# Development of the Viking Mars Lander Thermal Control Subsystem Design

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**Two Viking spacecraft, each consisting of a lander capsule and an orbiter, were launched toward Mars late in the summer of 1975. About a year later, the orbiters are to go into orbit around Mars and the landers are to descend to the surface for 90-day landed missions. The Viking mission presents several unique and severe requirements important to the lander thermal control subsystem design. The lander must withstand a wide variety of environmental and operational conditions during all phases of the mission, including prelaunch sterilization. On the surface of Mars, lander internal temperatures must be controlled under widely varying thermal environments and atmospheric conditions. The lander is designed primarily for Mars surface operation, with modifications and additions made as required for other mission phases. The lander thermal design is based on maximum use of passive techniques and is integrated into the overall vehicle design operation. Solutions to the unusual combinations of design problems and a summary of the results of full-scale model testing under simulated mission conditions are presented in this paper.**

## Introduction

THE Viking lander is an unmanned vehicle designed to make a soft landing on the surface of Mars; perform scientific experiments; photograph the surroundings; and transmit information back to Earth. Two identical Viking spacecraft, each consisting of a lander capsule coupled to an orbiter, were launched toward Mars during August and September of 1975, with the interplanetary cruise phase lasting ten to twelve months.[1] The spacecraft are scheduled to arrive at Mars between June 14 and Aug. 13, 1976, and then go into orbit for periods of ten to fifty days. The lander capsules are to separate from their orbiters and then descend to the surface for landings between 30 deg south and 45 deg north latitude. The landings are to occur between June 24 and October 2, 1976, for 90-day landed missions, so the latest scheduled end of mission is December 31, 1976. Each orbiter is to circle the planet, performing other experiments and observations, and providing an additional communications relay link between the lander and Earth.

This paper describes the development of the Viking lander thermal control subsystem design for Mars surface operation, the most critical part of the mission. This subsystem design is complicated by a combination of several unique or unusually severe conditions and requirements. The entire lander capsule must be enclosed in a bioshield and sterilized before launch, and must withstand a wide variety of thermal and other environments after launch. On the surface of Mars, the lander will be subjected to a very complex planetary environment including an atmosphere. The lander power supply consists of two radioisotope thermoelectric generator (RTG) units mounted on the lander, which dissipate large quantities of waste heat at all times. The many power system and electrical components will dissipate energy as heat over a wide range of duty cycles. The same thermal design should provide adequate control during all mission phases with a high degree of reliability, but for low cost, weight, and development risk.

## Thermal Control Requirements

The total lander capsule consists of the lander body, terminal descent, and deorbit/attitude control propulsion systems, decelerator, aeroshell, base cover and bioshield, as shown in Fig. 1. The vehicle that lands on Mars consists of the lander body with the terminal descent propulsion system attached. The components used during the descent and landing, but not during surface operation, the RTG units, and the antennas and instruments that must be in contact with the Mars surface environment are mounted on the outside of the lander, as shown in Fig. 2. All other components used on the Mars surface are mounted inside the lander body.

Before assembly, each component is sterilized and then rechecked at ambient temperatures. Before the final prelaunch checkout, the entire lander capsule is exposed to sterilization temperatures of 235°F for about 40 hr. During launch, entry and terminal descent phases, the capsule is subjected to a wide variety of thermal and dynamic loads. During cruise and orbital phases, the vehicle is exposed to vacuum and thermal conditions of space for about a year. Components and materials used on the lander must be compatible with all of these environments, and effects of all elements of lander design on the various mission phases must be considered. The lander must be vented during sterilization, launch, and atmospheric entry. The entire lander body must constitute a radio frequency shield around its contents. Very little electrical energy is available for thermal control during the lander mission.

The Mars surface environment[2] includes solar radiation, surface and atmospheric (sky) infrared radiation, atmospheric free and forced convection (wind), and blowing dust and sand, as indicated in Fig. 3 and Table 1. This environment is primarily a function of landing site latitude, time of year and time of day, but also depends on altitude and soil and atmospheric properties. The solar flux range is from 154 to 201 Btu/hr-ft$^2$; the mean daily surface temperature between −139°F and −34°F; the total surface temperature range from −184°F to +95°F; the daily surface temperature variation between 125°F and 215°F (Fig. 4); the effective sky temperature varies from −200°F to −281°F; and the steady windspeed may vary from zero to 130 fps. The power dissipated inside the lander may vary from 25 to 175 watts during a typical day, with the daily average between 60 and 75 watts.

## Thermal Design Approach

The thermal design[3,4] of the total lander capsule is based on designing the lander body primarily for the Mars surface environment as indicated in Fig. 5. Then modifications are made to this design and features added to the other parts of the vehicle as required to satisfy the other mission phases. The surface environments used for the lander design are stacked

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Index categories: Spacecraft Temperature Control Systems; Spacecraft Ground Testing and Simulation (including Components).

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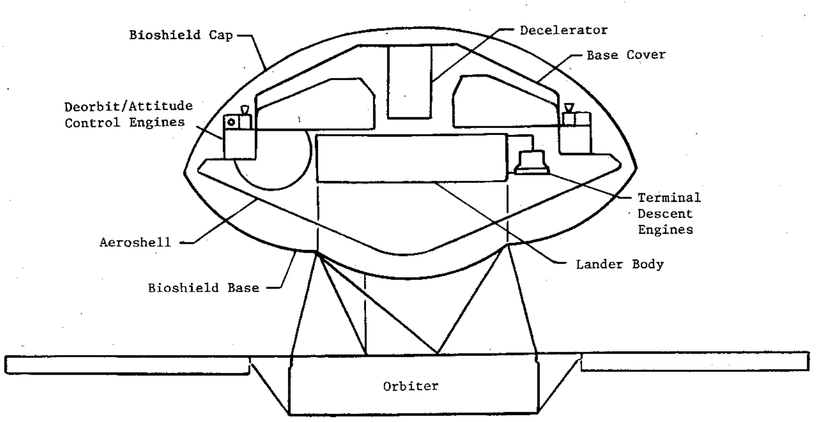


**Fig. 1 Viking lander capsule.**

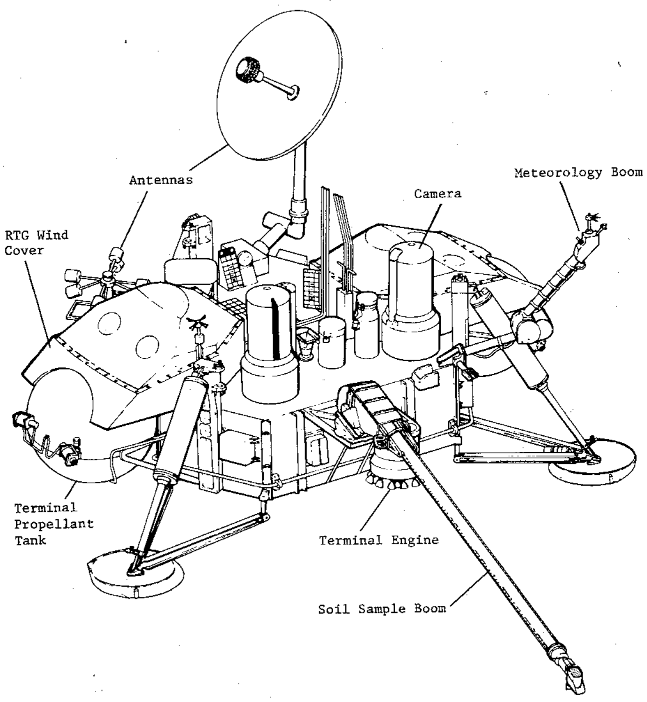


**Fig. 2 Viking lander on Mars.**

worst-case hot and cold conditions, since there are many possible combinations and not enough available data to apply to statistical approaches. Minimum temperature sensitivity of the lander to external environments is desired, since many environmental conditions are either not well known or are known to vary widely.

The vehicle thermal design makes maximum use of passive techniques, such as insulations, coatings, and high and low conductivity materials. A 25°F margin is desired for all passively controlled areas to allow for thermal analysis complexities and the continual changes to vehicle design and operation. The thermal control subsystem is designed to provide high reliability, testability, and minimum constraints on mission operation, as well as low cost, weight, and development risk. To meet these criteria, the thermal design is integrated into the overall vehicle design and operation. Constant surveillance of all other vehicle subsystem design changes and operating modes is required to determine possible thermal effects and to assure proper thermal performance.

The lander thermal design is based on locating most of the components required to operate during the landed mission in a single thermally controlled compartment. The heat capacity of this compartment is important in attenuating the temperature fluctuations caused by the wide diurnal variations in external environment and internal power dissipation. The lander body thermal design is passively balanced to maintain acceptable internal temperatures under the hot design case conditions. Additional heat required by the lander for the cold design case is obtained from the RTG units through two thermal switches. For moderate environments, thermal switches control heat flow to obtain the desired temperatures. Compartment conductance is designed to remove during a hot case night the heat accumulated during a hot day by maximum equipment power utilization, but not to remove during a cold case day and night more heat than can be provided by the minimum equipment operating cycle plus thermal switches.

The lander compartment is isolated from the external environment by insulation of varying thicknesses located inside the structural enclosure. This insulation is thickest on top, which is subject to the greatest temperature and heat flux variations, and thinnest on the bottom which is subject to the smallest environment variations. Thermal effects of penetrations through the insulation are minimized by fitting the insulation around each penetration and making this penetration of minimum cross section, maximum length, and low conductivity materials wherever possible. Most of the internal components are hard-mounted to a high conductance equipment mounting plate, and most internal surfaces have a high emittance coating to promote heat transfer across the compartment and minimize thermal gradients. The equipment plate is located at the top of the compartment to provide minimum path length for the thermal switches. This plate has varying thickness to provide greater heat transfer and heat capacity in critical areas with minimum weight. The specific

**Table 1 Mars surface design case environments**

| Time/Place<br>Event | Cold north solstice | Nominal late landing | Hot end of mission |
|---|---|---|---|
| Earth date, 1976 | July 4 | Oct. 2 | Dec. 31 |
| Lander latitude, deg | −30 | 0 | +1 |
| Lander altitude, km | −9 | −3 | +3 |
| Solar conditions | | | |
| Solar latitude, deg | +24.8 | +18.4 | +1.0 |
| Solar irradiance, Btu/hr-ft$^2$ | 154 | 171 | 201 |
| Irradiance time, hr | 10.2 | 12.3 | 12.3 |
| Surface properties | | | |
| Solar absorptivity | 0.75 | 0.77 | 0.85 |
| Infrared emissivity | 0.96 | 0.93 | 0.89 |
| Atmoossssspheric properties | | | |
| Pressure at site, mb | 20.2 | 6.8 | 2.85 |
| Composition, wt % $CO_2$ | 82 | 100 | 100 |
| wt % Ar | 18 | 0 | 0 |
| Temperatures | | | |
| Sky, effective °F | −281 | −221 | −200 |
| At surface, °F, max | −60 | +42 | +95 |
| mean | −139 | −67 | −34 |
| min | −184 | −141 | −121 |

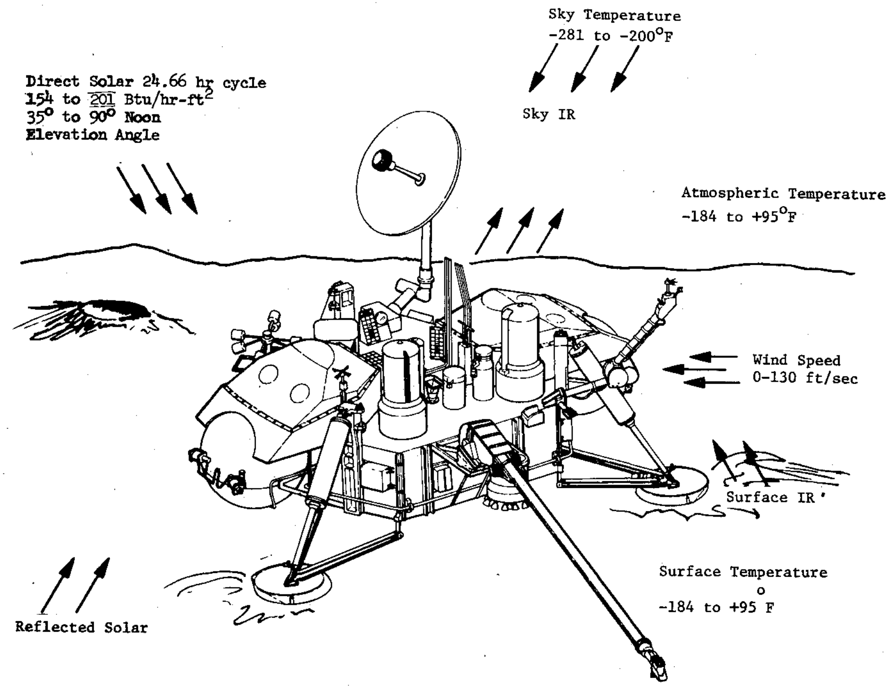


Fig. 3 Mars surface thermal environment.

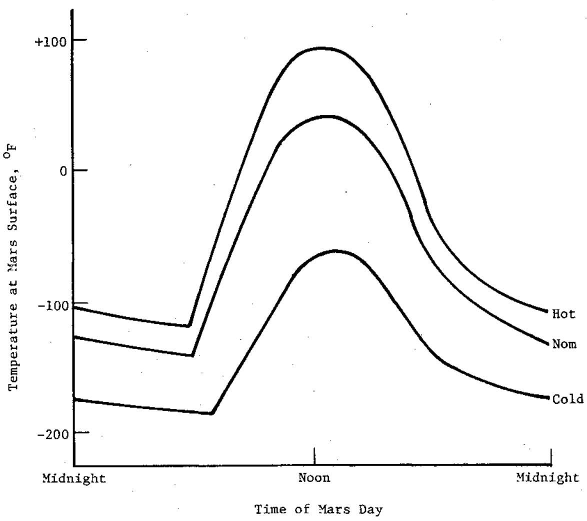


Fig. 4 Mars diurnal temperature cycles.

arrangement of internal components is partly determined by thermal considerations.

The basic lander body enclosure consists of three main side beams and top and bottom covers, all of aluminum. This structure must be vented to the atmosphere, but is designed to prevent penetration by wind, dust, or radio waves. Lander exterior coatings must resist erosion by wind and wind-blown dust and sand, and provide low solar absorption and high infrared emittance to reduce solar radiation effects. The terminal propulsion system, the components used during descent and landing but not during the landed mission, and the antennas and other landed mission components that must be outside the lander and are not sensitive to thermal conditions, are all mounted on the lander exterior. However, they have little effect on the lander thermal performance except to partially shade the lander from the sun and sky.

The RTG units are mounted on top of the lander and are isolated from both the equipment plate and the side beams. These units are also isolated from the external environment by windshields to prevent excessive heat loss to a cold, windy environment. The RTG windshield areas, radiating fin areas, and surface coatings are designed to prevent overheating in a hot calm environment. The thermally sensitive cameras, which must be mounted at least partly outside the compartment, require special thermal considerations.

## Thermal Control Design Details

Design features of the major areas affecting the thermal subsystem performance are described below. Of course, there are many minor details, too numerous to mention, that contribute to successful subsystem performance. However, the features described should provide a good overview of the many kinds of problems encountered in the thermal design of a Mars lander.

### Compartment Insulation

The thermally isolated compartment is completely enclosed by bulk insulation blankets of varying thicknesses, with 1-in. thick blankets on the bottom, 2-in. blankets on the sides, and up to 4¾ in. of insulation in three layers on top, as indicated in Fig. 5. The bottom and side blankets and two of the three top layers are made up of ½-in thick, low density, high performance fiberglass batts stacked to obtain the desired thickness. They are enclosed in Dacron covers to prevent contamination of other components by insulation particles and to aid in dimensional stability. The assembled blankets have a density of less than 1 $lb/ft^3$ and a thermal conductivity about 0.012 Btu/hr-ft-°F in a Mars atmosphere. Large blankets have strips of tape or ribbon sewn across the large areas with nylon ties through the insulation between tapes to control thickness. All penetrations and indentations in the insulation are cut out to fit and sealed by extensions of the cover material. Even relatively small areas are filled with small blankets cut to the required size and shape and are often sewn to adjacent large blankets. A ½-in. wide air gap between the interior equipment and the insulation inside surface is specified to further isolate the compartment and the individual components from the external environment.

The lowest of the three top insulation layers consists of many small 2-in. thick blankets that fill the spaces between the stiffeners and platforms extending upward from the equipment mounting plate. The second layer is made up of large 3-in. blankets that cover the entire lander top above the stiffeners (except for penetrations), and is slightly compressed when installed. The third layer of insulation is made up of ten

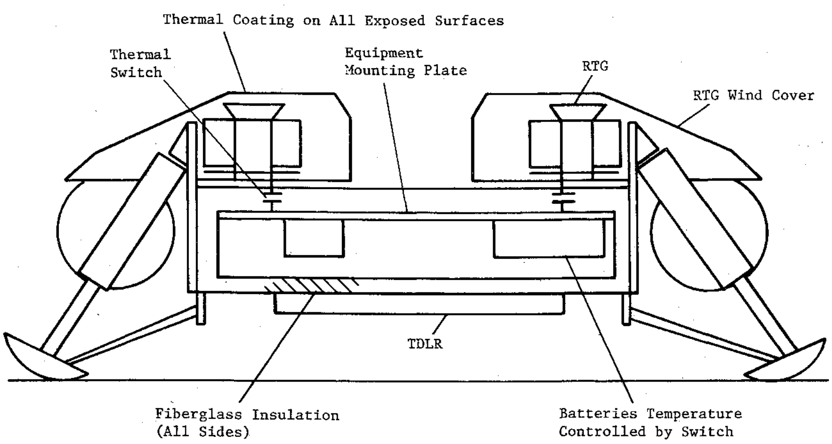


**Fig. 5 Mars surface thermal control.**

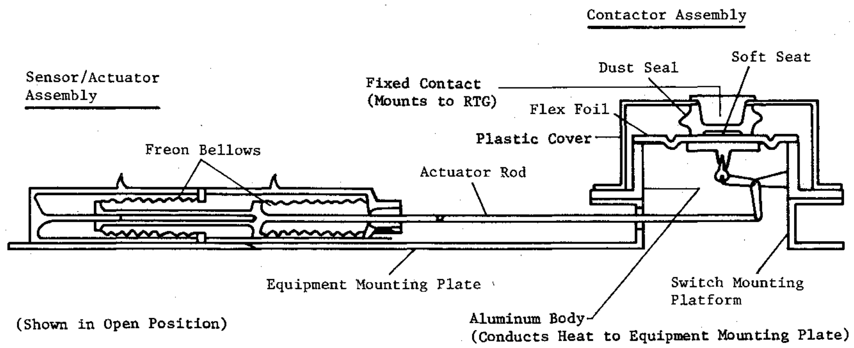


**Fig. 6 Viking thermal switch design.**

layers of double metallized radiation shields separated by Dacron netting, and is installed under the top cover in the area under and between the RTG wind covers. It was added primarily to reduce the heat leak from the RTG units into the lander during the cruise mode, but also reduces the undesired heat leak into the lander during a landed mission hot day.

Some areas near the middle of the bottom insulation blanket are actually only ½ in. thick. These areas are where the ½ in. high lander bottom cover stiffeners and the lander compartment vent are located. However, the center one-third of the lander bottom area is covered on the outside by the terminal descent and landing radar (TDLR). This component is essentially a box 30 in. square and 3 in. deep mounted below the bottom cover with about a ½ in. air gap. A wind fence extends down from the lander bottom around the TDLR to prevent wind and wind-blown dust from entering this air gap. Thus, even with the thinner insulation, the TDLR acts to further isolate this lander bottom area from the external environment under windy as well as calm conditions. A screen assembly including a fine-mesh aluminum screen acts as a filter and provides the required RF shielding over the vent opening, and supports the ½ in. thick insulation blanket away from the opening to assure adequate venting.

### Thermal Switches

Each of the two thermal switches consists of a contactor assembly and a sensor/actuator assembly connected by an actuator rod, as shown in Fig. 6. The contactor limits heat transfer between RTG units and the lander when the contacts are open, and transfers heat from RTG units to the lander and locally distributes the heat to the equipment plate when the contacts are closed. The contractor has a dust seal around the contacts to reduce the possibility of contamination, a movable soft seat to minimize sensitivity to particulate contamination and contact misalignment, and copper flex foil conductors to provide high conductance flexible heat transfer paths between the movable contact and fixed base.

The combination flexible foil conductor and soft seat is constructed of copper foil and tin. Rectangular pieces of thin copper foil, alternately stacked 90 deg to one another, are diffusion-bonded together to form a cruciform flexible conductor. A thick layer of tin is then cast onto the solid center square section to provide the desired combination of conductivity, rigidity and flatness to the movable contact. A single layer of soft tin foil covers the cast tin, to promote good full-surface contact with the machined aluminum upper (stationary) contact surface. This arrangement was developed to desensitize the contact conductance from the effects of particulate contamination and/or distortions in the contact surfaces.

The sensor/actuator senses the local equipment plate temperature by means of Freon vapor pressure acting on a piston within the actuator, which moves the actuator rod to open the contacts upon warming. The spring force of the bellows enclosing the Freon overcomes the vapor pressure to close the contacts upon cooling. Upon further cooling, the actuator applies force to the contacts in proportion to the degree of cooling to increase the contact heat transfer coefficient as the temperature drops. A double bellows seal is used to prevent Freon leakage from the actuator.

The thermal switches are the one new type of mechanism developed for the Viking lander thermal control subsystem. They have successfully completed all flight qualification and flight acceptance tests. They have very consistent performance, with open and closed position conductance of 0.1 and 5.0 Btu/hr-ft$^2$ for a control ratio of 50 to 1. An actual temperature/conductance curve for one of the flight article switches is shown in Fig. 7. The slight degree of hysteresis is due to friction within the copper flex foils. The switch contactor assemblies are mounted on raised platform areas of the equipment plate beneath the RTG units and are connected to the lower RTG mounting flanges. The sensor/actuator assemblies are mounted on the upper side of flat sections of the equipment plate about 10 in. from the contactors and in areas considered most critical for thermal control.

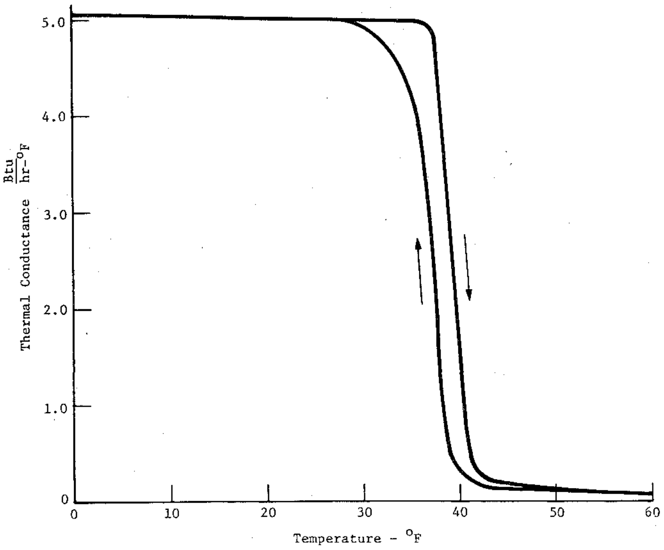


**Fig. 7 Thermal switch conductance/temperature characteristic.**

### Equipment Mounting Plate

The equipment mounting plate is made from a single piece of high conductivity aluminum which provides good thermal conductance across the compartment. In addition to the structural requirements for stiffeners, metal thicknesses and component mounting provisions, a number of additional requirements are based on thermal considerations. The thermal switch contactor mounting platforms are required to shorten the switch conductance distance, to aid in distributing heat to the equipment plate, and to provide room for the actuator rods. The areas around the thermal switch actuators and contactors, and certain high power usage components such as transmitters, are made thicker to increase heat conductance and provide additional heat capacity. Very flat surfaces are required on the plate to provide good high conductance mounting for many of the components. A length of tubing located to pass near all major components is cemented to the plate lower surface for use with cooling water in removing component checkout heat during the prelaunch phase.

Thermal considerations had considerable effect on the arrangement of components. For instance, the four batteries are required to be close to the same temperature and to maintain minimum temperatures 40° higher than many other components. Therefore, they are packaged in pairs and mounted together under one thermal switch. The two transmitters, which have high power usage when operating, are located some distance from each other to reduce the possibility of local overheating. The power converter, which has a continuous heat output during landed operation, is located near the middle of the equipment plate for even heat dissipation. In general, high heat dissipating components are interspersed among the more passive components to provide relatively even compartment heating.

### Structural Penetrations

The primary structural supports that penetrate the compartment insulation are the nine brackets supporting the equipment plate from the side beams, three rods supporting the terminal descent and landing radar (TDLR) below the lander bottom from the equipment plate, and two brackets supporting the inboard side of each RTG unit from the equipment plate. There are also support brackets that penetrate the insulation for the two soil processor and distributor assemblies (PDAs), the radar altimeter electronics (RAE) on top of the lander, and the pressure transducer inside the lander.

The major concern for most of the penetrations is heat loss to a cold environment, but the major concern for the RTG support brackets and the equipment plate brackets near the RTG units is heat leaking in from a hot environment. In either case, minimizing conductance should reduce compartment sensitivity to the external environments. Most structural penetrations are made of titanium, which has high strength and the lowest conductivity of any structural metal. Several of the smaller ones are made of fiberglass re-enforced phenolic. The equipment plate support brackets are titanium channel bipods and tripods for high strength and stability with low conductance.

The TDLR supports are titanium tubes with the bottom ends covered by extensions of the bottom insulation, and the exposed surfaces having very low emissivities. The soil PDA units are supported from their associated experiments inside the lander by fiberglass-phenolic cylinders filled with insulation. The RAE is supported from the equipment plate by titanium brackets and the pressure transducer is supported from a side beam by a fiberglass-phenolic bracket. The thin lander top cover is supported from the equipment plate in certain critical areas by small diameter plastic posts, and several other similar requirements are met in a similar manner. In each case, the lander insulation is cut and fitted around the penetration to minimize degradation of the adjacent insulation performance.

### Other Penetrations

The electrical and transmitter cables penetrating the insulation are a major heat leak source because the required high electrical conductivity materials also have high thermal conductivities. In addition to the copper conductors, many of the cables include copper or aluminum shielding, which greatly increases the cable conductance. To reduce penetration conductance to a reasonably small value, the cable bundles must be insulated for about two feet of length, including the side insulation penetration. Since there is room for not more than a ft. of this length inside the compartment, the cable bundles must be isolated from the side beams to permit additional insulated cable length outside the lander, as shown in Fig. 8.

The cable connectors between the inside and outside cables are mounted on enclosed standoffs filled with insulation and mounted on the outside of the side beams. Since re-enforced plastic standoffs would have required a metallic covering such as aluminum foil to provide the required RF shielding, they are made of thin titanium which is simpler and provides a lower total conductance. The cable pass-through holes in the side beams are made large enough to permit a ½-in. thick fiberglass blanket around each cable bundle between the connector and compartment insulation inside surface. The titanium standoffs and the bundles for a distance of about 8 in. out from the standoffs are also covered with ½-in. thick blankets secured and sealed against the wind. Inside the insulated compartment, the cable bundles are isolated by a radiation shield wrap around each bundle and a separation of at least a ½-in. from adjacent equipment for a distance of a foot in from the side insulation.

The science experiments and prelaunch cooling and venting systems require a number of tubes penetrating the insulation. Wherever possible, the tubes are made of plastic such as Teflon. Where metal tubes are required by pressure or purity considerations, small diameter, thin-walled stainless steel tubing, or an equivalent, is used. Where bulkhead fittings are required at the side beams, they are also of stainless steel which has a conductivity about twice that of titanium, but is much more readily available.

### RTG Configuration and Installation

The Viking lander RTG power supply consists of two modified SNAP 19 units, externally mounted as shown in Fig. 5. Each RTG contains a plutonium oxide 675 watt (nominal) heat source, and generates a minimum of 35 W of electrical power. Major design modifications to the SNAP 19 include:

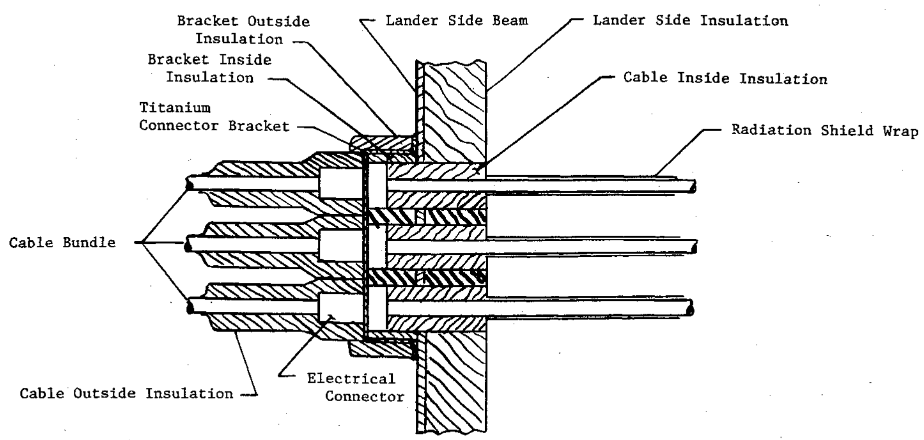


**Fig. 8 Cable side beam penetration.**

an increase in fin size to allow operation in hotter environments; redesign of case thickness to promote conduction of heat to the upper and lower flanges; and redesign of the end flanges to adapt to Viking mounting interfaces. The thermal design of the RTG installation is constrained by the somewhat conflicting requirements to prevent overheating in hot environments while maintaining sufficiently high temperature in cold environments to provide a heat source for the lander compartment.

Excess heat is removed during prelaunch operations by chilled water flowing through end cap coolers mounted to the RTG end flanges. These end cap coolers are aluminum castings enclosing stainless steel tubing. The bottom coolers also serve as RTG mounting flanges and as conductive couplings to the associated thermal switches. The RTG units are enclosed within reenforced plastic wind covers to prevent excessive cooling by the convective effects of high velocity (up to 130 fps) low temperature Mars surface winds. These wind covers are made as large as space permits to maximize radiation heat rejection in hot environments. Conical and annular fins are attached to the upper and lower end cap coolers, respectively, to provide supplemental radiation areas for heat rejection during interplanetary cruise and Mars surface hot environments. All RTG fin and wind cover surfaces have high emissivity coatings to promote radiation.

The RTG units and their wind covers are supported by the lower end cap coolers acting as mounting flanges. These mounting flanges are in turn supported from the equipment plate by titanium bipods and tripods, and from the side beams with fiberglass-phenolic spacers that act as thermal isolators. Soft tin gaskets are placed between each RTG end flange and the adjacent end cap cooler, and between each lower end cap cooler (RTG mounting flange) and the associated thermal switch to minimize thermal resistance at these interfaces. End cap coolers are bolted to RTG end flanges with Belleville washers to preclude loosening of these joints by temperature cycling. The spaces between the bottoms of the wind covers and the lander top cover are sealed sufficiently to exclude wind from these areas and further decrease lander sensitivity to wind.

This RTG installation concept serves to enhance the RTG thermal coupling to the thermal switches, minimize any other thermal coupling between the RTG units and the lander, promote radiant heat rejection to the environment, and attenuate Mars wind effects on both RTG units and lander.

### Camera Installation

Two cameras are physically outside of the lander body, but are within the total insulation enclosure. Each lander camera assembly consists of a camera head mounted on a cylindrical mast enclosing the camera electronics. This assembly is enclosed in a 2-in. thick layer of bulk insulation except for the lens opening and mast base. Each assembly is supported above the lander top cover by a short cylindrical aluminum mast, and is supplied with power through an electrical cable inside the support mast. This support mast is mounted on top of the equipment mounting plate, and the upper end above the lander top cover is enclosed in bulk insulation to match that over the camera assembly. The aluminum lander top cover is connected to each aluminum support mast by a thin titanium ring which provides the required RF shielding with minimum heat leak from the mast to the cover.

The camera head contains a low-powered (4-W) thermostatically controlled heater that provides just enough heat to keep the insulated camera above its lower acceptance limit of $-50°F$ in the coldest environment. Under these cold conditions, the heat leak from the lander interior up the support mast and power cable is designed to be just enough to prevent the two masts from draining heat away from the camera. This design maintains the cameras above their minimum temperature with very little electrical power and very little heat drain from the lander.

### Thermal Coatings

The radiation properties established for most external lander surfaces are an infrared emissivity ($\epsilon$) of 0.85 minimum and a solar absorptivity ($\alpha_s$) of $0.5 \pm 0.05$. The high $\epsilon$ value is required to provide sufficient radiant heat rejection to prevent lander equipment overheating under interplanetary cruise and Mars surface hot case conditions. The absorptivity is defined to satisfy two opposing considerations: 1) a low absorptivity is desired to minimize temperature rise during Mars hot daytime hours, and 2) a high absorptivity is desired to minimize surface glare, which is degrading to camera operation. Although these properties are requirements for external coating, the thermal subsystem is designed to perform with external surfaces covered with dust, which may have $\alpha_s$ values up to 0.85.

Another factor considered in selection of the external coating is resistance to windblown sand and dust erosion. Several different finishes and coatings were subjected to tests under the specified combinations of wind velocity (up to 130 fps), temperature ($-195°F$ to $+95°F$), and dust particle size (1-50$\mu$) and flux ($8 \times 10^{-4}$ gm/cm$^2$-sec). The only coating to completely withstand this environment was a silicone type (General Electric RTV 511) applied to a thickness of 0.010 in. This coating is used on external insulation fabric covers as well as on metal and plastic surfaces. The selected coating resists erosion by being relatively soft and flexible. Harder coatings were severely eroded by the test environments.

To minimize temperature gradients indise the lander, all exposed surfaces of the equipment mounting plate and most components are coated with a flat black paint (glyceryl phthalate) which has a minimum specified emissivity of 0.85. The excepted surfaces are those of the tape recorder and biology mechanical subsystem which are required to be relatively isolated from the compartment environment.

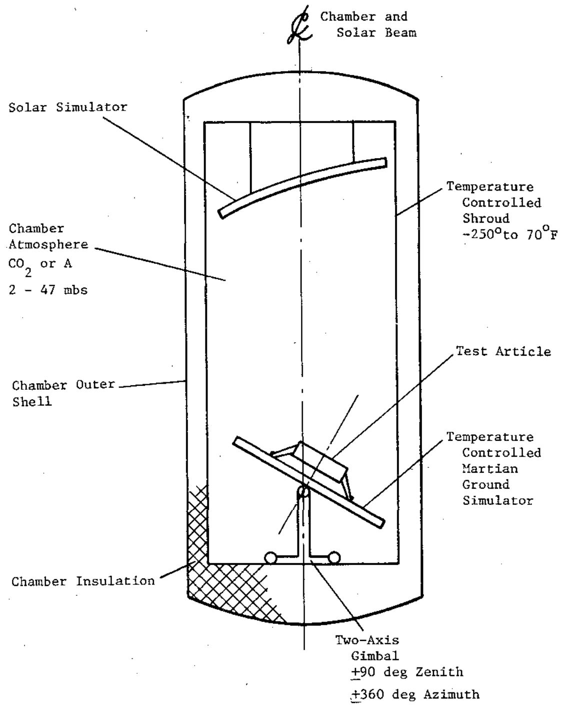


Fig. 9 Test configuration schematic.

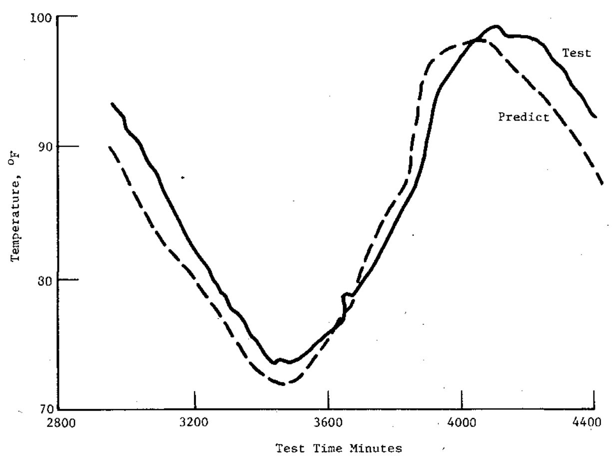


Fig. 10 Battery temperature history, Mars hot.

### Terminal Propulsion System Thermal Control

The primary terminal descent propulsion system thermal problem is prevention of hydrazine propellant freezing before system operation during the terminal descent and landing on Mars. The two spherical propellant tanks have contoured heater blankets bonded to the external surfaces. The heaters are controlled by mechanical thermostats to maintain propellant tank temperatures between 75°F and 86°F during the cruise phase. Two completely redundant circuits are used on each tank, with each circuit having a heater element and two thermostats in series. The propellant tanks, roll engines, and propellant valves are all enclosed in an aluminized mylar multilayer insulation blanket. Power requirements for the pre-landing operation of the other lander components necessitate disabling the heater circuits about 35 hr before the terminal descent phase. The effective thermal diffusivity of the insulated tanks limits the propellant temperature drop to 28°F during this period. Thus, the minimum propellant temperature is 47°F compared to the propellant freezing point of 35°F.

All of the propellant lines outside the heated tank insulation enclosure are dry until just before the terminal descent engine firing, so it is not necessary to protect them against freezing

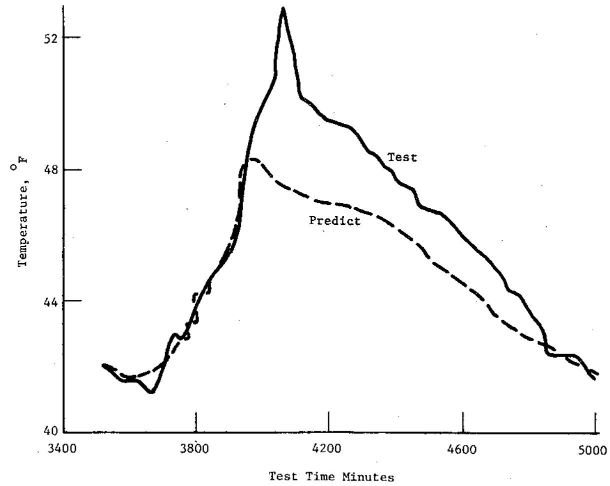


Fig. 11 Battery temperature history, Mars cold.

before engine firing. Even though the initial line temperature may be as low as −25°F, freezing is still not a problem at the onset of propellant flow because of the low heat capacity of the lines compared to that of the propellant. During propellant outflow, the propellant temperature is maintain by the ½-in. thick fiberglass insulation blankets around the lines.

### Lander Thermal Performance

Operational performance of the Viking lander thermal control subsystem has been demonstrated in simulated Mars surface environments using the thermal effects test model (TETM).[5] The TETM lander structure was built to flight vehicle design drawings and specifications. Thermal simulators were designed and built to represent electronics components, science experiments, and deployable mechanisms. Each simulator was designed to the same size, shape, heat capacity, and heat generation characteristics as the component it represented. Thermal coatings were applied to proper surfaces in accordance with flight vehicle process specifications. Flight-type insulation, propulsion subsystem heaters and thermostats were used. The thermal switches used in the TETM were development units that were operationally the same as flight units, but did differ in some design details.

The landed portion of the Viking mission was simulated in the Martin Marietta Denver Division 29×65 ft thermal vacuum chamber.[6] The lander was mounted on a gimbaled ground plane simulator, as illustrated in Fig. 9. The Mars surface diurnal thermal environments were simulated by means of variable temperature cold wall, ground plane simulator heaters, low pressure atmosphere and a 16-ft diam solar beam. A total of 648 thermocouples were monitored and recorded during the tests.

Three different diurnal cycle simulations were performed. Each cycle was performed three or four times to attain repetitive temperature profiles. Run 231 simulated the coldest environment expected. Runs 223 and 227 were both simulations of the hot design environments; the difference in the two runs being that in Run 223 the solar beam intensity was increased to 170% of the maximum value expected on the Mars surface during the Viking mission to simulate the higher absorptivity of dust covering the lander body. The resulting environment of Run 223 was hotter than desired, because the increased solar flux caused the ground plane simulator temperature to rise above its specified value. The actual hot design case environment is slightly less severe than that attained in Run 223. A complete discussion of the Viking Mars surface thermal test program, including the rationale used in defining chamber environments, is presented in Ref. 5.

A summary of component temperatures together with applicable qualification limits is presented in Table 2. Typical

Table 2 Mars surface test temperature summary component mounting interfaces

| Component | Qualification temperatures | Run 223 max temperature | Run 227 max temperature | Run 231 min temperature |
|---|---|---|---|---|
| Battery 1/2 | 30°F to 100°F | 103°F | 86°F | 49°F |
| Battery 3/4 | 30°F to 100°F | 102°F | 90°F | 46°F |
| Command control unit (CCU) | −15°F to 125°F | 92°F | 75°F | 4°F |
| Data acquisition and processing unit (DAPU) | −25°F to 125°F | 102°F | 82°F | 24°F |
| Data storage memory (DSM) | −25°F to 125°F | 105°F | 96°F | 64°F |
| Guidance control and Sequencing computer (GCSC) | −25°F to 125°F | 96°F | 80°F | 14°F |
| Microwave components assembly (MCA) | −15°F to 125°F | 94°F | 94°F | 10°F |
| Meteorology electronics assembly (MEA) | −25°F to 125°F | 106°F | 84°F | 24°F |
| Power control distribution assembly (PCDA) | −25°F to 125°F | 94°F | 78°F | 10°F |
| Pressure transducer | −65°F to 130°F | 100°F | 78°F | −18°F |
| Seismometer electronics assembly (SEA) | −25°F to 125°F | 93°F | 76°F | 0°F |
| Soil sampler control assembly (SSCA) | −25°F to 125°F | 100°F | 80°F | 0°F |
| Transponder 1 and 2 | −15°F to 125°F | 92°F | 78°F | 24°F |
| Traveling wave tube amplifier (TWTA 1) | −15°F to 130°F | 104°F | 96°F | 14°F |
| Traveling wave tube amplifier (TWTA 2) | −15°F to 130°F | 104°F | 96°F | 14°F |
| Ultra high frequency transmitter (UHF) | −15°F to 125°F | 106°F | 95°F | 22°F |
| Biology experiment | −15°F to 95°F[a] | 105°F | 82°F | −4°F |
| Camera | −35°F to 118°F[a] | 105°F | 79°F | −14°F |
| Gas chromatograph/mass spectrometer (GCMS) | −18°F to 110°F[a] | 102°F | 76°F | −8°F |
| X-ray fluorescence (XRFS) | −25°F to 118°F[a] | 98°F | 78°F | 10°F |

[a] Qual limits for these components are defined by diurnal curves for hot and cold cases. Only minimum and maximum points are given.

diurnal temperature histories for one of the battery measurements are shown in Figs. 10 and 11. In comparing test results with pretest predictions, it was found that most temperatures correlated within 10°F for the cold test 231, and within 5°F for the hot tests 223 and 227. The major concern during the landed operations is the absence of adequate margin in battery and biology experiment temperatures under hot extreme environments.

## Conclusions

A planetary lander is required to meet a variety of conditions, such as prelaunch sterilization, long-term cruise in space, and a wide range of planetary environments. Therefore, the thermal design of such a lander must be an integral part of the overall vehicle design and operation. Extensive use should be made of any available heat sources and sinks, all structural members should be designed to provide maximum or minimum heat conduction as required; and the use of passive techniques should be maximized.

For the Viking mission, the Mars surface environments present the greatest problems, and therefore have the greatest impact on lander design. The lander interior must be isolated well enough from the surface conditions to make it insensitive to the wide variations in surface and atmospheric temperatures, solar flux and wind velocity, but still allow the internally generated heat to be dissipated. The thermal system developed for the Viking lander meets all of the requirements while placing few constraints on mission performance.

The most critical active component is the thermal switch which was developed specifically for this application. It has successfully undergone exhaustive development and qualification programs and has successfully performed in the full scale vehicle thermal tests. All other critical components and materials and the entire thermal design have also been proven by subjecting the full scale test vehicle to a wide range of simulated mission conditions.

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