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APOLLO EXPERIENCE REPORT - THERMAL PROTECTION FROM ENGINE-PLUME ENVIRONMENTS

by J. Thomas Taylor

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16. Abstract

Portions of the combined Apollo spacecraft (the command and service module and the lunar module) are subjected to the impingement of hot exhaust gases from the various propulsion systems of the modules. In this report, the configurations of the vehicles and the sources of impinging engine plumes are described. A typical Apollo mission is outlined. Protection and design-verification methods are discussed. Finally, recommendations are made for future spacecraft programs.

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THERMAL PROTECTION FROM ENGINE-PLUME ENVIRONMENTS

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SUMMARY

Portions of the combined Apollo spacecraft (the command and service module and the lunar module) are subjected to the impingement of hot exhaust gases from the various propulsion systems of the modules. The operational requirements of these propulsion systems and the peculiarities of the total design of each spacecraft module determined the design approach for plume-impingement protection. The design verification of the plume protection was accomplished by analyses or by test or by both, depending on the configuration complexity, confidence in analysis, and allowable design conservatism. The successful completion of several Apollo flights has proved the adequacy of the plume-protection designs.

INTRODUCTION

The maneuverability of spacecraft is dependent upon the ability to apply a propulsive force to the vehicle. Such propulsive forces on the Apollo spacecraft result from the expansion of hot gases that, in some cases, partially impinge on the spacecraft, thereby presenting a heating source to the spacecraft. In this report, these rocket exhaust plumes are discussed, and the mission requirements for these plumes, the resulting heating of the vehicles, and (primarily) the methods used to protect the vehicles are described.

The command and service module (CSM) and the lunar module (LM) have two and three sources, respectively, of plume impingement. Basic spacecraft-design constraints to plume-protection design methods are discussed. Design features resulting from engine-nozzle thermal radiation and engine temperatures are not discussed in this report because they are dependent on engine and nozzle designs and not on the plume environment or characteristics.

An analytical definition of the chemical processes that occur during the rocket-engine-propellant combustion and the combustion-product expansion through the nozzle is highly complex. This complexity, in addition to the complex flow fields resulting from the LM geometry, required that scale-model testing be performed to determine heating rates; selected test data are presented. In the case of the LM reaction control system (RCS) plume impingement, full-scale portions of the LM vehicle were tested in the Manned Spacecraft Center (MSC) Space Environment Simulation Laboratory (SESL) to verify the scale-model tests performed by the LM prime contractor. Full-scale and scale-model test data are presented.

VEHICLE CONFIGURATION AND PLUME SOURCES

The Apollo spacecraft (fig. 1) consists of a command module (CM), a service module (SM), and an LM, each of which is associated with a special function of the lunar mission. Each vehicle configuration will be discussed separately.

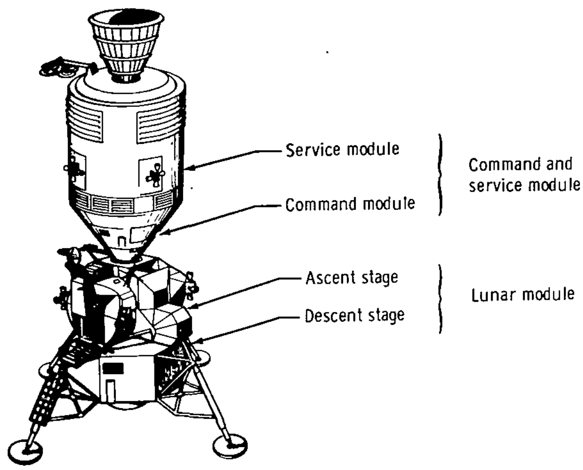


Figure 1. - Apollo spacecraft in docked configuration.

Command Module

The CM (fig. 1) is the only part of the spacecraft that returns to earth; therefore it is required to function as an atmospheric-entry and landing vehicle. The CM serves as the control center and crew quarters during most of the mission.

In the launch configuration (fig. 2), a launch escape system (LES) is located directly above the CM and is jettisoned during launch at an altitude of approximately 250 000 feet. During tower jettison, there is no appreciable plume impingement on the spacecraft. The CM thermal protection system (ablator) protects the crew if the escape rockets must be fired.

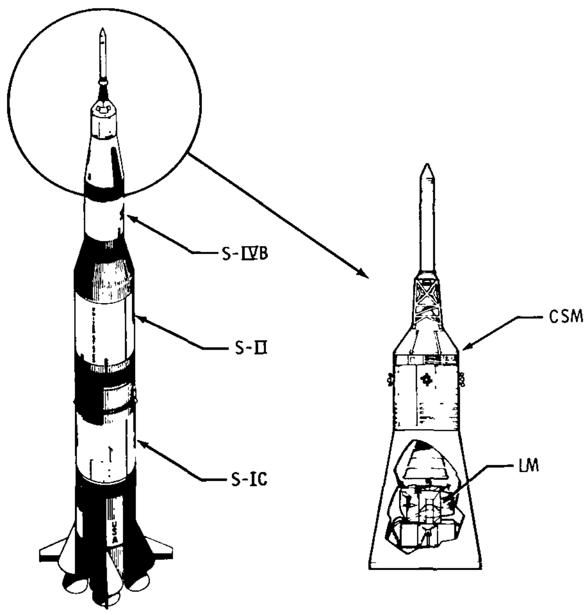


Figure 2. - Apollo launch configuration.

The CM has its own attitude control system for use after separation from the SM during atmospheric entry. Six reaction-control engines and a redundant six-engine system are provided for this control; these engines are located as shown in figure 3. The plumes from these engines do not impose any thermal-design problems because they do not impinge significantly on any surface of the CM or other vehicles. During atmospheric entry, the firing time is

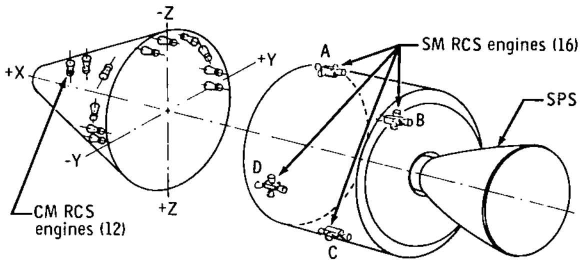


Figure 3. - Command and service module engine location.

too short to affect the entry heating significantly (ref. 1). The plumes from the LES and the RCS will not be discussed further because they do not affect the spacecraft thermal design.

Service Module

The SM is a cylinder with a diameter of 154 inches and a length of 155 inches. It contains consumables and the electrical power subsystem (except for entry batteries in the CM), RCS radiators, the main propulsion subsystem, and the attitude control subsystem. The SM provides attitude control for the entire Apollo spacecraft during most of the mission by means of the RCS, located as shown in figure 3. Plume gases from the RCS impinge on the SM skin and, in the docking configuration, on the LM. Therefore, plume-impingement protection is required for both the SM and the LM; this will be discussed in a subsequent section. The service propulsion system (SPS) is housed in the SM structure and provides the main propulsion for the spacecraft. The SPS exhaust plume does not impinge on any portion of the spacecraft; and, consequently, no associated thermal protection is required. An SM base heat shield is provided, however, to protect the structure from plume and nozzle radiation.

Lunar Module

The LM (fig. 4) includes an ascent stage (AS) and a descent stage (DS). The AS of the LM serves as the control center and living quarters for the two-man crew when the LM is activated. The AS contains two propulsion systems: the ascent propulsion system (APS) and the RCS for attitude control during independent flight of the LM. Both systems are a source of plume impingement and affect the thermal design of the LM. The RCS not only heavily influences the thermal protection system of the LM but also that of the CM, because the plumes from the forward-firing engines impinge on the CM thermal-control coating when the LM is docked to the CSM.

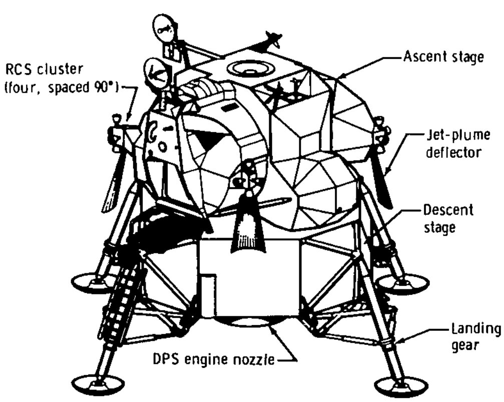


Figure 4. - Lunar module configuration.

The DS contains the landing mechanism or landing gear, auxiliary crew consumables, batteries, stowage for scientific and extravehicular equipment, and the descent propulsion system (DPS) used to slow the LM to an acceptable landing velocity (fig. 4). The plume from the DPS impinges on a portion of the landing-gear probe during descent firing. As the LM approaches the lunar surface, the plume is deflected upward by the lunar surface and impinges on the bottom surface of the DS and most of the landing gear, thereby requiring these areas to have additional thermal protection.

MISSION DESCRIPTION

The Apollo spacecraft is injected into earth orbit by the Saturn V (S-V) launch vehicle in the configuration shown in figure 2. After earth orbit has been achieved, the CSM and LM are injected into translunar coast by the Saturn IVB (S-IVB) stage of the launch vehicle. The two spacecraft are still in the launch configuration at this time. After a successful translunar injection, the CSM separates from the launch configuration, and the spacecraft lunar module adapter (SLA) section (housing the LM) is jettisoned. The CSM performs a transposition maneuver and, using the SM RCS for closure and attitude control, docks with the LM. Both the LM and SM receive plume impingement from the SM RCS engines during transposition and docking. The docked spacecraft are then separated from the S-IVB booster stage by springs.

During translunar coast, the LM is inactive. The spacecraft is oriented so that the longitudinal axis is within $\pm 20^{\circ}$ of perpendicular to the rays of the sun. The vehicle is then rotated about the longitudinal axis from 1 to 3 revolutions per hour to distribute the solar heating. This maneuver is known as the passive thermal control (PTC) mode and is initiated with the SM RCS. The spacecraft maintains the PTC mode throughout translunar coast except for short-duration attitude holds such as those for midcourse corrections, television transmission, navigational sightings, and so forth. The SM RCS is used to maintain attitude during attitude holds and to reinitiate the PTC mode as required during the entire mission.

The SPS injects the Apollo spacecraft into lunar orbit. After lunar orbit has been attained, the LM crew transfers to the LM and performs a final systems activation and checkout. After the LM checkout procedures have been completed, the SM RCS engines are fired and the CSM separates from the LM. At the appropriate time, the DPS engine is fired, using the LM RCS for attitude correction, for descent-orbit insertion (DOI) and is then fired a second time for the actual descent to the lunar surface. During this phase, the LM RCS is used extensively for attitude correction and for any required landing-site redesignation. Both the AS and the DS are subjected to plume impingement. As the LM approaches the lunar surface, at approximately 15 feet altitude, the DPS engine plume reflects from the lunar surface and impinges on the DS landing gear and base-heat-shield area.

After the lunar stay, the AS is injected into lunar orbit by the APS engine. The separation of the AS from the DS results in high initial pressures and heating rates on the top deck of the DS and on the bottom of the AS until enough separation between the stages is obtained to allow free plume expansion without reflection from the DS or lunar surface. During lunar ascent, the LM RCS is fired repeatedly for attitude control.

After lunar orbit has been achieved, the LM and the CSM dock. At this point, either the CSM or the LM has the capability to accomplish the docking. Therefore, one of the spacecraft can be subjected to the RCS plume of the other in a docked or near-docked configuration.

While in lunar orbit, the AS is jettisoned from the CSM. The SPS is fired to inject the CSM into a transearth-coast trajectory. During this phase, CSM attitudes are similar to those of the translunar-coast phase. Just before entry, the CM separates from the SM. During entry, the CM RCS provides attitude control for aerodynamic flight through the atmosphere.

The foregoing general mission description provides an insight into the various plume-impingement occurrences and into the interactions between the vehicles and stages. Design criteria resulting from the mission requirements are discussed later.

PROTECTION METHODS AND DESIGN VERIFICATION

Several methods of plume-impingement protection are used on the Apollo spacecraft. These are ablation, heat sinks, multilayer radiation shields, and deflectors. The choice of designs is predicated on the basic structure, thermal-control design, allowable temperatures, frequency and level of heating, and weight.

Command Module

As pointed out previously, the CM is subjected to SM RCS, LM RCS, and CM RCS plume impingement. However, the heat-shield design for entry conditions (ablation) has sufficient heat-sink capability so that heating from the SM RCS and the CM RCS is negligible. Heating from the LM RCS on the CM is negligible from the standpoint of affecting the heat-shield material, but effects on the CM-surface thermal-control coating are sufficiently significant to be a matter of concern.

When the LM is active and in the docked configuration, the firing of the LM RCS engines results in plume impingement on the CM thermal-control coating. The CM coating is an aluminized polyimide tape with a solar-absorptance-to-infrared-emittance ratio of 0.4 and a hemispherical emittance of 0.4. Significant degradation in the CM coating properties would cause the CM ablator temperature to approach the maximum and minimum temperature limits during long attitude holds, resulting in wider internal-cabin-temperature excursions and possible degradation of the heat-shield structural integrity.

During full-scale integrated thermal-vacuum testing of the CSM in the SESL, it was discovered that gases trapped under the coating expanded and formed bubbles under the CM tape when exposed to vacuum conditions. Because the thermal capacity of the tape is small, plume impingement could have caused severe damage to the bubbled tape, resulting in undesirable surface properties. Efforts to solve the bubbling problem by perforating the tape, both before and after application, did not improve the situation. Subsequent testing to assess the problem showed that the condition was acceptable, as discussed in the following section.

Command Module Design Verification

The CM coating was required to withstand 6 seconds of LM RCS plume impingement while the CSM and the LM were in a docked or near-docked configuration. Maximum heating from the plume was estimated analytically to be 1.33 Btu/ft^2-sec. Tests were conducted on five test panels in the MSC 10-megawatt arc-heated wind tunnel. The test conditions are presented in table I. Four of these panels were exposed to more severe conditions than required.

TABLE I. - APOLLO CM THERMAL-CONTROL TAPE

PLUME-IMPINGEMENT TEST CONDITIONS

Test panel number	Initial temperature, °F	Test duration, sec	Heating rate, Btu/ft^2-sec	Test chamber pressure, mm Hg	Stream total enthalpy, Btu/lb
1	210	15.0	0.60	~0.5	13 000
2	210	15.0	.77	~.5	13 000
3	210	15.0	1.30	~.5	13 000
4	60	30.0	.79	~.5	13 000
5	200	6.5	1.00	~.5	13 000

As in the full-scale test, bubbles formed under the tape. The test results indicated that the tape was capable of withstanding the plume heating without significantly affecting the physical integrity or thermal-control characteristics of the tape. Additional test specimens were subjected to a continuous 6 seconds of full-scale RCS plume impingement in the SESL. The specimen locations, relative to the RCS engine, were representative of the CM surface and LM RCS engine (scaled) geometry. The axial separation distance was selected so as to assure that the coating experienced a heating rate of 1.3 Btu/ft^2-sec. During post-test inspection of the samples, it was noted that there was very slight discoloration of the surface and no physical damage to the areas that had bubbled. Post-test measurement of the thermal-control-coating emittance and solar absorptance showed negligible increases of 0.002 to 0.052 in emittance and 0.002 and 0.009 in absorptance properties.

Service Module

The SM, which is affected only by plumes from the SM RCS, uses a combined heat-sink/ablation method. Cork material is applied to the areas where the basic structure does not have sufficient heat sink to prevent the allowable honeycomb-structure temperature limits from being exceeded. The initial cork thickness was based on design-trajectory boost heating rates and provided sufficient protection for the normal RCS duty cycles encountered during a mission. However, additional cork was added for the contingency case where the SPS is inoperative and the SM RCS must be used for earth deorbit. In this case, the cork serves as an ablator for the 750 seconds of firing required.

Heating rates on the SM were determined analytically by converting free-plume data to flat-plate heating rates. A heating map of RCS plume impingement on the SM is shown in figure 5. The cork pattern and thickness are shown in figure 6 for a representative section of the SM. The thickest cork, 0.155 inch, is located directly under the engine nozzles and on those areas experiencing the high heating rates (1.3 Btu/ft^2-sec) from the +X and -X firing engines. These were the only areas requiring additional cork, other than for boost, to accommodate the 750-second RCS deorbit contingency.

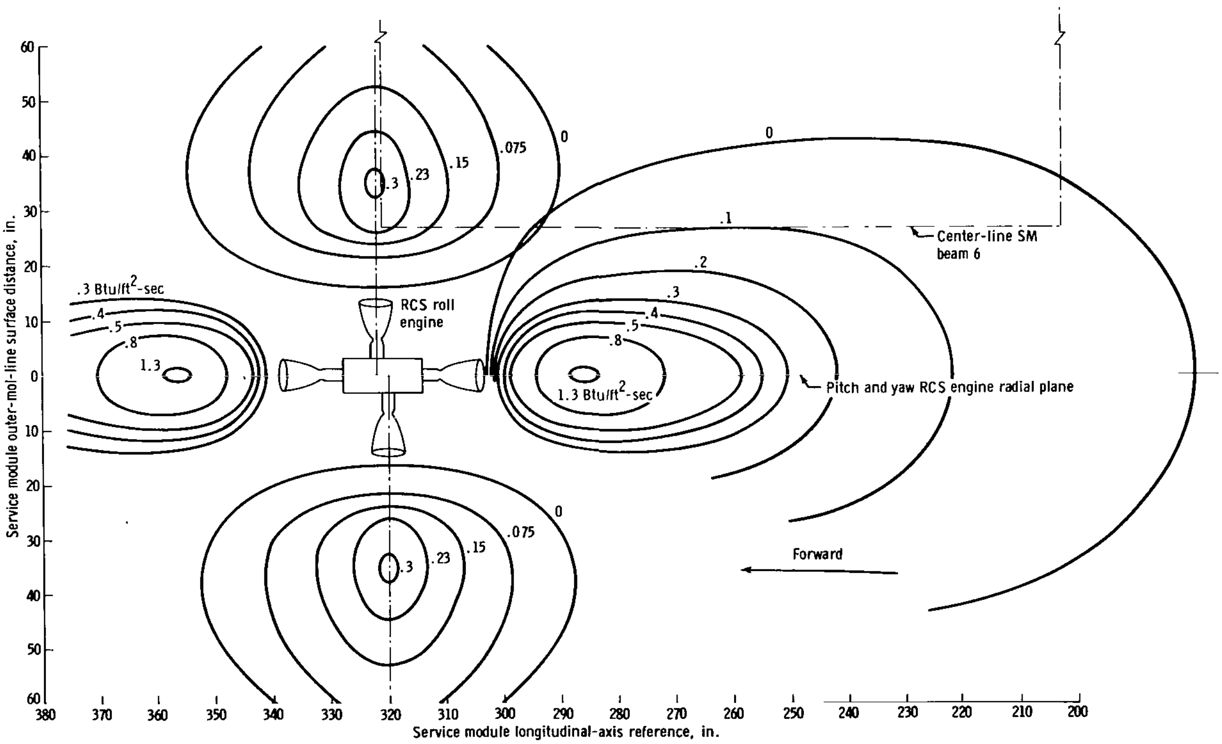


Figure 5. - Service module RCS plume heating map.

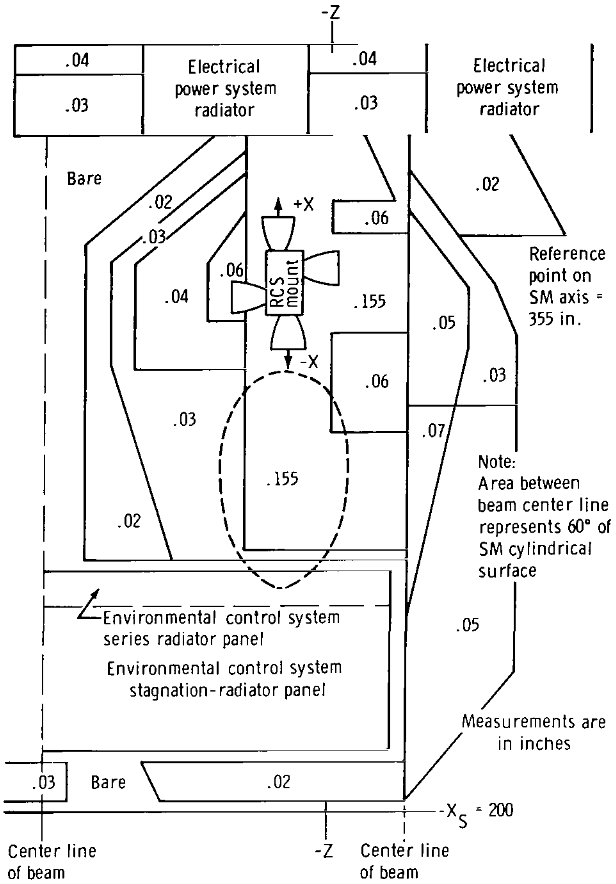


Figure 6. - Typical SM cork pattern.

The allowable SM aluminum temperature during boost and RCS impingement is 400° F. Because the SM does not perform any function during entry but must provide deorbit capability, the temperature constraint was relaxed for the deorbit contingency. The cork was sized to prevent structural and insulation burnthrough in order to prevent plume heating of the propellant tanks beyond allowable limits.

Service Module Design Verification

Temperature predictions for the maximum heating, 1.3 Btu/ft^2-sec, are shown in figure 7. Material properties used in the design are presented in table II. In the analysis, it was assumed that the cork would perform as a subliming ablator. However, tests conducted in the MSC 10-megawatt arc-heated wind tunnel showed that the cork performed as a charring ablator, which would result in lower structural

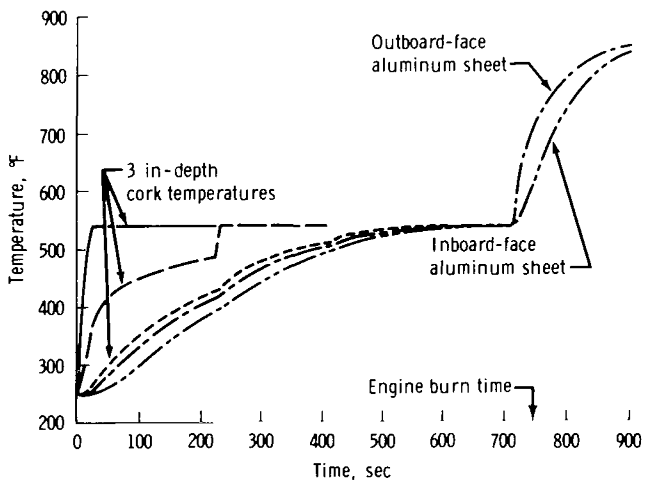


Figure 7. - Reaction control system panel plume-impingement temperature history (data point B, heating rate = 1.3 Btu/sec-ft^2, 750-second RCS deorbit).

TABLE II. - SERVICE MODULE MATERIAL PROPERTIES

Material	Density, lb/in^3	Specific heat, Btu/lb, °F	Thermal conductivity, Btu/hr-ft, °F
Cork (MBU 130-020), type 1	0.0173	0.47	0.047
2024 (T81) aluminum honeycomb, outboard-face sheet	.100	.22	86.0
7178 (T6) aluminum honeycomb, inboard-face sheet	.102	.22	70.0
5052 aluminum honeycomb core	[a].00468 or [b].0026	.22	80.0

[a]With 0.003-inch-thick foil core.

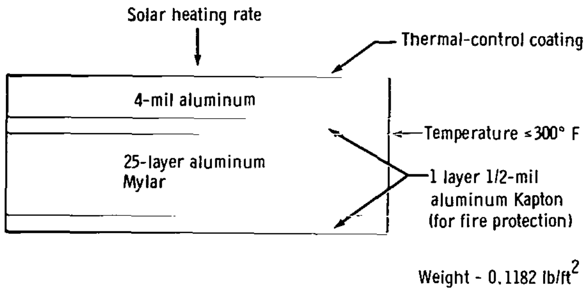
[b]With 0.001-inch-thick foil core.

temperatures than predicted. Based on these test results, additional analyses showed that the maximum temperature that the charred cork would reach was approximately 800° F. It was concluded that structural failure would not occur.

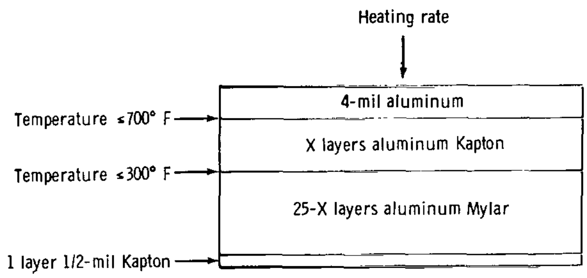
Lunar Module

The LM, which receives plume impingement at one time or another over nearly 100 percent of its surface, has heat-sink, multilayer-radiation-shield, and plume-deflector designs. The basic LM thermal design is one of isolation; that is, the available sensible heat at launch, primarily from propellant, is conserved throughout the mission by isolation from the space environment. Because the LM is not subjected to the boost environment (aerodynamic or thermal), nonstructural, lightweight materials are applied to the outer surfaces of the vehicle without incurring large weight penalties. The basic design (fig. 8) has 25 layers of 1/8-mil aluminized polyester film encapsulated in inner and outer 1/2-mil layers of aluminized polyimide film and an outer micrometeoroid shield of aluminum.

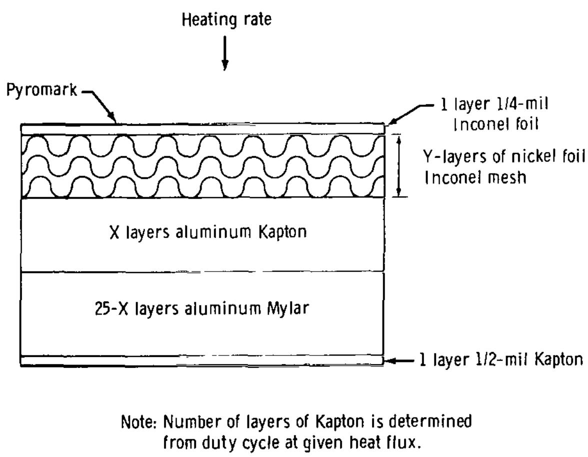
As discussed previously, the LM is subjected to plume impingement from the SM RCS, the LM RCS, the LM APS, and the LM DPS. The basic design just described was modified as required to provide minimum-weight protection from the various plume sources without degradation or reduction of the basic insulation requirements. This modification was achieved by substituting layers of polyimide film, nickel foil, Inconel mesh, and Inconel foil for all or part of the basic multilayer blanket and aluminum outer skin (fig. 8). Each insulation blanket layup was tailored to meet particular plume-source duty cycles. The materials and pertinent properties used in the LM thermal-protection design are presented in table III.



(a) Basic LM thermal design.



(b) Reaction control system plume-impingement thermal-shield design, heating rate ≤ 0.5 Btu/ft^2-sec (design same for heating > 0.5 but < 1.0 Btu/ft^2-sec except for 8-mil aluminum).



(c) Reaction control system plume-impingement thermal-shield design, heating rate ≥ 1.0 Btu/ft^2-sec.

Figure 8.- Lunar module thermal-protection designs.

TABLE III. - LUNAR MODULE MATERIAL PROPERTIES

Material	Thickness, in.	Density, lb/ft^3	Specific heat, Btu/lb, °F	Emittance	Solar absorptance	Allowable maximum temperature, °F
Inconel	0.125×10^{-2}	555	0.05 to 0.138 (0 to 2300° F)	0.85 (black pyromark finish)	0.93 (black pyromark finish)	2300
Inconel mesh	$.249 \times 10^{-1}$	4.327	.05 to .138 (0 to 2300° F)	NA[a]	NA	2300
Nickel	$.500 \times 10^{-3}$	555	.01 to .145 (0 to 2300° F)	--	Not used on outer surfaces	2300 (used beneath Inconel)
Aluminized polyimide (H-film)	$.500 \times 10^{-3}$ $.999 \times 10^{-3}$ $.200 \times 10^{-2}$	89.0	.30	[b].49 [b].54 [b].65 [c].06	[b].33 [b].36 [b].40 [c].14	750 (based on 2 percent material shrinkage)
Aluminized polyester (Mylar)	$.150 \times 10^{-3}$	86	.315	[c].06 [b].5	[c].14 [b].14	375 (based on 2 percent material shrinkage)
Anodized aluminum	$.36 \times 10^{-2}$ $.648 \times 10^{-2}$	172.0	.23	.3	.42	--

[a]Not applicable.
[b]Film side.
[c]Aluminum side.

To minimize conservatism and thereby reduce weight, it was necessary to establish duty cycles based on simulations of the various maneuvers, such as CSM/LM docking and lunar descent, that require the use of the RCS. These simulations account for the systems interaction, vehicle dynamics, and crew capabilities that are pertinent to the performance of the particular maneuver. The plume-impingement design criteria for the LM are given in table IV. Before the first lunar landing (LM-5), differences existed among the plume-protection designs of each of the vehicles. These differences existed for several reasons: the mission did not necessitate conditions that required maximum protection; launch weight was not critical; and adequate simulation data were not available to support hardware schedules.

Simulations of the lunar landing resulted in LM RCS duty cycles that required increased plume protection. The increased RCS duty cycle for the lunar landing did not invalidate all plume-protection blanket layups but affected particular blankets and the structural integrity of several AS and DS blanket attachments. To minimize the resulting weight impact, methods were investigated to minimize or eliminate plume impingement from the downward-firing LM RCS engine. This investigation resulted in implementing a device to deflect the plume away from the vehicle on LM-5, the first LM to land on the moon. The design of this device is discussed later in this document.

TABLE IV.- LUNAR MODULE PLUME-IMPINGEMENT CRITERIA[a]

System	Stage	Vehicle configuration		Remarks
		Unstaged, sec	Staged, sec	
SM RCS	AS	5	--	SM RCS continuous firing during CSM/SLA separation
	DS	5	--	
	AS	7	7	Docking maneuvers
	DS	7	--	
LM RCS	AS	30 (continuous upward-firing jet)	440 (pulsed 50-percent duty cycle)	The 440 seconds apply only to the AS during powered ascent from the lunar surface.
		15 (continuous downward-firing jet)	--	
	DS	15 (continuous downward-firing jet)	--	--

[a]For LM-5 and subsequent lunar landers, the unstaged-downward-firing-jet requirement increased to 23 seconds at a 19.2-percent duty cycle for 120 seconds.

The previous discussion has been directed toward vehicle protection from direct plume impingement. Although the basic design approach is the same, the problem of DPS plume impingement during lunar landing, a problem peculiar to the LM, deserves attention. As pointed out in the mission description, the DPS plume impinges directly on only the lower portion of the landing probe until the engine nozzle is approximately 15 feet above the lunar surface. At this time, the plume deflects off the lunar surface and begins to impinge on the LM landing gear and base heat shield.

The convective heating rate to the secondary strut of the landing gear, as determined from shock-tunnel tests, is shown in figure 9 as a function of engine nozzle height above the lunar surface for the DPS "fire until touchdown" (FUT) mode. In addition to the convective heating, solar-, lunar-, and nozzle-radiation heat loads were considered.

The same data are presented in figure 10, but the effects of descent velocity compared with rock or platform heights (depicting terrain variations) and engine shutoff delay (astronaut response) on total heating are depicted.

Design criteria defined for protection from FUT heating are shown in table V, together with the actual conditions experienced during the first lunar landing, Apollo 11. Plume protection for the LM-5 secondary strut, which is representative of the landing gear and probe, is shown in figure 11. After Apollo 11, additional efforts were made to reduce the weight of the FUT heating protection. The lower temperature polyimide films were replaced with thin, high-temperature shields of Inconel and nickel. This approach allowed for greater heat rejection by radiation and thereby a reduction in the net heat absorbed by the basic insulation blanket and gear. The effect of this substitution on absorbed heat is shown in figure 12.

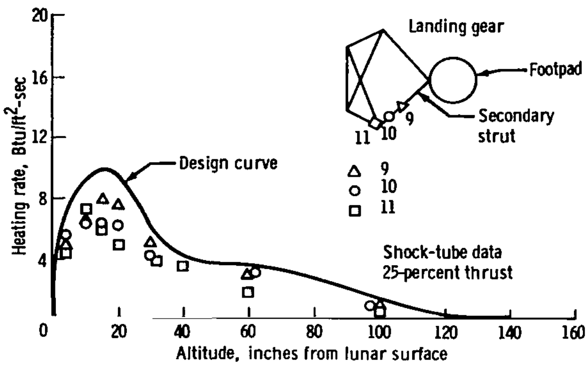


Figure 9. - Fire-until-touchdown convective heating, secondary strut.

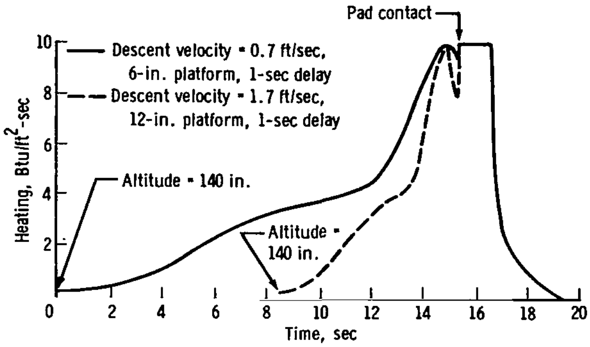
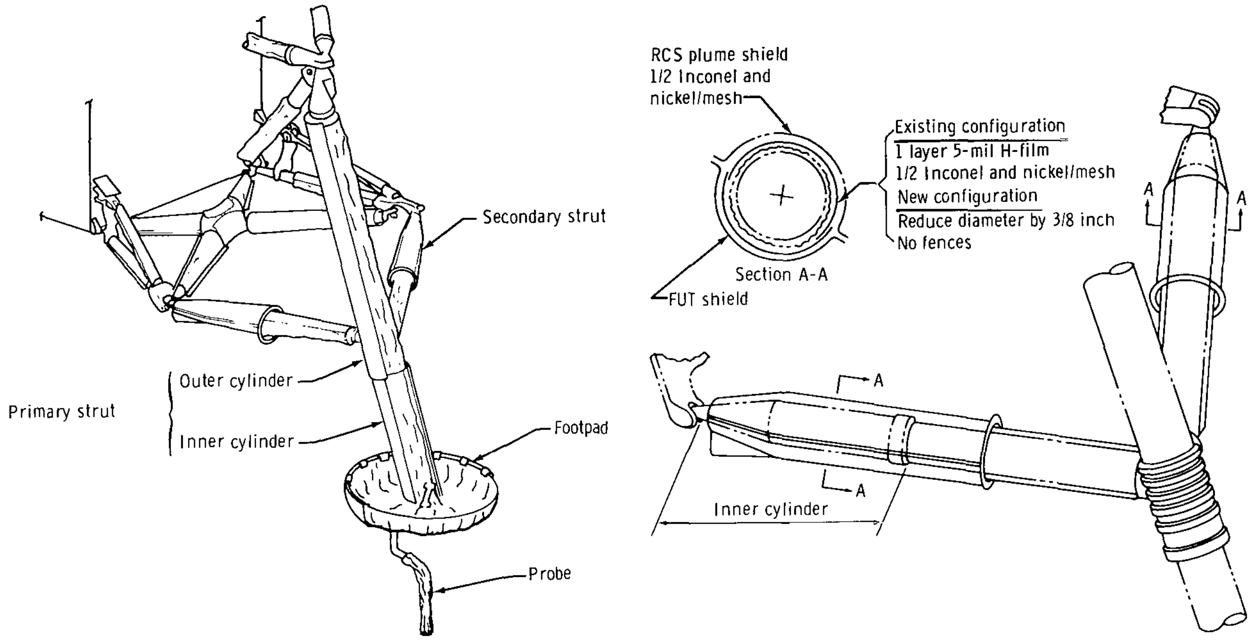


Figure 10. - Fire-until-touchdown convective heating, secondary strut, comparing effects of descent velocity with rock or platform heights and engine shutoff delay.

TABLE V. - FIRE-UNTIL-TOUCHDOWN DESIGN COMPARED WITH LM-5

Parameter	Design conditions	LM-5 flight conditions
Sink rate from 15 to 0 feet, ft/sec	0.7	0 to 1.8
Engine OFF (after pad contact), sec	1	1.6
Thrust, percent	[a]25	[b]23 and [a]10
Total heating, Btu/ft^2	85	55
Convection to contact plus 1 sec, Btu/ft^2	68	50
Convection from engine tailoff, Btu/ft^2	5	5
Lunar-surface radiation, Btu/ft^2	12	0
Analysis (heat absorbed), Btu/ft^2	85	55
Heat sink (temperature rise of H-film), Btu/ft^2	30	30
Charring (20 Btu/layer), Btu/ft^2	[c]55	[d]25

[a]To engine OFF.
[b]To touchdown.
[c]Three layers charred.
[d]One layer charred.



(a) Overall landing gear. (b) Secondary strut.

Figure 11. - Lunar module landing gear.

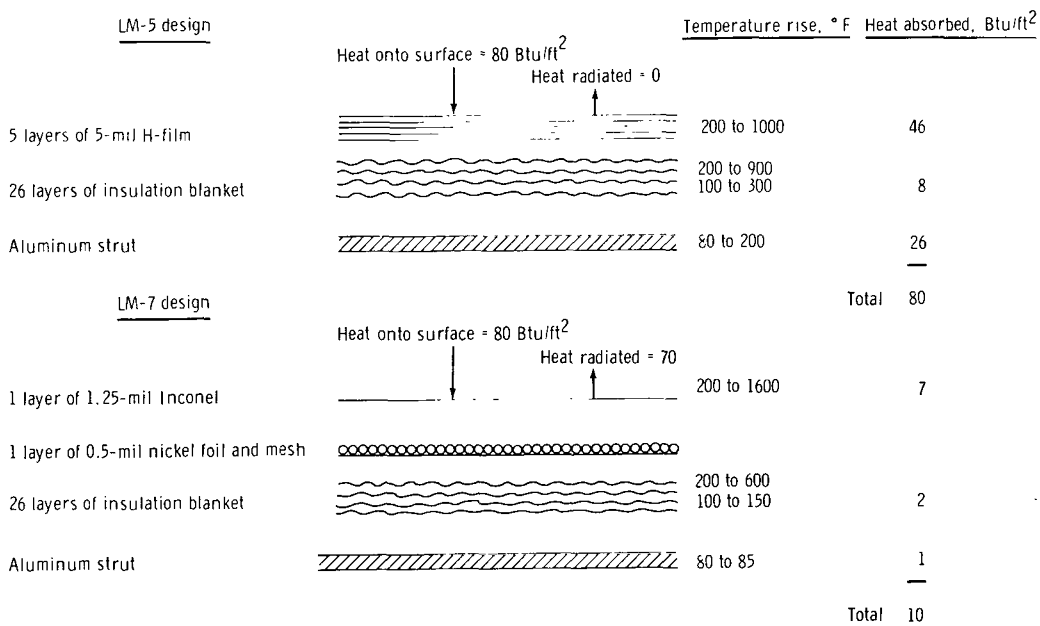


Figure 12. - Primary-strut outer-cylinder design comparison.

Design Verification of LM Plume-Protection Blankets

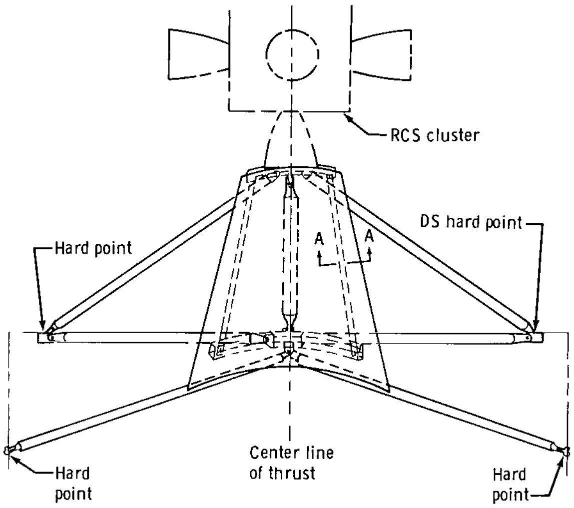
Verification of the plume-protection blankets was accomplished by tests of 2- by 2-foot blanket specimens. The specimens were subjected to simulated plume heating from quartz-lamp arrays in a vacuum chamber.

Before the flight of LM-1, no simulation of the effects of the plume-gas pressure on blanket thermal performance had been considered. Postflight analysis indicated that the outer layer did not reach expected temperature levels; however, inner-blanket temperatures were somewhat higher than expected, indicating degraded insulation performance. Additional blanket tests were performed using hot carbon dioxide gas impingement to simulate the plume. Also included in the test were comparisons of flight-thermocouple and laboratory-thermocouple response to determine the effects of thermocouple mass on the measurement of thin-foil temperature. Gas pressures from 0.008 to 0.02 psia were measured. The magnitude of the pressures was shown to be a strong function of the geometry of the blanket seams. These data, although not providing absolute design criteria, allowed verification of the LM plume-protection design.

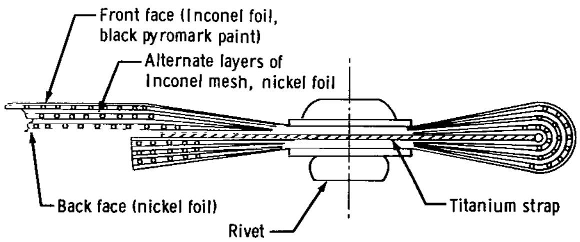
Verification of scale-model shock-tunnel heating-rate data was obtained from a full-scale RCS plume-impingement test conducted in the SESL in May 1969. A full-scale model of a section of the DS was subjected to plume impingement to obtain heating-rate and plume-pressure data.

Plume Deflector

To increase flexibility in lunar-landing-site real-time redesignation by the crew (requiring increased RCS firing), the plume deflector was chosen instead of redesigning the already fabricated LM plume-protection shields or the associated structure (or both). A view of the LM with three of the four deflectors visible is shown in figure 4. The deflector (fig. 13) is an open-section 47° arc length of a truncated 90° circular cone mounted below the downward-firing RCS engines and canted 10° outboard with respect to the engine center line. The deflector is constructed of Inconel, Inconel mesh, and nickel foil. Layer buildup is tailored to the heating profile of the deflector. A typical cross section is presented in figure 13(a). The deflector is attached to titanium



(a) Overall view.



(b) Section A-A.

Figure 13. - Plume deflector.

catenary straps on either side and to box beams at either end and is supported by six support struts and a center support rib.

The basic plume-protection design was based on a 15-second continuous RCS engine firing or equivalent duty cycle; however, landing simulation resulted in an increased requirement of 23 seconds at a 19.2-percent duty cycle during the final 120 seconds of lunar landing. The increased plume-impingement capability, resulting from addition of the plume deflector, is presented in figure 14. The break in the curve at 340 seconds is determined by the AS S-band steerable antenna, which is not protected by the deflectors.

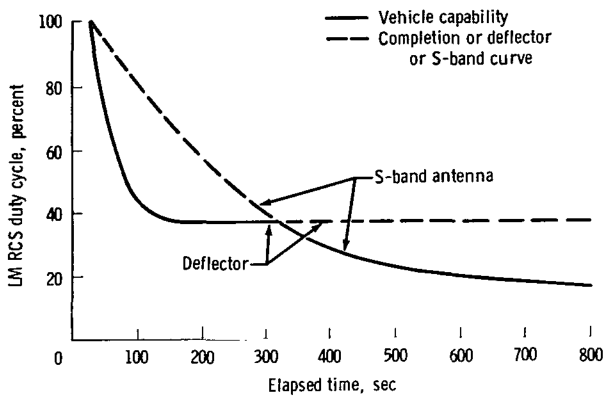


Figure 14. - Plume impingement capability, LM-5.

Deflector Design Verification

A full-scale RCS plume-impingement test was conducted in the SESL. The primary purposes of the test were to validate scaled plume-heating-rate data used in the LM plume-protection design and to subject the flight plume deflector to the design-landing RCS duty cycle. The test also included specimens used to determine free-plume characteristics, plume heating and surface pressures on LM surfaces, and the thermal performance of LM thermal-insulation blankets. The deflector was subjected to the RCS duty cycle shown in tables VI and VII. The deflector was subjected to the complete duty cycle without any structural or thermal failures. Thermocouple locations are shown in figure 15. Test data and analytical predictions for two front-face thermocouples and one back-face thermocouple are shown in figure 16. Although some front-face peak-temperature predictions are higher than the data shown, the heating and cooling rates agree very closely. Allowable front- and back-face temperatures were 2300° and 850° F, respectively. A back-face thermocouple response and predictions for two levels of interfoil pressure are shown in figure 16(a), which illustrates the effect of interfoil pressure levels on blanket performance. Complete test results are presented in reference 2. Correlation of the thermal mathematical model with test data allowed analytical verification of the deflector for a 40-percent duty cycle, as compared to the 19.2-percent design requirement.

TABLE VI. - LUNAR MODULE RCS DUTY CYCLE FROM START OF ULLAGE TO 500 FEET ALTITUDE

Simulation	OFF, sec	ON, sec
Ullage	0	4.0
	4	4.0
Initial gimbal mistrim	2	.1
Throttle up to full-throttle position	24	1.6
Throttle down	346	.75
Radar update	30	.3
Radar update	10	.2
Radar update	10	.2
Radar update	10	.2
Radar update	34	.4
Redesignation	25	.5
Redesignation	25	.5
Redesignation	34	.5
	55	
Totals	609	13.25

TABLE VII. - LUNAR MODULE RCS DUTY CYCLE FROM 500 FEET TO TOUCHDOWN[a]

ON pulse duration, sec	Number of ON pulses	Total ON time, sec
2.0	1	2.00
1.5	1	1.50
1.25	2	2.50
1.00	2	2.00
.75	3	2.25
.50	11	5.50
.20	17	3.40
.10	39	3.90
Totals	76	23.05

[a]Twenty-three seconds of engine-ON time during a period of 120 seconds or a duty cycle of 19.2 percent. The longest ON time will be 2 seconds, and the longest OFF time will be 6 seconds.

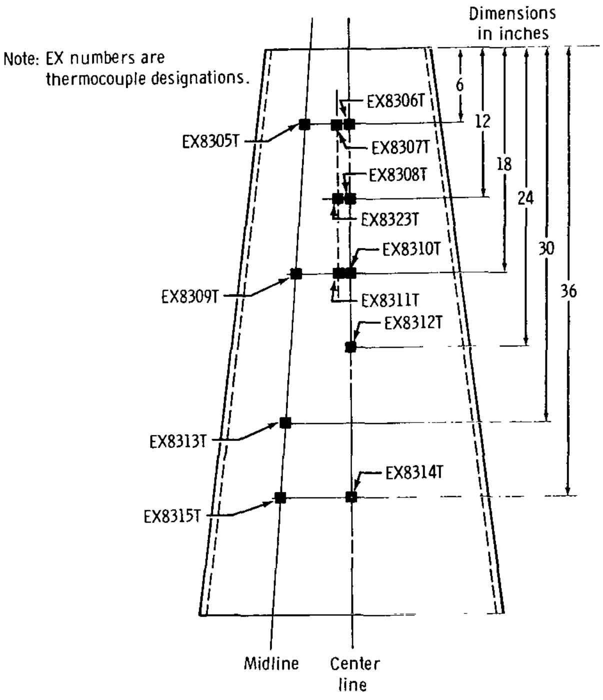
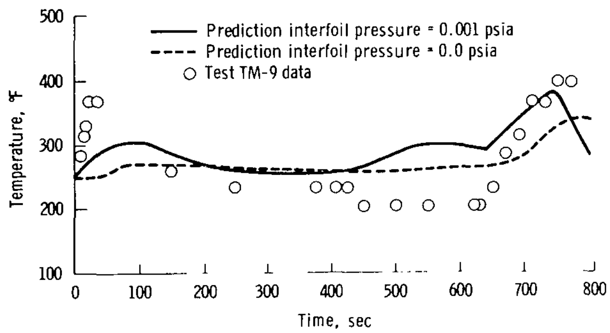
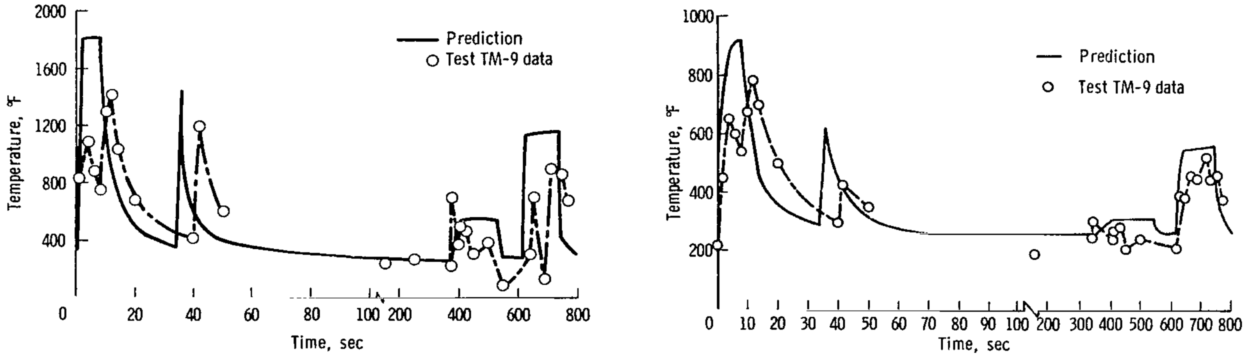


Figure 15. - Plume-deflector thermocouple locations for test TM-9.



(a) Back-face thermocouple number EX8307T.

Figure 16. - Test data and analytical predictions for two front-face thermocouples and one back-face thermocouple.



(b) Front-face thermocouple number EX8306T.

(c) Front-face thermocouple number EX8314T.

Figure 16. - Concluded.

CONCLUDING REMARKS

The methods and materials used for protecting spacecraft structures, components, and crew are very dependent on the basic spacecraft design and total environment. One major difference exists between the environments to which the command and service module and the lunar module are subjected: that of direct exposure to the boost aerodynamic environment. The command and service module, which is exposed to the aerodynamic environment, required plume-protection materials that could withstand the resulting structural loads and heating without degradation of plume-protection capabilities. By contrast, the lunar module was not exposed to the aerodynamic environment. Therefore, lightweight nonstructural materials could be used for insulation and plume protection on the outer surfaces of the vehicle.

To minimize the weight and complexity of the plume-protection design, it is necessary to define accurately heating rates and design criteria, such as engine duty cycles, at an early stage of the design and development of the spacecraft. The command and service module had a relatively simple surface configuration for which heating rates could be determined analytically with a reasonable degree of confidence. On the other hand, the lunar module had a complex arrangement of surfaces for which heating rates had to be determined by tests. This determination was accomplished through scale-model testing in the contractor shock tunnel and verified by full-scale tests in the Manned Spacecraft Center Space Environment Simulation Laboratory. This verification came relatively late in the program because engine duty-cycle requirements were redefined to accomplish the lunar landing. These tests were necessitated by the need to minimize weight and by the impact of insulation-panel redesign on vehicle delivery and launch schedules.

Obviously, early definition and verification of design parameters and criteria are ideal and should be strived for in initial program definition. Apollo Program experience has proved the requirement for much emphasis on the thermal-design discipline and early design verification. It is the author's opinion that thermal test, and analyses and supporting technological requirements should be defined in detail in the beginning, as they are in other areas such as structural design. The sensitivity of thermal analysis and the use of new materials require that caution be taken in minimizing the importance of relatively simple tests for obtaining data that can readily verify or replace extensive analyses.

The necessity for accurate design analysis and performance-prediction capability is becoming more and more important as new and more complex vehicle designs are implemented. The rule-of-thumb and handbook approach to thermal design of earth-bound systems is not yet applicable to spacecraft and spacecraft systems. Therefore, it is mandatory that the approach to thermal design and testing be as rigorous and complete as that of other spacecraft-design disciplines.

Limitations on the flight data taken to assess plume-protection-design performance required that much confidence be gained in the analytical tools and the correlation of analytical models with ground-test data. In addition, precise definitions of propulsion system performance and mission requirements and high-fidelity simulations were necessary for formulating the design requirements and criteria. The resulting designs were proved adequate throughout the step-by-step increase in mission complexity that culminated in the successful completion of the lunar-landing mission.

Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas, October 20, 1971
914-11-20-12-72

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