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THERMAL SUPPORT FOR SPACE SHUTTLE
INTERIM FINAL REPORT

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CR-123813

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Prepared for National Aeronautics and Space Administration
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Details of illustrations in this document may be better studied on microfiche

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FOREWORD

This Interim Final report presents the results of work performed by personnel of the Lockheed-Huntsville Research & Engineering Center for the Astronautics Laboratory of NASA-Marshall Space Flight Center under Contract NAS8-25569, "Thermal Support for Space Shuttle." This work was conducted in two phases.

The NASA Contracting Officer's Representative (COR) for Phase I of this contract was Mr. R. R. Fisher, S&E-ASTN-PTB. The period of performance for this phase was from February to September of 1970. The COR for Phase II of this work is Dr. K. E. McCoy, S&E-ASTN-PTC. The period of performance for this phase is from February 1971 to the present. Work is still in progress on this phase.

The Lockheed-Huntsville Project Engineer for this contract is William G. Dean.

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Section 1
INTRODUCTION AND SUMMARY

The primary purpose of this contract is to provide thermal support of the design of the Thermal Protection System (TPS) for the Space Shuttle vehicle. The work to date has been conducted over a wide range of problems. However, these can be grouped generally into two phases:

Phase I: Analyses in Support of MSFC "Point Design" Shuttle Configuration: This work involved two tasks as follows:

- Generation of temperature boundaries of the space shuttle "point design."
- Leading edge and nosecap TPS material investigation, and

Phase II: Support of Three TPS Test Facilities as Follows:

- MSFC 36 x 36-Inch Panel Radiant Lamp Test Facility
- MSFC Structural/Thermal Test Facility (STFF), and
- MSFC Hot-Gas O_2/H_2 Burner Test Facility.

This document summarizes the work done to date on each of these phases. It is designated as an "interim final report" (since the contract is still in progress) and combines the results of efforts under both phases to date.

Approximately 40 documents have been published under this contract, most of which are discussed in this report. Also a discussion of several miscellaneous tasks which do not fit any of the above categories is presented, and a list of thermal analytical models developed under the contract is given.

Section 2
TECHNICAL DISCUSSION

The work effort under this contract to date has consisted of a wide range of tasks. For discussion purposes these tasks are grouped under the following categories.

2.1 GENERATION OF TEMPERATURE BOUNDARIES FOR THE "POINT DESIGN" SHUTTLE CONFIGURATION

An analysis was made on the MSFC Space Shuttle orbiter and booster vehicles to generate temperature boundaries for these vehicles during atmospheric reentry. The boundaries were defined as the lowest attitude at which a vehicle can fly without exceeding the specified temperature at a given body point. The boundaries were generated using the following as parameters: allowable wall temperature, three angles of attack, two transition Reynolds numbers, three heating rate methods and a heating rate margin of safety. A discussion was presented of the effect of each parameter on the allowable altitude for each vehicle. An analysis was also made to estimate the effect on allowable altitude of flowfield assumptions and shape correction factors. It is concluded that the heating rate method, the vehicle angle of attack, and the allowable temperature are the parameters that most affect the temperature boundaries. This work is documented in Ref. 1. The work was done using the computer program of Ref. 2.

2.2 LEADING EDGE AND NOSECAP MATERIALS INVESTIGATION

The purpose of this task was to: (1) survey available materials for nosecap and leading edge application, and obtain available thermal and ablation properties; (2) survey available computer programs for analyzing these materials; and (3) apply these materials and programs to determine leading edge, nosecap, and selected protuberance region thermal protection system designs and weights.

A study was conducted in which various thermal protection system materials were applied to nosecap and leading edge areas of the MSFC Space Shuttle orbiter and booster vehicles. Results are documented in Ref. 3. Material thicknesses and weights were determined for each body area. The results showed the nonreusable ablators to be the most efficient (from a weight standpoint) of those materials studied. The Manned Spacecraft Center carbon/carbon material, backed by an insulation such as Dynaquartz, was also shown to be a very attractive material if it can be developed to the point of being fully reusable. An increase in allowable backface temperature (between the thermal protection system material and the substructure) of from 600 to 1000°F was shown to yield a thermal protection system weight decrease of about 40 to 50%. The computer program used for most of the analyses was the NASA-MSC Charring Ablation program of Ref. 4.

2.3 SUPPORT OF MSFC TPS PANEL TEST FACILITY DESIGN AND TESTING

The MSFC TPS Panel Test Facility is shown in Figs. 1 and 2. This is a radiant heating test facility for testing Shuttle-type TPS panels. It consists of a vacuum-tight panel holder for holding a panel 36x36 inches and a radiant array with tungsten element tubular quartz lamps. The lamp reflector is water-cooled, polished aluminum. The facility is capable of heating an area of approximately 28x28 inches to a temperature of approximately 2500°F. Research, Inc., designed and built the facility and their complete description is given in Ref. 5. Lockheed-Huntsville provided some of the thermal analyses in support of the design of this facility.

Four panels have been tested to date in this facility. These are: (1) an L-605 calibration/checkout panel; (2) an L-605 Shuttle-type panel; (3) a Rene' 41 Shuttle-type panel; and (4) a titanium Shuttle-type panel. Lockheed supported the instrumentation, test planning, testing, and data analyses of each of these panels. On the calibration panel, analyses were performed to predict the lateral heating rate uniformity in two directions on the panel surface. Thermocouples were used, data were taken, and results were compared with predictions. The uniformity proved to be quite good as predicted by the analyses. This panel also had a Microquartz insulation package which was instrumented with thermocouples through its thickness.

A series of tests was run on this panel at three pressures: (1) 1 atm; (2) 10 torr; and (3) 2×10^{-4} torr. Data from these tests were used to verify the thermal performance of the Microquartz/Inconel insulation package. This work was completed several months ago but documentation was delayed due to more urgent tasks. Documentation is now being completed in the form of Ref. 6. As a result of this work, it was concluded that published Microquartz thermal conductivity data could be used in conjunction with appropriate contact resistances between the Microquartz and Inconel to obtain an accurate thermal model of the insulation package. Figure 3 shows typical results of Ref. 6 in the form of temperature-time history comparisons from experimental and analytical results.

The L-605 panel, the Rene' 41 panel, and the titanium panel were all tested for 100 thermal cycles. Figure 4 shows the L-605 panel. It consists primarily of an outer corrugated skin with stiffeners, standoff clips, instrumentation package, aluminum angles and a flat-plate aluminum "tank wall simulator." The panel is designed to simulate the initial Space Shuttle booster metallic TPS and hydrogen tank configuration. It is approximately 36x36 inches. The Rene' 41 and titanium panels are similar to the L-605 panel except that the outer skin and standoff clips are changed from L-605 to Rene' 41 and titanium, respectively. Also the L-605 and Rene' 41 panels have a 1.0-inch thick Microquartz insulation package while the titanium panel has a 0.5-inch thick panel.

Tests were conducted at atmospheric pressure in the clean room and at vacuum conditions in the MSFC 14-foot vacuum chamber in MSFC Bldg. 4557. Several types of tests were conducted such as those shown on the following page.

1. Raise panel outer skin to maximum temperature and hold until inside primary aluminum temperature reaches 200°F.

2. Cyclic temperature profile tests to simulate a flight temperature-time history.

3. TPS lap-joint gap leak tests at various pressure differential values.

4. Rain and moisture effects tests.

The maximum allowable outer surface temperatures used were:

L-605	1600°F
Rene'41	1200°F
Titanium	600°F.

Several non-destructive evaluations (NDEs) were made on each panel during the process of accumulating 100 cycles. Also, emissivity values were measured for use in analytical models of the panels.

All these panel tests were conducted by NASA personnel as well as data recording and reduction. Lockheed assisted in the planning of types and methods of testing, with instrumentation recommendations, with test monitoring, and with experimental/analytical comparisons. Test plan inputs were written and instrumentation drawings were made. Approximately 95 thermocouples were used on each panel. Also, on-site support and monitoring were provided for some of the initial testing and when special test problems arose.

A large analytical model of the L-605 panel installed in the test facility was developed for predicting temperatures. It consists of approximately 400 nodes and 2400 resistors. The model considers both radiation and conduction and was run on the Lockheed Thermal Analyzer (Ref. 7) and Lockheed LOHARP program (Ref. 8). The analytical model results have been compared with the experimental data in Ref. 9 with very gratifying results. Figure 5 shows some typical comparisons from Ref. 9.

This L-605 model was modified to become the Rene'41 panel model and the results of this new model were compared to the Rene' 41 panel data in Ref. 10. Again, results compared quite well. Figure 6 shows typical results of this work.

A study was made of the L-605 panel data to determine if any noticeable thermal degradation occurred due to the exposure to 100 thermal cycles. It was concluded (Ref. 9) that there was no degradation within the accuracy of the data taken.

Also a Columbium "panel" was tested on 23 November 1971 to determine if the facility would reach 2500°F. This "panel" consisted of a single sheet of Columbium with 12 thermocouples on the back, and a Microquartz insulation package approximately 2.0 inches thick. The panel reached 2500°F in approximately two minutes as shown on Fig. 7. This temperature was reached without going to full power. However, some of the radiant quartz layers burned out. This was thought to be due to the "smoke" from impurities being burned out of the Microquartz insulation.

2.4 STRUCTURAL/THERMAL TEST FACILITY (STTF) SUPPORT

The work done in support of this facility falls into three categories which are discussed separately as follows:

- Thermal support of STTF design
- Support of Prototype I, Test Article 1
- Support of STTF Checkout Article.

2.4.1 Thermal Support of STTF Design

2.4.1.1 STTF Description

Figure 8 is a sketch of the STTF. The facility is presently under construction on the S-II Test Stand area at NASA-MSFC. It is a radiant heating test apparatus capable of testing large structures up to a maximum size as shown in the cross section of Fig. 9. The radiant array consists of a large number of modular units on adjustable stanchions for allowing testing of different test article shapes. Three rows of modular units are used for a total test region height of approximately 126 inches. A typical modular unit is shown in Fig. 10. It consists of a water-cooled, polished aluminum reflector with end reflectors and lamp supports. Tubular quartz, tungsten filament lamps are used.

The facility is designed to provide test temperatures on a thin-skin metallic TPS as follows:

- Bottom Region, 2500°F
- Side Region, 1800°F
- Top Region, 1200°F.

These regions are shown on Fig. 9. Also, cooling-air jets are to be used in the facility to provide forced cooling to the outer TPS surface to simulate rapid cooldown during a flight.

A complete description of this facility is given in Ref. 11.

2.4.1.2 STTF Design Support

Lockheed provided thermal design support to NASA-MSFC during the monitoring of the design of the STTF. Of particular concern was the problem of being sure that enough power would be provided to reach the specified temperatures. Factors which had to be considered were: (1) "chimney effects" or natural convection losses; (2) radiant array "efficiency"; (3) wind losses; and (4) adequacy of the forced cooling.

Several thermal models of the modular units were made for the purpose of supporting the facility design. (See Table 1 for a list of these thermal models.) The final model developed (used for prediction of power requirements) is documented in Ref. 12. It contains each lamp element and each quartz envelope as an individual node. A Columbium TPS with Microquartz insulation was included in the model to simulate the test article. Natural and forced convection were also included in the model. The methods used are being documented in Refs. 13 and 14 for natural and forced convection, respectively. This model was used to predict power requirements for the top, bottom, and side regions and to predict the cooldown capabilities of the air jets. The cooldown rates were compared with those given as requirements in Ref. 11.

A summary of the results is given in Table 2, where predicted power requirements are compared with the power being provided by Research, Inc. It was concluded that more than enough power is being provided. Results of the cooldown work showed that the cooldown rates can be met by natural convection alone down to a TPS temperature of 500 to 900°F depending on the location. Below this temperature the air jets will cool the TPS at an adequate rate down to a temperature of approximately 200°F. It is felt that 200°F would be a low enough temperature for simulation purposes. If not, the larger jets should be provided which could be accomplished by replacing the air orifices in the modular unit reflectors with larger orifices. It is understood that these orifices are removable.

In addition to the power requirements, cursory analyses were done to estimate the cooling-water requirements for the radiant array. It was concluded that the water flow rate being provided should be sufficient.

2.4.2 Support of Prototype I, Test Article 1

2.4.2.1 Prototype I, Test Article 1 Description

One of the first test articles planned for the STTF is known as Prototype I, Test Article 1. Figure 11 shows an isometric, cutaway view of this structure which consists of a cylindrical LH_2 tank, having hemispherical domes and cylindrical skirts, all of which are covered with TPS panels. The TPS is shown in "Detail A" of Fig. 12. It consists of a corrugated outer skin, with standoff clips, Microquartz insulation, and a titanium support plate. The bottom of the cylindrical tank is supported by shear panels, giving a D-shape cross section (as shown on Fig. 13).

2.4.2.2 Prototype I, Test Article 1 Analytical Support

Support by Lockheed of this test article consisted of: (1) temperature-time history prediction, for various test conditions; (2) temperature gradient predictions; (3) structural analyses; and (4) instrumentation recommendations.

For the temperature predictions two sections through the structure were considered as shown on Fig. 11. The first section was at the middle of the tank between rings 6 and 7. The second section was near the tank dome/tank skirt, and tank cylindrical section junction between rings 3 and 4. Figure 13 shows the section between rings 6 and 7. At this location a number of two-dimensional thermal models were developed and used for temperature calculations (See Figs. 13 and 14 for the locations of these various model locations.) Figure 15 shows an isometric view of the structure and location of "cuts" that were used in the modeling. Note that two sets of models were developed, one at ring 6 and one between rings 6 and 7. A total of eight models was developed. Those at ring 6 were designated as Regions 1-SP through 3-SP and Region 4, and those between rings 6 and 7 were designated as Regions 1-A through 4-A. Figure 16 shows a typical nodal arrangement for one of these models (Region 3-SP). The models extend from the outer TPS skin all the way into the LH_2 inside the tank. These two-dimensional models were used to predict temperature-time histories for several test conditions which were used in the stress analyses of Refs. 15 and 16. The results from these thermal models are documented in Refs. 17 through 25.

The locations of the models developed in the Tank/Dome/Skirt area of the Prototype I, Test Article 1 structure are shown in Fig. 17. These models were designated as Regions I through V. The complexity of the structure, as seen by the cross-sectional view of Fig. 18 required development of three-dimensional models rather than two-dimensional as in the middle of the tank at ring 6. These models are also 40 inches long where the two-dimensional models are 20 inches long. They included detailed radiation view factors and radiation connectors as calculated by the program of Ref. 8. All two- and three-dimensional models were run on the Lockheed Thermal Analyzer program of Ref. 7. Results of the three-dimensional models are documented in Ref. 26. Figure 19 shows a typical nodal network for one of these models.

A model of a single shear panel at ring 6 was developed early in the program before the two-dimensional results were available. This was done to get a quick estimate on the shear panel temperatures to see if there were any particular problem areas. This work is documented in Ref. 27.

A model of a single standoff clip with conduction and radiation was also developed. This was designated the "Heat Short" model and is documented in Ref. 28. Results of this model were used in the TPS stress analysis of Ref. 15.

In addition to the thermal analyses work, Lockheed also supported the Prototype I, Test Article 1 with instrumentation recommendations. A total of 612 thermocouple locations was specified initially for taking data during thermal testing. This was later reduced to 448 when the calibration article was introduced into the program (see Section 2.4.3).

An analysis was also done to calculate the LH_2 boiloff rate from the tank during a typical test. Results are shown on Fig. 20.

Lockheed was also involved in reviewing and commenting on the various versions of the test plans for this test article.

2.4.2.3 Thermal Protection System Stress Analysis

A structural analysis was performed on the TPS center section of the TPS Prototype I, Test Article 1. This analysis determined the allowable stresses for the components at room and elevated temperatures, solved for the maximum combined mechanical and thermal stress levels occuring during the thermal test program, and computed the margins of safety to ensure the structural integrity of the test article. Safety factors of 1.1 on the yield and 1.4 on the ultimate were used in the margin of safety calculations. No safety factor was applied to the thermal component of the stresses.

The TPS structure analyzed was typical of the region between rings 4 and 8, well within the area of applied heat. The applied heat rates were assumed to be uniform longitudinally, while varying symmetrically around the circumference. Due to the symmetry of the structure and applied heat rates, only one-fourth of the panel structure was modeled, thus, allowing for much greater detail in the finite element computer models.

The above work is documented in Ref. 15.

2.4.2.4 Primary Structure Stress Analysis

A structural analysis of the center tank section of the Prototype I, Test Article 1 was performed. This analysis determined the allowable stresses for the components at room and elevated temperatures, solved for the maximum combined mechanical and thermal stress levels occurring during the thermal test program, and computed the margins of safety to ensure the structural integrity of the test article. Safety factors of 1.1 on the yield and 1.4 on the ultimate were used in the margin of safety calculations.

The center tank structure was considered to be the region between rings 4 and 8, well within the area of applied heat. The applied heat rates were assumed to be uniform longitudinally, while varying symmetrically around the circumference. Due to the symmetry of the structure and applied heat rates, only one-half of the structure need to be modeled, thus, allowing for much greater detail in the finite element computer model.

The center tank structure was analyzed for the following local conditions:

- Load Condition A — Maximum Heating Rates ($t = 324$ seconds)
- Load Condition B — Maxium Temperature Levels ($t = 560$ seconds)
- Load Condition C — Flight Profile Heating Rates ($t = 600$ seconds)
- Load Condition D — Tank Proof Pressure ($P = 63.0$ psi).

This work is documented in Ref. 16.

2.4.3 Structural Thermal Test Facility Calibration Article Support

Lockheed also participated in the selection of a test article for the checkout of the Structural Thermal Test Facility (STTF). The problem involved was that of being able to checkout the 2500°F capability of the facility without having to build an expensive test item of a 2500°F material such as Columbium. Several concepts were considered such as a large movable Columbium panel, a water-filled concept, etc. However, the article selected consisted of L-605 "panels" or shingle mounted on a steel frame. It is the "maximum envelope" shape, that is, the largest shape the STTF is capable of testing. A test plan is now being formulated on the checkout procedure to be used with the test article/facility. Instrumentation was recommended which consisted of 398 thermocouples, eight "total $\dot{q}$" type calorimeters, and 20 radiometers.

2.5 NASA-MSFC HOT GAS TEST FACILITY SUPPORT

2.5.1 Facility Description

The Hot Gas Facility is shown in Figs. 21 and 22. It consists of the basic parts shown on the following page.

- Combustion chamber
- Throat section
- Nozzle (four sections; see Fig. 22)
- Front diffuser (top and bottom sections)
- Rear diffuser (six sections)
- Diffuser centerbody
- Centerbody support section
- Leading edge specimen.

The facility burns gaseous O_2/H_2 at an oxygen-to-fuel ratio of approximately 12 to 1. The Mach number at the nozzle exit plane is about 4.0. It runs at a chamber pressure of up to 140 to 150 psia. All parts of the facility are water-cooled.

2.5.2 Facility Design and Analyses Support

Lockheed was involved in the thermal design of the Hot Gas facility in the following areas:

- Flowfield calculations for various chamber pressures
- Heating rate calculations on all surfaces exposed to the flow from the ejector face to the diffuser exit.
- Cooling water flow rates for various pressures in approximately 11 water circuits
- Cooling water temperatures (boiling calculations were necessary in some areas)
- Facility wall temperature in some areas exposed to the hot gas flow field.

As a result of the cooling water temperature predictions, some of the water flow circuits were changed from the initial design to lower the bulk water temperature. Several runs have now been made in this facility and most of the calculated water temperature agreed favorably with the measured values. An exception to this is the rear diffuser region where the measured values are quite a bit higher. This is presently being checked out and is thought to be due to shock/boundary interactions inside the diffuser.

Work was also done to determine if the facility would run with the leading edge model at angle of attack. It was concluded that the facility would probably run with 10 degrees but not at 30 degrees (the two values investigated).

Also, an investigation was made to determine the temperature on a movie camera window mounted on the side of the facility.

A summary of the work on the Hot-Gas Facility is documented as follows:

- For the flow fields (Refs. 29 and 30)
- For the leading edge at angle of attack (Ref. 31)
- For heating rates (Refs. 30 and 32)
- For cooling water analyses (Refs. 33 through 36).

2.5.3 Hot Gas Facility Test Specimen Analytical Support

In addition to the facility design support, analyses were also done on test articles. Three test articles were involved:

- A water-cooled leading edge calibration model
- A carbon/carbon shuttle wing leading edge specimen made by McDonnell Douglas
- Panel-type insulation and/or ablative materials.

On the calibration model, the cooling water requirements and temperatures were calculated. This work is documented in Ref. 33. Instrumentation recommendations were also made for pressures and heating rate measurements, as given in Ref. 37. Data from this model are now being compared with the analytical results with fair agreement. However, the data are preliminary at this time.

In support of the carbon/carbon leading edge specimen, several thermal analytical models have been made. These are:

- Initial hand calculations (Ref. 38)
- Two-dimensional model (Ref. 39)
- Three-dimensional model (Ref. 40)
- Three-dimensional model (with refinements after inspection of the model when delivered to MSFC) (Ref. 41; see Fig. 23).

The first models were crude but provided quick numbers for support of the facility design, that is to check to see if the facility could provide the needed temperatures on the specimen. As time progressed the models were improved and refined. The latest models show that the carbon/carbon leading edge should reach 2580^{o}F for a chamber pressure of 142 psia, and 2410^{o}F for a chamber pressure of 76 psia. The Rene' 41 plate on top of the model should reach 1560 and 1450^{o}F for P_c = 142 and 76 psia, respectively.

Work on the panel specimen is just beginning. It is planned that these tests would be run with the panel of "spray-on" foam insulation mounted on top of the centerbody, just downstream of the leading edge specimen. Work is also now in progress to provide a means of simulating lower heating rates during an ascent trajectory of the shuttle. Two means are being investigated: adding either gaseous N_2 or liquid water to the flow in the combustion chamber. It is also hoped that this can be used to program the heating rate time history to simulate a flight profile.

2.5.4 Pilot Facility Support

In addition to the Hot Gas facility support, work was also done on a smaller O_2/H_2, O/F = 12 to 1 facility. This was done to check the validity of analytical methods being used. Boundary layer temperature profiles and heating rates inside the nozzle were predicted and compared with experimental results. Agreement was favorable as documented in Refs. 29 and 42.

2.6 OTHER MISCELLANEOUS TASKS

Several other tasks were performed which did not fall into any of the preceding categories. These include:

- A thermal analysis of a heat-short/fastener area on an HCF panel attachment design. This analysis showed that the design was not acceptable.
- A study to determine the effect of emissivity values on internal heat transfer rate between internal surfaces such as Microquartz/L-605, and titanium support sheet/ LH_2 tank wall. This work is documented in Ref. 43.
- Instrumentation recommendation for a "Langley" Test Panel. It is planned that 42.5 x 60-inch Rene' TPS panel will be tested in the Langley 8-foot Structures Tunnel.

Instrumentation recommendations for this panel have now been completed. One-hundred twenty nine thermocouples were recommended.

- A study of the flow field resulting from the liftoff of a space shuttle configuration using two large solid rocket motors (SRMs) in combination with the main orbiter engine.

For the latter study the analysis will include liftoff to SRM booster stage separation. It is anticipated that starting and operation of the orbiter main engines on the launch pad will be difficult. This problem arises because the orbiter engines are configured to operate at high altitudes and thus have high area ratio exhaust nozzles. The exit pressure on the nozzles is much lower than sea level ambient conditions, even at full chamber pressure. Therefore, the nozzle always operates in an overexpanded mode at sea level. The starting transient in the chamber pressure creates an even more severe condition. To permit the nozzle pressure to equilibrate with the ambient pressure, a shock forms in the nozzle flow field and produces a subsequent separation of flow at some point on the nozzle wall. Experimental results obtained with the J-2 engine have shown that: (1) the separation occurs in a random fashion within the nozzle; (2) the separation region is constantly moving, sometimes in a spinning mode; (3) high side loads on the engine bell are produced; and (4) detrimental heating on nozzle walls is produced when the internal shock contacts the nozzle wall. It is anticipated that these problems will be encountered with the orbiter main engines.

Various corrective measures for this problem have been discussed. These include: (1) nozzle plugs of various configurations and materials; (2) constant area extensions to the nozzles; and (3) boundary layer injection and/or suction. A study has been initiated to conduct cursory analyses of the nozzle plug concept being considered. An analysis is being conducted to predict the flow field resulting from a blunt-nose plug being located in an orbiter nozzle. To accomplish this study, a representative blunt nose and plug afterbody shape and plug location will be selected. The flow field adjacent to the blunt nose region will then be solved. This information will then be utilized to predict the flow field in the orbiter nozzle with the plug in place. The results of the study will be: (1) the pressure distribution on the nozzle and plug walls; (2) the location of the shock emanating from the blunt nose in the flow field; and (3) engine performance with the plug in place.

2.7 SUMMARY OF THERMAL MODELS

Table 1 gives a list of the thermal models developed to date under this contract.

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Table 1
LIST OF THERMAL MODELS GENERATED FOR SPACE SHUTTLE SUPPORT CONTRACT

Model Name/Description
1. L605 Test Panel (~2000 resistors)
a. Mod. I (turn 90 degrees)
2. Heat short through flexible clip (3-D radiation and conduction)
a. Mod. for Prototype I (general)
b. Mod. for Prototype I (L-605/shear panel area)
c. Mod. for Prototype I (Rene'/shear panel area)
d. Mod. for Prototype I (Rene'/tank)
e. Mod. for Prototype I (Ti/tank area)
3. 2-D through L605 calibration panel (direction No. 1)
4. 2-D through L605 calibration panel (direction No. 2)
5. 1-D through L605 calibration panel
6. 1-D through Prototype I
a. Zone IV (1390°F)
b. Zone III, V (1500°F)
c. Zone II, VI (1200°F)
d. Zone I (1600°F)
e. LH_2 fill and hold
7. 2-D at splice between Rene'/Ti Prototype I
a. "Cut 1"
b. "Cut 2"

Table 1 (Continued)

8. "Big Picture" of Prototype I (150 nodes)
   a. Mod. with more nodes
   b. Mod. with convective effects
9. HFC Heat-Short/Fastener
10. Carbon/Carbon Leading Edge in nozzle duct
    a. Hand-calculation model
    b. 2-D model with external/internal insulation (18 nodes)
    c. 3-D model with 85 nodes
11. Prototype I facility calibration maximum envelope model (boilerplate)
12. Hot Gas Facility movie window
13. Shear panel at Ring 6
14. 2-D Region 1 model. L605 at Prototype I bottom centerline (1390°F outer skin temperature)
    a. Version 1 (cut through shear panel at Ring 6) Model 1-SP
    b. Version la (cut through shear panel area between Rings 6 - 7)
15. 2-D Region 2 model through Prototype I L605 at boundary between 1390 and 1500°F temperature zones
    a. Version 2 (cut through shear panel at Ring 6) Model 2-SP
    b. Version 2a (cut through shear panel area between Rings 6 - 7)
16. 2-D Region 3 model through Prototype I at Rene'/L605 splice
    a. Version 3 (cut through shear panel at Ring 6) Model 3-SP
    b. Version 3a (cut through shear panel area between Rings 6 - 7)

Table 1 (Concluded)

17. 3-D LH_2 Tank Dome/Skirt Model at Rene'/Ti splice on Prototype I

18. Same as 17 but at the Rene'/L605 splice

19. Same as 17 but at the L605, 1500/1390°F junction

20. Same as 17 but at the bottom centerline (L605, 1390°F)

21. Same as 17 but at the top centerline (Titanium)

22. Hot-Gas Facility nozzle fourth section (without water cooling)

23. Prototype I calibration structure response model

24. Prototype I radiant array modular unit

    Version 1: 55 lamps
    Version 2: 27 lamps
    Version 3: 10 lamps

25. Prototype I radiant array quartz lamp

26. Detailed tank/dome skirt model at end of lamp region

27. Model of curved bottom portion of radiant array

Table 2

COMPARISON OF ESTIMATED POWER REQUIREMENTS WITH POWER PROVIDED (FROM REF. 12)

			Required					Provided	Margin
Region	Zone No.	Modules per Zone	Maximum Power per Lamp (Btu/sec)	Lamps per Module	Maximum Power per Module (Btu/sec)	Maximum Power per Module (kW)	Maximum Power per Zone (kW)	Maximum Power Provided by R.I. (kW*)	"Safety Factor" = Provided Required ÷ by Power Provided
Bottom	B1	2 x 3 = 6	1.47	55	81	85.3	511	891	1.74
	B2	4 x 3 = 12					1022	1782	1.74
	B3	6 x 3 = 18					1533	2227	1.45**
	B4	4 x 3 = 12					1022	1782	1.74
	B5	2 x 3 = 6					511	891	1.74
				Total for Bottom			4599	7573	1.65
Side	S1	3 x 3 = 9	.964	27	26	27.4	246.5	656	2.66
	S2	3 x 3 = 9					246.5	656	2.66
	S3	3 x 3 = 9					246.5	656	2.66
	S4	3 x 3 = 9					246.5	656	2.66
				Total for Side			986.0	2624	2.66
Top	T1	6 x 3 = 18	.911	9	8.2	8.64	155.5	437	2.81
	T2	7 x 3 = 21					181.5	510	2.81
	T3	6 x 3 = 18					155.5	437	2.81
				Total for Top			492.5	1384	2.81

*From Ref. 11 (Table 3.5.1)

**Safety factor is lower because the power required in Zone B3 is more due to addition of three modular units (This was necessary because the maximum envelope shape size was increased.)

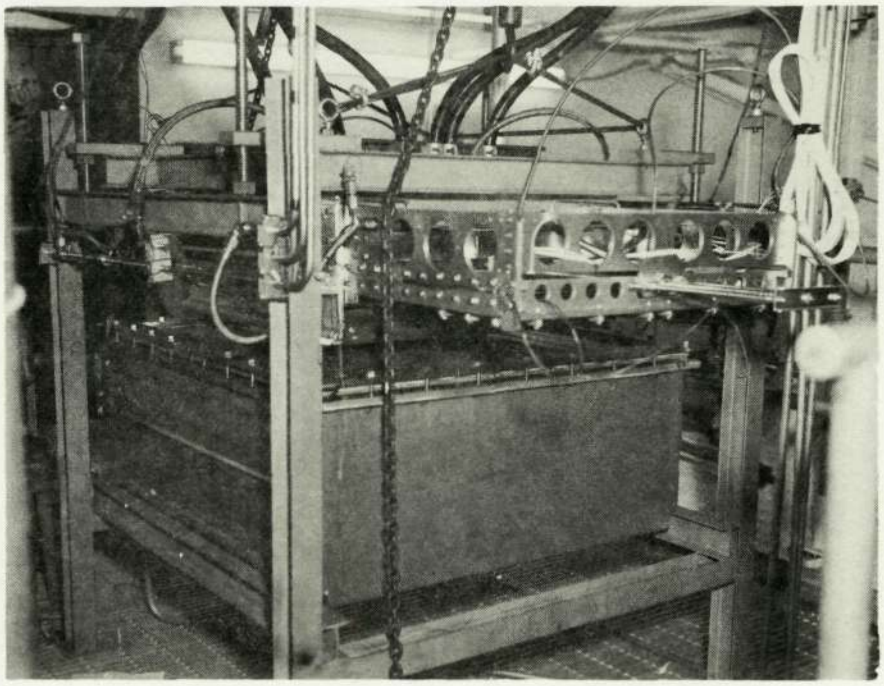


Fig. 1 - Overall View of NASA-MSFC TPS Panel Radiant Heating Test Facility

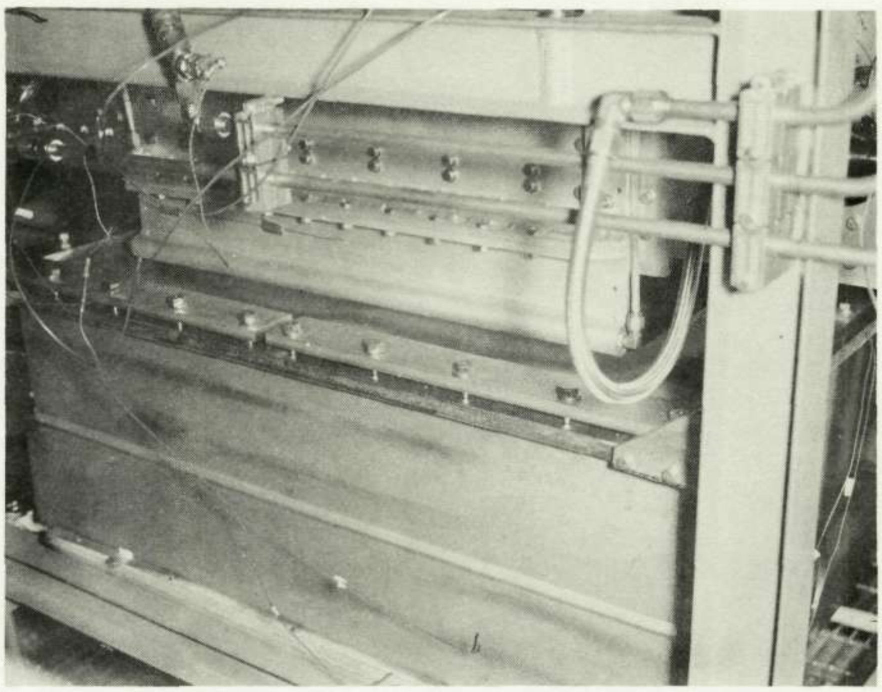


Fig. 2 - Closeup View of NASA-MSFC TPS Panel Radiant Heating Test Facility During a Typical Test

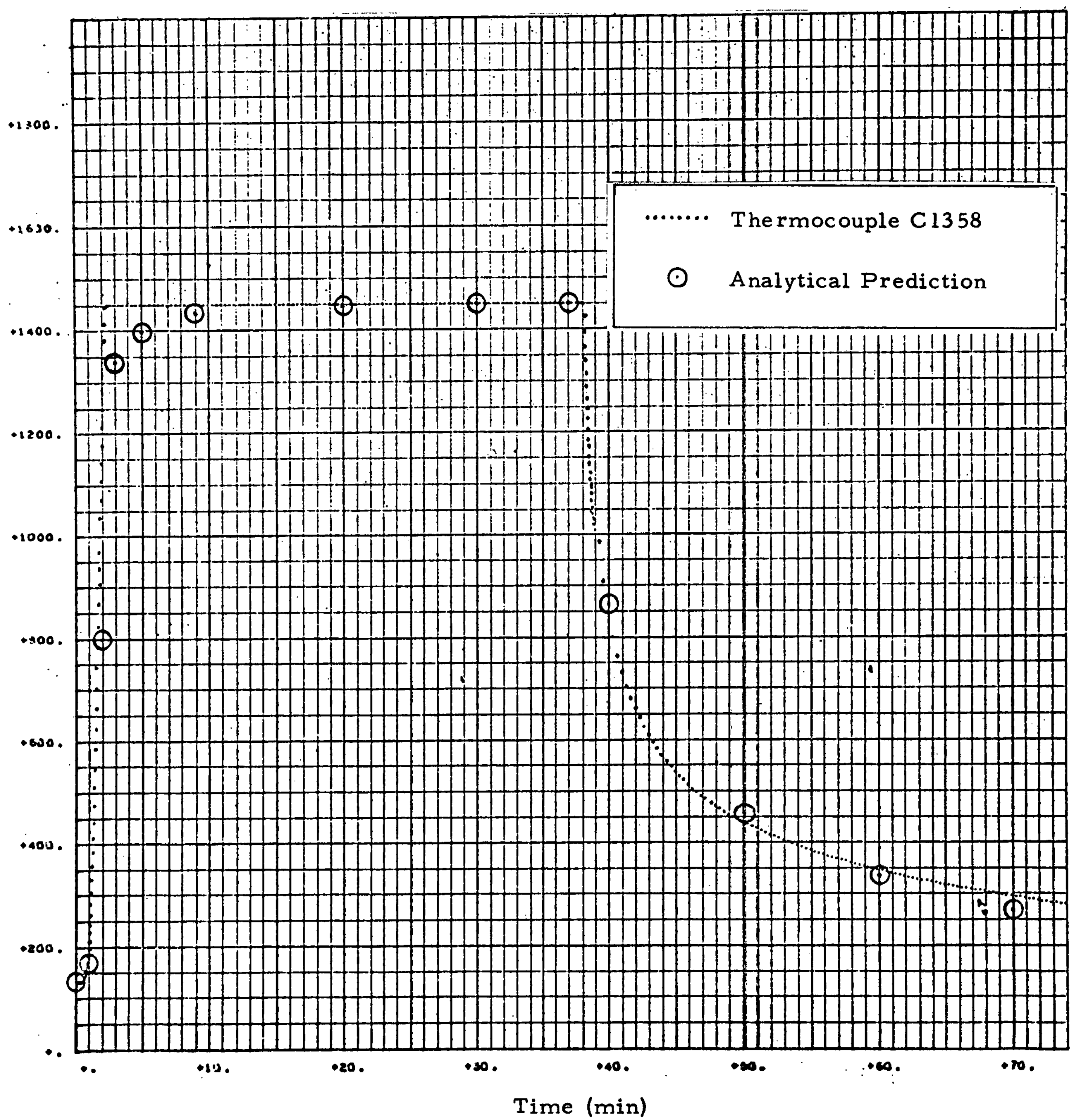
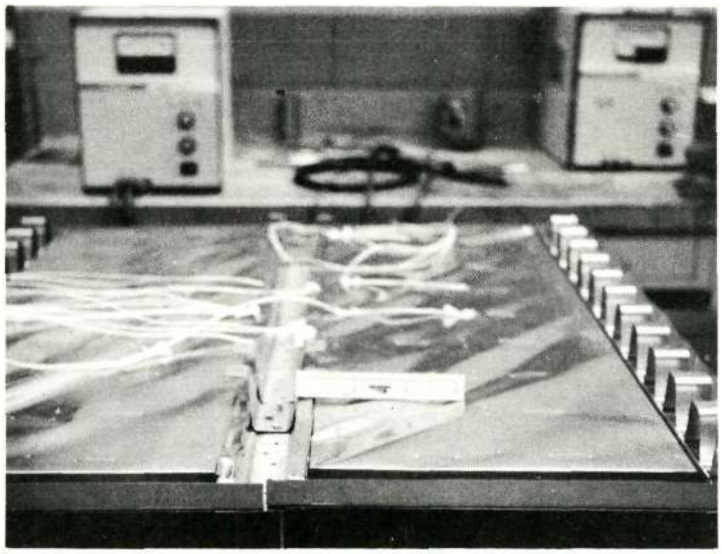


Fig. 3 - Typical Results from Ref. 6. (Inconel Temperature Profile at a Pressure of 2 x 10^{-4} Torr – TPS Panel Test Number 22)

TPS Panel Without L-605 Skin Showing Instrumentation



L-605 Skin with Stiffeners Welded in Position
(Skin upside down on table)

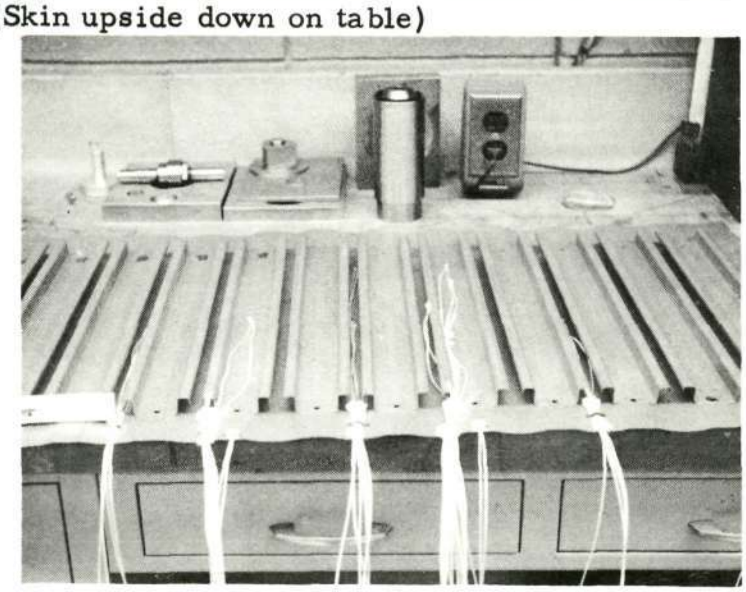


Fig. 4 - L-605 Panel Photographs Before Testing Showing Insulation Packages, Standoff Clips, Stiffeners, and Outer Skin

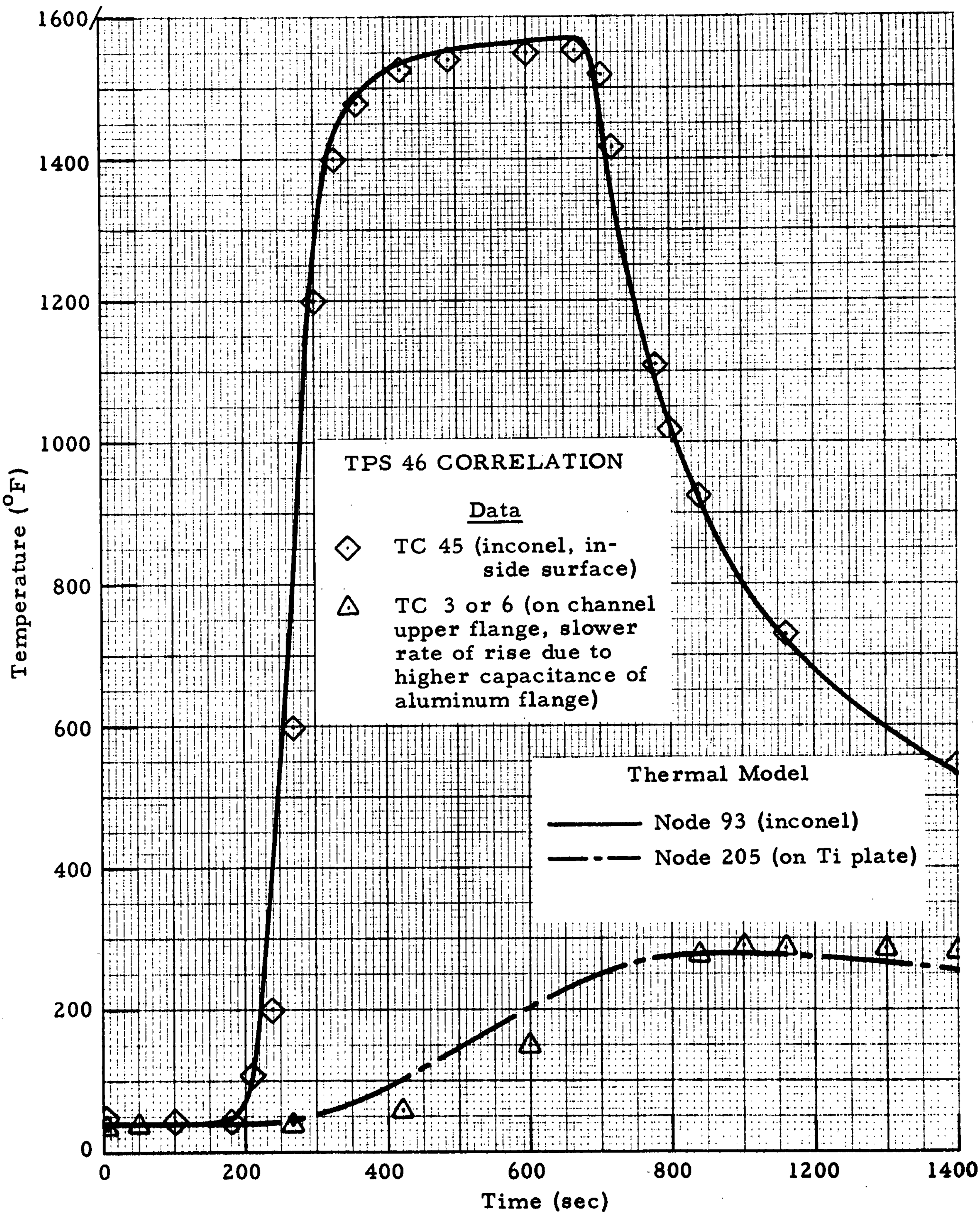


Fig. 5 - Comparison of Analytical L-605 Panel Analytical Model with TPS 46 Test Data (Vacuum Environment)

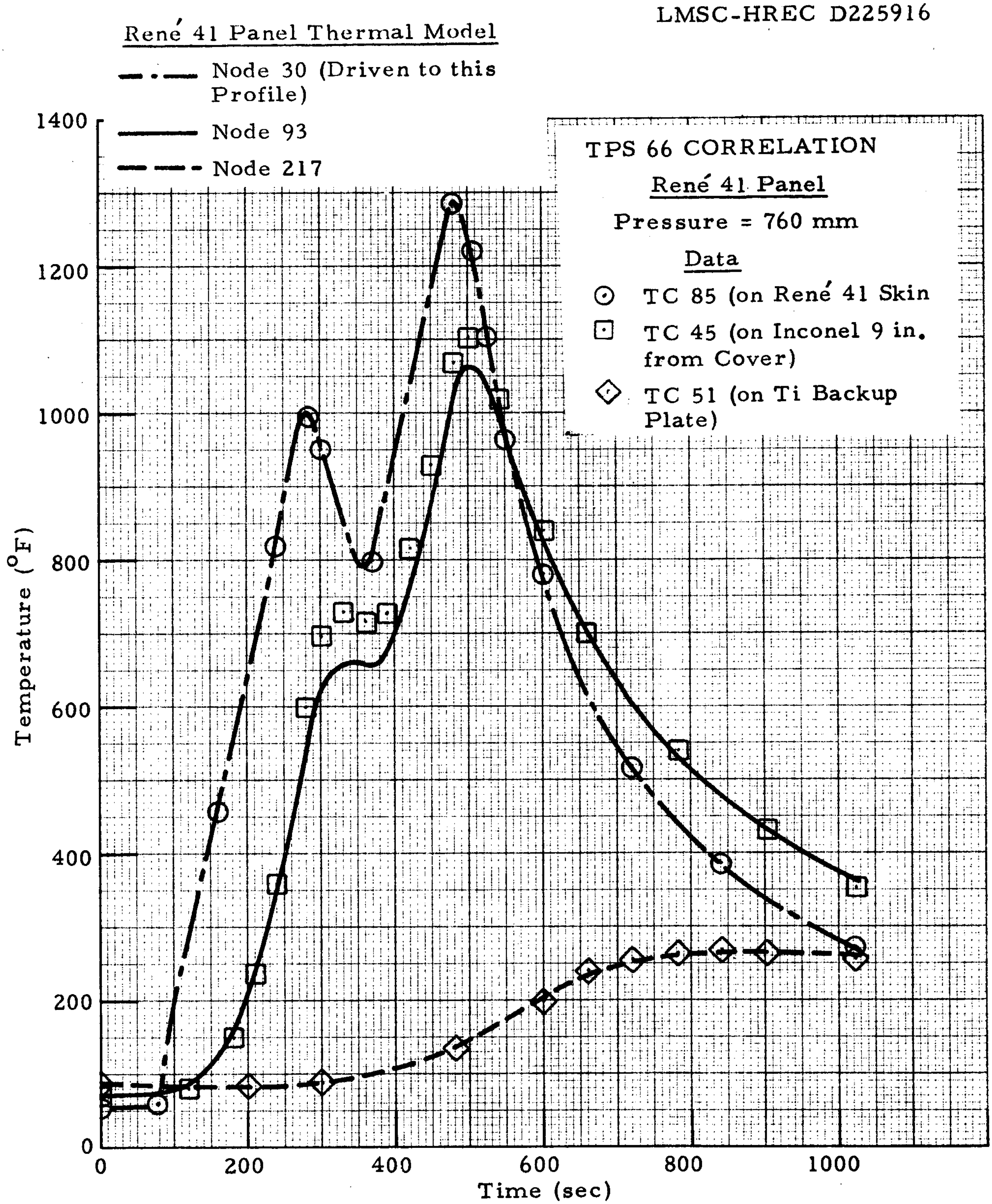


Fig. 6 - Comparison of Analytical Rene' 41 Panel Thermal Model with TPS 66 Test Data (Atmospheric Pressure)

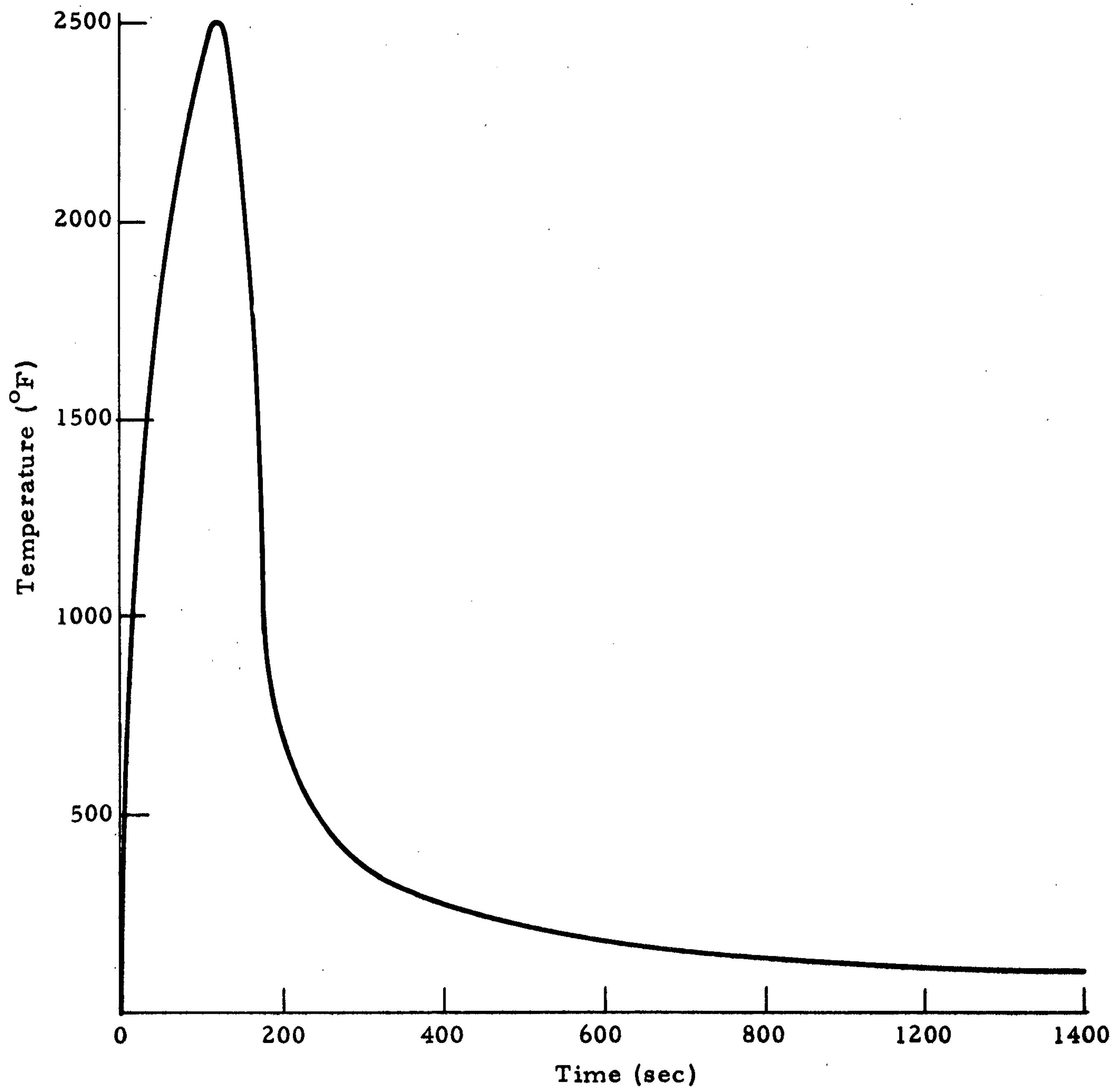


Fig. 7 - Columbium Panel Test Results Lamps at Normal Height Above Columbium (approximately 5-1/4 in.) TPS 53, Run 1

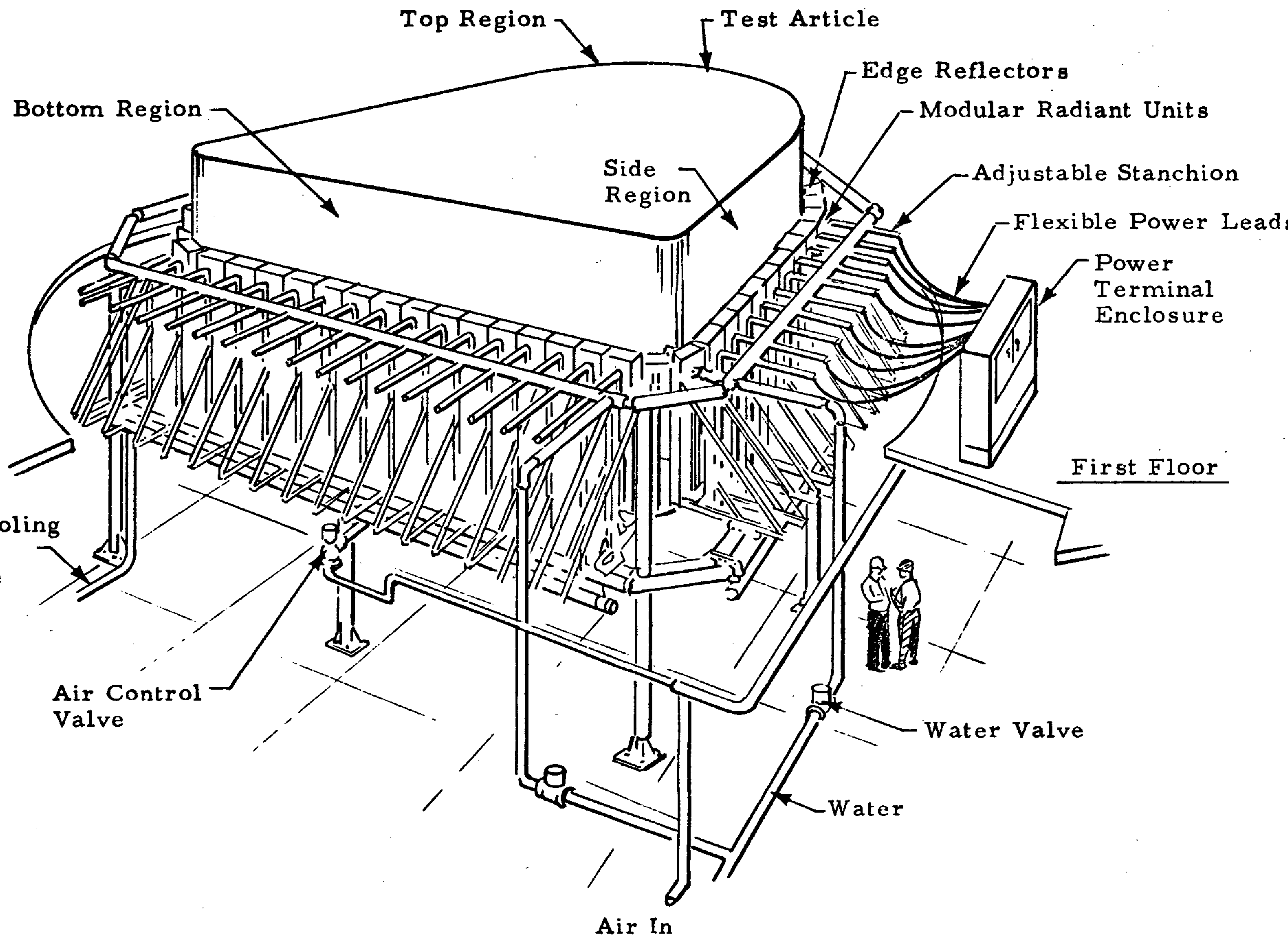


Fig. 8 - NASA-MSFC Structural/Thermal Test Facility (STTF)

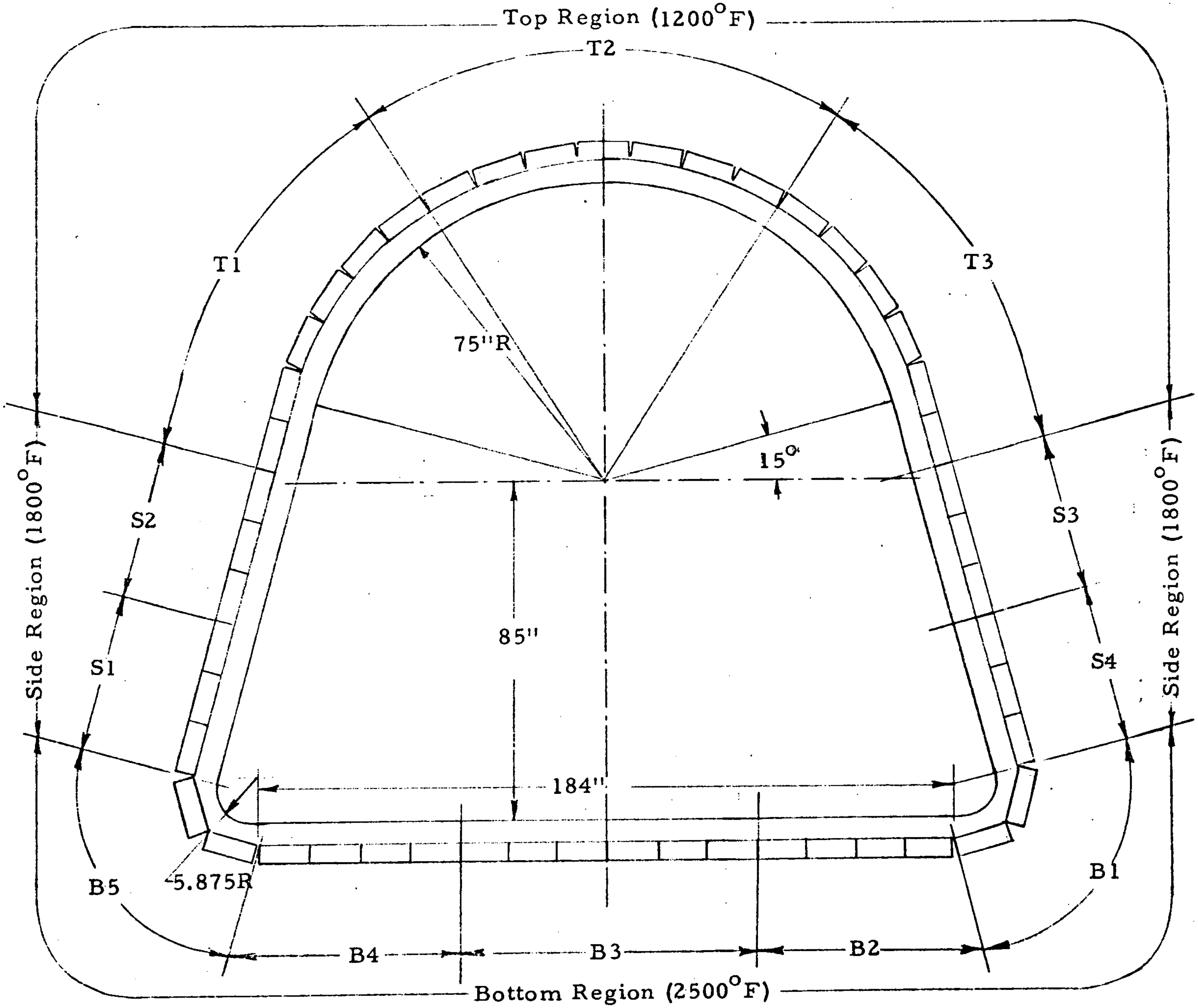


Fig. 9 - Maximum Envelope Prototype I Test Article 1 Showing the Three Regions and Their Zones

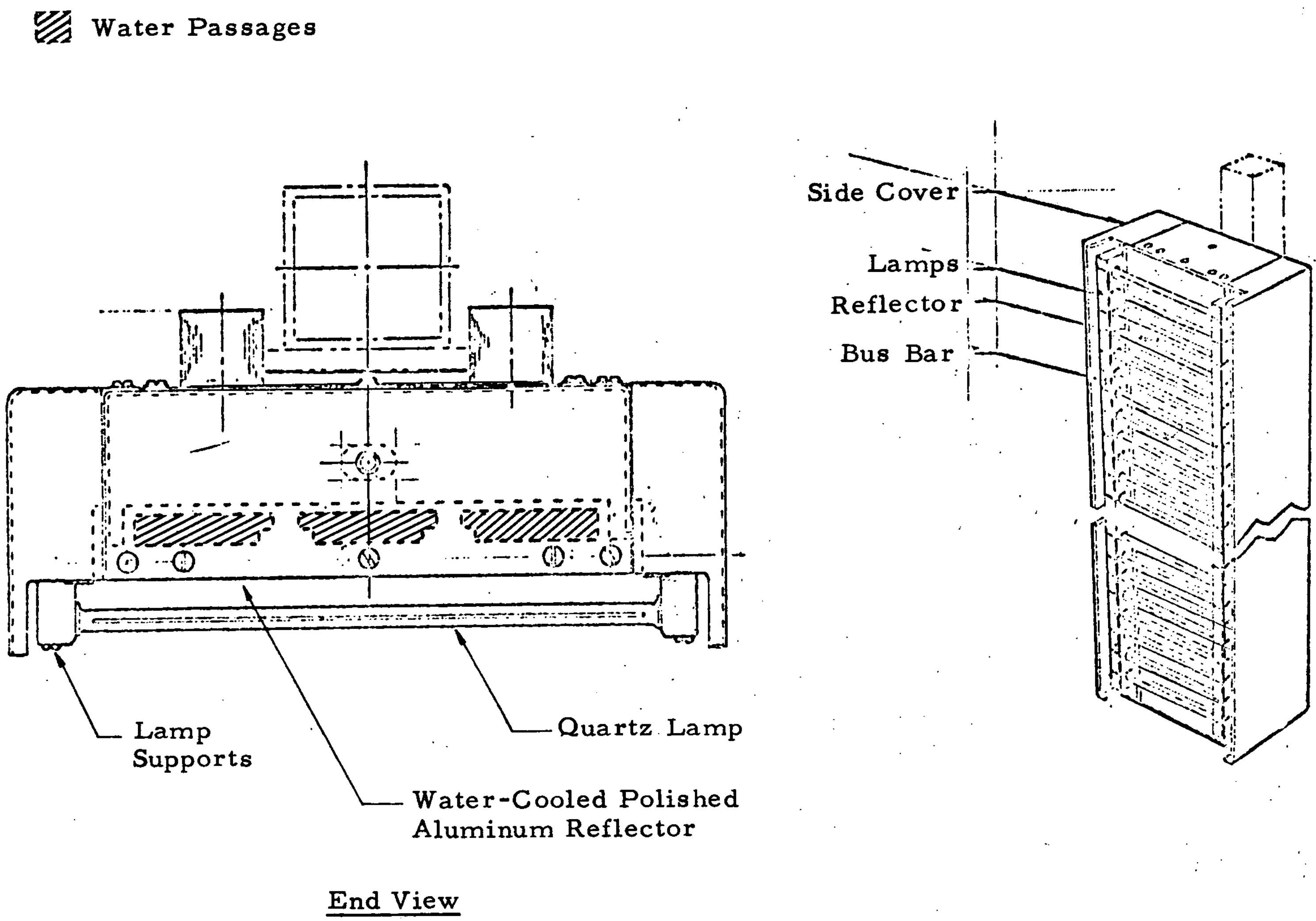


Fig. 10 - Typical Modular Radiant Heating Unit (Ref. 11)

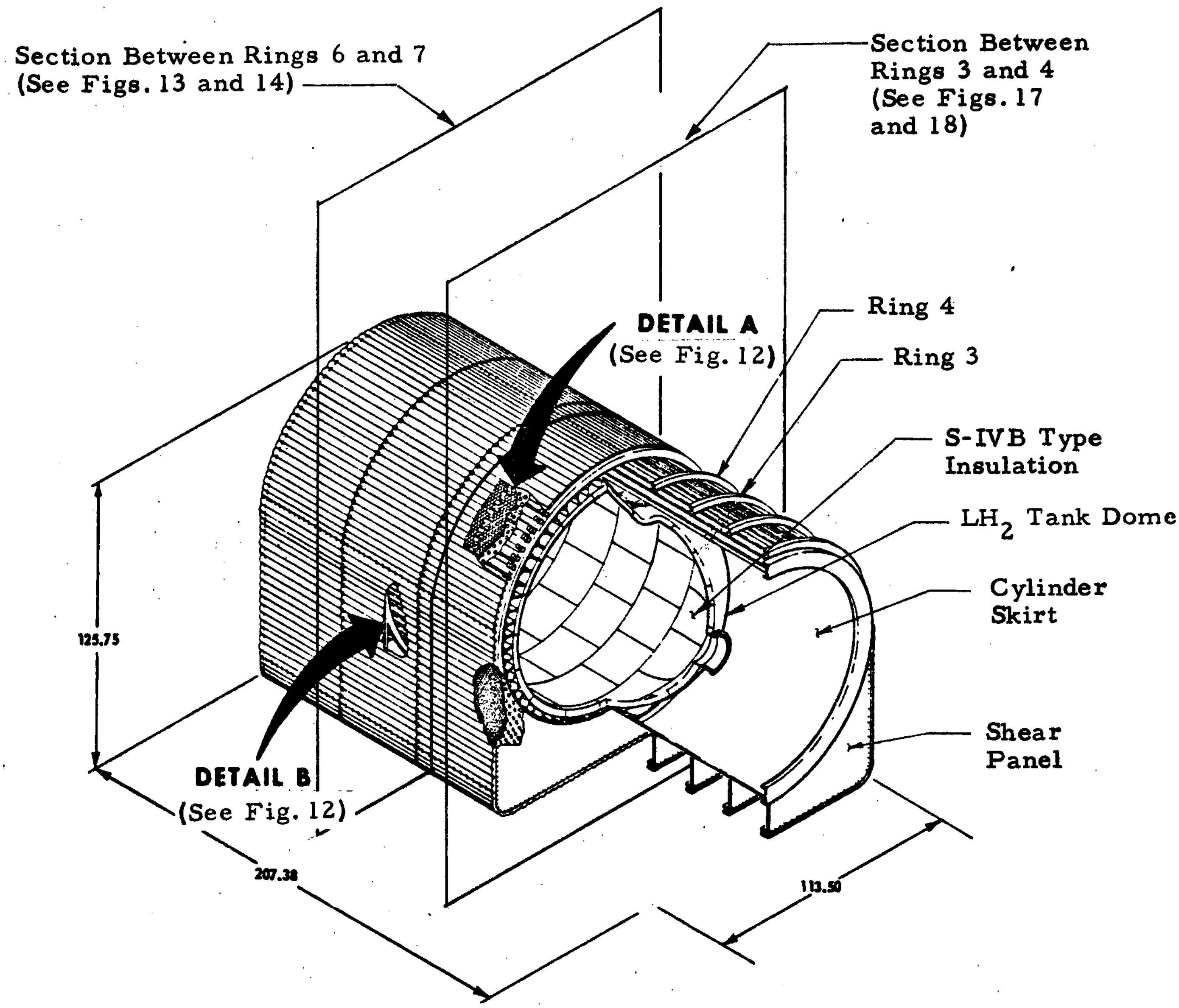


Fig. 11 - Isometric Cutaway View of Prototype I, Test Article 1

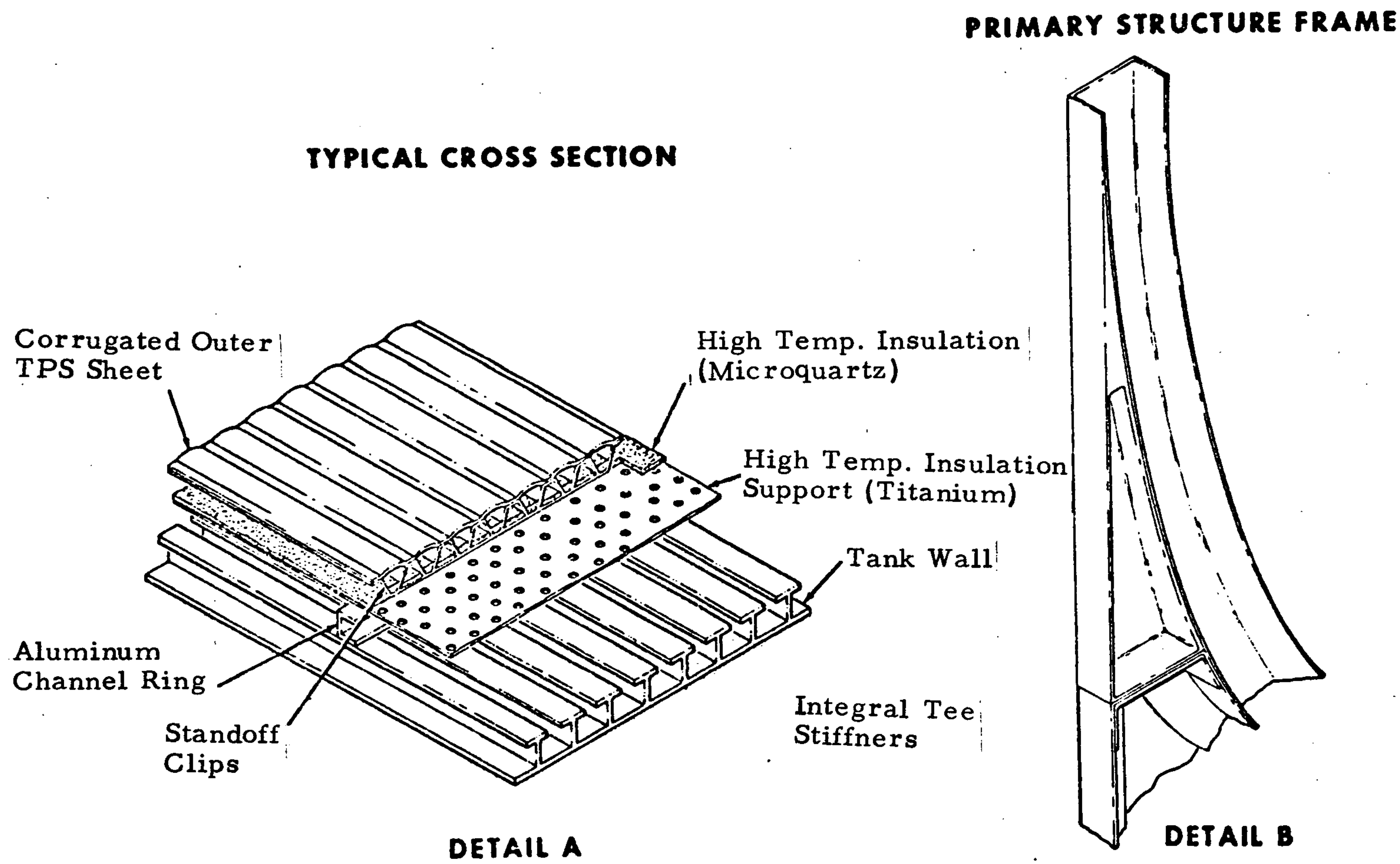


Fig. 12 - A Typical Cross-Section of a TPS Panel and Frame Structure for Prototype I, Test Article 1

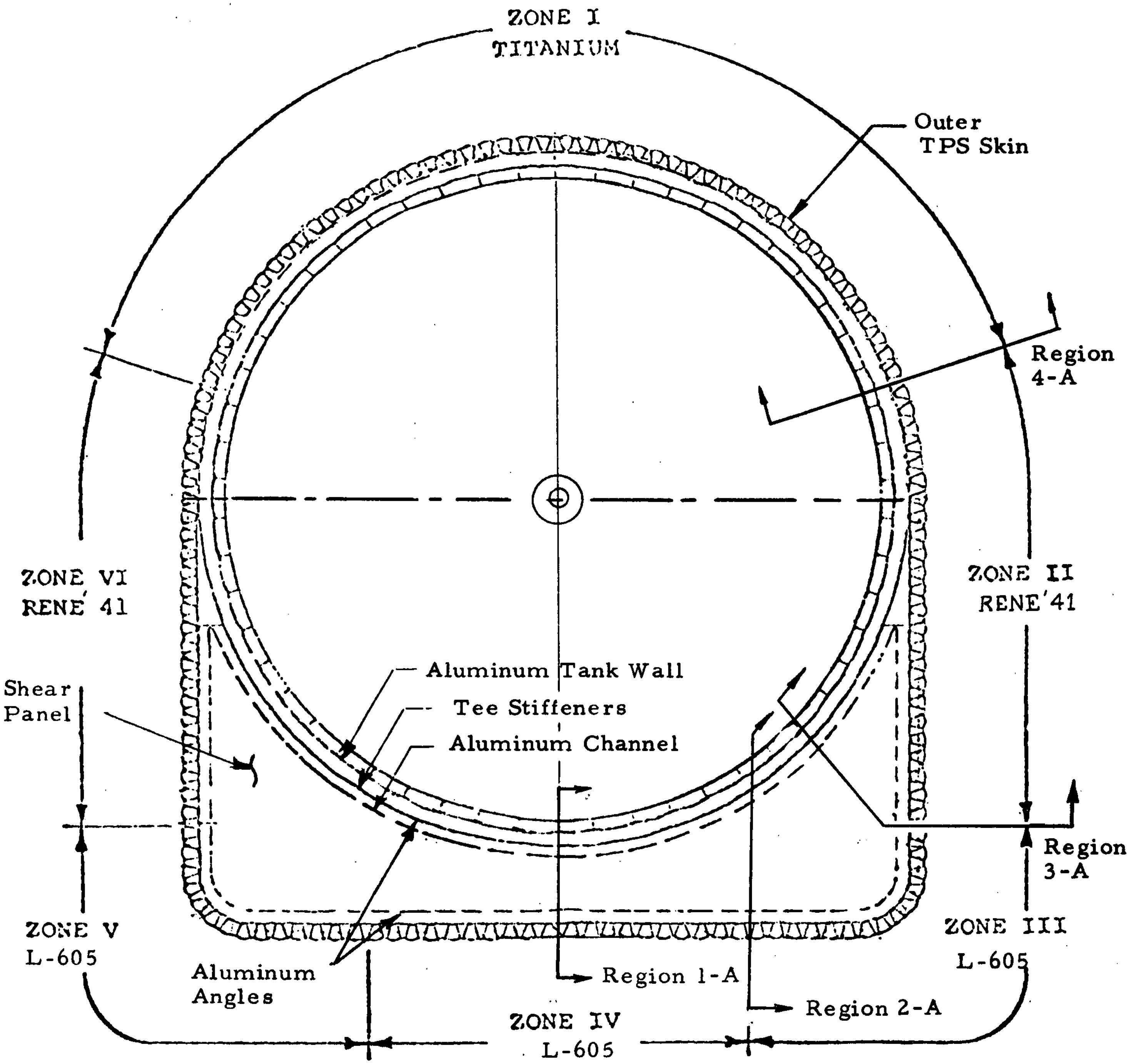


Fig. 13 - Cross-Sectional View of Prototype I - Test Article 1 Showing Location of Two-Dimensional Models Between Rings 6 and 7

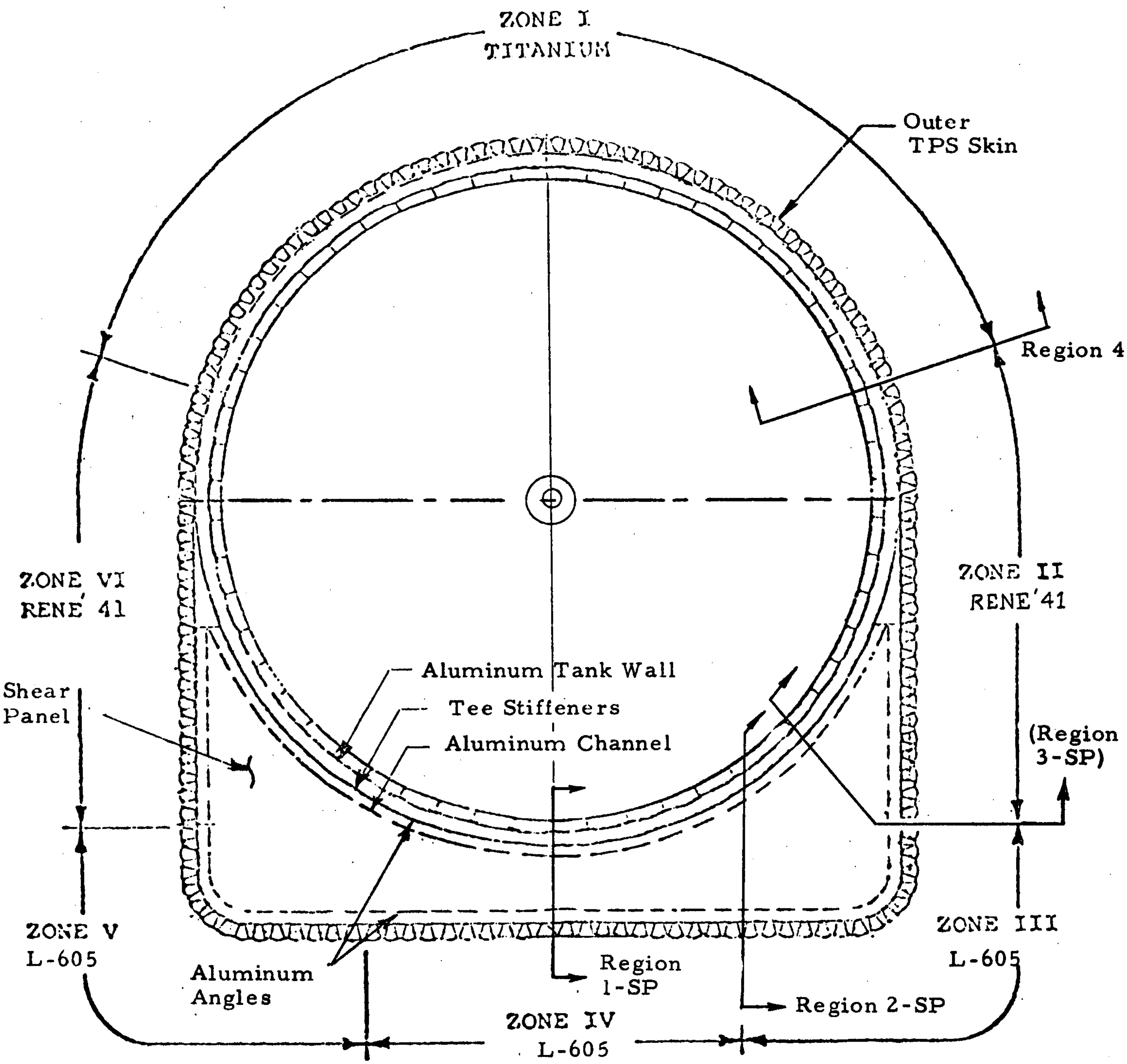


Fig. 14 - Cross-Sectional View of Prototype I - Test Article 1 Showing Location of Two-Dimensional Models at Ring 6

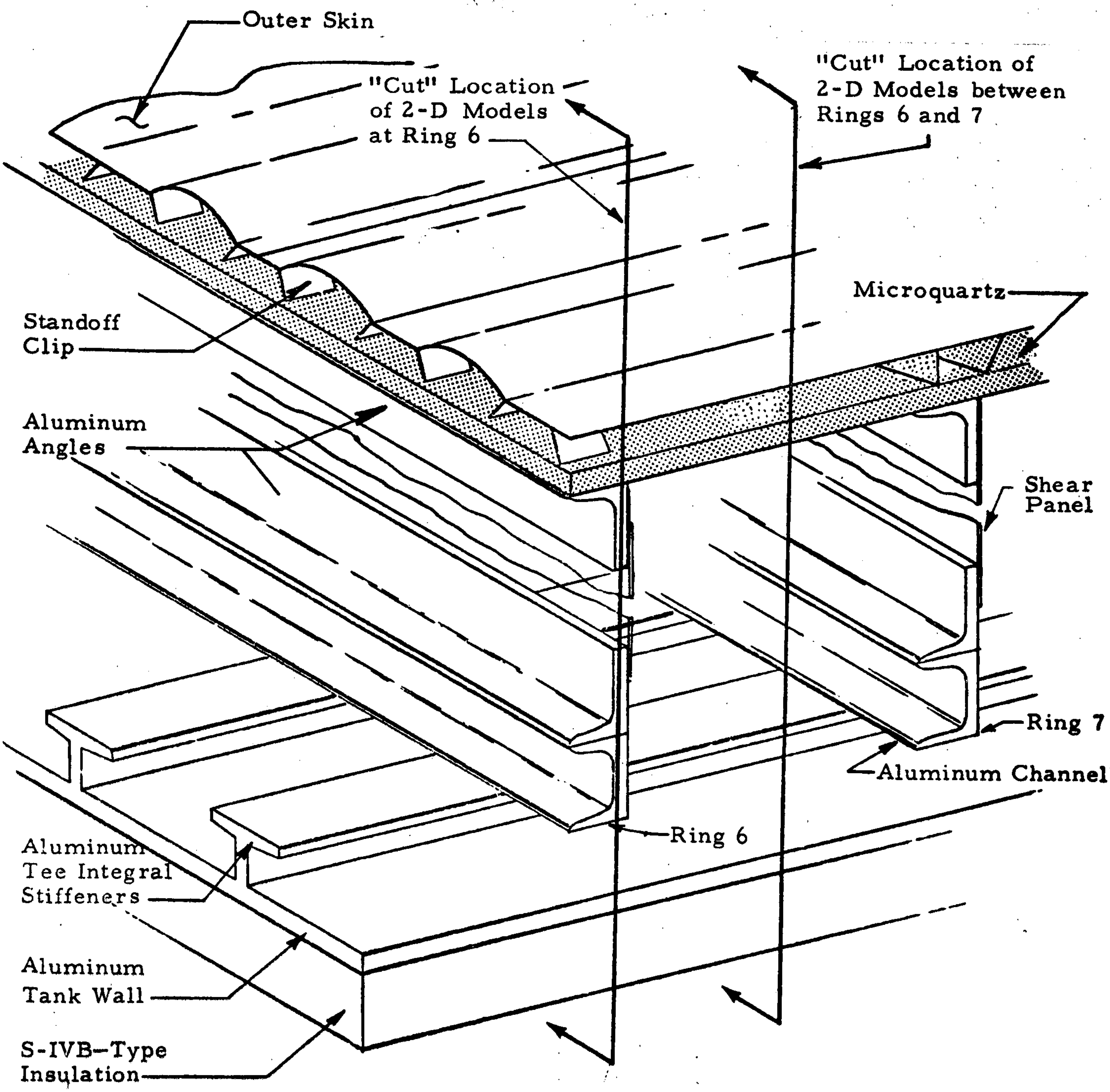


Fig. 15 - "Cut" Location for Two-Dimensional Model

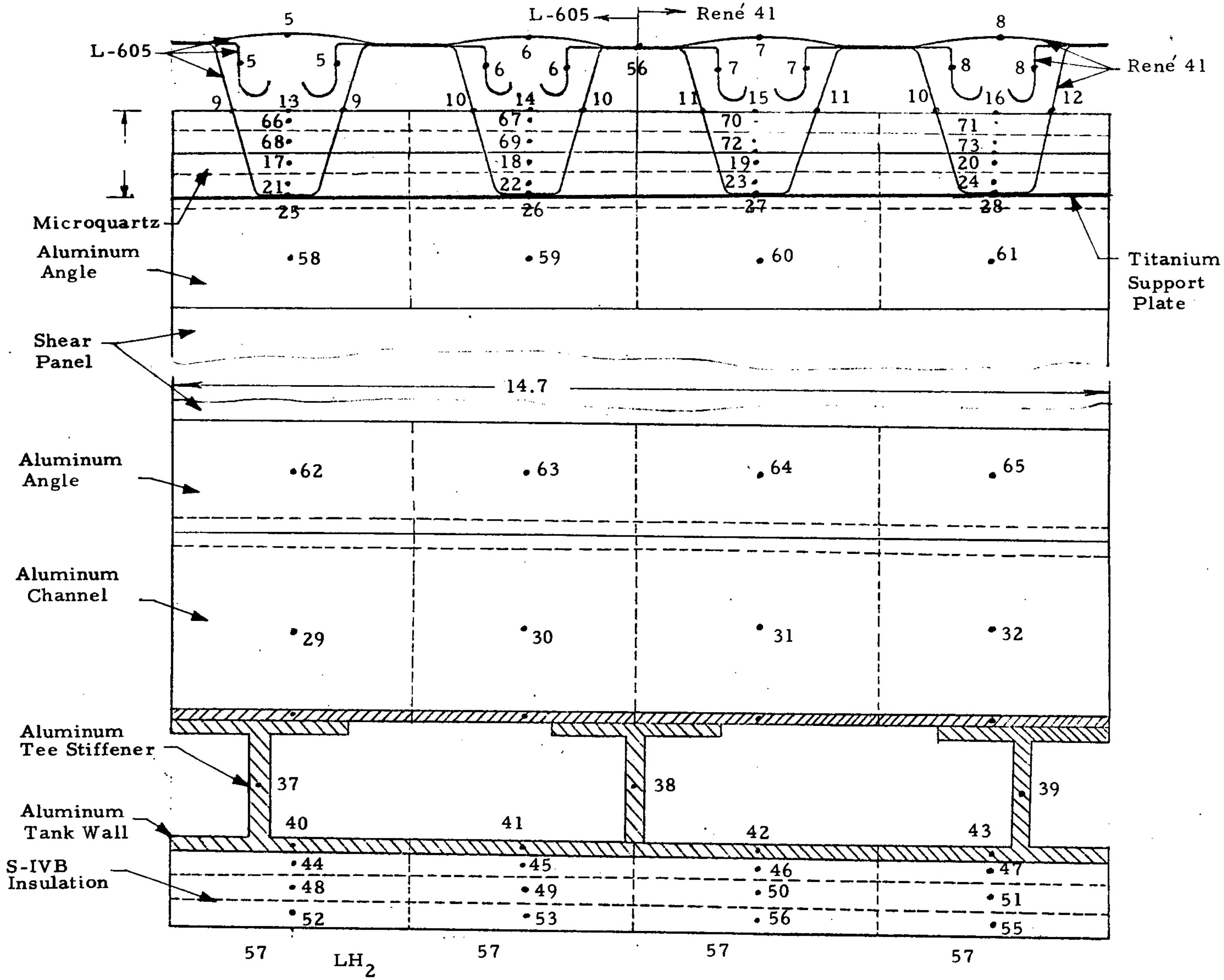


Fig. 16 - Typical Prototype I Two-Dimensional Model at Region 3-SP, Showing Node Locations (from Ref. 20)

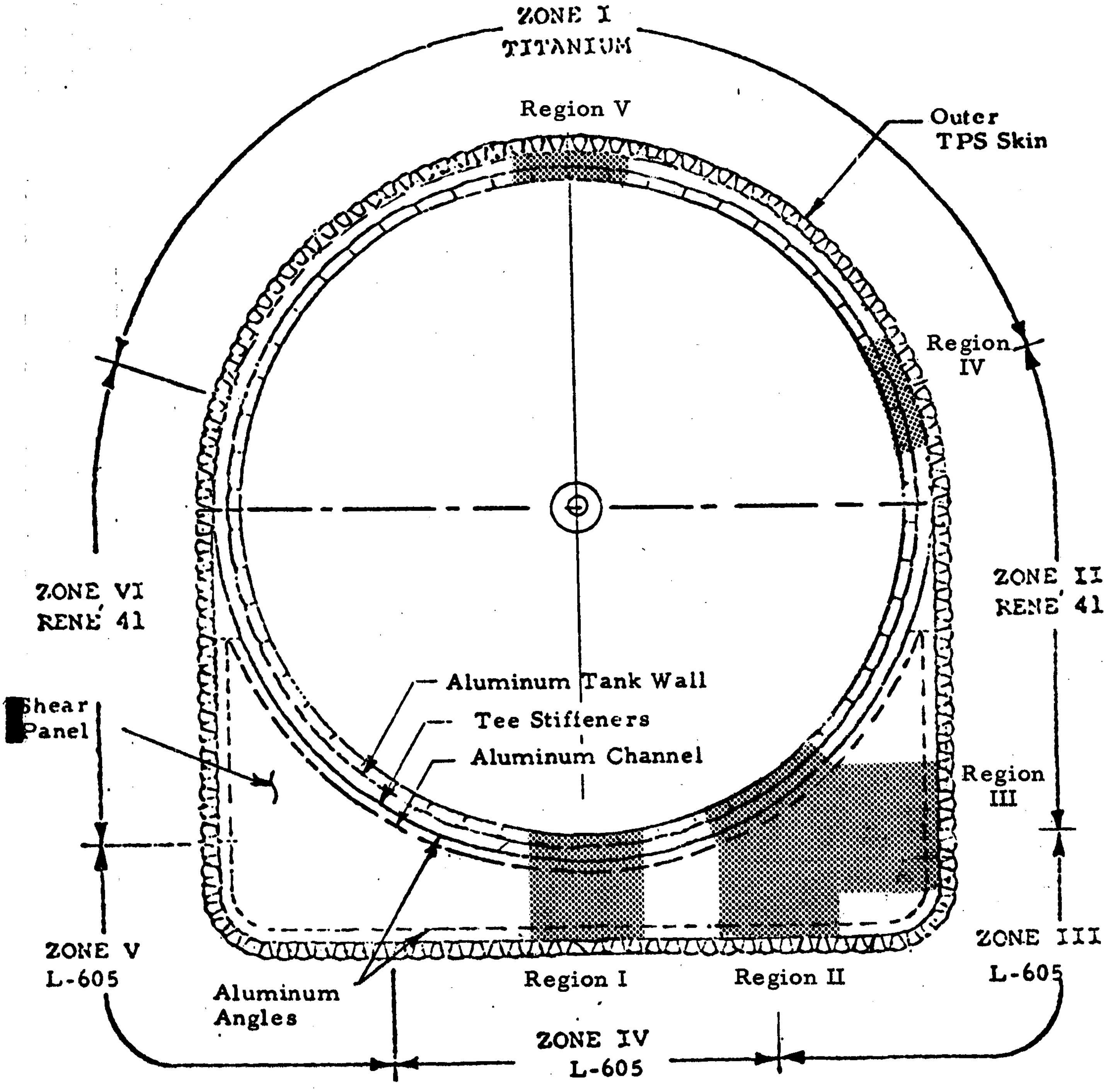


Fig. 17 - Cross-Sectional View of Prototype I, Test Article 1 Between Rings 3 and 4 Showing Location of Regions Where 3-D Models Were Developed

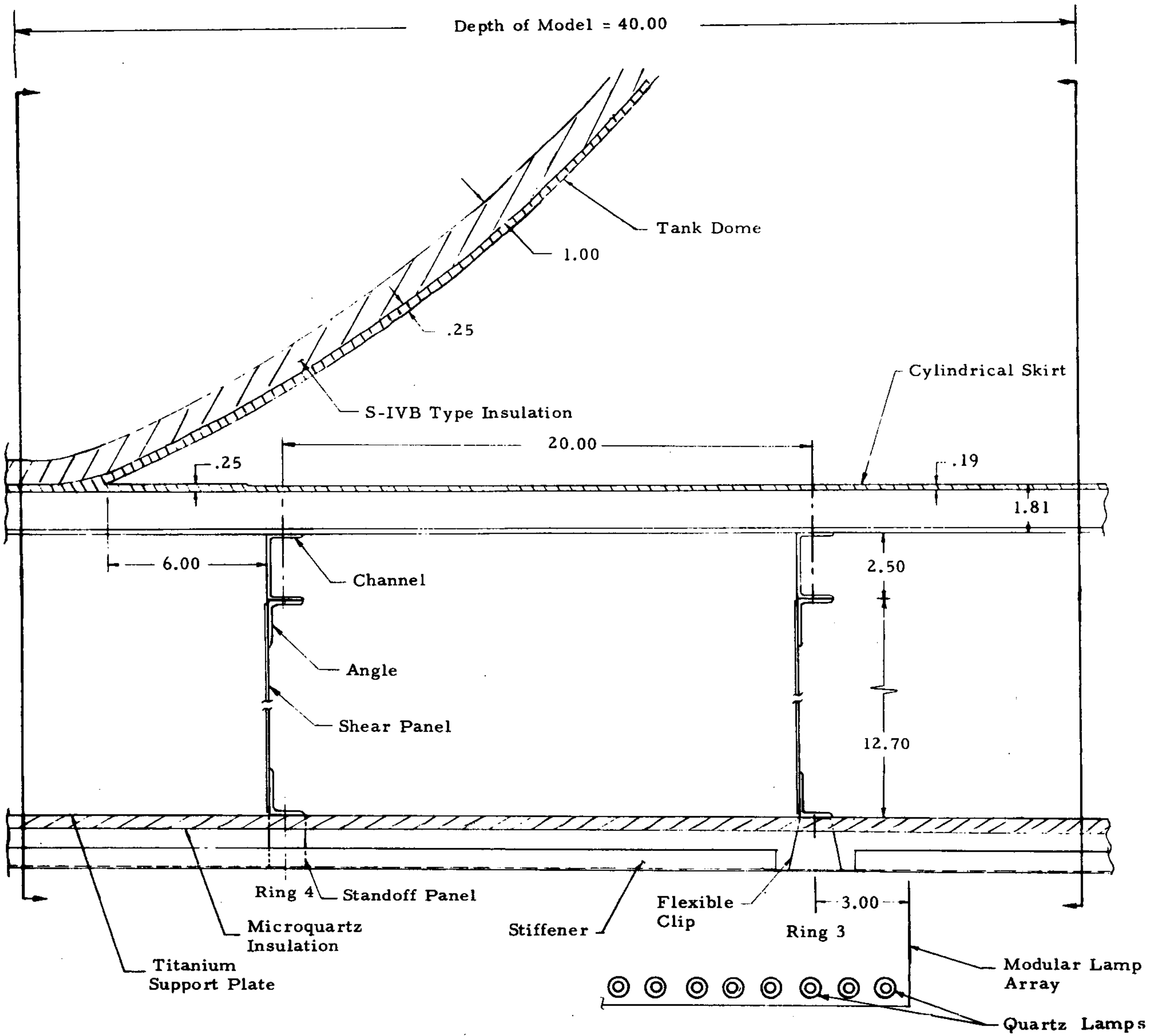


Fig. 18 - Typical Longitudinal Section Through Region I Showing Dome/Skirt Configuration and Modular Lamp Array Position

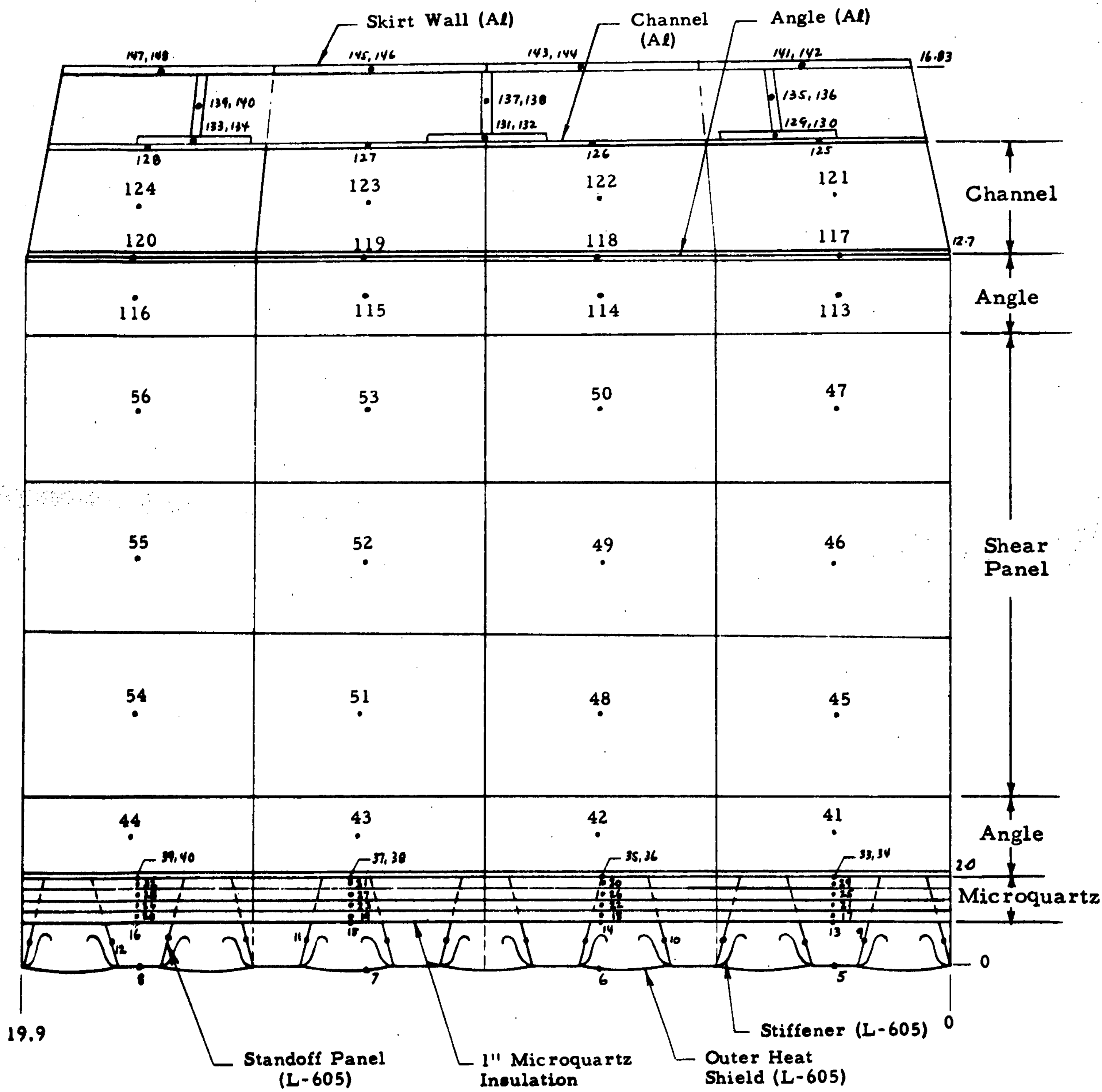


Fig. 19 - Typical Nodal Breakdown of Thermal Model at Ring 4, Region I, (from Ref. 26)

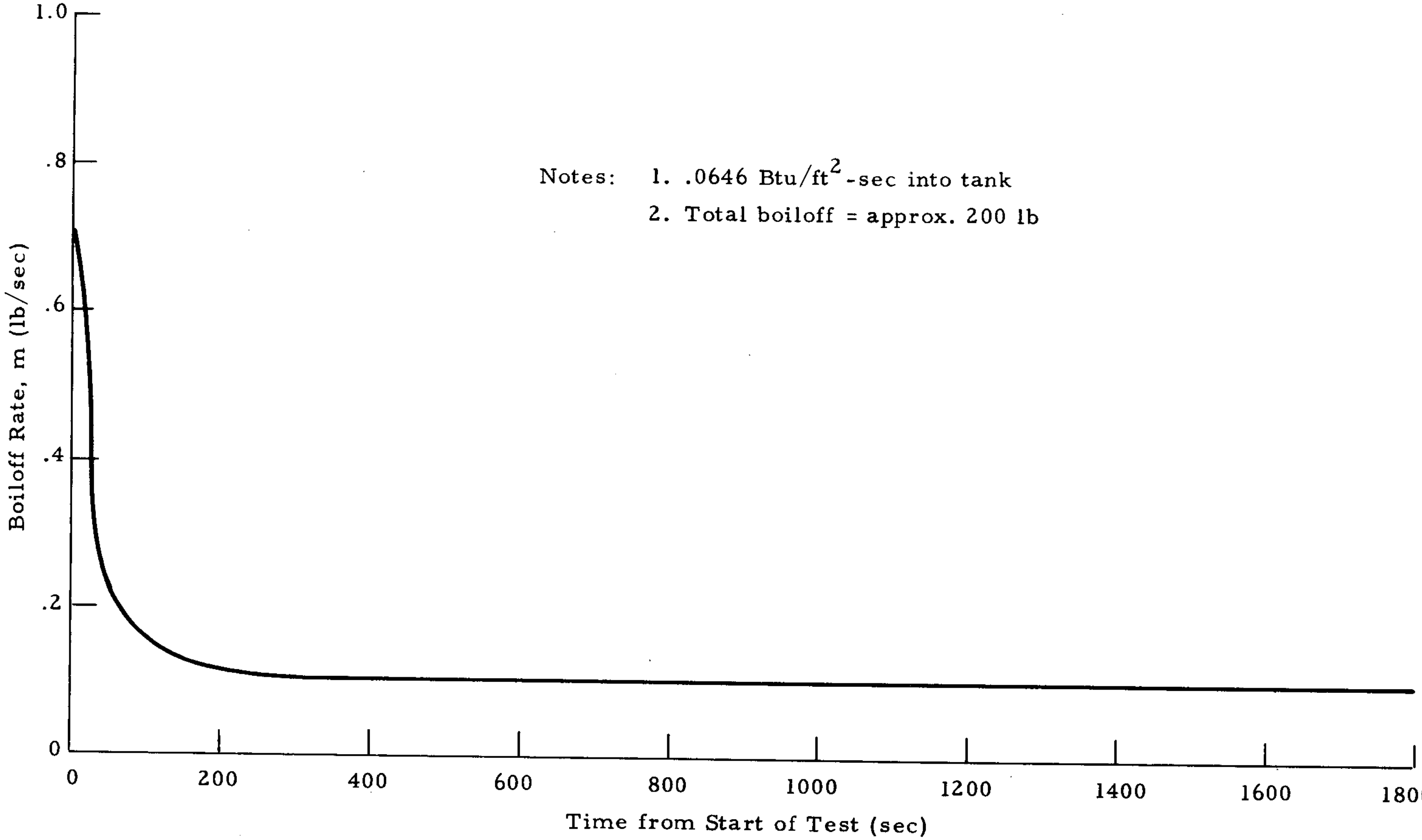


Fig. 20 - Liquid Hydrogen Boiloff Rate vs Time for Prototype I Test Article 1 Tank

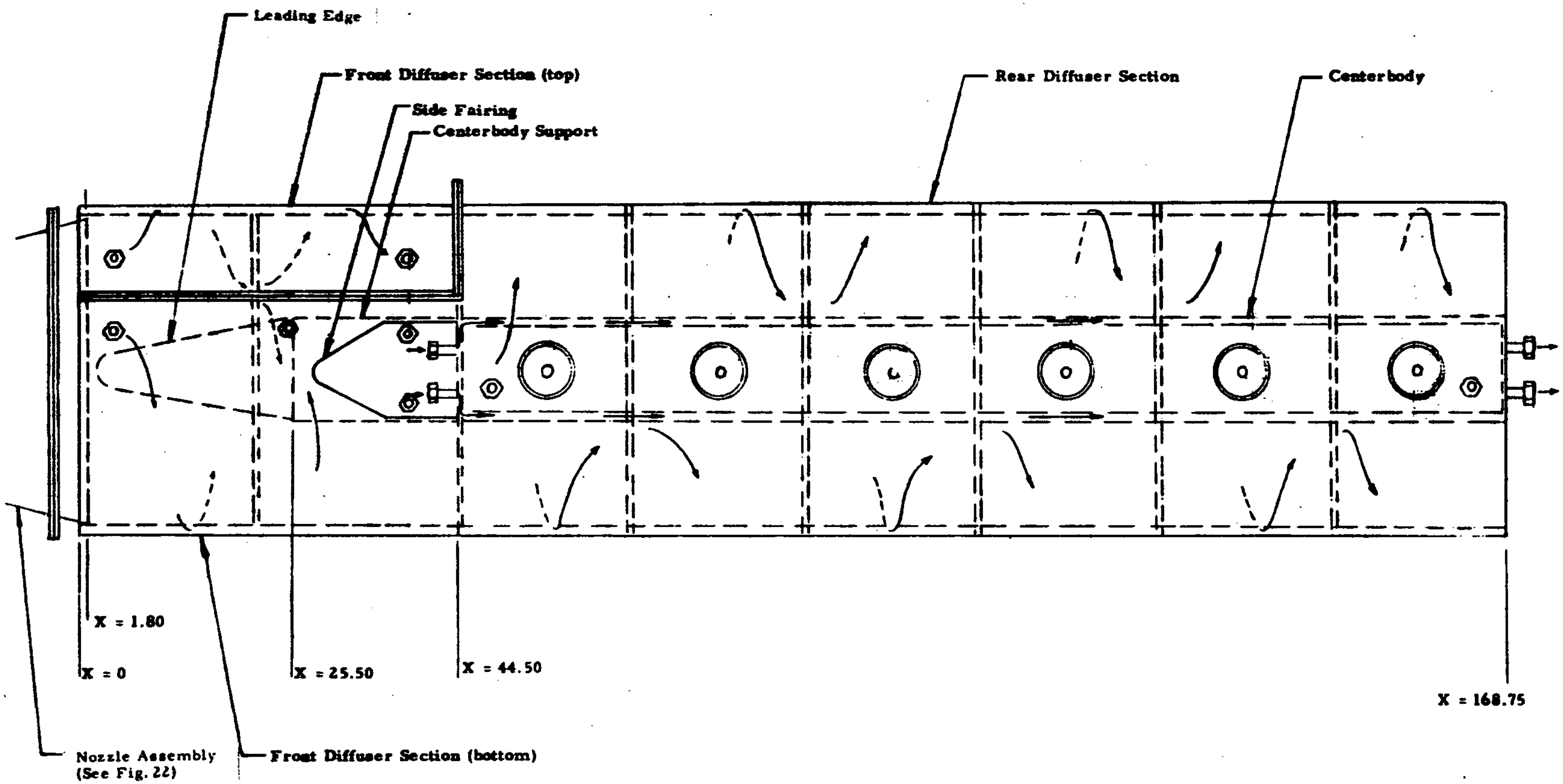


Fig. 21 - NASA-MSFC Hot Gas Test Facility Front and Rear Diffuser and Centerbody Assembly Showing Water Flow

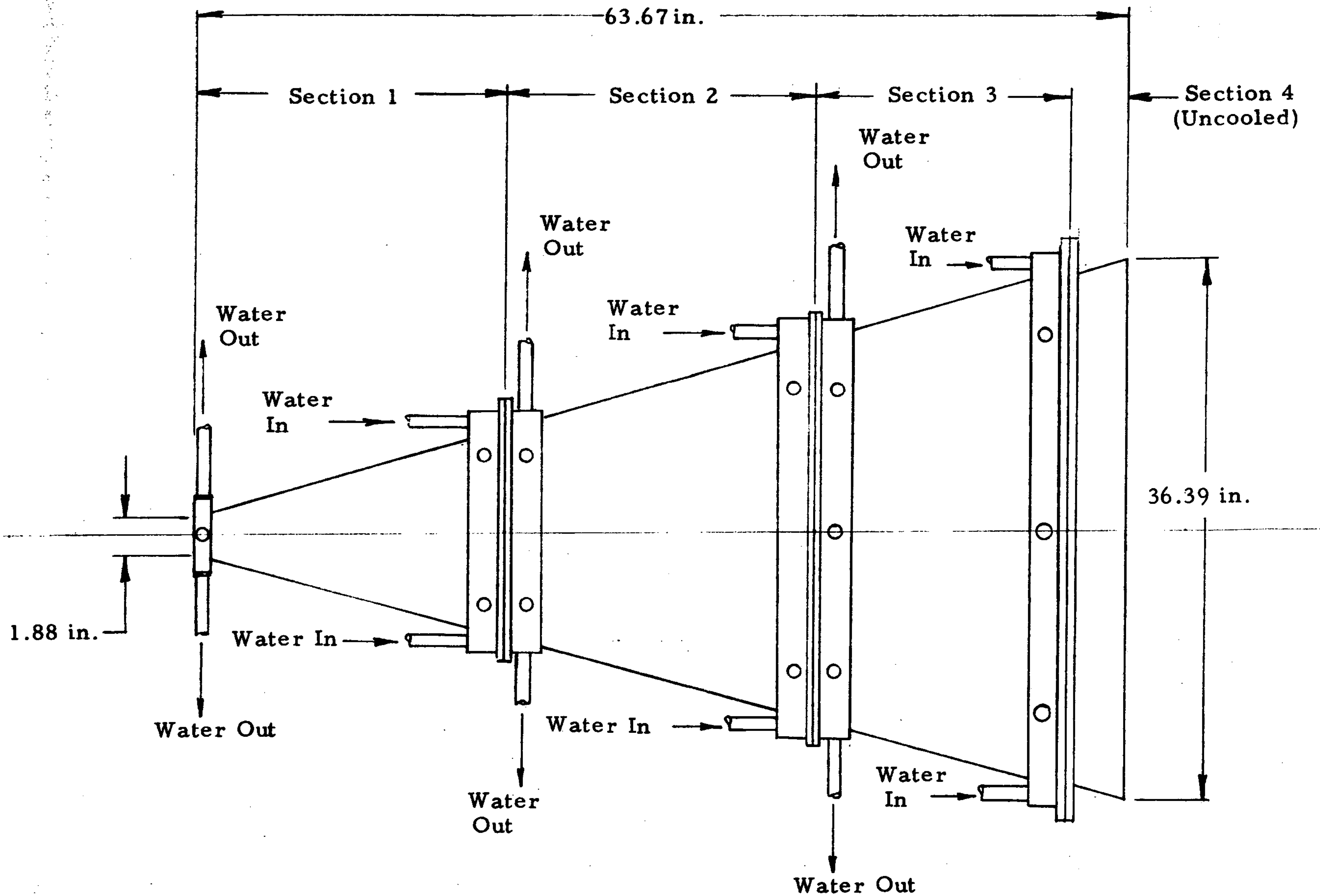


Fig. 22 - NASA-MSFC Hot Gas Test Facility Nozzle Assembly

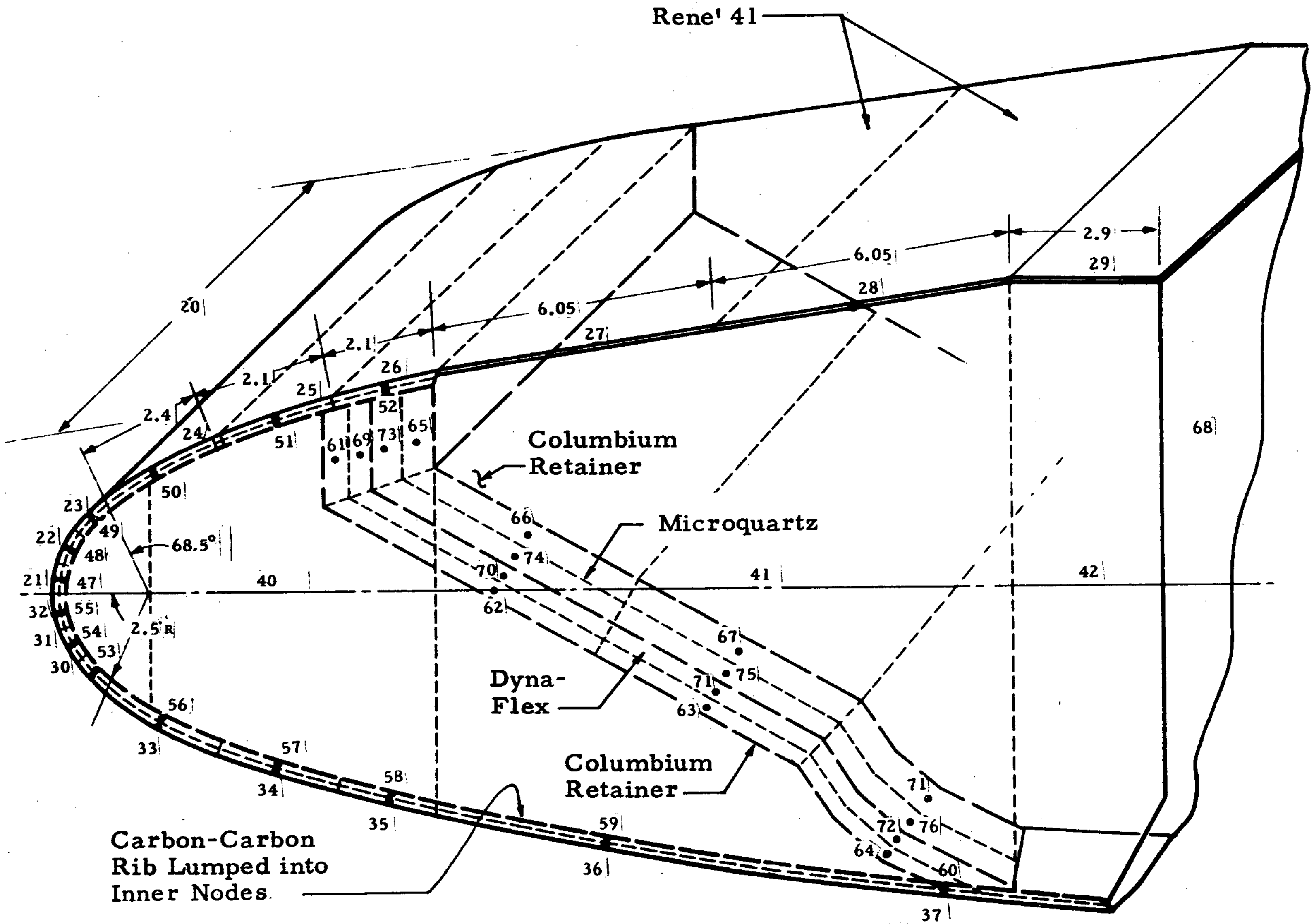


Fig. 23 - Nodal Breakdown of the Carbon/Carbon Leading Edge Three-Dimensional Model