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PERFORMANCE ANALYSIS OF ADVANCED SPACECRAFT TPS

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INTRODUCTION

Spacecraft entering the earth's atmosphere require a very sophisticated thermal protection system. The detail design must be tailored to each specific vehicle based on its planned mission profiles. For the Space Shuttle several types of ceramic heat shield materials were developed[1], both flexible and rigid. These worked very well. Nevertheless, improvements on the materials are continuosly evolving. Material properties that have been studied and improved are high temperature limits, material strength, surface ruggedness, and surface catalycity. This report presents the results of preliminary analyses concerning the thermal protection systems of future spacecraft and their sensitivity to the above material properties.

CATALYTIC SURFACE

Flight tests on the Space Shuttle have dramatically demonstrated the effect of surface catalycity on the local heat transfer rate[2]. In a recent arc-jet test, Wendell Love of the Ames Research Center, measured a significant temperature increase (Fig. 1) when he painted the surface of his low catalycity graphite model with a highly catalytic material. An analysis was undertaken to quantify the magnitude of this catalytic effect.

The analysis was done using a fairly sophisticated two-dimensional numerical model. This model was designed to give the net heat flux to the model surface using the measured temperature history as the boundary condition. The basic element in this analytical model is the heat balance equation shown in Fig. 2 for a typical surface element. The heat flux history was calculated for model runs with and without a catalytic coating on the model, but with the same arc flow conditions. The ratio of the heat flux with the catalytic coating to that without gives a measure of the catalytic effectiveness of the coating.

There were two major deficiencies in knowledge about the models. One was the exact depth of the thermocouple (reportably about .040 inch), and the other was the conductivity of the model material. The material was graphite, but handbooks show a fairly wide range in magnitude of thermal conductivity. All data show the same significant variation of conductivity with temperature. For this analysis it was assumed that because these uncertainties were the same for both models, the ratio of the heat fluxes for the two models would not be significantly affected. Subsequent sensitivity checks verify this. It was found that the use of a constant average thermal conductivity gave the unrealistic result that the arc flow enthalpy increased steadily with time although the arc parameters were all steady. Use of proper temperature dependent conductivities corrected this problem.

The results of the analysis are tabulated in the following table: (Results are averages for the portion of the data between 8 and 30 seconds. Prior to this time the arc starting transients cause problems. After this time the catalytic coating erosion was likely to be significant)

Analytical Model	Ratio of Heat Fluxes	
	Stagnation	Side
Nominal	1.37 +- .04	1.27 +- .04
Nominal but 1/2 Conductivity	1.38 +- .04	1.30 +- .04
Nominal but data corrected for finite depth of thermocouple	1.41 +- .04	1.28 +- .04
1-dimensional approximation (numerical model similar to 2-d)	1.52 +- .10	1.60 +- .08

It is apparent that the catalytic paint increased the heat flux to the stagnation region about 40% and to the side surface about 30%. This trend along the surface is as expected . As assumed, the ratioed results are insensitive to conductivity and to thermocouple depth. The absolute values for heat flux are very sensitive to these parameters, but the magnitudes fall within the expected range for the arc-jet test conditions. It is apparent that a one-dimensional model is not adequate. The ratio values for the 1-D model differ from those using the 2-D model and the expected trend of decreasing catalytic effect along the surface is reversed.

LH2 FUEL TANK INSULATION

Most, if not all future spacecraft will use liquid hydrogen for fuel. These fuel tanks must be protected from aerodynamic heating during the ascent and descent phases of the missions. For all spacecraft, there will be competing options for the thermal protection material used. A preliminary comparison is made here of the performance of two types of insulation for the cryogenic fuel tank for a typical vehicle and mission. One is the flexible ceramic fiber insulation, TABI and the other is a layered super insulation, MLI. TABI is a flexible ceramic blanket that is an evolution of the flexible ceramic blankets that were used sucessfully on the upper surfaces of the Space Shuttle. MLI is a multi layered blanket of thin metal foils separated by an open mesh cloth or by dimples in the foil. Sketches of the models used for the analysis are shown in Fig.3. In lieu of detailed information on the design of the tank walls, the insulation for the analytical model is fastened directly to a simple aluminum wall tank. The cryogenic is treated as a constant temperature heat sink. The heat transferred into the tank is accomodated by cryogenic boiloff. The criterion for comparison of the two insulation systems is the relative weights when the same amount of heat is transferred into the tank under identical heating conditions. Although simplified, these models should provide a reasonable first cut performance comparison between the two systems.

The heating environment imposed on the two systems is based on the the stagnation point heating rates calculated for a likely mission[3] (Fig. 4). For modeling simplicity, the calculated heat flux history is approximated by the dotted line profile. The flux to the tanks will of course be much less than at the stagnation point so calculations were made using fluxes that were 1 and 10 percent of the values in Fig. 4. If the flow remains laminar, the 1% level is most likely, but if the flow becomes turbulent, then the level may approach 10%.

The computer models are one-dimensional. The temperature dependent properties of AFRSI at .01 atm are used for the ceramic insulation, but because the Shuttle type ceramic insulations have similar thermal conductivities, the results will be representative of all of them. Because the mission altitude ranges from ground level to maximum mission altitude, the pressure dependancy of the conductivity should be included. This can be entered in the next step. The MLI is treated as providing an effective emittance to the inner surface of the cover plate and the outer surface of the tank wall. A series of calculations was made of the total heat flux into the tank with varying thickness for the ceramic insulation and with varying effective emissivity for the MLI model. The results of these calculations are shown in Figs. 5(a) and 5(b) for 1% and 10% of the stagnation point heating level. If we assume that the two systems are thermally equivalent if they allow the same total heat load to the tank during the mission, then we can determine the conditions for them to be equivalent from this figure. For example, for the 1% of stagnation heat case, a heat load of 2 kJ/cm is passed by both a 1.1 in. thick AFRSI tile and MLI with 0.055 effective emittance. Referring to the top of the figure, it can be seen that the surface density of the systems is 0.8 and 1.5 lb/ft respectively. Fig. 6 was generated in this manner for a range of acceptable heat loads. It is apparent from these figures that for the models and assumptions of this preliminary analysis, the TABI system is generally the lightest. For low allowed heat loads into the tanks the MLI system can be the lightest. However, it may not be possible to achieve these low heat loads in all cases because of the practical limits of building MLI blankets. Seams, holes and posts generaly limit the effective emissivity to values between .01 and .02. For reference: 1 kJ of heat will boil about 2 grams of LH2 at 1 atm. and the heat capacity of LH2 is about 1 joule/gm/K.

The weight of the TABI system in Figs. 5 and 6 is just that of the insulation plus an RTV bond. About 95% of the weight of the MLI system is that of the outer cover. The weight used is based on data from Ref. 4 for the measured weight of a fabricated cover for an Advanced Carbon-Carbon tile designed for advanced spacecraft. The weight of the cover support posts is included. The surface density is equivalent to a sheet of aluminum 0.11 in. thick. If the cover weight is actually different from this, then the MLI weight curves will be shifted by the weight difference.

Fig. 7 shows the liguid hydrogen boil off as a function of the TABI thickness. The sum of the TABI and boil off weights is a significant parameter in the design of the tank insulation thickness. For this case it has a minimum at about 4.5 inches of TABI. This calculation applies to portions of the tank where LH2 is adjacent to the surface for the full mission. In actuallity much of the tank surface loses this heat sink as the fuel is used. This factor will be considered in future analysis. Other factors to be considered are the benefits of lining the inside of the tank with foam and of developing a ceramic insulation with greatly reduced thermal conductivity.

IMPACT RESISTANCE

It is important that the heat shields for future spacecraft be able to withstand surface impacts. A rigid surface heat shield tile design is proposed by Riccitiello in Ref. 5. The tile consists of a rigid ceramic tile covered by a silicon-carbide cover that slips over the tile and is fastened in place with snaps. The rigid cover adds weight to the tile so a portion of the rigid tile is hollowed out and replaced with a low density felt. To be useful this tile must also be weight competitive with other ceramic tiles. As part of the current program an analysis was made to show that it can be. The procedure and results are presented in Ref. 5, and Fig. 8 is reproduced from that report. This figure was developed from the calculated entry heating environment of the proposed Entry Research Vehicle performing a set of missions with varying total heat load to the surface. The figure shows that there is some weight penalty for the rugged surface, but for high heat load missions the penalty is relatively small and may be acceptable.

COMPOSITE HEATSHIELDS

There can be an advantage to layering heatshield materials rather than using a single material[6,7]. In Ref. 7, it is shown that there is a potential advantage of combining MLI with the ceramic insulation, FRCI. Fig. 9 is taken from that report. It shows by example for a specific Entry Research Vehicle that a significant mass saving can be realized by using MLI in series with a 4 cm thick FRCI tile. This result was based on a reasonable but assumed performance of MLI (its effective emissivity). To establish that this benefit can be achieved an MLI blanket must be designed and tested. A detailed computer model has been constructed for a blanket with variable number of layers, surface emissivities, and leakage conductance. From this computer model, blankets can be designed to match any effective emittance requirement. Results from the application of this model will be presented in the next progress report.

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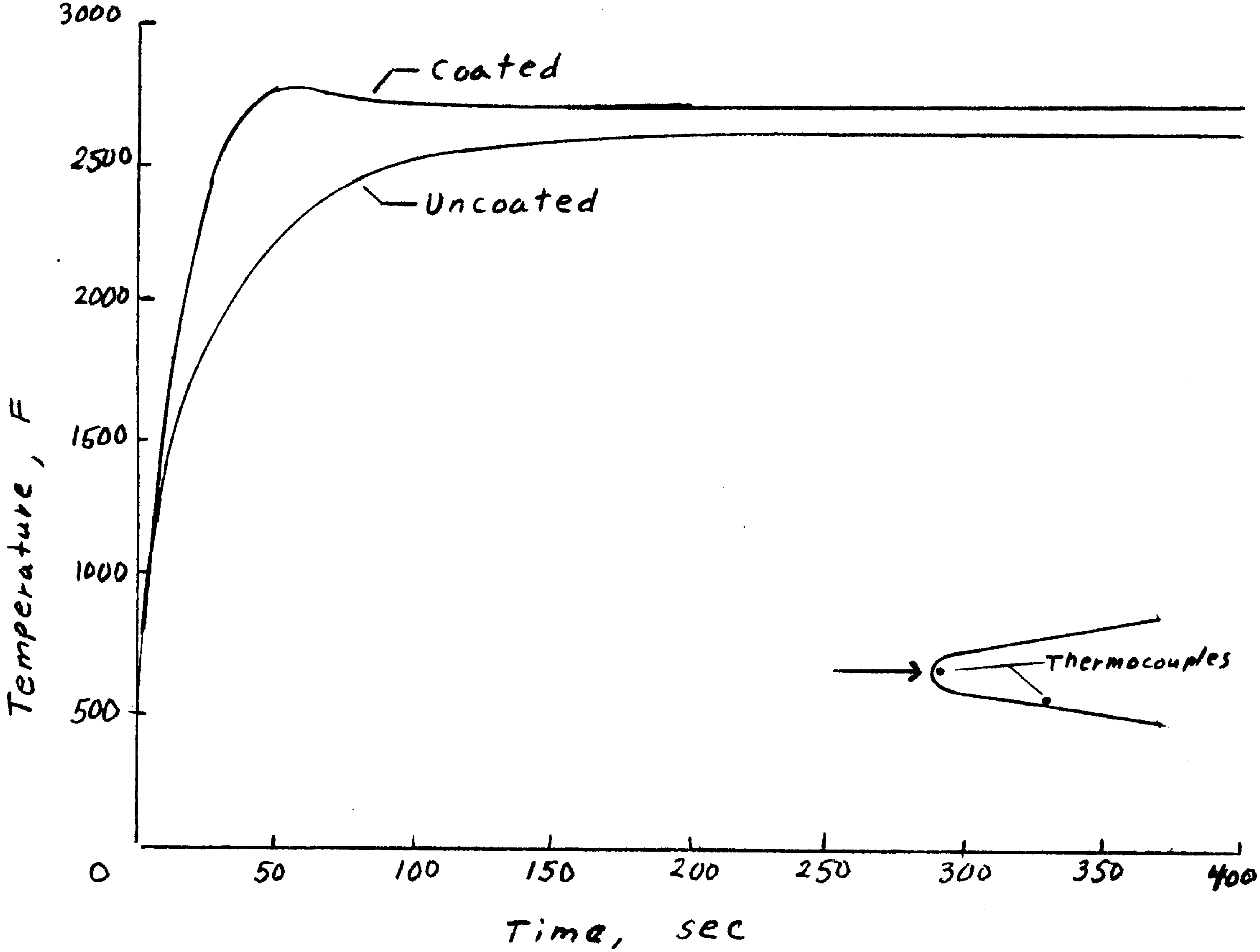
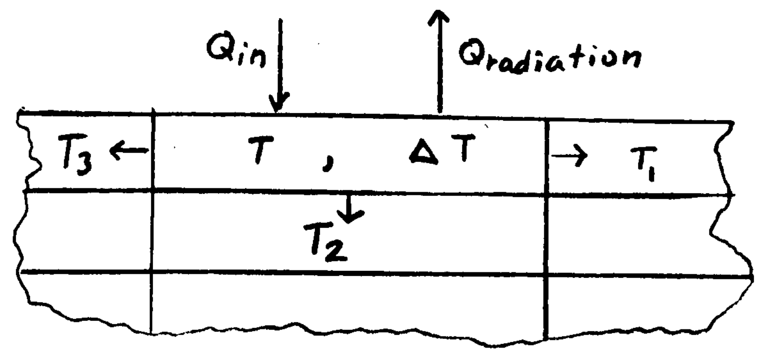


Fig. 1 - Effect of catalytic coating on stagnation point temperatures.



$$Q_{in} = \sigma \epsilon A_s T^4 + \rho V C_p \Delta T + \sum_n K_n (T_n - T)$$

Fig. 2 - Heat balance model for analysis.

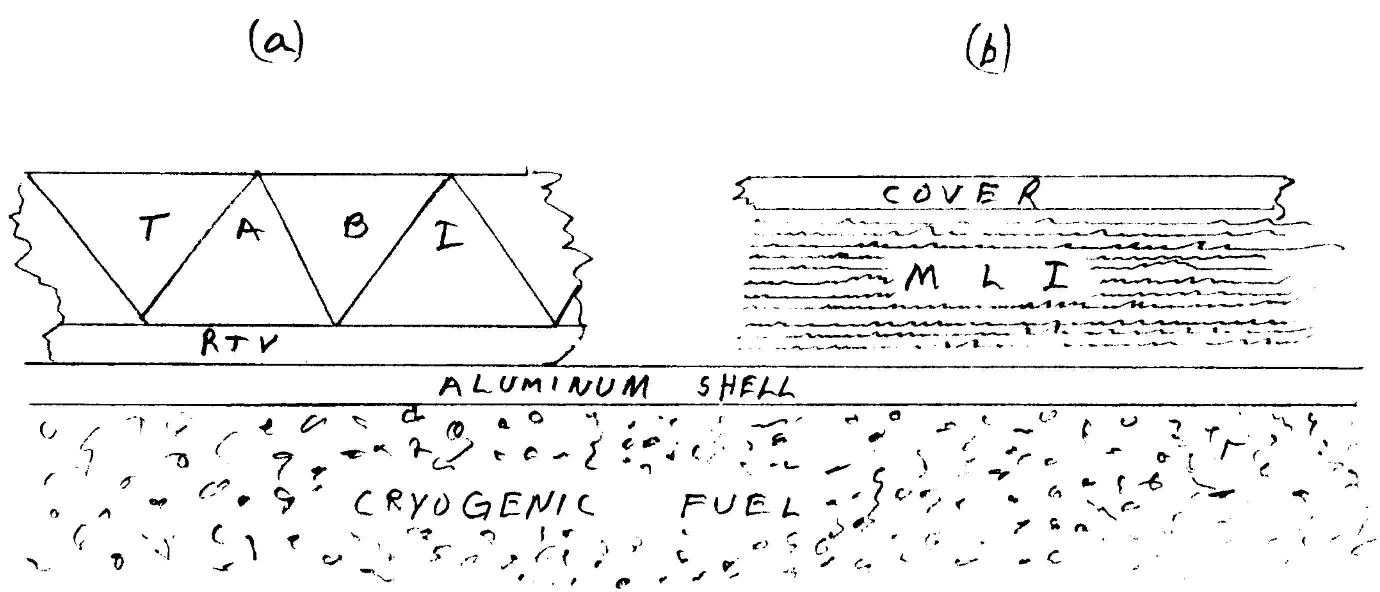


Fig. 3 – Insulation systems used in analysis.

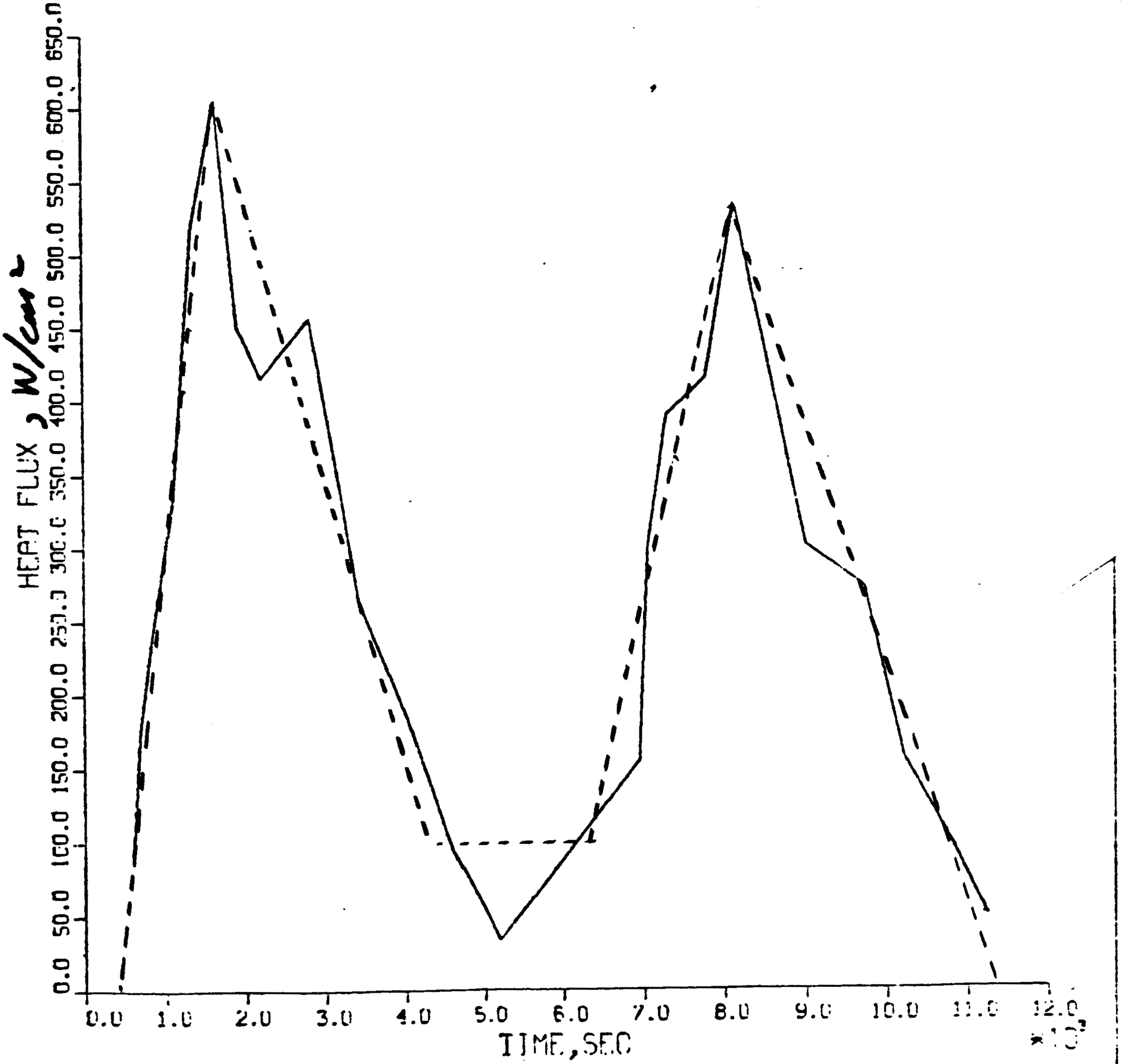


Fig. 4 – Stagnation point heat flux for NASP mission.

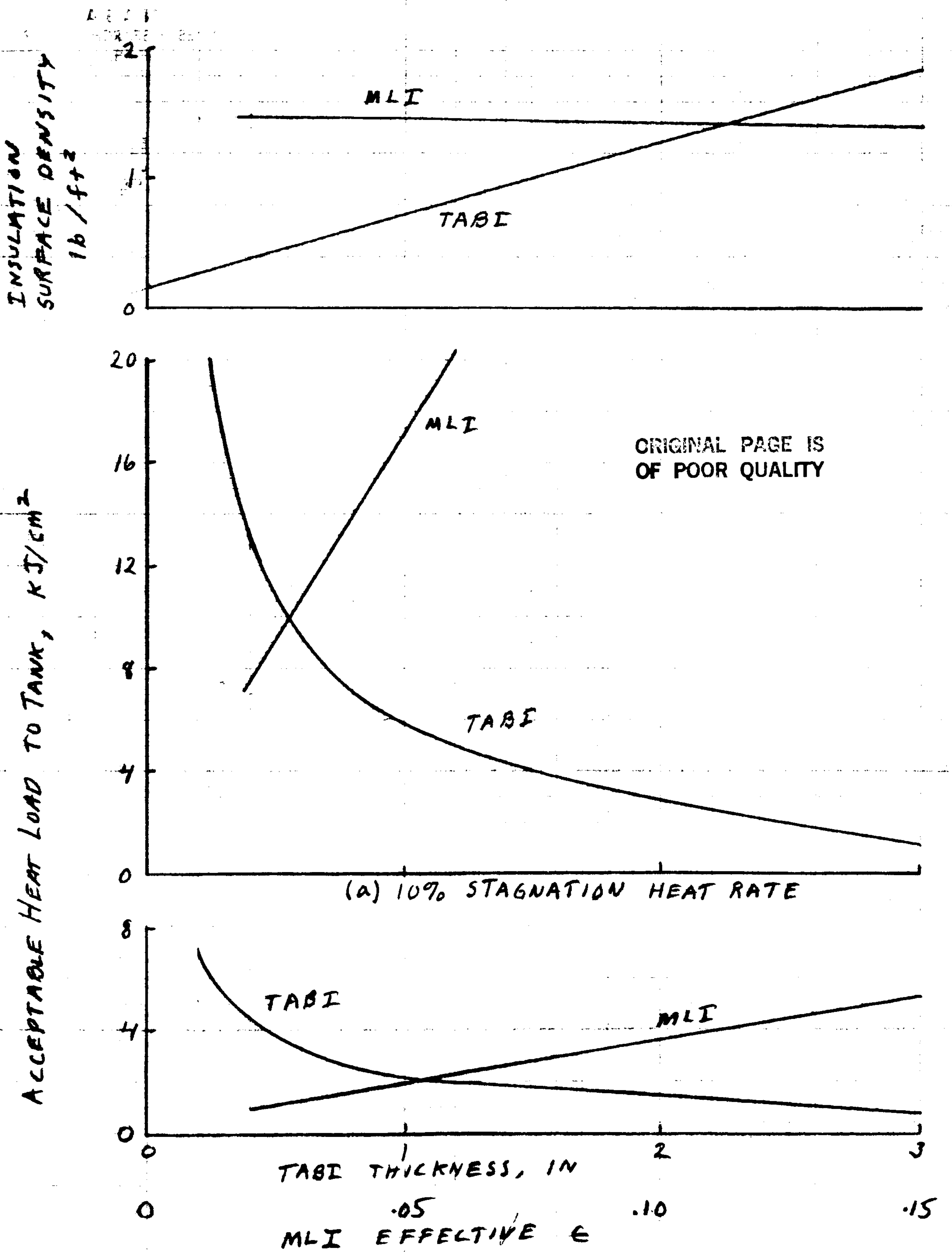


Fig. 5 – Total heat load to tank during mission ascent and descent.

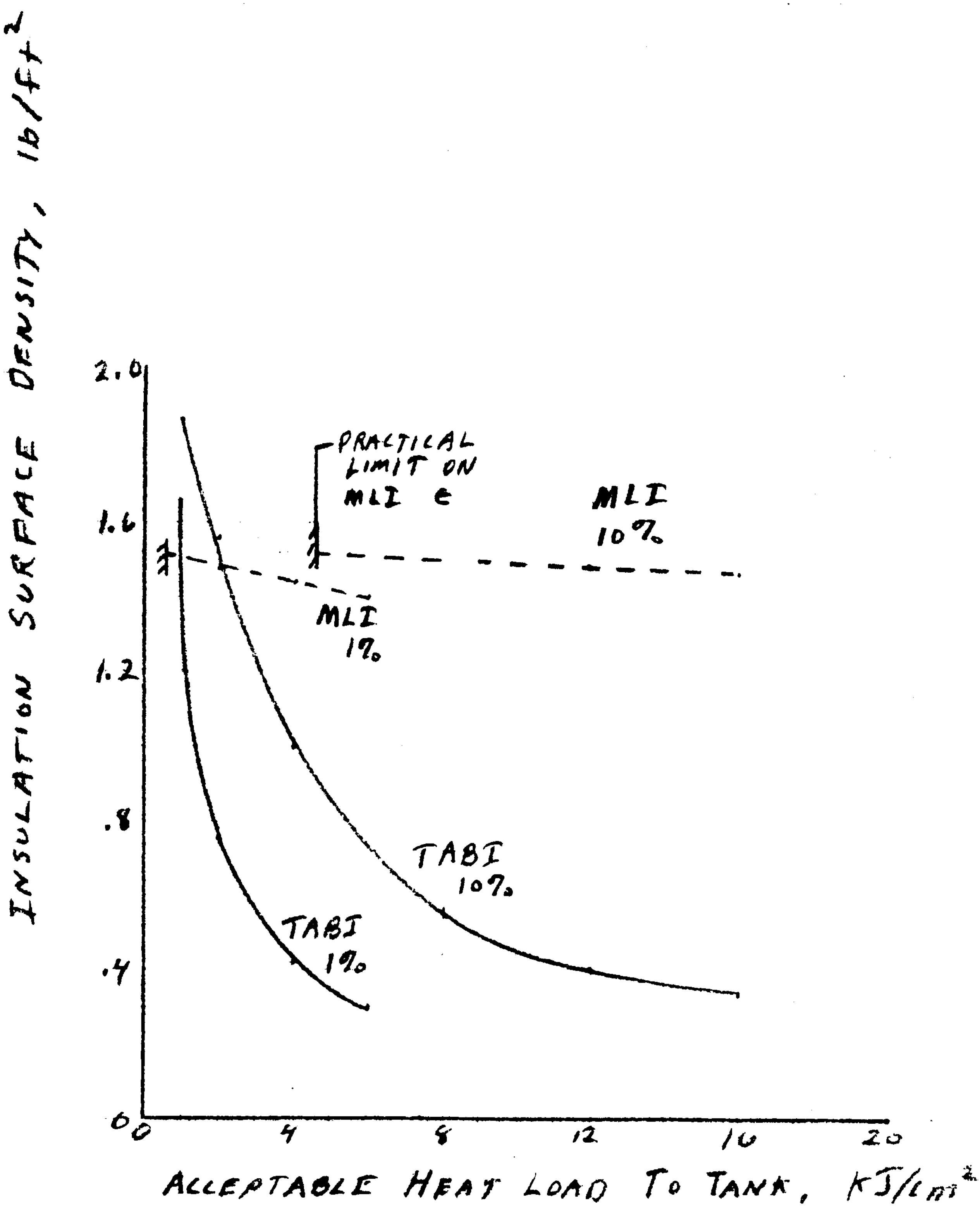


Fig. 6 – WEIGHT COMPARISON FOR SAME HEAT LOAD TO TANK.

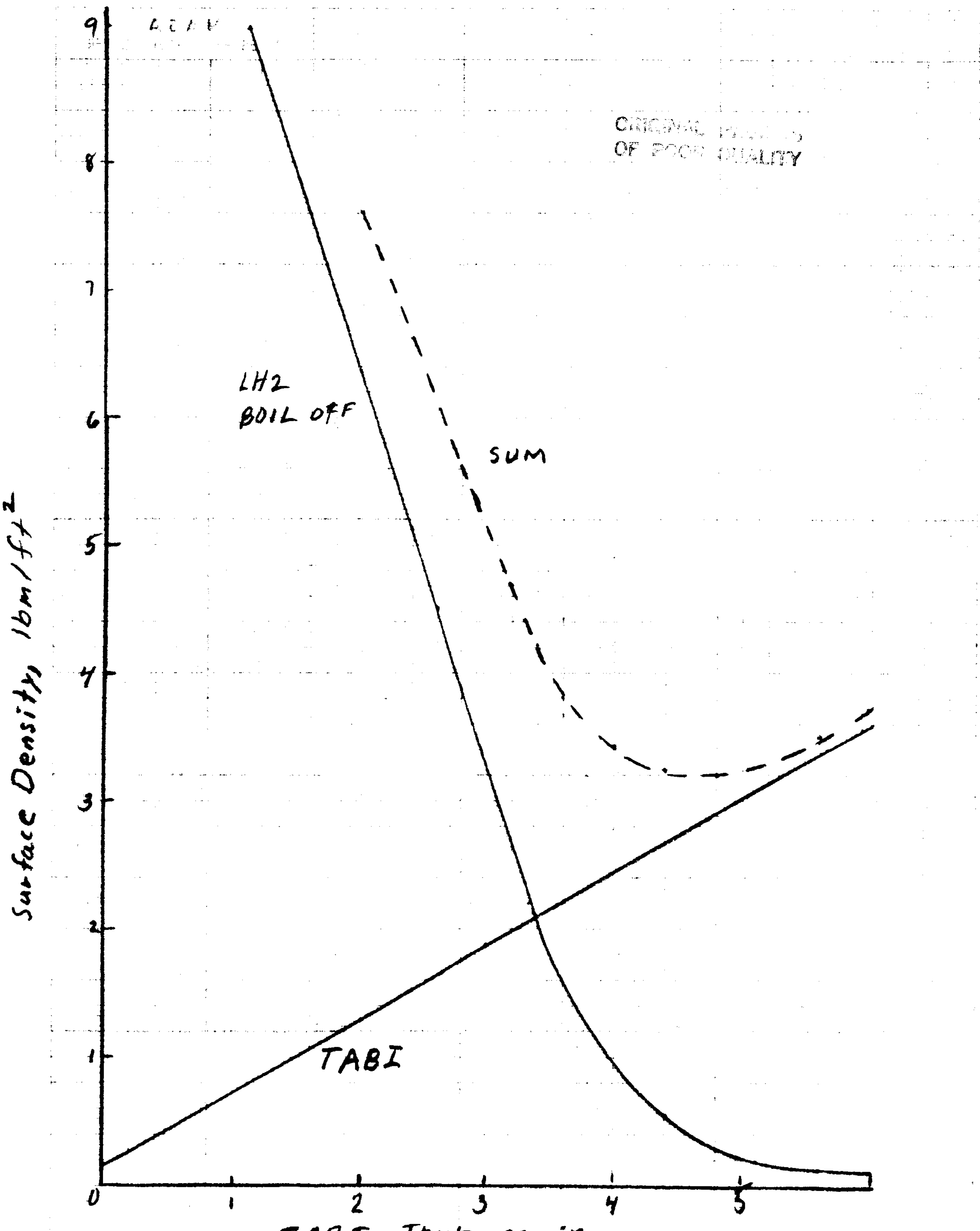


Fig. 7- Liquid hydrogen boil off during mission.

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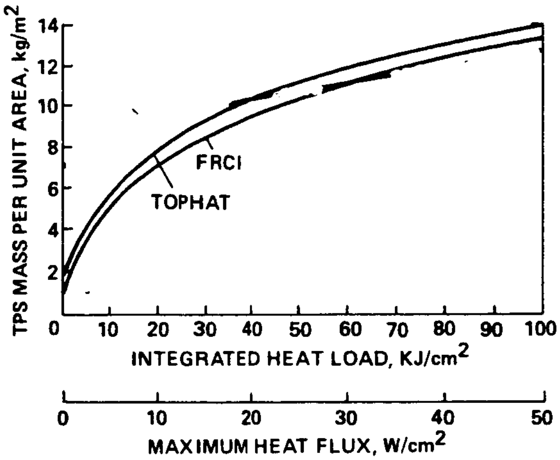

Fig. 8 Relative surface density requirements for three heatshield concepts.

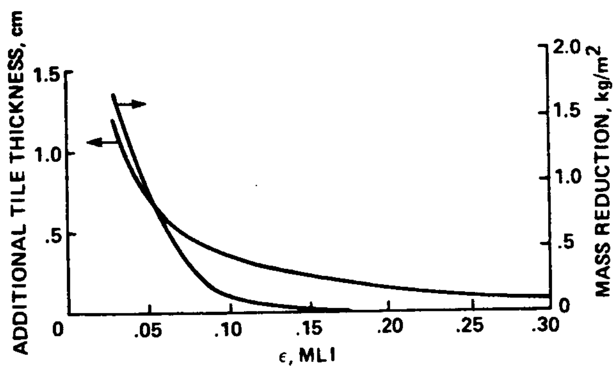

Fig. 9 FRCI-12 tile thickness required to match effectiveness of 4 cm thick tile with MLI and corresponding mass saving.

APPENDIX

Toughened Outer Surface Reuseable Surface Insulation for Advanced Thermal Protection Systems

by

Pitts, W. C. and Riccitiello, S. R.,

(Proposed paper for AIAA Journal of Spacecraft and Rockets)

INTRODUCTION

The Space Shuttle Thermal Protection System, TPS, utilizes a variety of heat shield materials, depending on the temperature of the local surface during entry. A coated, reinforced carbon/carbon composite is used on the nose cap and wing leading edges where the temperatures exceed 2300°F. High Temperature Reusable Surface Insulation[1], HRSI, is used on the lower surfaces, where the temperatures are between 1200°F and 2300°F while a Low Temperature Reusable Surface Insulation, LRSI, Flexible Reusable Surface Insulation, FRSI, and Advanced Flexible Reusable Surface Insulation, AFRSI, are used on the upper surfaces where the temperatures do not exceed 1200°F. All of these materials worked for the Shuttle missions, but there has been a continuing effort to make the overall heat shield system more durable.

The HRSI used on the lower surface of the Shuttle is a class of materials rather than a specific material. They are rigid, fibrous silica tiles, coated with a black (for emissivity control) reaction cured glass called RCG.[2,3] The coating provides the desired optical properties for the TPS, and also aids in the handling of the tiles during installation on the vehicle. Shuttle experience has shown that the RCG is very sensitive to low energy impacts from ice and other debris during launch and landing. The damage ranges from small cracks and chips of the surface to large holes that expose the white vitreous tile material which then melts on exposure to entry heating.

Advanced space transportation systems will use a form of rigid, reusable surface insulation as part of the TPS for high temperature areas other than leading edges. Since the environment will be more severe than that of the Space Shuttle and because minimum maintenance is required, the new insulation system will require a toughened outer surface, relative to the state of the art HRSI, that can withstand the rigors of handling as well as impacts from small tools and ice.

The problem with the Shuttle HRSI is that the glazed coating is an example of a thin brittle plate supported by a low density elastic substrate.[4] To aleviate this problem, a new tile cover called "Top Hat" was designed to replace the brittle plate with a toughened outer, ceramic/ceramic composite, structure. The "Top Hat", is composed of a ceramic, woven fabric which has a silicon-carbide coating deposited by chemical vapor deposition. Major considerations during the development of the "Top Hat" concept were weight and the optical and impact properties. The advanced TPS being described in this paper takes into account these factors and compares the "Top Hat" Advanced Thermal Protection Material System with the State of the Art HRSI material.

MATERIALS

The thermal-structural performance of the "Top Hat" concept will be compared with that of the HRSI used on the Space Shuttle. The basic HRSI materials, composition, and fabrication processes have been described elsewhere[5,6,7] and therefore will not be discussed in this paper. The major difference between the two insulation systems is the outer surface. The

Shuttle HRSI has a RCG glaze or coating with silicon boride dispersed within it as the emmissivity control agent. The coating is spray applied to the fibrous insulation and fired to give the glaze on the surface.

The "Top Hat" is a tile cover made as a separate unit which is fastened to the basic insulation by integral clips (Fig.1) so that the ceramic/ceramic composite top is stress/strain independent of the base component of the tile. This approach eliminates the thermal stress at high temperature that results from the difference in thermal expansion between the tile coating and the FRCI base.

The "Top Hat" is a woven fabric, infiltrated with silicon carbide by chemical vapor deposition. For this development study, all fabrics were made of "3M" AB312 alumina-boria, silica fiber. Other high temperature fibers such as "Nicalon" (TM Nipon Carbon) could be used also. The strength and impact resistance of the cover depends on the weave of the fabric. Three weaves were strength tested: a "baseline" weave by Hexcel, a ballistic weave by Hexcel, and a triaxis weave by Albany International. The "Top Hat" single ply outer cover with the AB312 fiber and "baseline" weave, has an area density of about 0.40 lb/ft^2 ,of which 0.30 lb/ft^2 is the SiC coating/infiltrate. The "Top Hat" maximum use temperature is limited by the maximum temperature capability of the fabric. For AB312 this limit is approximately 2800°F.

The form of the "Top Hat" tile insulation can be tailored to meet specific requirements. The form assembled for testing in the radiant panel and arc-jet facility is shown in Fig. 2. The base tile insulation used was FRCI-12[8] to which the "Top Hat" was held in place by means of the clips.The FRCI-12 (12 lb/ft^3) was selected over the state of the art LI-900 (9 lb/ft^3) insulation because of its superior physical properties. To make the weight of the "Top Hat" comparable to the reference Shuttle HRSI tile a fraction of the FRCI was replaced with a low density (6 lb/ft^3) felt material. The "3M" 440 felt was selected because it has the temperature capability to meet the 2800° F requirement. In this model the FRCI was isolated from the aluminum substrate with a strain isolation pad since this was the configuration used for the HRSI on the Shuttle. This allows a direct comparison of the "Top Hat" system data with available data for the Shuttle HRSI system. However, for flight applications, the strength of the FRCI allows a direct adhesive bonding of the tile to a polyimide graphite structure.

EXPERIMENTS

A series of tests were made in the process of developing the "Top Hat" system. First strength and toughness tests were made, then the thermal performance was evaluated.

Tensile Test

The tensile properties were measured for various silicon carbide infiltrates and fabric weaves that are potential "Top Hat" materials. An Instron Test Machine, model #4202, was used in combination with a custom-built high temperature furnace designed by Smith et al.[9] The tensile specimens of the coated fabrics were tested at room temperature, 1100^{O}F, and 1800^{O}F. The specimens were all single ply strips nine inches long and one inch wide, with a saw cut in the test section, leaving a 0.75 inch test width. A two-part polyester resin was used to harden both ends of the test specimens to prevent slippage or crushing in the test grips during the tensile pull.

Impact Test

A modified Gardner Laboratory impact tester, Fig. 3, was used to compare the impact resistance of the "Top Hat" coated fabric with that of RCG coating on the Shuttle HRSI. The hammer weight of the tester was .507 lb and the impact head diameter was .375 inch. The impact tester base was designed to hold the "Top Hat" material with or without a substrate such as LI900 or FRCI. The "Top Hat" material was clamped over the substrate and subjected to increasing impact levels to determine the threshold of damage.

Thermal Shock Test

Prior to testing a "Top Hat" thermal response model, a simple thermal shock test was performed to ascertain whether or not the coated fabric design would survive extreme thermal gradients. The test involved placing a full-scale "Top Hat" into a preheated furnace at 2000^{O}F then raising the temperature as rapidly as the furnace would allow to 2500^{O}F. The model was held at 2500^{O}F for about one minute, then quickly removed from the furnance and allowed to cool to room temperature. If no cracks or other damage were observed, the part was deemed ready for the radiant panel tests.

Radiant Panel Tests

This test consisted of placing the model in a radiant panel facility that used high density quartz lamps as the heat source. The model surface temperature was raised at a controlled rate to 2500^{O}F. The internal temperature response was measured and a post test examination was made of the model. If the model showed no damage and no anomolies were observed in the temperature data, then the model was readied for tests in the 20 megawatt arc-jet facility.

Arc-Jet Test

This test was conducted in the Ames Research Center semi-elliptic 20 megawatt arc-jet facility. The static pressure and heating rate can be controlled up to maximums of 10 mm Hg and 15 Btu/ft^{2}/sec respectively by varying the arc mass flow and arc current. For the present test, the arc conditions were controlled so that after a transient start, the model surface temperature was maintained to a predetermined, constant value. The test conditions used are shown in Table 1.

For this test, the model from the radiant panel test was placed in a model panel holder as shown in Fig. 4. The "Top Hat" tile was surrounded by state of the art HRSI tiles with one HRSI tile instrumented with surface thermocouples which were used as pilots during the test. The locations of the thermocouples in the "Top Hat" tile and the test control thermocouples in the HRSI tile are shown in Fig. 2.

RESULTS AND DISCUSSION

The tensile test results are shown in Table 2 for a variety of fabric weaves and area densities. Both the triaxis weave and the ballistic weave have superior tensile properties relative to the baseline weave, depending on the fiber direction. These data suggest that the good impact resistance of the baseline material can be improved on.

Since impact tolerance is of prime importance for any future TPS, a series of impact tests was run. Post impact pictures of the specimens are shown in Figs. 5(a) to 5(d). All pictures are to the same scale. Part (a) of this figure shows the high sensitivity of the state-of-the-art RCG coating to a moderate impact of 0.56 joules. None of the "Top Hat" composite materials were damaged by the same impact level. Fig. 5(b) shows this to be true for the "Top Hat" with the baseline weave. When the impact level was doubled the "Top Hat" materials were all damaged, but the ballistic weave material suffered the least damage of the three candidate materials. The comparison between the baseline and the ballistic weaves is shown by the back surface pictures in Figs. 5(c) and 5(d). Note that the ceramic/ceramic composite shows no crack propagation or spallation as the RCG coating did.

The results of the impact test are summerized in Fig. 6. The threshold energy for impact damage is shown as a function of the material surface density. The composite materials are 10 to 20 times more impact resistant than the RCG coating (0.006 joules vs. 0.06 to 0.12 joules) for a surface density of 0.4 lb/ft^2.

The final step in the "Top Hat" development was the thermal response test done in the arc-jet. The center region temperature histories for the arc-jet run no. 3 are shown in Fig. 7 for the Top Hat tile. These temperatures are compared with internal temperatures calculated for the same applied surface temperatures. The system thermal response is as expected for the material conductivities used. The temperatures along a tile tab are shown in Fig. 8. These data show no anomolies. The temperature at the top of the tab are higher than at the center because of edge effects, but for a given depth the temperatures at the tab and at the center are comparable. Post test examination showed no evidence of cracking of the tile tabs as a result of thermal stress between the FRCI and the cover during any of the runs.

The relative thickness of the 440 felt to that of the FRCI was selected for the arc-jet test to make the Top Hat tile weight comparable to that of the control HRSI. This is not necessarily the optimum thickness ratio for thermal performance in flight. For flight vehicle applications, this ratio will be determined from a tradeoff study between structure requirements and weight requirements. Fig. 9 shows the results of a calculation to show the dependence of tile mass on the relative thickness of the insulation material. For these calculations the Top Hat was assumed to be exposed to the heating environment of a proposed experimental AOTV called the ERV[10] (Entry Research Vehicle). For each thickness ratio, a determination was made of the thickness required to limit the back surface temperature to the structural temperature limit of polyimide graphite, 540 ^{o}F. For felt thickness less than 40%, the mass with the felt on top is larger than for the all FRCI tile. The reason for this is that, for the high temperatures near the surface, the effective conductivity of the felt is higher that that of FRCI. Consequently, the total tile thickness must be increased to meet the bottom surface temperature criterion. An alternative design would be to put the felt at the bottom of the tile where the temperatures are lower. The result of interchanging the felt and FRCI is shown by the lower curve of Fig 9.

Fig. 10 shows the result of a heat shield sizing calculation[11] for the lower surface of the ERV using Top Hat and FRCI HRSI. The integrated heat loads are for the range of heat loads over the lower surface of the vehicle.[10] The "Top Hat" design and felt thickness ratio of Fig. 2 were used for this analysis. This "Top Hat" system is less than 10% heavier, but much more durable. Optimization of the felt thickness ratio would reduce this mass difference.

CONCLUSIONS

The major conclusions derived from the "Top Hat" development program are:

1. The ceramic/ceramic "Top Hat" is 10 to 20 times more resistant to impact than the RCG coating for equivalent thicknesses.
2. The "Top Hat" ceramic/ceramics have higher temperature capability than RCG.
3. At equivalent weight the advanced system thermal response is similar to that of the state of the art HRSI material.

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Table 1. Arc-Jet Test Conditions

Run	Surface Temperature	Heat Rate
	(F)	(Btu/ft^2/sec)
1	1850	8
2	1930	9
3	2250	15

Table 2. Tensile test data.

Fabric Weave	Aerial Density (lb/ft^2)	Sample Dimension (in) Width	Thickness	Weave Direction	Tension to Break (lb) Room Temp	600^oF	1000^oF
Baseline	0.40	0.75	0.040	Fill	161	179	163
Baseline	0.40	0.75	0.040	Warp	138	178	144
Ballistic	0.35	0.75	0.045	Fill	241	194	146
Ballistic	0.35	0.75	0.045	Warp	245	259	238
Triaxis	0.32	0.75	0.035	60^o	237	234	165
Triaxis	0.32	0.75	0.035	90^o	180	202	130

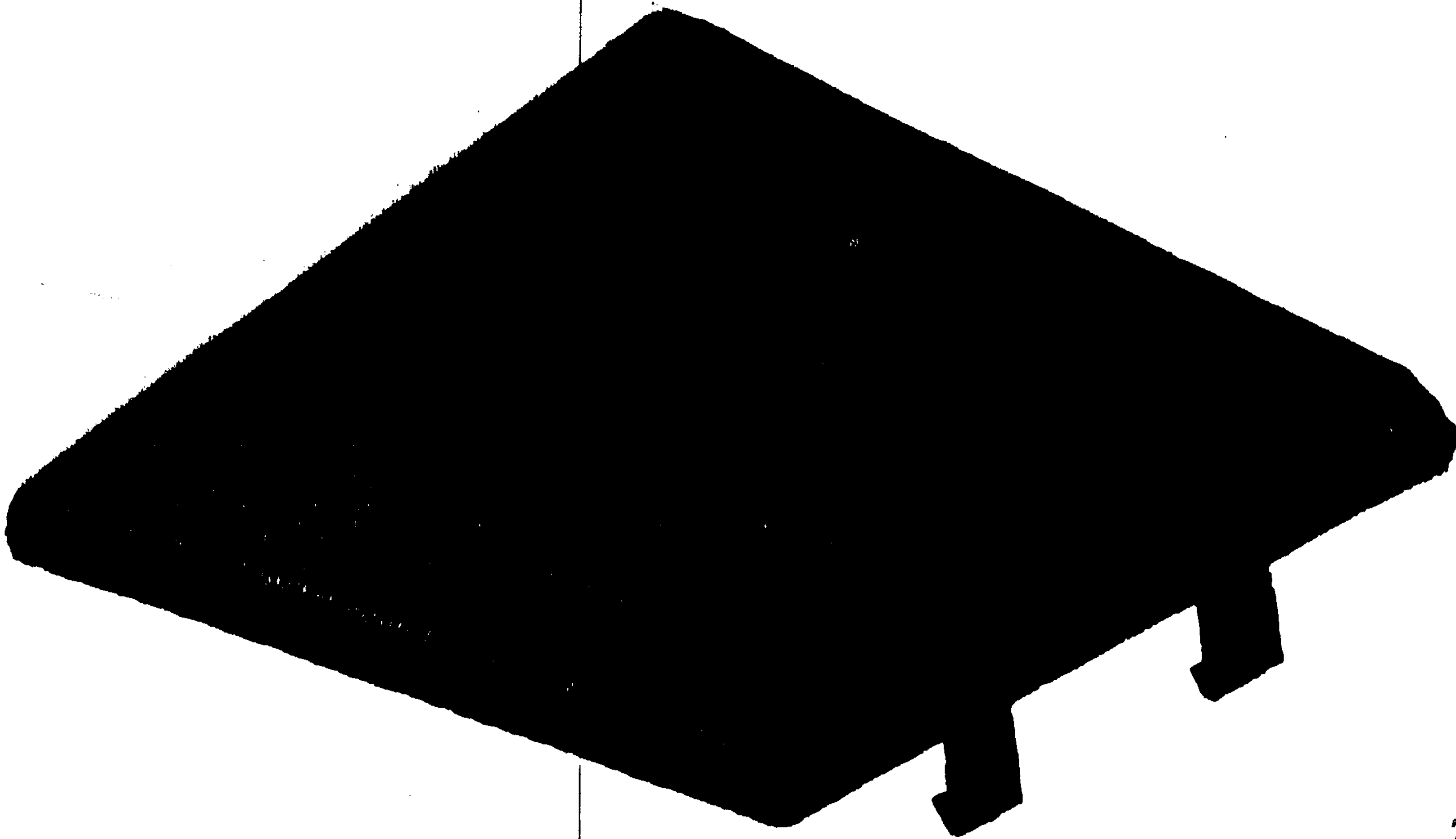


Fig. 1 Inner surface of "Top Hat" ceramic/ceramic tile cover.

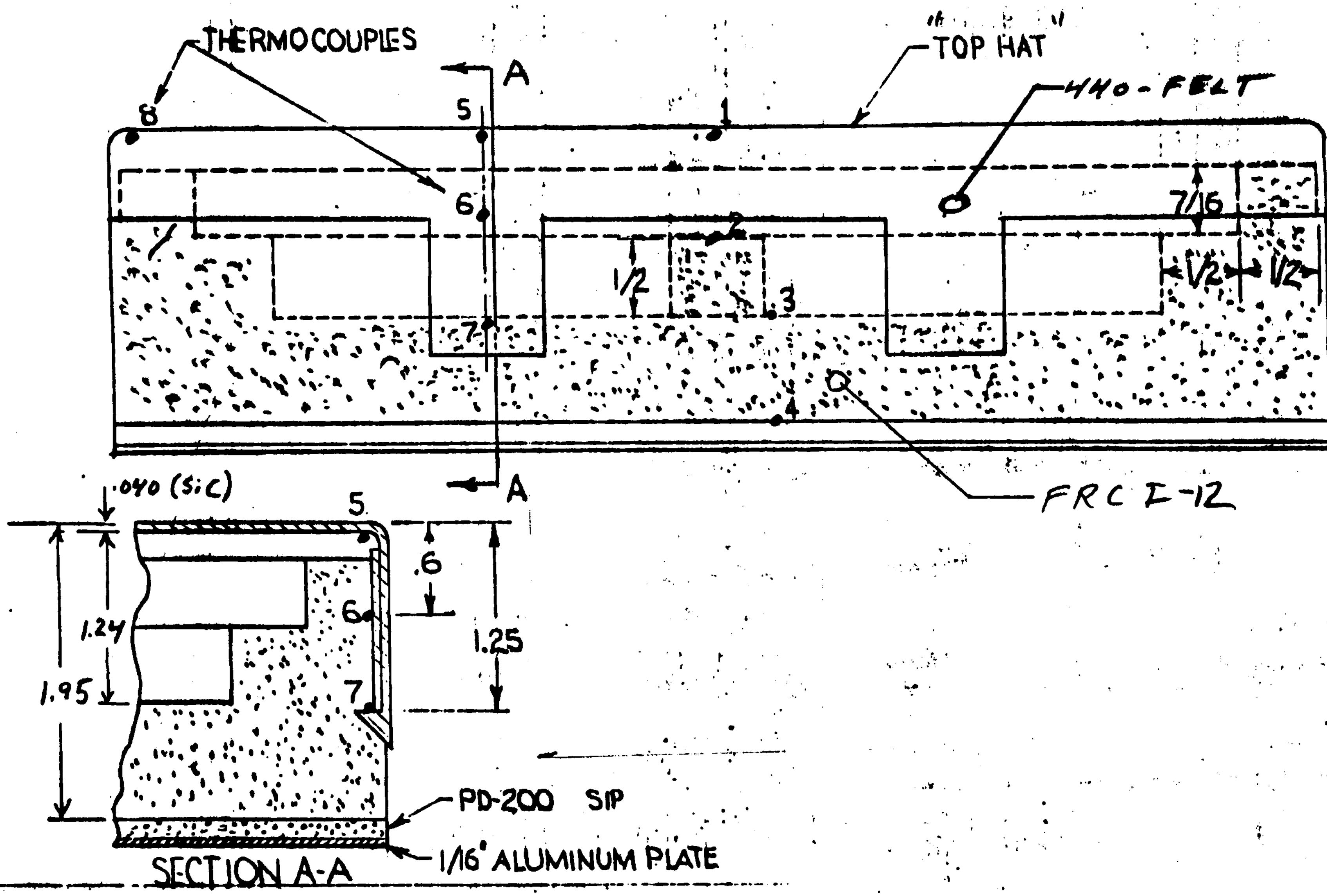


Fig. 2. Thermocouple locations in "Top Hat" model used for arc-jet test.

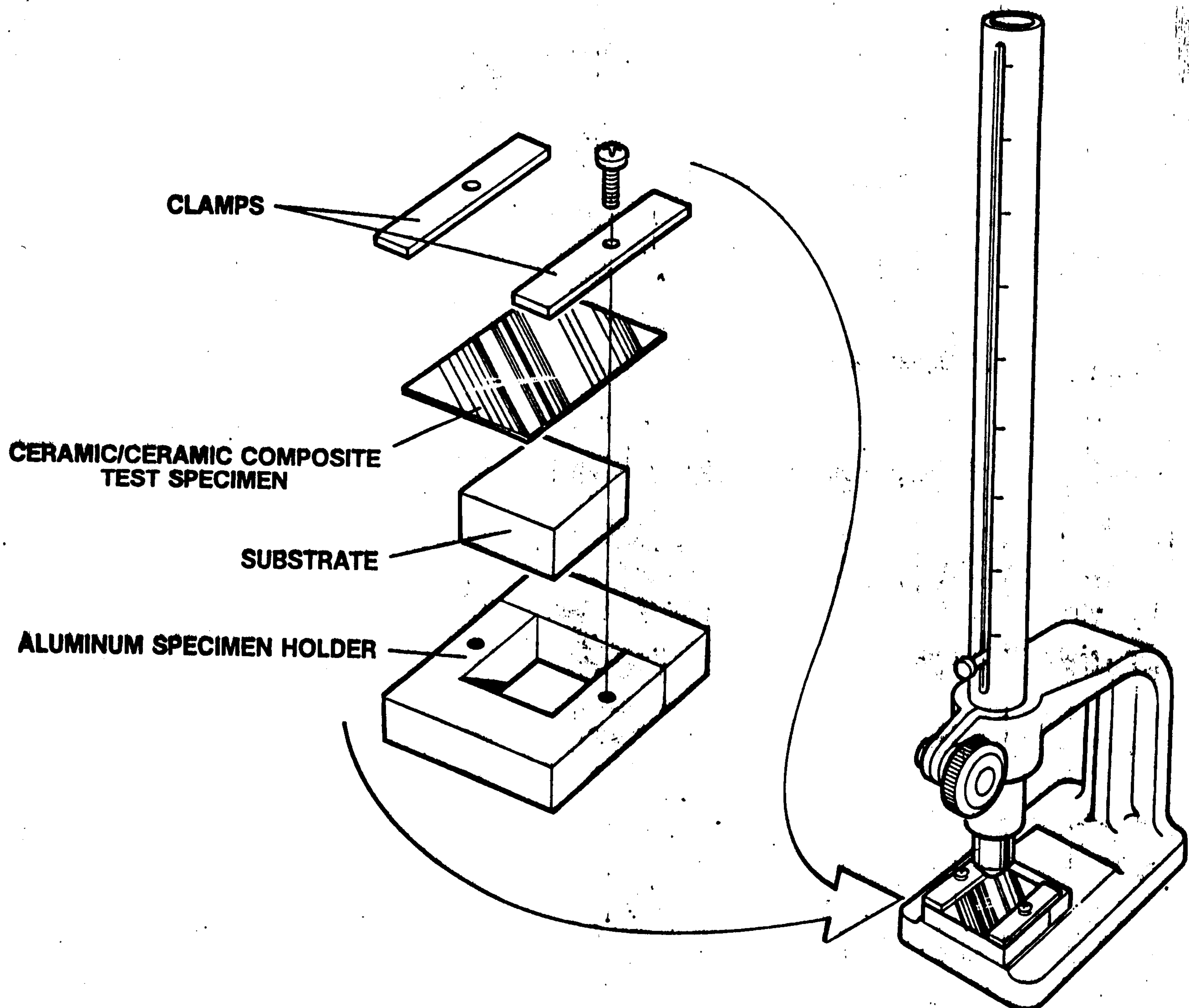


Fig. 3 Modified Gardner Laboratory impact tester.

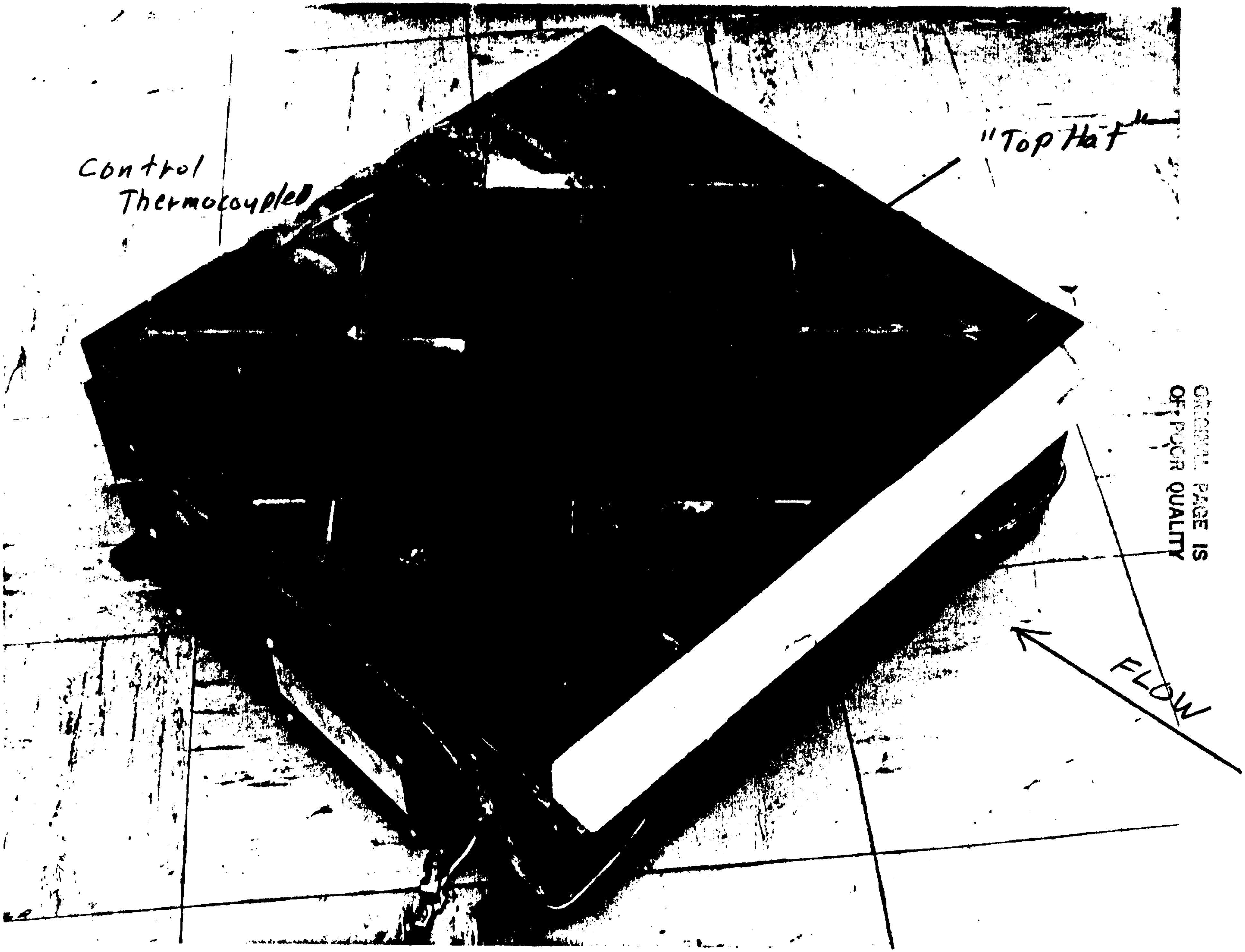


Fig. 4. – "Top Hat" model mounted on test panel with HRSI tiles.

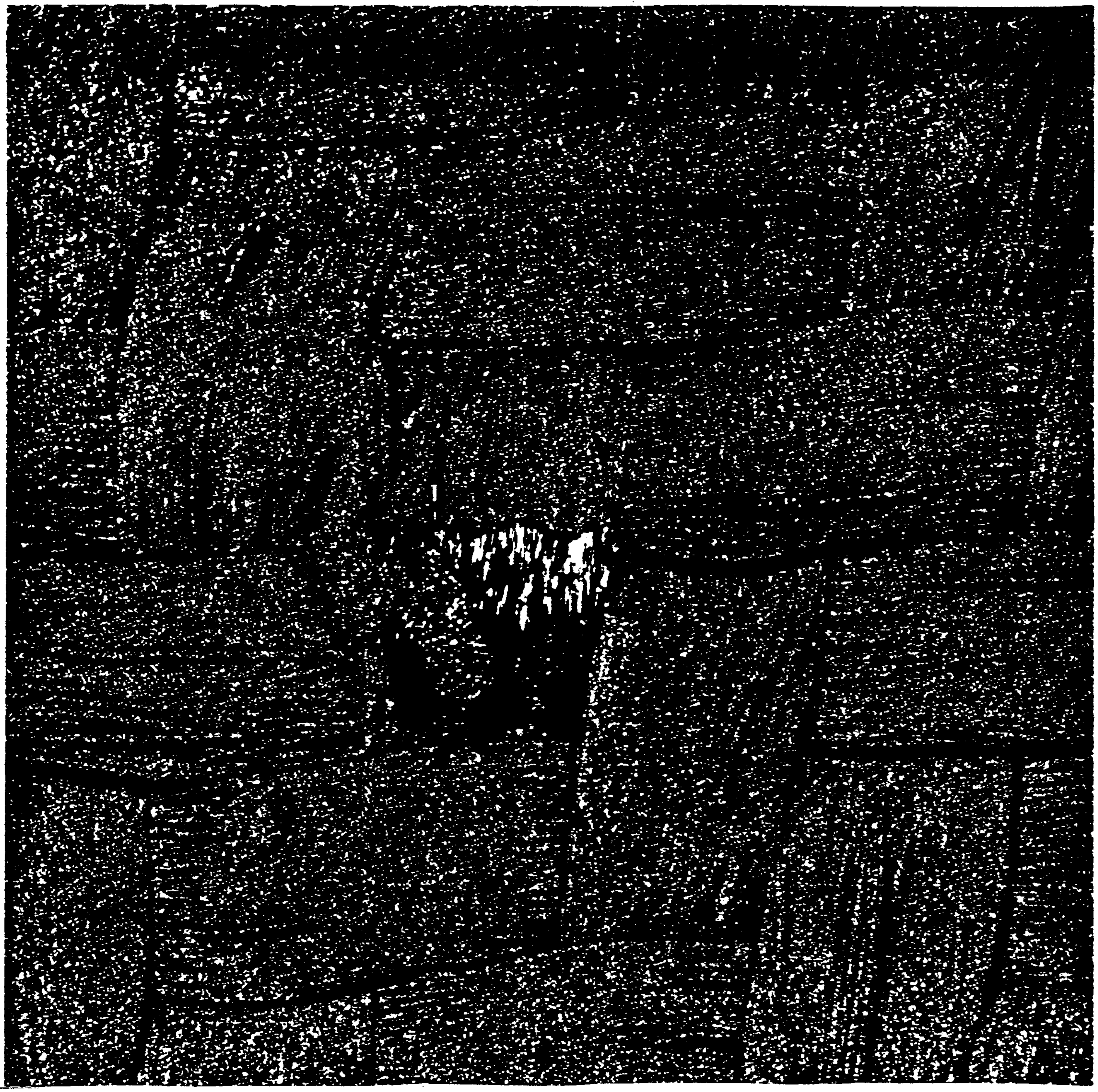
(a) State-of-the-art RCG coating, impact side, 0.56 joules impact

Fig. 5 Post test pictures of impacted tile covers.

5(a)

(b) "Top Hat" baseline weave, impact side, 0.56 joules impact.

(c) "Top Hat" baseline weave, back side, 0.113 joules impact



(d) "Top Hat" ballistic weave, back side, 0.113 joules impact

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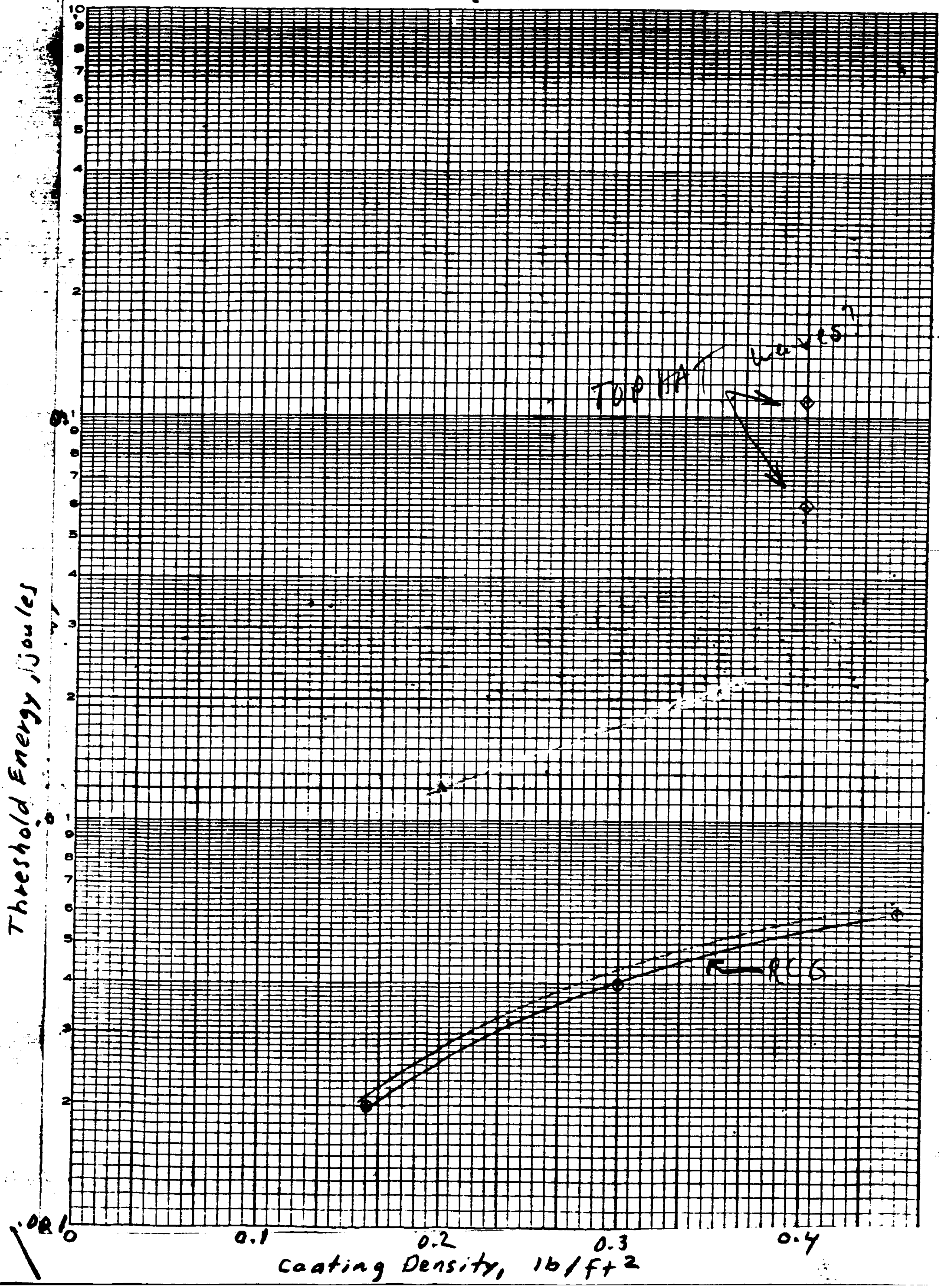
Fig. 6


Threshold Energy, joules
TOP HAT
waves?
RCG
0
0.1
0.2
0.3
0.4
Coating Density, lb/ft²

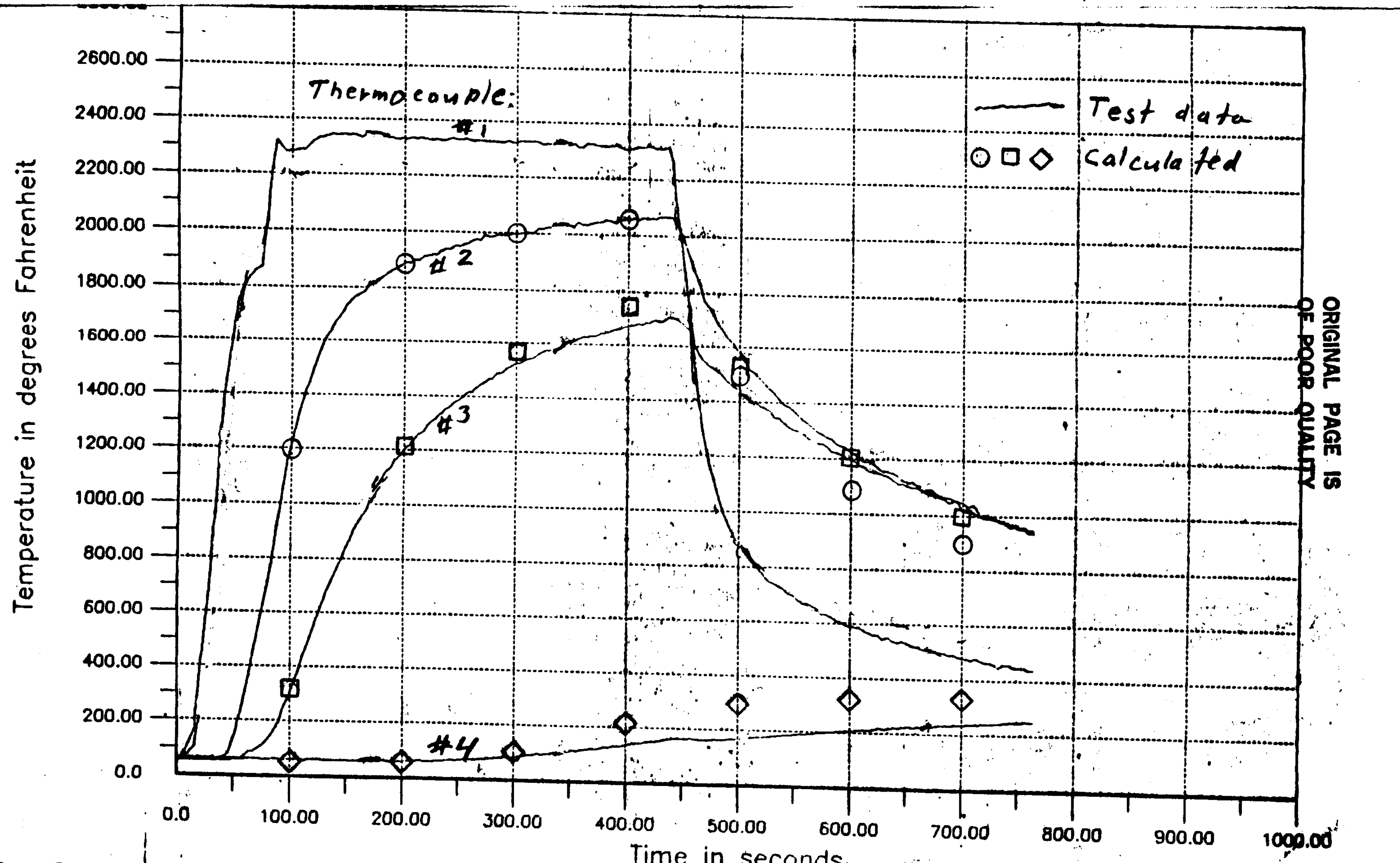


Fig. 7 Comparison of measured and calculated internal temperatures with surface temperatures matched

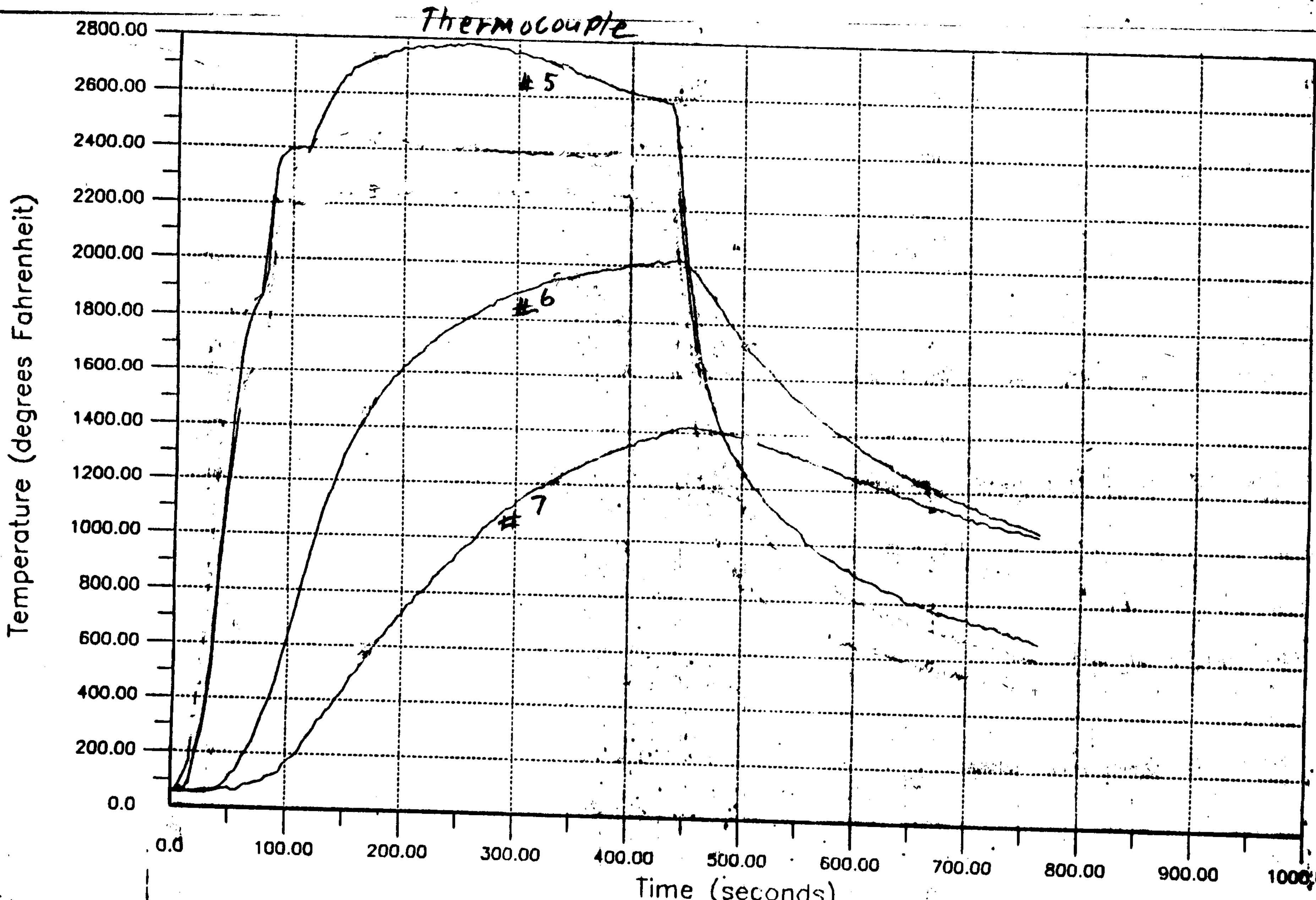


Fig. 8 Measured temperatures on the tab.

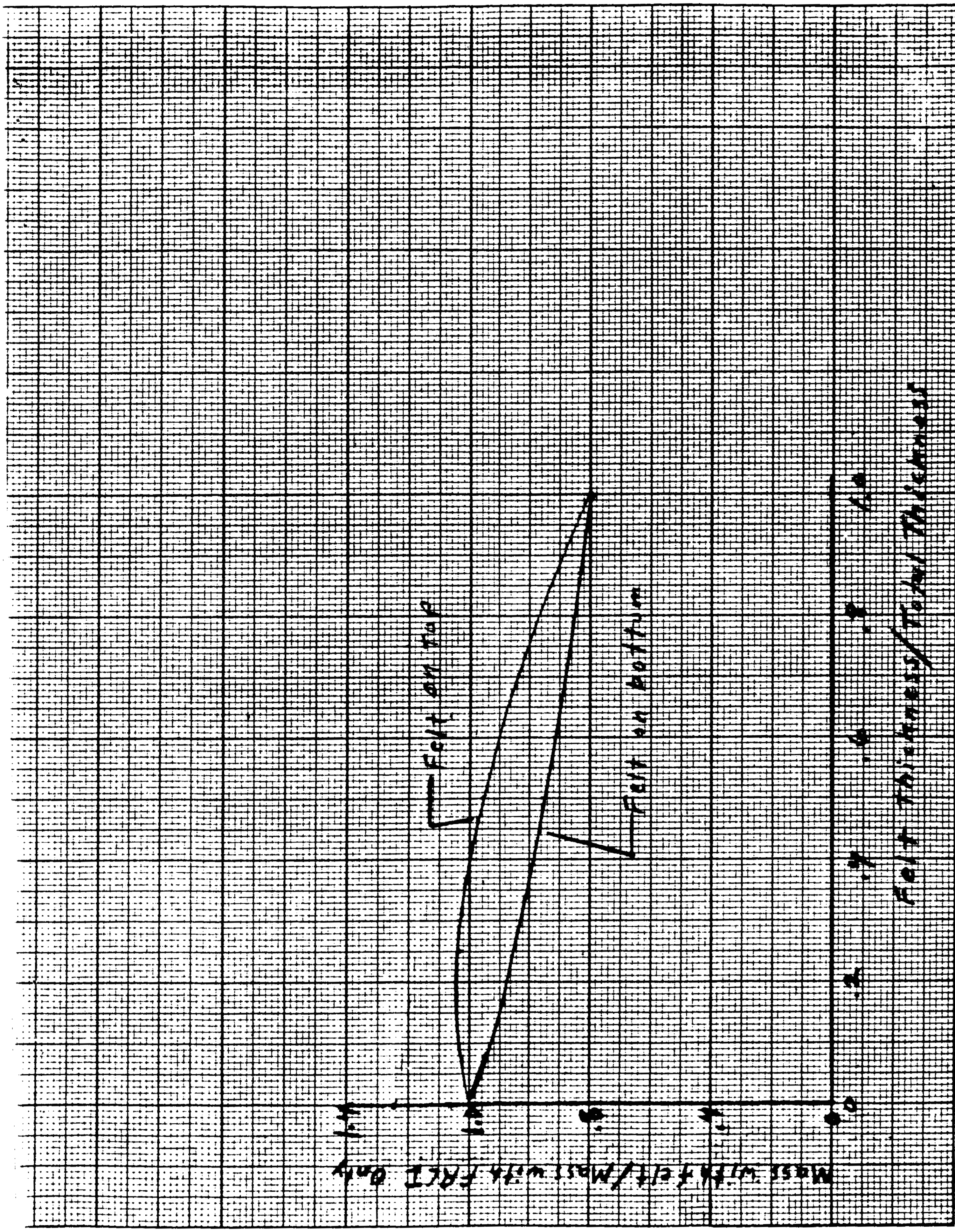


Fig. 9

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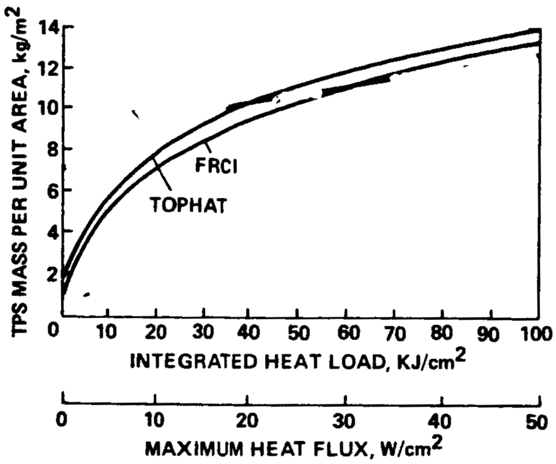

Fig.10 Relative surface density requirements for three heatshield concepts.