test program for Mars Pathfinder was developed and executed under the same challenging (faster, better, cheaper) guidelines that the rest of the spacecraft was under. Due to the nature of the spacecraft and the mission, the test was divided into two parts: STV- 1 for the cruise and EDL phases and STV-2 for the landed phase, each with very different thermal environments. This paper focuses on the STV-1 test,

The next three subsections of this paper discusses test design challenges encountered during the planning phase of the STV- 1 test program.

Simulating Off-Sun Conditions

The cruise phase of the test had to be designed to simulate the solar loads on a spin stabilized spacecraft that cruises at off-Sun angles ranging from 0° to 60°. Different angles at the same heliocentric distance result in worst-case thermal environments for different areas of the spacecraft. A novel approach, using a combination of infrared lamps and a solar simulator, was implemented to simulate spacecraft off-Sun conditions.

Minimizing Test Duration

The time allocated to the test was restricted to no more than 15 days with a 5 day contingency. The test was planned and designed to be completed in 11 days without chamber break. The additional 4 and the contingency where kept in the schedule to be used if a chamber break was necessary.

Designing a test that would fulfill all the objectives in 11 days was a challenge. The goal of minimizing non-essential test time (i.e. cooldown and warm-up) was achieved by not simulation the mission thermal timeline and by accelerating the spacecraft cooldown and warm-up.

The mission thermal timeline, starting from a hot environment near Earth and ending with a cold environment near Mars, was not followed during the test, Instead the test started with the cold Mars simulation and stepped up to the near-Earth warm environment. The change in the timeline was acceptable since the transient information on the spacecraft cooldown was not an objective of the test. Testing time was significantly reduced by accelerating the cooldown during the initial pump down phase and by accelerating the warm up periods between steady-states utilizing strategically placed test heaters and IR lamps.

Accelerating the Spacecraft Cooldown

The accelerated cooldown strategy was conceived to minimize the time needed for the spacecraft to achieve its first steady-state case - a worst-case cold condition at 0° degrees off-Sun. The cool down procedure began with pumping down the chamber to a hard vacuum, cooling down the shrouds to - 135°C and then back filling the chamber with GN_2 to a pressure of 8 torr. The cold walls and 8 torr pressure were then maintained until the spacecraft reached temperature levels specified by pre-test analysis. The addition of an 8 torr GN_2 atmosphere in the chamber allowed gas conduction and free convection effects to accelerate the spacecraft cool down. When the desired temperatures were reached, the chamber was again pumped down to a hard vacuum and the shrouds were flooded with LN_2. The design of the accelerated cooldown was extremely challenging from the point of view of the spacecraft safety and chamber integrity.

In order to accelerate the cooldown of more massive areas of the spacecraft, like the aeroshell, and insulated areas, like the Lander, less massive equipment had to be closely monitored and controlled with test heaters to avoid violating minimum temperature limits.

Chamber integrity was also an issue during the cooldown since the 25-foot vacuum chamber was not designed to hold pressure and cold wall simultaneously. Overcooking of the chamber pressure vessel became an issue simply because the 8 torr GN_2 allowed gas conduction and convection effects to greatly increase the thermal coupling between the cold shrouds (-1 35°C) and the chamber walls. Additionally ice formation in areas outside the chamber could act as an insulation barrier between the cold chamber walls and the warm air in the test lab aggravating the problem. Chamber facility management was concerned that under those conditions the chamber stainless steel shell could have dropped below -20°C causing embrittlement of the carbon steel welds. Since the welds connect the structural stiffener rings to the chamber shell, in a worst-case situation, failure of the welds could have cased the collapse of the chamber. Other concerns included overcooking of the door and door seals, the fused quartz solar window and the electrical and instrumentation feedthrough connections.

Since the 8 torr, -1 35°C environment simulation was also necessary to simulate Mars operations, a proof-of-concept test was design and performed without the spacecraft in the chamber. The test was performed 2 weeks prior to the start of the STV-1 test and consisted maintaining the chamber pressure at 8 torr and shrouds at -85°C and - 135°C. At each step, the temperature of the chamber was incrementally decreased to the desired temperature and held for 8 hours. The shell and carbon steel welds were monitored at several locations. Results demonstrated the chamber ability to stay at -85°C (steady-state) and at -135°C for at least 8 hours.

The test also demonstrated the chamber ability to perform a emergency pump down (from 8 torr to less than 0.001 torr) in less than one hour. Pumping the chamber down from 8 torr to hard vacuum immediately arrests the gas conduction and free convection cooling effects on the chamber, thus allowing the chamber temperature to rise back into a safe region.

Steps on the chamber facilities procedure were added, as precaution, to monitor temperature and visual] y inspect the chamber real time during the accelerated cooldown. It was agreed that any time the accelerated cooldown could be interrupted by pumping down to vacuum and flooding the walls, if recommended by the chamber facilities. Results of the accelerated cooldown are reported in the section on STV- 1 test results.

TEST DESCRIPTION

IR lamps Implementation

The most realistic simulation of the cruise thermal environment would have involved spinning the flight spacecraft at 2 RPM (the nominal cruise spin rate) and rotating the spacecraft spin axis from 0° to 60° off-Sun, Due to schedule, budget and safety constraints, this type of simulation was not possible and an alternative plan was developed. During the STV test, an infrared (IR) quartz lamp array applied thermal loads on the HRS radiator and backshell to simulate time-averaged, solar loads that the spinning spacecraft would experience when it was flying off-Sun. The spacecraft is constrained to maximum off-Sun angles of 60° at Earth (0.99 AU) and410 at Mars (1.55 AU).

Figure 5 shows the entire IR lamp array structure assembled in the 25-foot diameter thermal vacuum chamber. The lamp array structure consisted of five, 9-foot tall, modular support structures that were bolted together around a pentagonal shape at the bottom of the chamber. The support structures held IR lamps on a central 360° ring, with vertically-mounted bulbs to illuminate the backshell and horizontally-mounted bulbs to illuminate the HRS radiator. Safety lamps, located above the spacecraft solar array and below the heatshield, were added to the lamp array structure to facilitate warming of the spacecraft when needed (e.g., during spacecraft warm-up to prevent contamination products from condensing on spacecraft surfaces or in the event of a solar simulator failure). The lamp array support structure and frames were designed to minimize obstruction of the spacecraft view to the side and floor shrouds.

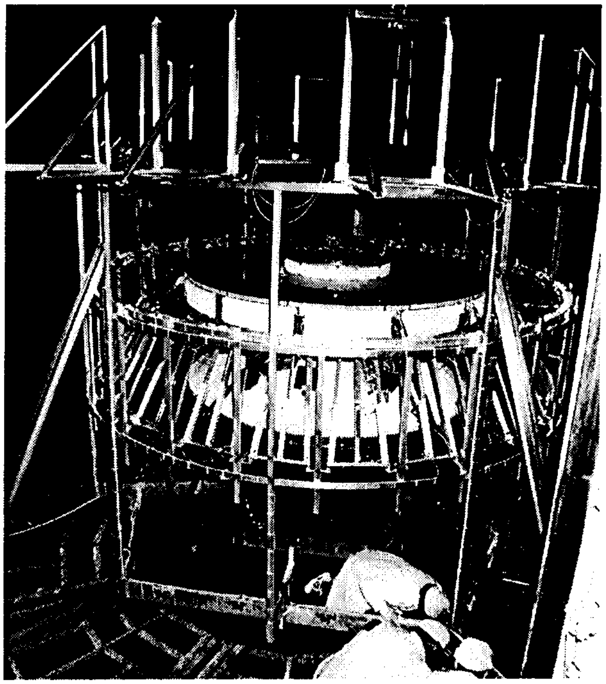

Figure 5. MPF and IR Lamp Array inside JPL's 25-foot Solar Simulator

The backshell lamp array consisted of 38, 14" long bulbs mounted vertically (side by side, approximately 12" apart) on pivoting "A-frames." The pivot feature allowed the angle of the bulb relative to the backshell cone to be adjusted. The radiator array had 30, 14" long bulbs mounted horizontally (end to end, approximately 15" apart). Both arrays had 1.5" wide reflectors positioned 1.5" behind the bulbs.

A special test was performed to assess the flux uniformity of the lamp array and to determine the proper dimensional characteristics of the array (e.g., angle of the pivoted bulbs, distance between bulbs, distance from bulb to spacecraft surface, etc.). The flux uniformity test was done in air, with a 1/5 section of the entire lamp array. A 1/4 sector wire frame, scale model of the spacecraft backshell and HRS radiator was fabricated and covered with black Kapton blanket material skin (Figure 6). The lamp array illuminated the front side of the scale model to the maximum flux levels that would be seen in the STV test. An IR camera viewed the back side of the scale model and recorded the resulting temperature distribution as a color temperature map. Results indicated that the final array design would deliver the desired flux levels with a flux uniformity of ±1O%.

Even at the highest required flux levels, the system was operated at only 40% of its maximum capacity. The desired maximum absorbed fluxes into the backshell and HRS radiator, for the 60° off-Sun case at 0.99 AU, were 333 W/m^2 and 67 W/m^2, respectively.

Since the spectral characteristics of the IR lamps change with applied voltage (some of the flux is in the IR region and some in the solar region), it was necessary to use calorimeters to control the absorbed heat loads into spacecraft surfaces. Calorimeter absorbing surfaces were coated with the same material as the spacecraft surfaces that were illuminated by the lamp arrays. The HRS radiator was covered with NS43G white paint, the backshell was covered with Flamemaster white paint and cruise stage equipment was covered with second-surface aluminized Kapton, the thermal blanket outer layer. The calorimeters were calibrated, in a vacuum, so that a conversion between calorimeter temperature and absorbed flux level could be made.

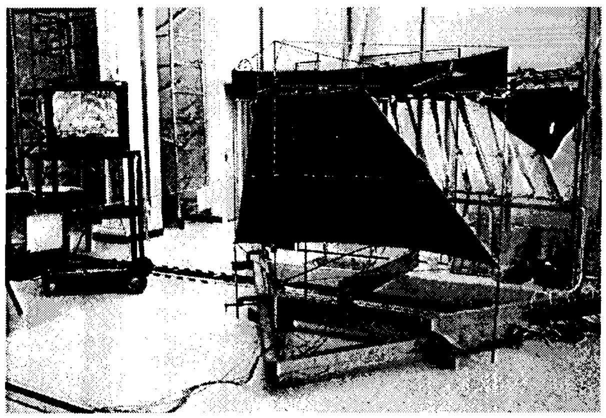

Figure 6. IR Lamp Array Mapping

During the STV test, the IR lamps were computer controlled, using a feedback loop between the calorimeter and the IR lamp power supply to maintain the desired absorbed flux levels on the backshell and HRS radiator. There were 6 calorimeters on the backshell (5 up the height of one side and one on the opposite side) and 2 calorimeters on opposite sides of the HRS radiator. Output from the backshell array was adjusted such that the total absorbed load into the backshell (summed up from the 5 backshell calorimeters) matched the predicted flight value at the proper off-Sun angle and AU distance. Two additional calorimeters were also placed near cruise stage equipment to monitor the absorbed loads from the backshell array into the thermal blankets.

Test Set-up

The spacecraft was tested in the JPL's 25-foot Space Simulator chamber (Figure 7). The chamber is about 85 feet (25.9 m) high and 27 feet (8.2 m) in diameter. The chamber shrouds are painted black on all surfaces facing the test article and can be controlled over a range of -185°C to 200°C using liquid and gaseous nitrogen.

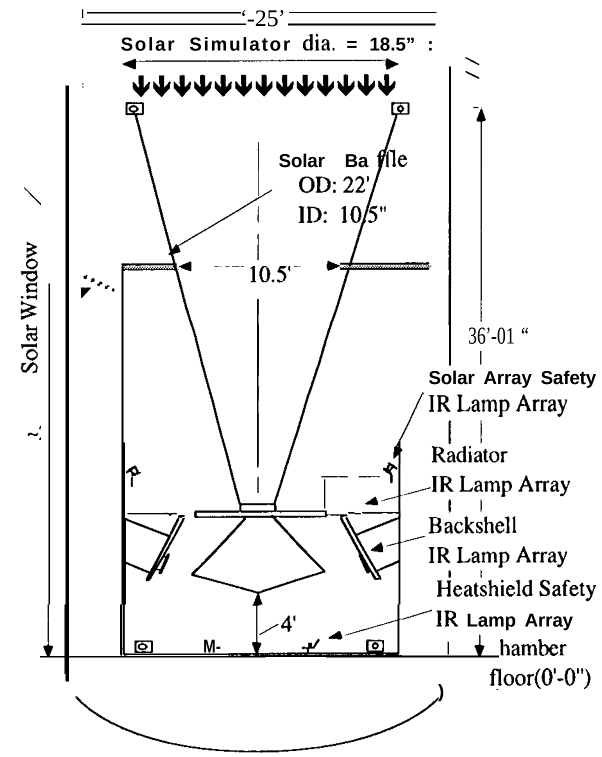

Figure 7A. MPF STV- 1 Configuration

The chamber is equipped with a solar simulator consisting of an array of 37 Xenon arc lamps, an integrating lens unit which mixes the light from the lamps to form a uniform beam, a fused-quartz penetration window, and a 23-foot diameter collimating mirror mounted at the top of the chamber. The chamber, capable of a solar simulation up to 2.7 Suns (-3690 W/m^2), delivered a maximum of about one Sun (1400 W/m^2) during the MPF STV- 1 test.

Efforts were made to install the spacecraft in the chamber in the most flight-like manner. The spacecraft was suspended inside the 25-foot chamber using three 3/8" diameter stainless steel cables connected to the Launch Vehicle Adaptor (LVA). The LVA was hard mounted to the spacecraft launch adaptor and it was not flight hardware. Thermal isolation between the LVA and the spacecraft could not be accomplished in this configuration. The LVA was blanketed to minimize additional losses/gains from/to the spacecraft due to the increase in area.

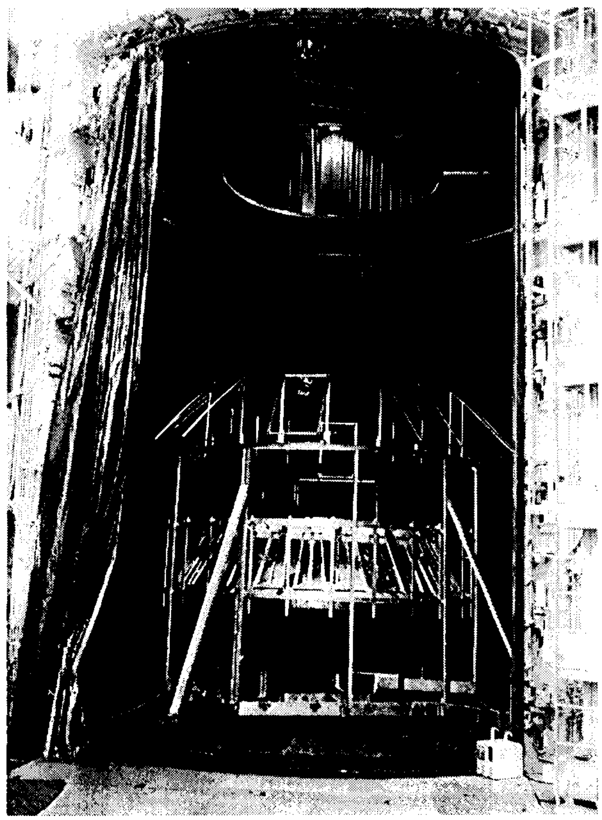

Figure 7B. MPF STV- 1 Configuration

A number of the non-flight, direct-access cables had to be connected to the spacecraft in STV- 1. To minimize conduction cable losses that could affect the spacecraft local temperatures all cables were blanketed with a 10-layer multi-layer blanket over a minimum distance of 5-feet. Guard heaters were also added to reduce conduction losses through the large umbilical cables. The guard heaters, located close to the spacecraft, were controlled to maintain the gradient between the local cable and spacecraft temperature to less than 1 “C.

A solar baffle reduced the diameter of the chamber solar beam from 19 feet to 10.5 feet. The bean reduction was necessary to minimize the illumination of the IR lamp array’s aluminum structure and the chamber floor. Even though the floor was painted black, 2% reflection back onto the spacecraft (especially on the heatshield) was possible. Reducing solar illumination of the chamber floor also reduced the LN_2 consumption in the floor shroud,

Test Instrumentation

Approximately 350 thermocouples were used in the test. All thermocouples were Chromel - Constantan, Type E with wire sizes varying from 26 to 30 gage,

A total of 36 DC power supplies were used in the test to control the test heaters, a pressure transducer, and an over-temperature cut-off for the IR lamps controller. Test heaters were used to accelerate warm-up transients when transient data was not required and as safety precaution for flight hardware.

The Space Simulator System data acquisition system was used to control the power in the test heaters and to record and display data for the test heaters and thermocouples. All the instrumentation cables were blanketed.

Test Matrix

The test consisted of a cold phase, an EDL simulation, and a hot phase. The test profile is summarized in Figures 8 and 9.

Due to the nature of the mission a single test case would not simulate the worst-case cold (or hot) environment for all major assemblies in the spacecraft. In both cold and hot environments, the worst-case temperatures in different regions of the spacecraft, were driven by the spacecraft off-Sun angle, In the hot case, for example, the Sun-pointed attitude at Earth produced extreme hot temperatures for the cruise stage, z-axis mounted sun sensors and shunt radiator, but not for the backshell or lander mounted equipment. In the cold case, a large off-Sun angle (410, at Mars produced extreme cold temperatures for the solar array, z-axis mounted sun sensors and shunt radiator, but not for the backshell or lander mounted equipment.

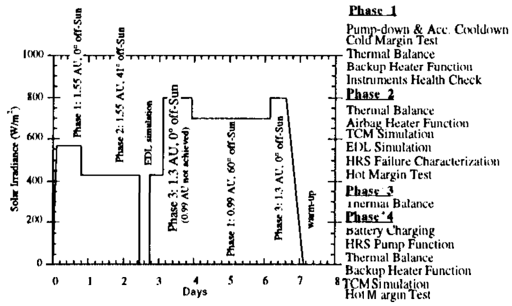

Figure 8. STV- 1 Test Profile

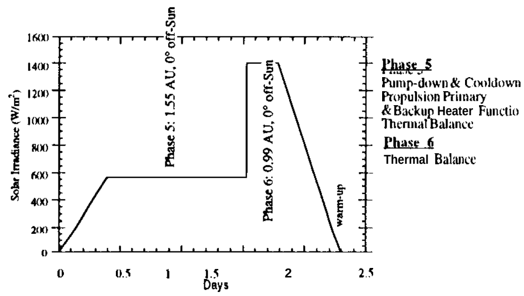

Figure 9. STV-1 Test Profile

Two cold cases were simulated to verify the thermal design at the extreme cold condition of Mars arrival (1.55 AU). The simulations were done at a Sun pointed attitude (worst-case cold for the backshell and lander) and at 410 off-Sun angle (worst-case cold for solar array, sun sensors and shunt radiator).

The EDL simulation was performed at the end of the 1.55 AU, 410 off-Sun thermal balance test, the appropriate initial condition for the start of EDL. Even tough the aerodynamic heating was not simulated during the EDL portion of the test, results for the equipment shelf electronics were deemed valid due to the short duration of the heating pulse, less than 2 minutes, and the high degree of insulation between the heatshield and the lander. Warm-up rates for equipment shelf electronics after the HRS is disconnected were derived during the test. The test was also used to verify the EDL heater sizes.

Two hot cases were done to verify the design at the extreme hot conditions near Earth (0.99 AU). Two off-Sun angles were simulated, 0° (worst-case hot for the solar array, sun sensors and shunt radiator) and 60° (worst-case hot for the backshell and Lander).

Other test cases were added between the thermal balance cases. Most of the time these tests did not add a significant amount of time to the overall test, but did contribute significantly to our understanding of the overall spacecraft system and thermal design.

STV-1 TEST RESULTS

Accelerated Cooldown

Two cooldown phases were completed during the STV-1 test: the initial accelerated cooldown (8 torr and shrouds at -135°C) done at the beginning of the test and the standard cooldown (< 10-5 torr and LN_2 shrouds) done after the chamber break. A comparison of the cooldown data indicates the advantages of an accelerated cool down.

Comparison of the cooldown rate for one of the most massive pieces of the spacecraft, the heatshield, showed the accelerated cooldown increased the cooling rates by approximately a factor of three, as shown in the following table.

	accelerated cooldown rate [°C/hr]	standard cooldown rate [°C/hr]
during shroud cooldown (-3 hrs)	15	4.5
steady-state shrouds		
(first 3 hrs)	10	3
[next 3 hrs]	4	3.5

STV-1

The STV test was broken up into two parts separated by a chamber break. The first part (STV-1) covered the worst-case cold environments (O" and 410 off-Sun at 1.55 AU) and one of the hot cases (60° off-Sun at 0.99 AU). All spacecraft temperatures during STV-1 were within allowable flight temperature limits except for some undertemperature conditions on the propellant line filters. The undertemperature conditions were suspected, and verified during the chamber break, to be caused by blanket workmanship.

The second worst-case, hot environment condition (O" off-Sun at 0.99 AU), had to be aborted when the cruise stage solar array and sun sensor heads approached temperatures close to their maximum AFT limits at only 85% of the desired maximum solar flux at 0.99 AU. A micrometeoroid protection shield added last minute to the back of the solar array was responsible for the higher temperatures. The shield was not part of the original spacecraft configuration and it was added under the condition that it could be removed if it caused the solar array to overheat during the STV test.

Clearly some modifications were needed to the propulsion lines and solar array thermal designs before the spacecraft could be deemed flight ready. A chamber break was deemed necessary and it was used to make these modifications prior to are-test (STV- 1.1).

Modifications During Chamber Break

The following is a summary of the changes done in the 5-day chamber break:

In order to reduce the solar array temperature:

- All the propellant line micrometeoroid protection was removed.
- Black Kapton tape was added to the cruise stage bare aluminum ribs to increase their radiation heat loss to space.
- Most of the unpopulated areas on the top of the solar array were covered with Silver coated Teflon tape to reduce local hot spots.
- The IR lamp A-frames were removed to facilitate access to the spacecraft and to reduce radiation blockage to the shrouds.

In order to increase temperatures on the propellant lines filters:

- The existing 15-layer embossed Kapton blankets which covered the propellant lines filter area were removed and new blankets were installed. The new blankets consisted of two, 10-layer aluminized Mylar/Dacron net MLI overlapped to minimize heat leaks. The new blankets were also wrapped with a SSAK outer layer.

- Underneath the new blanket, strips of aluminum foil (approximately 0.01 2" thick) were taped to the lines to enhance lateral conduction and to better distribute the heat along the line.

- Auxiliary test heaters were added to the lines as a back up solution in case the re-blanketing efforts were not successful.

- Additional thermocouples were installed to gather more information during the re-test phase of the test.

- Semi-cylindrical radiation shields ("umbrellas") were added to the thermostats which controlled the propellant line heaters. The umbrellas were made out of 5 mil double-aluminized Mylar, heat-formed to shape. The highly reflective umbrellas increased the thermostats view to space and to decouple the thermostats from the warm solar array.

STV-1.1

After the modifications were made, the spacecraft was re-tested (STV- 1.1) in a Sun-pointed orientation in the worst-case hot (0.99 AU) and cold (1.55 AU) environments. The re-test was successful.

The peak solar array and sun sensor temperatures in the hot case were at least 10"C cooler than their maximum allowable flight temperature (AFT) limits. Modifications made to the solar array during the chamber break reduced its average temperature by 6°C and the hot spot temperatures by 27°C. Modifications made to the propellant lines during the chamber break resulted in dramatic improvements in thermal performance; sections of lines that previously were below the minimum AFT limit were now exhibiting 24°C to 28°C of margin above the minimum limit.

In general, the thermal design performed quite well during the STV testing. This was the first time that the entire HRS had been tested in a vacuum environment. The HRS performed flawlessly during the entire test; maintaining the lander electronics equipment shelf between 6°C and -8°C during the entire range of mission extreme hot and cold environments.

POST TEST MODIFICATIONS

After STV- 1.1, there remained some concerns about the performance of the propulsion line thermal design. In the worst-case cold environment, with both the primary and backup thermostats enabled, the line temperatures were well above the minimum AFT limit. However, test results indicated that, in the event of a primary thermostat failure, one of the propellant lines could develop a cold spot of 6°C (only 4°C above the freezing point of hydrazine). Because of this and related concerns, an extensive analysis of the propulsion line thermal design was performed and the following design modifications were made:

- The new Mylar/Dacron net blankets and aluminum foil heat spreaders were employed everywhere on the entire length of the propellant lines. Mylar/Dacron net blankets outperform embossed Kapton blankets on bends in the propellant lines.

- Additional blankets were added on each propellant line stand-off and at the interface between the line blanket and each stand-off. These areas were identified as being potential sources of high heat leaks.

- Semi-cylindrical radiation shields ("umbrellas") were added to all lines, except in areas that had good views to space, This helped minimize the heat load that thermostats receive from the solar array. The design philosophy was to bias the thermostats, which controlled the propellant line heaters, cold to protect against freezing propellant in a line.

Due to cost and schedule impact, no other full scale test was done to verify these changes. The soundness of the propellant line thermal design after the post-test modifications was verified by analysis, This was a departure from the "traditional" way of doing thermal design at JPL and represented the acceptance of a significant amount of risk by the project. This is the type of risk that will have to be taken in future "better, faster, cheaper" projects.

FLIGHT EXPERIENCE

At the time that this paper was written, the spacecraft had traveled approximately half the distance between the Earth and Mars.

The spacecraft has already experienced the worst-case hot environment during cruise (at 0.99 AU) and the thermal design has performed very successfully. There have been no violations of AFT limits and the HRS is maintaining electronic equipment inside the lander close to O°C. Based on STV test data and post-test analyses, the spacecraft thermal design is expected to maintain all spacecraft temperatures well above their AFT limits in the worst-case cold environment (coming up at Mars, 1.55 AU).

In addition to verifying the performance of the spacecraft thermal design, results of the STV test have been used to make decisions about how to operate the spacecraft in flight. A trajectory correction maneuver simulation that was done in the STV test was very useful in determining how warm propulsion drive electronics would get during a long maneuver. The EDL test results have allowed us to plan the timing of heater turn-on prior to entry and also to determine initial lander temperatures for landed operations simulations.

SUMMARY

The Mars Pathfinder STV-1 test completed all its objectives and produced data that is deemed invaluable for the operation of the spacecraft in flight. The first of a series of "better, chepear, faster" STV tests completed at JPL, the test was completed within shedule and budget. Test cost, for both STV- 1 and STV-2 tests, where initially estimated at $850K. Actual test cost, for both tests, is itemized in the table below:

	STV- 1	STV-2	Total
Engineering and Set-up	$53K	$21K	$74K
Chamber Test	$470K	$143K	$61 3K
Instrumentation	$ 69K	*	$ 69K
Solar Shade Bake-out	$ 28K	$0	$ 28K
Total	$620K	$ 164K	$784K

* Instrumentation charges not separately indendified

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