NASA CONTRACTOR REPORT

NASA CR-2579

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ABLATIVE HEAT SHIELD DESIGN FOR SPACE SHUTTLE, FINAL REPORT

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • OCTOBER 1975
5.

1. Report No. NASA CR-2579	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Ablative Heat Shield Design for Space Shuttle		5. Report Date OCTOBER 1975
		6. Performing Organization Code
7. Author(s) R. W. Seiferth		8. Performing Organization Report No. MCR-73-147
		10. Work Unit No.
9. Performing Organization Name and Address Martin Marietta Corporation Denver Division Denver, Colorado 80201		11. Contract or Grant No. NAS1-11592
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington D.C., 20546		13. Type of Report and Period Covered Executive Summary May 1972 - August 1973
		14. Sponsoring Agency Code

15. Supplementary Notes

Details of this Executive Summary are in Final Report, CR-132282 (Same Title)

FINAL REPORT

16. Abstract

Ablator heat shield configuration optimization studies were conducted for the Orbiter. Ablator and RSI trajectories for design studies were shaped to take advantage of the low conductance of ceramic RSI and high temperature capability of ablators. Comparative weights were established for the RSI system and for direct bond and mechanically attached ablator systems. Ablator system costs were determined for fabrication, installation and refurbishment. Cost penalties were assigned for payload weight penalties, if any. The direct bond ablator is lowest in weight and cost. A mechanically attached ablator using a magnesium subpanel is highly competitive for both weight and cost.

17. Key Words (Suggested by Author(s)) Space Shuttle, Heat Shield, Ablators, Design Criteria, Environmental Testing, Coatings, Cost Analyses, Trajectory Shaping, Reusable Surface Insulation	18. Distribution Statement Unclassified - Unlimited Subject Category 39

19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 28	22. Price* $3.75

* For sale by the National Technical Information Service, Springfield, Virginia 22151

FOREWORD

This report was prepared by Martin Marietta Corporation, Denver Division, Denver, Colorado, under Contract NAS1-11592, "Ablative Heat Shield Design for Space Shuttle."

The program was performed by members of the Ablative Systems Section, the Advanced Structures and Materials Department, the Mechanical Design Engineering Department and the Systems Analysis Department. The program manager for Martin Marietta from the start of the contract on May 17, 1972 to September 5, 1972 was Mr. Daniel V. Sallis; Mr. Rolf W. Seiferth was program manager from September 5, 1972 through August 31, 1973.

This final report covers the contract period from May 17, 1972 through August 31, 1973.

Many persons contributed during the performance of the program and in the preparation of this report.

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ABLATIVE HEAT SHIELD DESIGN FOR SPACE SHUTTLE

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SUMMARY

State-of-the-art ablative materials were used to design a thermal protection system (TPS) for the Space Shuttle Orbiter. An "ablator trajectory" was developed within the bounds of 2.5-g acceleration and 300 kW/m^2 (26 Btu/ft^2-sec) heating rate at the reference point 15.24 m (50 ft) aft of the fuselage nose on the bottom centerline. An "RSI trajectory" was also developed for design comparison purposes. This trajectory was shaped to minimize heating rate within the limits of skipout during reentry. Heating rates and total heats were developed for the total Orbiter. Ablative heat shield designs were derived for numerous locations on the Orbiter using direct bond and mechanically attached concepts. A reusable surface insulation (RSI) TPS was also developed for weight comparison purposes. Radiant heat tests were conducted on mechanically attached ablator specimens to verify design concepts.

A cost analysis was prepared for the various heat shield concepts. Weight was considered as a cost factor by determining a cost per pound to orbit using the "Preliminary Traffic Model for the Space Shuttle," (Shuttle Utilization Planning Office, NASA-MSFC, November 9, 1972). Ablator TPS operation was assumed for the first five years of Shuttle service. Cost data were derived for the operational phase and for reliability, which was treated as a quality assurance item. The sum of the weight costs, operational costs, and reliability costs was used to rate the various heat shield concepts and select this optimum ablator configuration.

The direct bond ablator system had the lowest weight and program cost of all the systems examined. Mechanically attached plates with ablator bonded to them are very competitive for both weight and cost.

A detailed description of this work is given in NASA CR-132282, "Ablative Heat Shield Design for Space Shuttle," by Rolf W. Seiferth.

I. INTRODUCTION

Ablators are a well-established system of thermal protection, having been used on such vehicles as Apollo, Gemini, Viking spacecraft, X-15, Titan, PRIME, and others. The need for ablator refurbishment following each flight is a drawback of this thermal protection system (TPS) and has led NASA and industry into the development of reusable surface insulation (RSI) ceramics. The RSI system of thermal protection has been baselined for use on the Shuttle Orbiter.

Much work needs to be done to qualify the RSI for Space Shuttle application and, to quote E. S. Love (ref. 1) from the Tenth Von Karman Lecture, "Ablators offer a confident fall-back solution (temporary) for both leading edges and large surface areas, should development of the baseline approaches lag."

In the past, ablator systems have been bonded directly onto the structures they are designed to protect. While this approach is both low in weight and cost effective, it has the drawback for the Shuttle Orbiter of taking up critical turnaround time for refurbishment between flights, and, during refurbishment, creates a problem of debris and dust control.

This program investigated Shuttle Orbiter TPS design concepts using available state-of-the-art ablators. An end objective of the program was to obtain ablator TPS weight and cost estimates based on detailed, verified heat shield designs. RSI cost estimates were not part of this program. Direct bond ablator and RSI designs were prepared for weight comparison purposes. A key part of the effort dealt with methods of mechanically attaching prepared ablator panels onto the Orbiter. Radiant heat tests were conducted to verify the design concepts.

The program was divided into five tasks:

Task 1 - Design Criteria;

Task 2 - Flight Environment;

Task 3 - Heat Shield Designs;

Task 4 - Design Verification;

Task 5 - Weight and Cost Analysis.

II. DESIGN CRITERIA (TASK 1)

Design criteria were prepared to develop valid heat shield configurations for the total environment from prelaunch through reentry and landing. Criteria were prepared for the trajectory definition and for the thermal and stress analysis.

Factors of Safety

Safety factors on loads and pressures for prelaunch through deorbit are based on engineering practice developed for boosters and spacecraft and for entry and atmospheric flight. The safety factors are those commonly used in the design of aircraft (table 1).

TABLE 1.- FACTORS OF SAFETY

Location	Ultimate design condition	(Ultimate or design factor of safety) times	(Limit load or nominal heating load)
Total Orbiter	Through orbit	1.4	Limit load
Total Orbiter	Entry	1.5	Limit load
Leading edges and lower forward surfaces	Ascent and entry	1.15	Nominal heating load
Lower surfaces aft		1.25	
Upper surfaces		1.50	

Backface (Structural) Temperature Limits

The backface temperature limits are as follows:

Ablator bond line	533°K (500°F)
Maximum structural temperatures	
At start of entry	311°K (100°F)
At completion of entry	450°K (350°F)

Ablator Strain Limits

Ablators have characteristically low mechanical properties. Although values for charred materials are difficult to ascertain, the following room temperature properties are typical:

Strength	480 kN/m^2 (70 psi)
Modulus	20 700 kN/m^2 (3000 psi)

In addition, the following induced strain limits have been developed for the Viking Project:

	Virgin	Charred
Tension strain	1.0%	0.6%
Compression strain,	1.0%	1.0%

Strength Analysis

The ablation material is not considered load carrying, but will be included in thermal and mechanical deflection analysis to determine the strain in the ablation material. The subpanel will be capable of carrying design loads without the ablator, and without exceeding the following surface waviness deflection criteria:

$H = 0.0125L$ limit

L = panel wave length

H = maximum deflection (wave height)

III. FLIGHT ENVIRONMENT (TASK 2)

Orbiter flight environments were defined for the boost, orbit, and entry conditions. Airloads and thermal histories were established, including interference heating during ascent. Heating load distributions accounted for the variances in time at which transition from laminar to turbulent flow occurs. Flow transition near the fuselage nose and wing leading edge occurs several hundred seconds following peak heating. Fully turbulent flow was assumed to occur at a location twice the length of the transition onset length. Transition from laminar flow was considered to occur when:

$$\frac{Re_\theta}{M_L\left(\frac{Re_L}{X}\right)^{0.2}} = 10$$

where

Re_θ = local Reynolds number based on momentum thickness

M_L = local Mach number

Re_L/X = local unit Reynolds number

Trajectory Shaping

Design entry trajectories were established for the ablator TPS and the RSI TPS. The ablator trajectory was shaped to take advantage of ablator's high heating rate capability, as indicated in figure 1. The maximum heat rate trajectory within the acceleration limit of 2.5 g yields a $\dot{q}$ = 0.802 MW/m^2 (70.7 Btu/ft^2-sec) at the reference point. With a thermal design factor of safety of 1.15, the design heat rate equals 0.938 MW/m^2 (81.3 Btu/ft^2-sec). Since SLA-561* ablator is limited to a maximum heating rate of 0.692 MW/m^2 (60 Btu/ft^2-sec), the entire lower surface of the Orbiter would require the higher densities of ESA-3560HF* and ESA 5500*. Reducing the entry angle and heating rate to 0.300 MW/m^2 (26 Btu/ft^2-sec) at the reference point permits use of low density SLA-561 over 98% of the Orbiter surface area. The $\dot{q}$ = 0.300 MW/m^2 (26 Btu/ft^2-sec) at the reference point was therefore designated as the ablator design trajectory. The RSI trajectory was shaped to take advantage of the ceramic TPS low thermal conductivity for total heat insulation. By reducing entry angle to the skipout limit, heat rate is minimized and total heat maximized resulting in an efficient RSI TPS design. Local static pressure histories were prepared for the Orbiter for ascent and entry. The latter history is presented in figure 2. These pressures are used for the ablator subpanel design.

Heating Rate Distributions

Heating rate distributions normalized to the reference point a at the lower fuselage centerline 15.24 m (50 ft) aft of the fuselage nose were prepared for the ablator and RSI trajectories (figs. 3 and 4, respectively).

*Elastomeric ablators developed by the Martin Marietta Corporation.

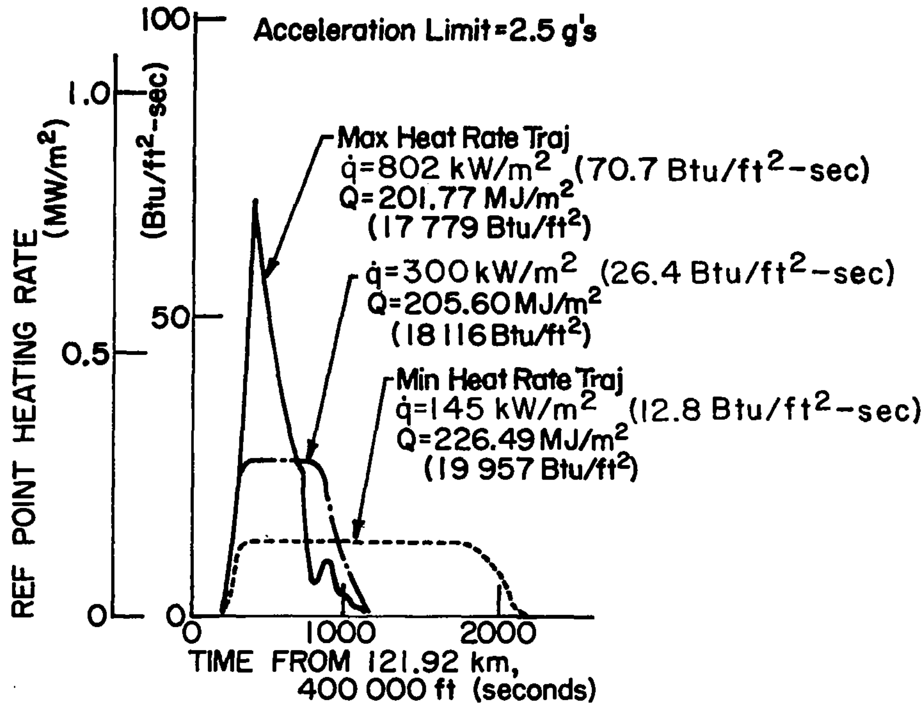

Figure 1.- Trajectory Shaping

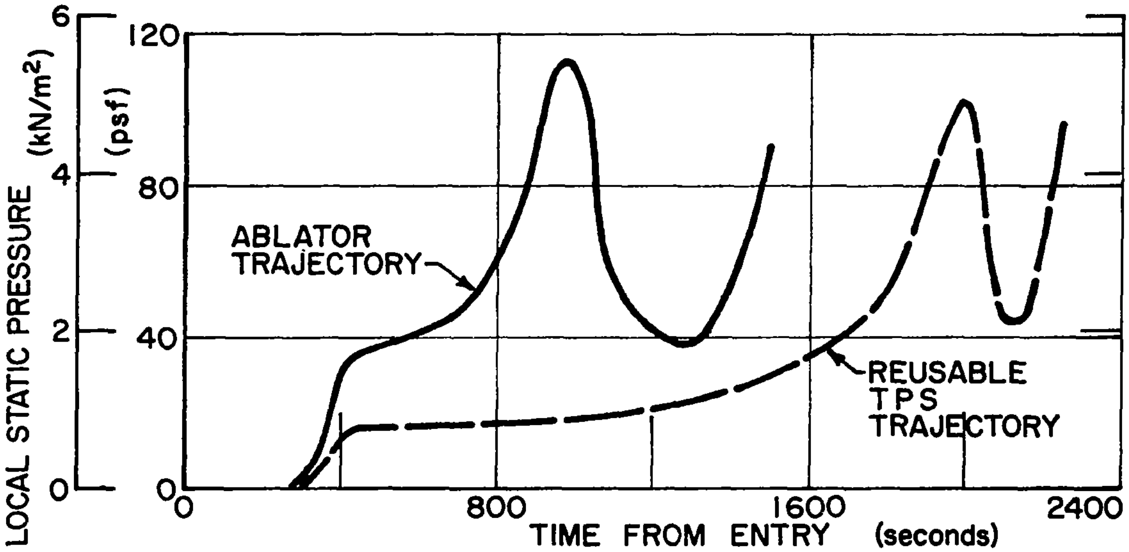

Figure 2.- Entry Local Static Pressure History at Body Lower Centerline Reference Location

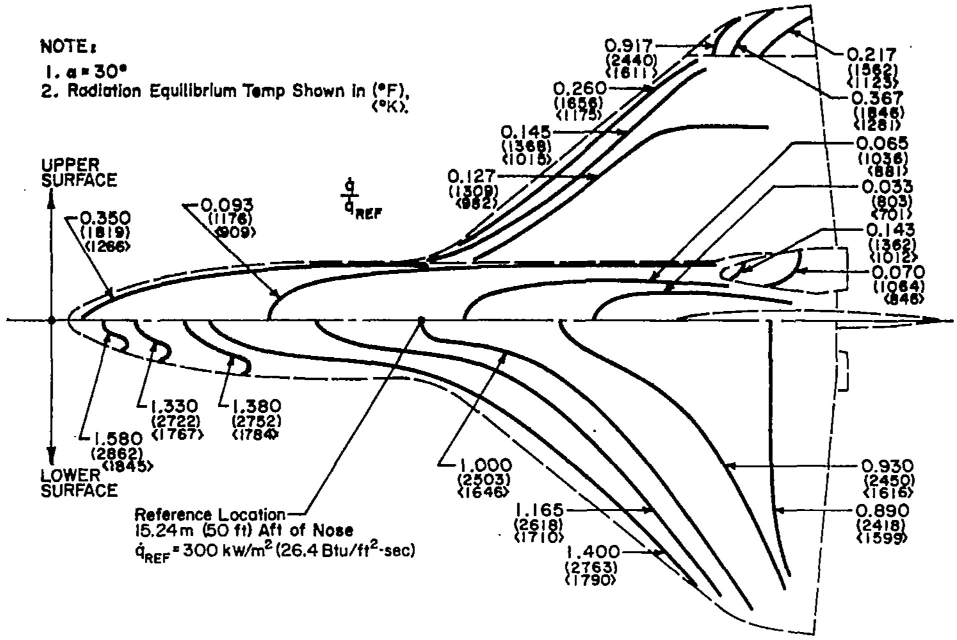

Figure 3.- Entry Heating Rate - Ablator Trajectory
(Without Factors of Safety)

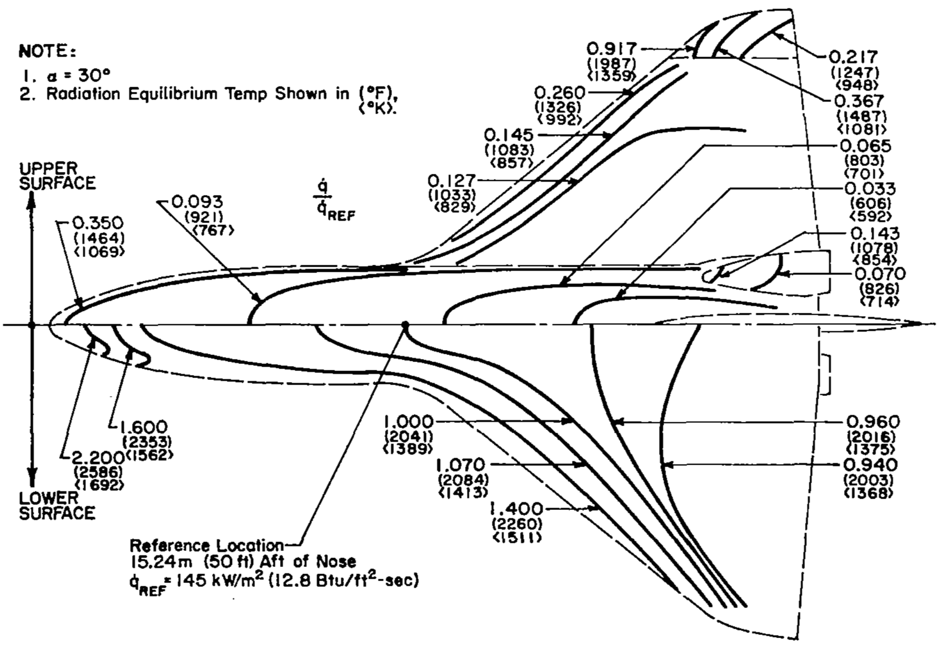

Figure 4.- Entry Heating Rate - RSI Trajectory
(Without Factors of Safety)

IV. HEAT SHIELD DESIGNS (TASK 3)

The contractual program investigated numerous locations on the Orbiter to establish heat shield configurations, weights, and costs. The baseline method of ablator attachment was direct bond. Alternative configurations investigated were:

1) Ablator bonded to a subpanel plate and mechanically attached directly to the Orbiter structure;
2) Ablator bonded to a subpanel honeycomb panel and mechanically attached directly to the Orbiter structure;
3) Ablator bonded to a subpanel honeycomb panel and mechanically attached through standoff fittings to the Orbiter structure.

The various ablator attachment configurations are shown in figure 5.

Each configuration was examined for 12.7-cm (5 in.), 25.4-cm (10 in.), 38.9-cm (15 in.), and 50.8-cm (20 in.) attachment spacing for weight and cost optimization. In addition, four materials of construction for the subpanel were studied: aluminum 2024-T81, Lockalloy, magnesium HM-21A, and graphite polyimide.

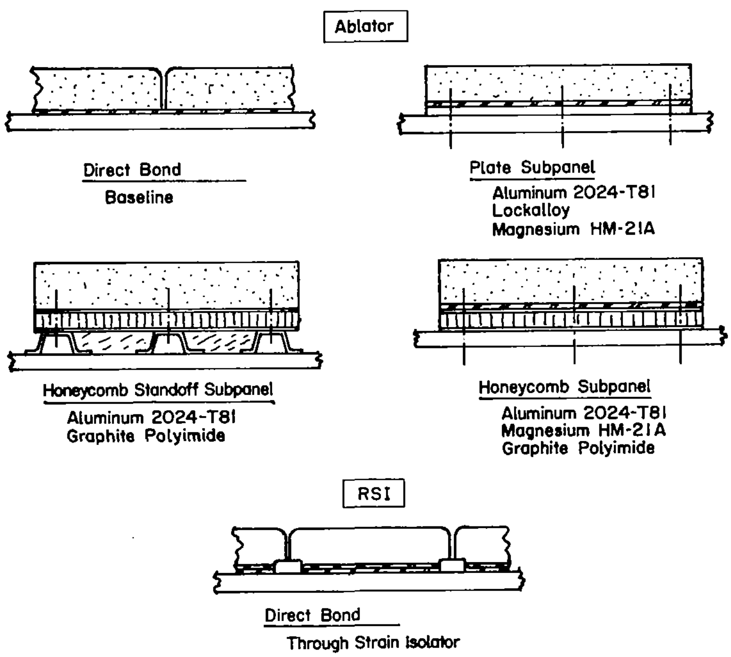

Figure 5.- Attachment Configurations Investigated

Thermal Analysis

Thermal analysis for ablator sizing was carried out with the Martin Marietta Thermochemical Ablation Program (TCAP III). Data input to this program include trajectory data, i.e., velocity, altitude, heating rate, and recovery enthalpy; thermophysical properties for the ablator material and backup structure materials; ablation kinetics; and geometry of the model being analyzed. Analysis results include time-temperature distributions throughout the model and a time-density profile through the ablative material.

Analysis for sizing the RSI material and for developing backup structure modeling techniques was carried out with the Martin Marietta Three Dimensional Heat Transfer program. Data input and analysis results are similar to TCAP III except that no considerations are made for an ablation process. Both programs allow for variations of conductivity with pressure as well as temperature.

Ablator thickness design charts resulting from this effort are typified in figure 6. The required ablator thickness in this figure is based on an entry start temperature of 311°K (100°F) and a limit on maximum structural temperature after entry of 450°K (350°F). An additional ground rule assumes that the heating factor, $F_{\dot{q}}$, is the ratio of maximum local heating rate-to-reference heating rate at every point on the vehicle and is equal to a similar ratio for total heats, i.e., $\left(\dot{q}_{loc}/\dot{q}_{ref}\right)_{max} = Q_{loc}/Q_{ref}$. The figure is useful in that, knowing the total heat capacity of the local structure, subpanel, bondlines, etc., to be protected, $\left(\sum_i^n \tau\rho C_p\right)$, the ablator thickness can be determined for any point on the Orbiter surface.

An example is given, based on a point on the vehicle bottom centerline that is 15.24 m (50 ft) aft of the nose. Here the heating ratio (fig. 3) is 1.00, amplified by the appropriate factor from from table 1 to yield a factor of 1.15. The nonablator components at this location and their physical and thermal descriptions are presented in table 2. Correspondingly $\sum_i^n \tau\rho C_p = 10.3\ \mathrm{kJ/m^2\ °K}$ (0.00351 Btu/in.^{2} °F). Entering figure 6 with this argument and $F_{\dot{q}} = 1.15$ yields a required thickness of SLA-561 of 4.32 cm (1.70 in.).

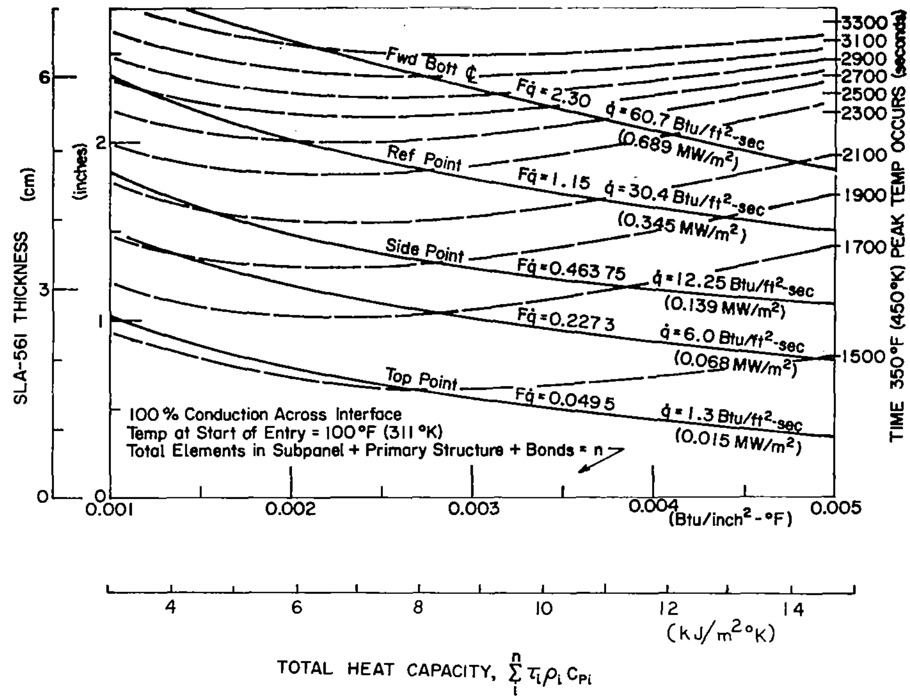

Figure 6.- SLA-561 Ablator Design Chart

TABLE 2.- THERMAL CHARACTERISTICS OF NONABLATOR COMPONENTS

n	Item	Material	τ Thickness		ρ Density		C_p Heat Capacity	
			cm	in.	kg/m^3	lb/in.^{3}	kJ/kg °K	Btu/lb °F
1	Orbiter Structure	Aluminum	0.262	0.103	2768	0.100	0.941	0.225
2	Ablator Bond	RTV Adhesive	0.076	0.030	1495	0.054	1.255	0.300
3	Subpanel	Aluminum	0.079	0.031	2768	0.100	0.941	0.225

Stress Analysis

The plate and honeycomb subpanels are designed by local aerodynamic airloads (see table 3). Orbiter structural strains are isolated from the subpanels by using a mechanical fastener in an oversized hole. The strains and deflections of the subpanels must be limited to prevent induced strains in the ablator from exceeding 1% and to limit surface waviness to less than 0.0125 times the attachment spacing. Subpanel overall dimensions are limited by handling capabilities. A 107-cm (42 in.) by 107-cm (42 in.) panel size was selected for this program. Since total heat shield weight and cost optimizes with small attachment spacing [less than 25.4 cm (10 in.)], the analytical methods for internal loads predictions were selected from references 2 and 3.

TABLE 3.- DESIGN LOADING CONDITIONS

Configuration	Design condition	Design load	Design criteria reference
Plate subpanel mechanically attached directly to the Orbiter structure	Plate bending stiffness Ablator strain limit of 1%	$\delta = 0.0125\ \ell$ Airload (limit) = 3.45 kN/m^2 (0.5 psi) Airload (ult) = 4.85 kN/m^2 (0.7 psi)	Service life strength analysis Service life ablator strain Environments pressures
Honeycomb subpanel mechanically attached directly to the Orbiter structure	Intracell buckling of face sheet		Environments pressures
Honeycomb subpanel mechanically attached through standoff fittings to the Orbiter structure	Intracell buckling of face sheet Ablator strain limit of 1% Panel flutter	Airload (limit) = 20.7 to 27.6 kN/m^2 (3 to 4 psi) Airload (ult) = 29.0 to 38.6 kN/m^2 (4.2 to 5.6 psi) $I = C\ell^3/E$	Environments pressures Service life ablator strain Environments acoustics

The effects of the studies are shown in figure 7; table 4 summarizes the optimum fastener spacing and configurations for a large part of the Orbiter.

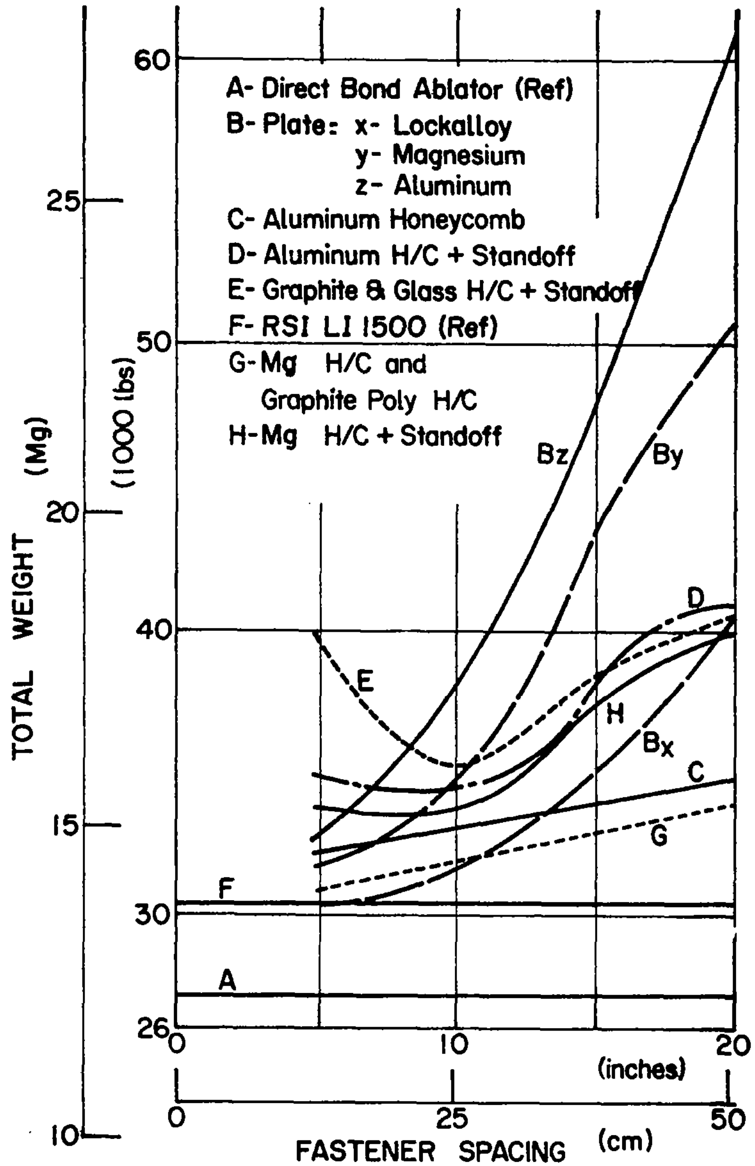

Figure 7.- TPS Attachment Configuration Weight vs Fastener Spacing

TABLE 4.- SUBPANEL GAGES REQUIRED FOR DIRECT ATTACHMENT APPLICATION

Subpanel & thickness identification	Attachment fastener spacing, cm (in.)	Material	Vehicle station: Nose cap (a), cm (in.)	Vehicle station: 584 to 2032 (230 to 800), cm (in.)
Plate thickness	12.7 (5.0)	Aluminum	0.965 (0.380)	0.079 (0.031)
		Magnesium	---- (----)	0.091 (0.036)
		Lockalloy	---- (----)	0.058 (0.023)
Honeycomb, each face thickness	25.4 (10.0)	Aluminum	0.025 (0.010)	0.013 (0.005)
		Magnesium	---- (----)	0.020 (0.008)
		Graphite	0.041 (0.016)	0.020 (0.008)

[a]Attachment fastener spacing does not apply for nose cap

Total Heat Shield Configuration Weights

The baseline TPS weights were calculated in detail. A ratio process was used for the major components of the alternate designs. These yielded the total TPS weights shown in figure 7.

V. DESIGN VERIFICATION (TASK 4)

A test program was conducted to evaluate the mechanically attached heat shield concept. The aluminum subpanel system at 25.4-cm (10-in.) fastener spacing was selected as being representative of most of the direct mechanically attached configurations. Testing was performed in three parts as indicated in table 5.

TABLE 5.- VERIFICATION TEST PROGRAM

Heat shield component	Test objective	Test environment/test facility	Test specimens	Results
Open gap/ attachments (panel test)	Primary: Feasibility of self-closing gap concept Secondary: a. Motion of subpanel plate under stud bolt b. Temperature distribution around stud bolt counterbore	Reentry Heating/ Structures Laboratory, Radiant Heat Facility	1 specimen, approximate size 1.12 x 0.56 m (44 x 22 in.)	• Full heating profile not achieved • Indications of incomplete gap sealing • Indications of inadequate subpanel motion • No abnormally high heating in counterbore
Gap sealer (component test)	Resiliency after high temperature deformations	Dead load compression and uniform heating/ Advanced Structures and Materials Ceramics Laboratory	17 specimens, approximate size 5.08 x 15.24 x 0.254 to 1.29 cm (2 x 6 x 0.1 to 0.5 in.)	• Blanket material showed adequate resiliency • Rope material nonresilient
Sealed gap/ attachments (panel test)	Primary: Feasibility of sealed-gap concept Secondary: a. Motion of subpanel plate under stud bolt b. Temperature distribution around stud bolt counterbore	Ascent and Reentry Heating/Structures Laboratory, Radiant Heat Facility	1 specimen, approximate size 1.12 x 0.56 m (44 x 22 in.)	• Ascent heating - no apparent effect on panel • Descent heating - some backface temperature > 450°K (350°F) - high temperature in gap - indications of inadequate subpanel motion - high temperature in counterbore

Open Gap Test

Ablator panels can be installed with a limited width gap at the joint between adjacent panels 0.318 cm for 107x107 cm panels (1/8 in. for 42x42 in.). Due to the thermal coefficient of expansion of the ablator, the gap closes with increasing temperature, limiting aerodynamic heating at the bottom of the gap. The entry heating pulse was simulated using radiant heat lamps. Although heating was not uniform, some gap closure with acceptable temperature limits at bottom of gap was noted. The attaching fasteners tended to bind, inhibiting free thermal motion of the subpanel. Temperature in the fastener cavity in the ablator was not excessive.

Gap Sealer Tests

Candidate materials were tested in ovens under deformation at elevated temperatures. Springback or recovery of deformation following temperature reduction was measured. Five to ten percent springback was desired following heating. Only the Fiberfrax* blanket compressed normal to the fiber direction was satisfactory.

Sealed Gap Test

Ablator panels can be installed with a sealer in the joint between adjacent panels. During ascent heating the gap closes partially, then opens while in orbit, closing again at entry. The selected sealer was Fiberfrax blanket, which is resilient enough to take these thermal cycles. The test panel was exposed to ascent and entry heating using radiant heat lamps. The sealer did not perform satisfactorily since temperatures were higher in the bottom of the gap than under the ablator. Fiberfrax blanket is stratified and the thermal conductivity is therefore transversely isotropic with the low values across the gap and the high values parallel to the gap.

*Product of the Carborundum Company, Niagara Falls, New York.

VI. WEIGHT AND COST ANALYSIS (TASK 5)

This program has developed design concepts and weight data for several competing methods for ablator attachment to the Orbiter structure. It remains only to select the best attachment system on a weight and cost basis. Competing heat shield weights were converted to costs and added to operational costs to establish the total program costs. The ablator configuration choice on the basis of minimum program costs thus includes the effect of heat shield weight.

Payload Weight Penalties

To aid in the optimum ablator heat shield selection, an ablator heat shield was assumed for the first five years or 151 flights of Shuttle operations--1979 through 1983. The payloads planned for that period were identified from the MSFC traffic model. Each of the competing heat shield design weights was compared to the payloads of the traffic model and payload weight penalties, if any were determined. To ascribe a dollar value to these weight penalties, a cost per pound to orbit was derived. All program costs were apportioned against all the payload weight. A DDT&E cost of $5150M* was apportioned to 445 flights for a cost per flight of $11.57M or $1747.5M for the first 151 flights. Added to this is an operational cost of $10.5M** per flight or $1585.5M for the first 151 flights. The total cost for the first five years of 151 flights is $3333M. The first 151 flights carry a total of 2.07 Gg (4.56M lb) of payload. Dividing $3333M by 2.07 Gg (4.56M lb) yields a unit cost to orbit of $1612/kg ($731/lb). This does not include the costs of the payloads themselves.

The payload weight penalties for each of the competing ablator systems was multipled by this cost per unit weight to orbit of $1612/kg ($731/lb) and added to the operational costs of each ablator system to establish a total program cost. The RSI heat shield weight was used as a baseline to establish payload weight penalties. In many cases, the full payload weight capability of the Orbiter was not used due to volume constraints. This reduced payload was used to establish payload weight penalties (table 6), for three typical missions.

*Estimated costs used during Shuttle Phase C-D pre-proposal briefings.

**Estimated cost given by Bastian Hello, RI, at AIAA/ASME/SAE 14th Structures, Structural Dynamics, and Materials Conference, Williamsburg, VA, March 20-22, 1973.

TABLE 6.- THREE MISSIONS OF THE 1981 PERIOD

Flight no. (a) (payload no.)	15 (NCN-10)		18 (NE2-44)		20 (NEO-16)	
Payload loading factor	0.31		0.38		0.83	
Mission capability	29 484 kg	65 000 lb	20 412 kg	45 000 lb	20 412 kg	45 000 lb
Payload bay load	9 140 kg	20 150 lb	7 757 kg	17 100 lb	16 942 kg	37 350 lb
Unused capacity	20 344 kg	44 850 lb	12 655 kg	27 900 lb	3 470 kg	7 650 lb
RSI design weight	13 717 kg	30 240 lb	13 717 kg	30 240 lb	13 717 kg	30 240 lb
Mission standard weight	34 061 kg	75 090 lb	26 372 kg	58 140 lb	17 187 kg	37 890 lb
38.1-cm (15 in.) fastener spacing, direct attach aluminum pate TPS system weight	22 128 kg	48 782 lb	22 128 kg	48 782 lb	22 128 kg	48 782 lb
Weight penalty	-0-	-0-	-0-	-0-	4 941 kg	10 892 lb
@ $1612/kg ($731/lb)	-0-	-0-	-0-	-0-	$7 962 052	($7 962 052)

Note: All weight penalties for all flights are added for the first 5 years (151 flights). For the case of 38.1 cm (15 in.) spacing direct attach aluminum plate:

Year	Penalty $K
1979	-0-
1980	-0-
1981	$ 15 924
1982	-0-
1983	348 484
Total	$364 408K

$$\boxed{\text{Average Weight Penalty (This TPS System)}} = \frac{\$364\ 408\text{K}}{151} = \$2413\text{K/Flight}$$

[a]Reference 1 nomenclature.

All the payload penalties for all flights were added, resulting in $364M cost penalties for the aluminum subpanel with 38-cm (15 in.) spacing, the example in table 6.

Operational Costs, TPS

Operational costs are the fabrication, refurbishment, and quality assurance tasks incurred during the 151 flight program. These costs include:

1) Ablator slab raw materials and fabrication including scrappage (this accounts for 3/4 of the operational cost);

2) Subpanel raw materials and assembly costs, including scrappage;

3) Assembly of ablator slab to the subpanel;

4) Installation of ablator panel assembly--tools and labor;

5) Removal of used ablator panels--tools and labor;

6) Repair materials and labor due to damage from handling during packing, shipping, storage and installation;

7) Bond line inspection;

8) Mechanical fastener inspection;

9) Subpanel fabrication inspection;

10) Refurbishment cleanliness inspection;

11) Inspection for damage following ablator installation;

12) Inspection of repaired areas.

The total operations costs, table 7, show the direct bond heat shield and the aluminum and magnesium subpanel mechanically attached heat shields to be very competitive with the lowest cost 25.4-cm (10 in.) fastener spacing subpanels. This is due to the reduction in ablative materials required and lower installation costs due to smaller numbers of fasteners. An illustration of the proportional effects of elements of TPS operational costs is presented in figure 8.

Total Program Costs

The inclusion of the heat shield weight as a payload weight penalty cost has a direct effect on determining the optimum heat shield selection (table 8). The direct bond system has the lowest total program cost, followed closely by the magnesium and aluminum subpanel plate systems at 12.7-cm (5 in.) fastener spacing. The low operational cost magnesium and aluminum subpanels at 25.4-cm (10 in.) spacing are prohibitively heavy as is shown by the $34.9M and $76.9M payload weight penalties. For the direct bond system, the effect of dust and debris and turnaround time was not evaluated.

TABLE 7.- OPERATIONAL COSTS

($M) for five years (151 flights)								
Configuration: Subpanel	Configuration: Fastener spacing	Ablator slab	Sub-panel	Panel assembly	Installation & removal	Repair	Quality assurance	Total
Direct bond	N. A.	125.7	----	----	27.9	4.4	6.8	164.8
Aluminum plate	12.7 cm (5 in.)	120.7	1.2	8.7	21.9	6.6	9.2	168.3
Magnesium plate	12.7 cm (5 in.)	122.1	1.8	8.7	22.4	6.8	10.0	173.8
Lockalloy plate	12.7 cm (5 in.)	120.7	31.1	10.4	25.6	6.8	10.6	205.2
Aluminum honeycomb	12.7 cm (5 in.)	124.9	8.7	8.7	25.7	7.0	12.1	187.1
Magnesium honeycomb	12.7 cm (5 in.)	121.8	12.2	8.7	28.1	6.9	13.0	190.7
Graphite composite honeycomb	12.7 cm (5 in.)	122.3	15.0	8.7	25.7	6.8	12.0	190.5
Aluminum plate	25.4 cm (10 in.)	117.2	1.1	8.5	15.2	6.2	6.5	154.7
Magnesium plate	25.4 cm (10 in.)	116.2	1.9	8.5	16.7	6.2	7.1	156.6
Lockalloy plate	25.4 cm (10 in.)	116.3	49.1	10.2	17.9	6.3	7.7	207.5
Aluminum honeycomb	25.4 cm (10 in.)	124.9	7.8	8.5	18.0	6.7	9.0	174.9
Magnesium honeycomb	25.4 cm (10 in.)	121.8	12.0	8.5	19.4	6.6	9.6	177.9
Graphite composite honeycomb	25.4 cm (10 in.)	122.3	17.2	8.5	18.0	6.6	9.1	181.7

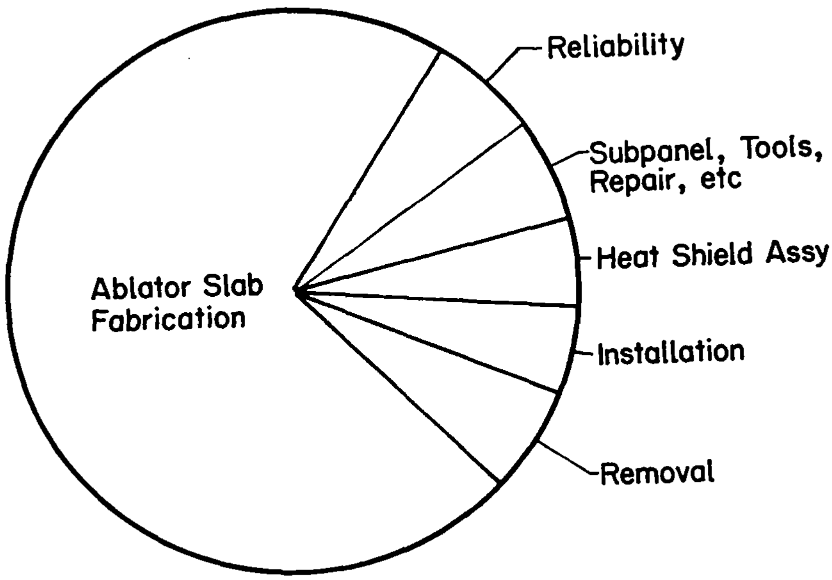

Figure 8.- Relative Operational Costs of a Typical Thermal Protection System [12.7-cm (5 in.) Aluminum Plate Subpanel]

TABLE 8.- TOTAL PROGRAM COSTS

($M) for 5 years (151 flights)					
Configuration		Heat shield weight, kg (lb)	Payload weight penalty	Operational costs	Total program costs
Subpanel	Fastener spacing				
Direct bond	N. A.	12 337 (27 199)	0	164.8	164.8
Aluminum plate	12.7 cm (5 in.)	14 777 (32 577)	10.6	168.3	178.9
Magnesium plate		14 335 (31 602)	3.6	173.8	177.4
Lockalloy plate		13 650 (30 092)	0	205.2	205.2
Aluminum honeycomb		14 587 (32 158)	7.3	187.1	194.4
Magnesium honeycomb		14 008 (30 882)	0.5	190.7	191.2
Graphite composite honeycomb		14 059 (30 994)	0.6	190.5	191.1
Aluminum plate	25.4 cm (10 in.)	17 147 (37 803)	76.9	154.7	231.6
Magnesium plate		15 757 (34 738)	34.9	156.6	191.5
Lockalloy plate		14 311 (31 549)	3.3	207.5	210.8
Aluminum honeycomb		14 855 (32 749)	12.0	174.9	186.9
Magnesium honeycomb		14 429 (31 811)	5.0	177.9	182.9
Graphite composite honeycomb		14 418 (31 772)	4.7	181.7	186.4

Considering these items qualitatively, the slightly higher cost of direct attached magnesium and aluminum plate systems at 12.7-cm (5 in.) fastener spacing are attractive alternatives. Mechanically attached heat shields must be selected prior to Orbiter CDR to permit inclusion of anchor nuts in the basic design. Adding this fastener hardware after the original engineering release becomes increasingly difficult and at the point where Orbiter hardware has been fabricated without heat shield anchor nuts, adding them becomes prohibitive.

VII. DISCUSSION OF PROGRAM RESULTS

Based on heat shield weight alone, the first six lowest weight ablator systems are ranked:

Direct bond	12 337 kg (27 199 lb)
Lockalloy subpanel at 12.7 cm (5 in.)	13 650 kg (30 092 lb)
Magnesium honeycomb at 12.7 cm (5 in.)	14 008 kg (30 882 lb)
Graphite composite honeycomb at 12.7 cm (5 in.)	14 059 kg (30 994 lb)
Lockalloy subpanel at 25.4 cm (10 in.)	14 311 kg (31 549 lb)
Magnesium plate at 12.7 cm (5 in.)	14 335 kg (31 602 lb)

Based on total program costs, the first six ablator systems are ranked:

Direct bond	\$164.8M
Magnesium plate at 12.7 cm (5 in.)	\$177.4M
Aluminum plate at 12.7 cm (5 in.)	\$178.9M
Magnesium honeycomb at 25.4 cm (10 in.)	\$182.9M
Graphite composite honeycomb at 25.4 cm (10 in.)	\$186.4M
Aluminum honeycomb at 25.4 cm (10 in.)	\$186.9M

The only repeaters in both lists are the direct bond and the magnesium plate subpanel at 12.7-cm (5 in.) attachment spacing.

While the direct bond ablator has the lowest heat shield weight and program cost, the concerns with dust control and minimization of turnaround time during the refurbishment period makes the second alternative attractive. Magnesium HM-21A subpanel at 12.7-cm (5 in.) fastener spacing is less than 5% heavier than the baselined RSI and only 8% higher in total program cost than the direct bond heat shield.

VIII. CONCLUSIONS AND RECOMMENDATIONS

Conclusions

- The design criteria for ablative thermal protection systems on a Space Shuttle Orbiter are comprehensive and complete in scope.

- A range of entry trajectories is available that fully uses an ablative TPS--all within 2½ g limitations. At one end of this spectrum is a short time, high peak heating rate entry that would demand considerable usage of dense ablator materials. Extending the time duration of entry reduces the heating conditions to levels which permits lightweight ablators over most of the vehicle.

- Direct bonding of an all-ablator TPS (low density SLA-561) to the Orbiter structure yielded the lowest TPS weight of all the heat shield systems evaluated [weight factor (WF) = TPS(i)/TPS(RSI) = 0.90]. An RSI TPS was next lowest (WF = 1.00), followed by a series of designs involving mechanically attached subpanels supporting SLA-561 (WF = 1.00 to 2.00). Fastener spacing was influential in the total weight of the latter designs.

- A feasible cost model, involving a weight penalty of $1610/kg ($731/lb), was derived based on an apportionment of program costs to the first 151 flights (assumed duration of utilization of all-ablator TPS) and the total payload weight carried in these flights. This penalty was employed in every instance where the total heat shield weight exceeded a given parameter.

- The direct bond ablator system had the lowest program cost of all the ablator configurations examined (the RSI system was not costed). No weight penalty (dollars) was required for this system.

- The next best cost ablative system, magnesium HM-21A subplates, directly attached, would incur $12 million more than the direct bonded arrangement. This was closely followed by the similar system using 2024-T81 aluminum ($14 million more).

- In the three candidate ablator designs highlighted above, approximately 3/4 of the TPS operational cost involves the fabrication of the ablator slab. The other quarter encompasses assembly, installation, removal, tooling, repair, and inspection.

A typical TPS operational cost is approximately 10% of the total program's estimated operational cost.

- The use of a nonablator, insulative material in the gaps between panels tended to make the structure along these lines hotter than the remainder, as demonstrated in a large scale test.

- A test to investigate the feasibility of experiencing gap closure before high heating was encountered was inconclusive because of poor heat distribution in the test assembly.

- A concept of a fastener design that would provide some degree of movement between ablator subpanels and the structure was established.

- An early decision in the design of an ablative TPS must be made concerning the incorporation of anchor nuts in the structure of an Orbiter to accommodate fasteners.

Recommendations

- Cost reductions with respect to ablative systems should concentrate on the basic slab fabrication--materials, processes, inspection, etc.

- Additional effort should be expended to find an acceptable gap sealer; i.e., caulking, etc.

- Additional investigations should be made on the concept of self-sealing of gaps before the high heat time period.

- The fastener presented should be reevaluated for greater tolerances and, possibly, Teflon coating.

- The feasibility of reuse of silicone ablators installed in low heat regions should be further examined.

NASA-Langley, 1975 CR-2579

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