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SPACE SHUTTLE
THERMAL SCALE MODELING
APPLICATION STUDY

Lockheed

MISSILES & SPACE COMPANY, INC.
A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CA

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SPACE SHUTTLE
THERMAL SCALE MODELING
APPLICATION STUDY

FINAL REPORT

CONTRACT NAS 9-12991

FEBRUARY 1973

BY

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LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION

CONTENTS

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CONTENTS
(Cont'd)

FOREWORD

This report is submitted to the NASA Manned Spacecraft Center by the Lockheed Palo Alto Research Laboratory, Lockheed Missiles & Space Company, Inc. The work was performed under Contract NAS 9-12991 and was administered by the Manned Spacecraft Center, with Mr. R. A. Vogt as technical monitor.

The final report provides a technical summary of the work performed from June 30, 1972 to October 10, 1972.

The authors wish to acknowledge K. W. McGee for his efforts in developing the Thermal Mathematical Model used in the program and R. K. Wedel for his part in establishing portions of the modeling criteria.

ILLUSTRATIONS

ILLUSTRATIONS
(Cont'd.)

NOMENCLATURE

A_I^*	--	Surface area ratio for incident radiant heat rates
A_i^*	--	Surface area ratio of element i
A_n^*	--	Area ratio normal to conductive heat flow
C^*	--	Ratio of model specific heat to prototype specific heat
C_p	--	Specific heat at constant pressure
G_r^*	--	Ratio of Grashof Numbers in the model and prototype
g^*	--	Ratio of gravitational fields
h^*	--	Ratio of convective heat transfer coefficient in model to that in prototype
I^*	--	Ratio of incident radiant heat flux
K_n^*	--	Ratio of model normal thermal conductivity to prototype conductivity
K_g^*	--	Ratio of conductivity of gas in model to the gas in the prototype
L^*	--	Length ratio of model to prototype
M^*	--	Ratio of molecular weights
P^*	--	Ratio of pressures
Pr^*	--	Ratio of Prandtl Numbers
Q^*	--	Ratio of energy transfer
Re^*	--	Ratio of Reynolds Numbers
t^*	--	Ratio of insulation thickness
$\bar{t}$	--	Equivalent skin thickness
T_m	--	Model temperature
T_p	--	Prototype temperature
T_g	--	Gas temperature
V^*	--	Volume ratio, Velocity ratio

Greek Letters

β -- Orbit orientation

ϵ -- Emittance

α_s - Solar absorptance

ρ - Density

ρ^* - Ratio of densities

θ^* - Time scale ratio

μ^* - Ratio of fluid viscosities

Subscripts

L/T -- Laminar flow in model and turbulent flow in prototype

c -- Convective coefficient

sub -- Substrate

ins -- Insulation

n -- Normal direction

Superscript

* -- Indicates the ratio of properties on the scale model to properties on the prototype.

Section 1

Summary

The objective of this contractual effort was to assess the capabilities and sensitivies of a Space Shuttle Thermal Scale Model (TSM) to support the Shuttle Orbiter integrated thermal control design, define a test article and preliminary test plan, and provide cost and schedule information for final design and fabrication of the TSM. To accomplish this objective the study was divided into five major tasks as follows:

Task 1: Shuttle Vehicle Design Review and Study Plan
Task 2: Thermal Scale Model Design
Task 3: Preliminary Test Plan Development
Task 4: Shuttle Thermal Scale Model Cost and Schedule
Task 5: Program Reports and Documentation

The program was originally scheduled to be a 7-month effort beginning on June 30, 1972, and ending on January 30, 1973. On October 10, 1972, approximately 3 months and 10 days after receipt of contract, LMSC was notified that the program was being cancelled at the convenience of the Government. During the period of performance prior to termination, the activity on Task 1 had been completed and work has started on Task 2 which involved detailed design of the thermal scale model. Technical progress had included development of a coarse thermal mathematical model for the entire Shuttle Orbiter, temperature calculations for the extreme hot, cold, and worst transient orbital cases, and development of scale model design and test requirements. In addition, the level of detail for modeling various subsystems was established, several potential applications for the scale model were identified, and a list of trade studies to aid the TSM detailed design was developed. The project was on schedule and within budget at the time of termination.

The purpose of this report is to document the progress on the project and to present the results of the completed studies. Since the project was not

completed, many of the questions regarding the application and sensitivity of a TSM for use in Shuttle thermal design remain unanswered. However, several significant items, especially those involving TSM design criteria for modeling convection in the crew cabin and modeling of the conduction-radiation heat flow paths throughout the vehicle, were established.

Results of the initial studies indicate that a thermal scale model with an overall scale ratio of 1/3 would be feasible to construct and would provide valuable information during preliminary thermal design of the Shuttle and would serve as a useful tool throughout the entire thermal design and verification program. The studies also show that a 1/3 TSM could be tested in the NASA/MSC Chamber A test facility under solar simulation with rearrangement of the present side and top solar lamp bank systems. For a TSM to be of maximum benefit to the Shuttle design effort, final design and fabrication of the test article must be completed early in the program such that test results can be obtained prior to the PDR.

Included in this report as Appendix A is the Analysis and Study Plan which was generated at the conclusion of Task I, but was not released due to contract termination. Appendix B contains an outline of a design review meeting held with NASA/MSC to review the results of Task I and to present the approach to be followed in accomplishing future work on the contract. The outline contains, in summary form, the major considerations in modeling the conduction-radiation-convection fields, the level of detail for modeling various systems, preliminary test requirements, and a list of potential applications for the TSM.

Section 2

INTRODUCTION

2.1 Background and Program Objectives

The Space Shuttle concept definition of Phase B defined a number of critical thermal control problems. These problems include: crew compartment temperature control and thermal loading, external insulation bond line temperatures and rate of temperature change, payload bay heat balance and temperature control, OMS and RCS propellant tanks and RCS thruster temperature control, thermal radiator performance, landing gear minimum temperatures, elevon actuator temperature control, minimum temperature of hydraulic fluids throughout the vehicle, and payload/orbiter thermal interface effects. These and other potential thermal problems can be investigated by analytical means if a thermal mathematical model (TMM) can be developed that provides exact definition of the internal and external heat flow fields. Verification of the analytical design on a total systems basis is not possible, however, due to the massive size of the Shuttle vehicle, as compared to existing thermal vacuum test facilities. Thus, a different approach to verification of the thermal design, as compared with that employed for previous spacecraft, is necessary. This combination of circumstances leads to consideration of a small-scale thermal model.

The potential for use of small-scale thermal models during spacecraft preliminary design and in design verification studies has been under investigation for a number of years. A variety of analytical and experimental programs has demonstrated the advantages and limitations of using such models. These programs involved a wide range of complexity -- from studies of simple geometries that lend themselves to rigorous analytical verification of test results, to studies of complete spacecraft where every effort was made to include all design details of the prototype vehicle. A summary of findings from all studies completed to date leads to the conclusion that a thermal scale modeling program can be a valuable tool for assessment of the strengths and weaknesses of a thermal design approach.

With this background in mind, the present program was undertaken to determine the utility of a thermal scale model in the overall Space Shuttle thermal design and thermal verification program. Specifically, the application of a TSM to assist in the development of a thermal math model and to aid in design of critical thermal control areas for the Orbiter were to be studied. In addition, the study had the objectives of determining the utility of a TSM for thermal verification testing of the final Shuttle design and the application of a TSM for thermal testing of the Shuttle in combination with typical payloads.

2.2 Technical Content

Reference 1 presents the plan by which the Shuttle thermal scale modeling application study was to be accomplished. The program plan document (Ref. 1) describes the specific tasks and subtasks and the approach for accomplishing the program objectives. In terms of the major tasks, the technical work planned for this study is outlined as follows:

Task 1: Shuttle Vehicle Design Review and Study Plan

- o Review of Phase B Shuttle Orbiter configuration
- o Establish Thermal Analysis Requirements
- o Development of prototype preliminary TMM
- o Establish preliminary TSM design requirements
- o Establish preliminary TSM test requirements
- o Define role of TSM in the Shuttle thermal design plan
- o Preparation of Study plan for use in Task 2

Task 2: Thermal Scale Model Design

- o Finalize prototype TMM
- o Thermal analysis using prototype TMM
- o Trade studies to define TSM design criteria
- o Finalize TSM design requirements
- o Establish test facility requirements
- o Develop TMM for scale model
- o Detailed design of TSM
- o Documentation of TSM design and specifications

Task 3: Preliminary Test Plans

- o Develop test requirements document
- o Establish post test analysis requirements
- o Develop requirements for correlation of test results

Task 4: Shuttle TSM Cost and Schedule

- o Definition of tasks and preliminary WBS for final design and fabrication
- o Develop preliminary task schedule
- o Establish cost estimates for final design and fabrication of Shuttle TSM

2.3 End Product

The end product planned for this contractual effort was to be a complete documentation of the application of a TSM to Space Shuttle Orbiter thermal design. Specifically, a complete set of preliminary engineering design drawings for a Shuttle Orbiter TSM was planned. Scaling factors, model materials, assessment and identification of the level of detail in the area of subsystems, assessment of applicability to the Space Shuttle Orbiter integrated thermal control design, and identification of sensitivities in terms of the model's ability for predicting prototype thermal behavior were to be specifically addressed. Details of test plans, test facility requirements, and schedules and total cost of a TSM final design and fabrication program were to be established.

Because of project termination before completion, the final objectives of the study were not realized. Work completed to date of cancellation included those items listed under Task 1. This effort included preparation of an analysis and study plan for use in accomplishing Task 2. The contract was terminated before the plan could be formally issued; therefore, in order to document all work accomplished, the study plan is included in Appendix A of this report. In addition, work had begun on Task 2 with refinement of the TMM and initiation of TSM detail design. Results of this effort are reported herein. No work had been done on Tasks 3 or 4 to date of contract termination.

Section 3

SHUTTLE PHASE B ORBITER CONFIGURATION REVIEW

3.1 Shuttle Orbiter Configuration

The proposed Shuttle configuration was reviewed in detail to familiarize personnel assigned to the program with the vehicle design and to establish the depth of thermal analysis required to support the scale model design activity. The vehicle's thermal design requirements were reviewed to identify critical thermal control areas, natural heat-flow boundaries, and areas with unique thermal response characteristics.

Per NASA and LMSC agreement, the Shuttle Orbiter concept as proposed by LMSC in the Space Shuttle Technical Proposal, Volume III, LMSC/D157364, May 12, 1972, was used as a basis for the thermal modeling application study.

Use of the LMSC Shuttle concept was selected because of the immediate availability of design information to LMSC personnel and due to the compressed schedule required for the TSM program. Taking this approach would not impede achieving the primary objective of the study which was to prove feasibility and utility of a TSM to support the integrated Shuttle thermal design/verification program. Major differences in critical thermal control areas between the LMSC concept and the final Shuttle design were to be considered during this study to assess their impact on the model's final configuration and test performance. NASA agreed to keep LMSC informed of major differences which should be considered during the modeling program.

The Shuttle concept, shown in Figs. 3.1 and 3.2, shows the major sections and systems of the vehicle. As shown in Fig. 3.1, the vehicle structure can be divided into six major sections. These include the forward fuselage, center fuselage, aft fuselage, wings, payload bay doors, and the fin and rudder

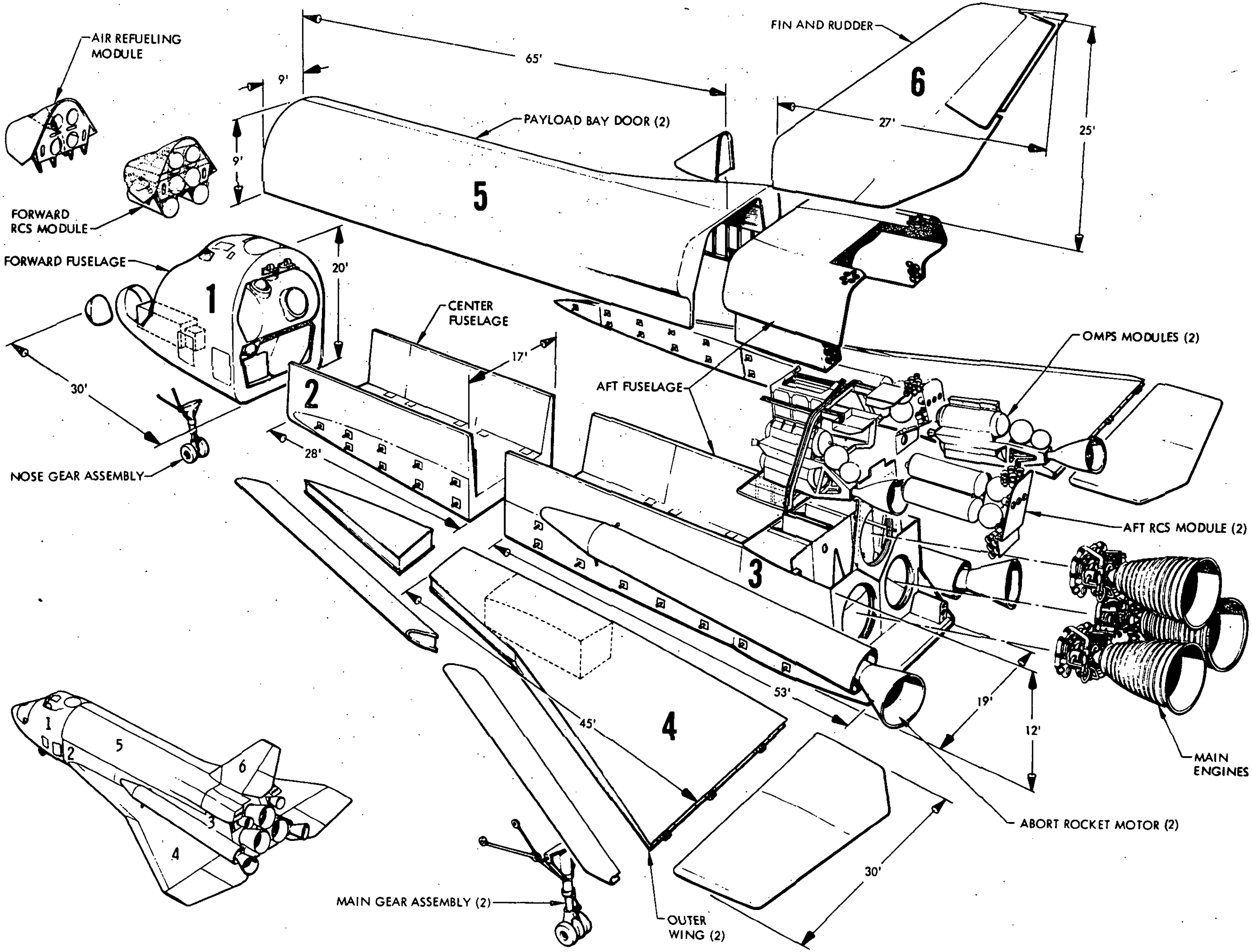


Figure 3-1. Air Transportable Orbiter Vehicle Breakdown Arrangement Provides Flexibility for Manufacture and Test Operations

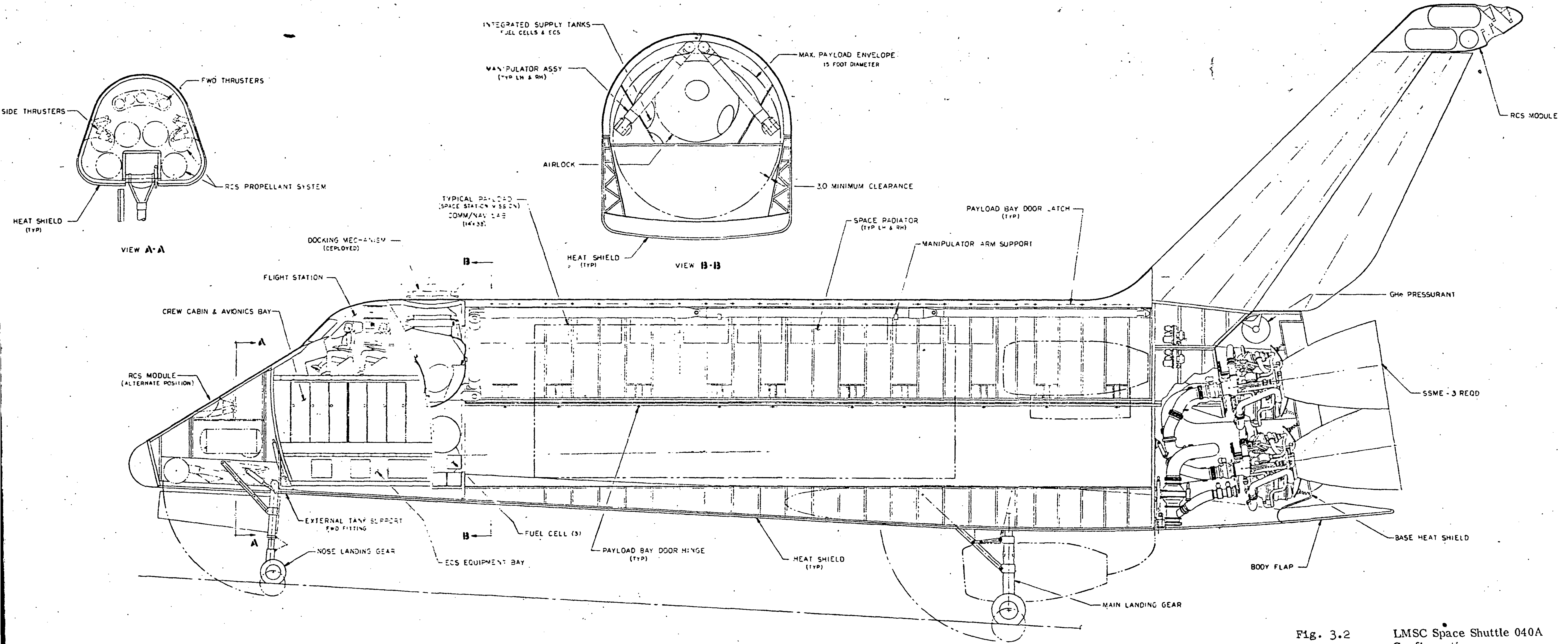


Fig. 3.2 LMSC Space Shuttle 040A Configuration

sections. Fig. 3.2 shows the location of major equipment and systems within the vehicle. The overall prototype Orbiter dimensions as given in Fig. 3.3 are 119.4 ft. long by 78.34 ft. wide by 52 ft. high.

The LMSC concept proposed the use of a honeycomb skin structure. However, for purposes of the TSM study a solid aluminum skin structure was assumed per NASA request, with equivalent $\bar{t}$ (average thicknesses) for this structure provided by NASA. The equivalent $\bar{t}$ values used in the analysis are shown in Fig. 3.4 with updated values given in Fig. 3.5. The $\bar{t}$ values include the skin and skin stiffeners. Material for the prototype skin structure was assumed to be 2024-T4. Other major structural materials included 6Al-4V Titanium for the landing gear and 4340 steel for the elevon and fin actuators.

LI-1500 with an infrared emittance of 0.84 and a solar absorptance of 0.80 was assumed for the external thermal protection system (TPS). Further details on this and other thermal control components are described in Section 4.

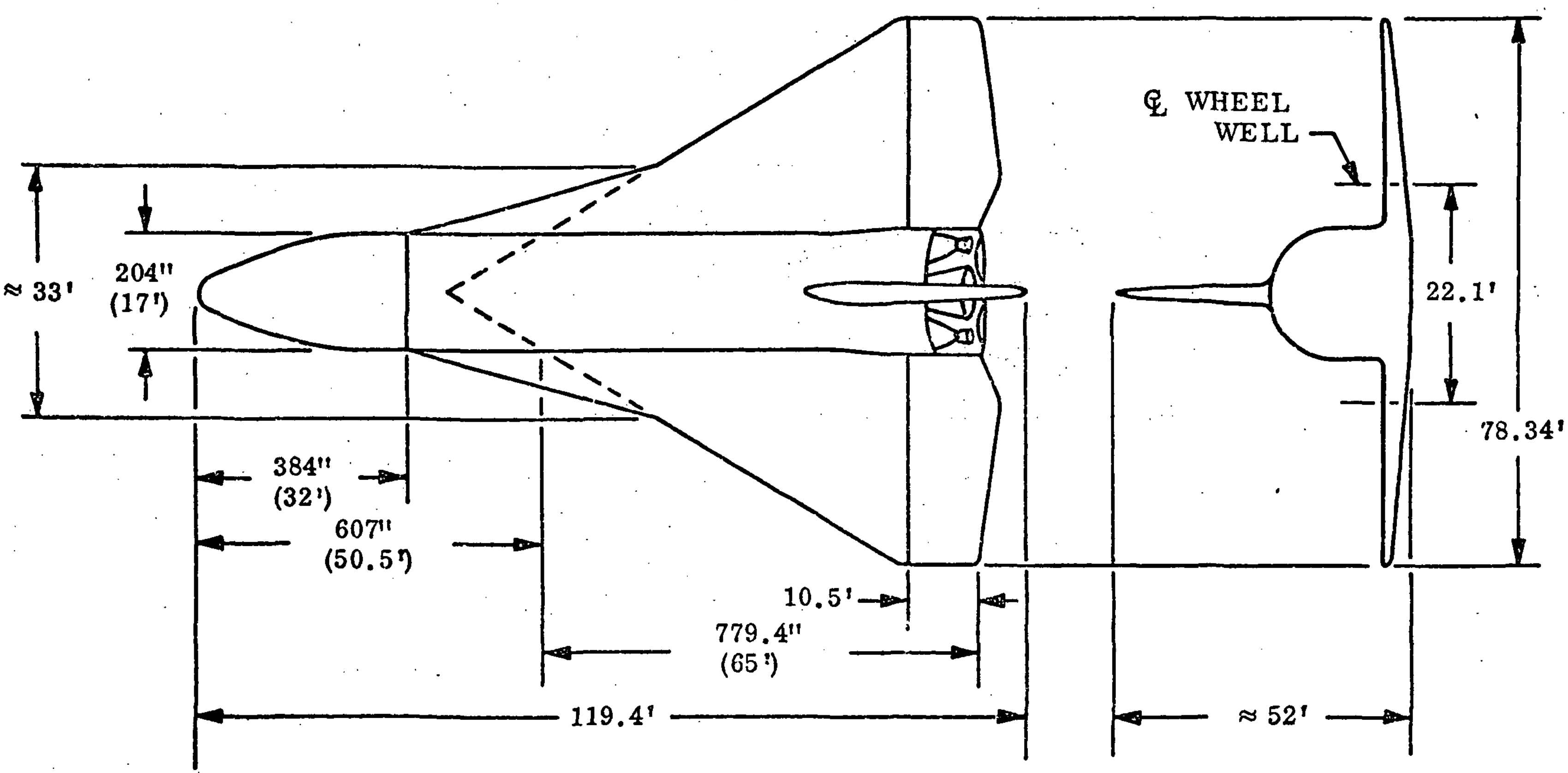
3.2 Critical Thermal Control Areas

Various critical thermal control areas for the Shuttle Orbiter were identified during the Task 1 studies. These areas are listed along with their temperature requirements in Section 4 of Appendix A.

3.3 Level of Detail for Modeling Systems

During Task 1 an assessment was made regarding the level of detail required for modeling various systems and subsystems within the Shuttle Orbiter. This level of detail is presented in Section 8.3 of Appendix A.

Fig. 3-3. SHUTTLE OVERALL DIMENSIONS (LMSC CONCEPT)



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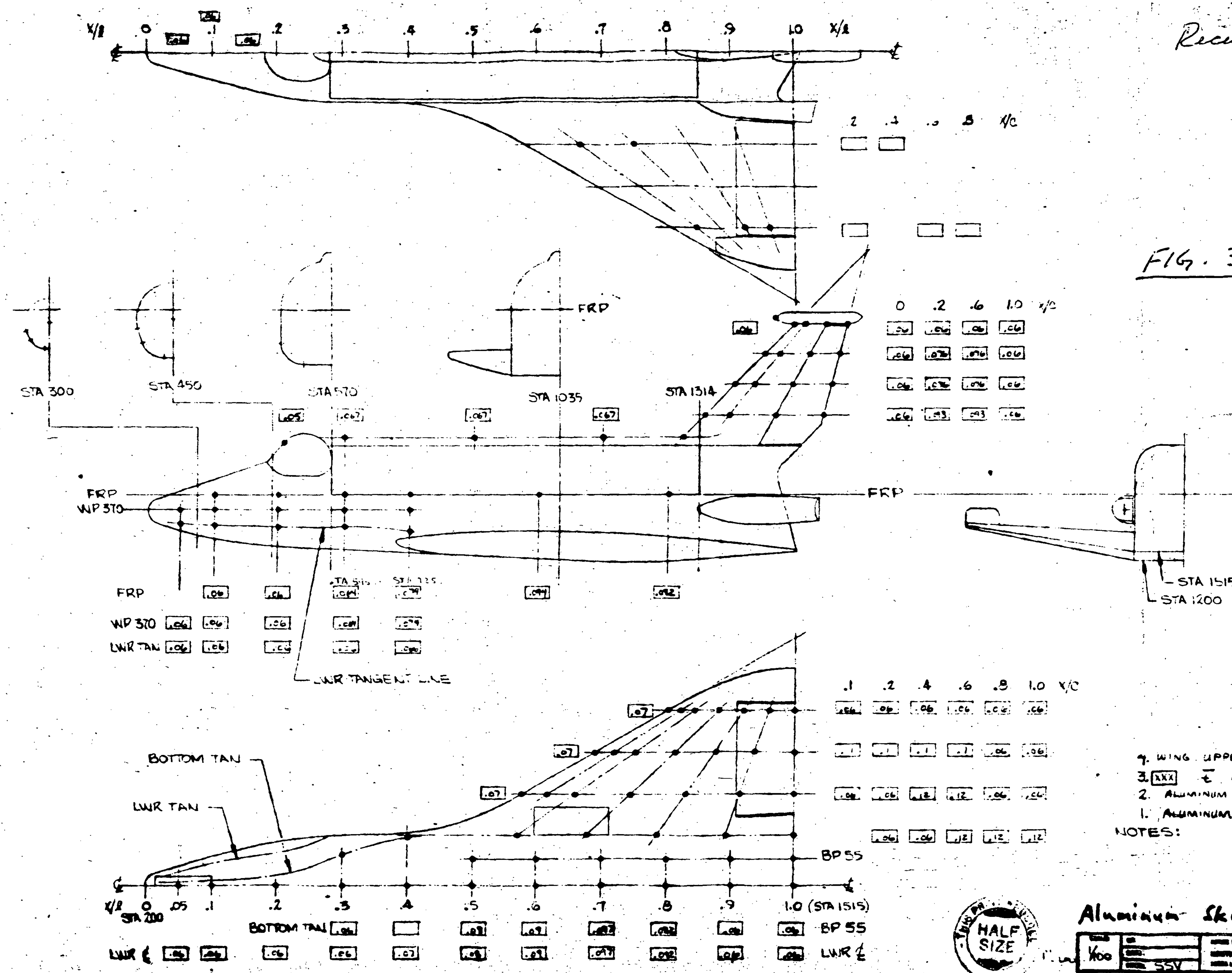

FIG. 3-4 ORIGINAL EQUIVALENT $\bar{t}$ VALUES

NOTES:
1. ALUMINUM PRIMARY STRUCTURE
2. ALUMINUM TEMP 350°F MAX
3. $\boxed{XXX}$ $\bar{t}$ ALUMINUM STRUCTURE — INCHES
4. WING UPPER AND LOWER SURFACES HAVE SAME $\bar{t}$'s

HALF SIZE

Aluminum Skin Thicknesses

1/100 SSV

NORTH AMERICAN ROCKWELL CORPORATION

~~ISOTHERM & MATERIAL REGIONS MAP~~

Bob Vogt 11-3-71

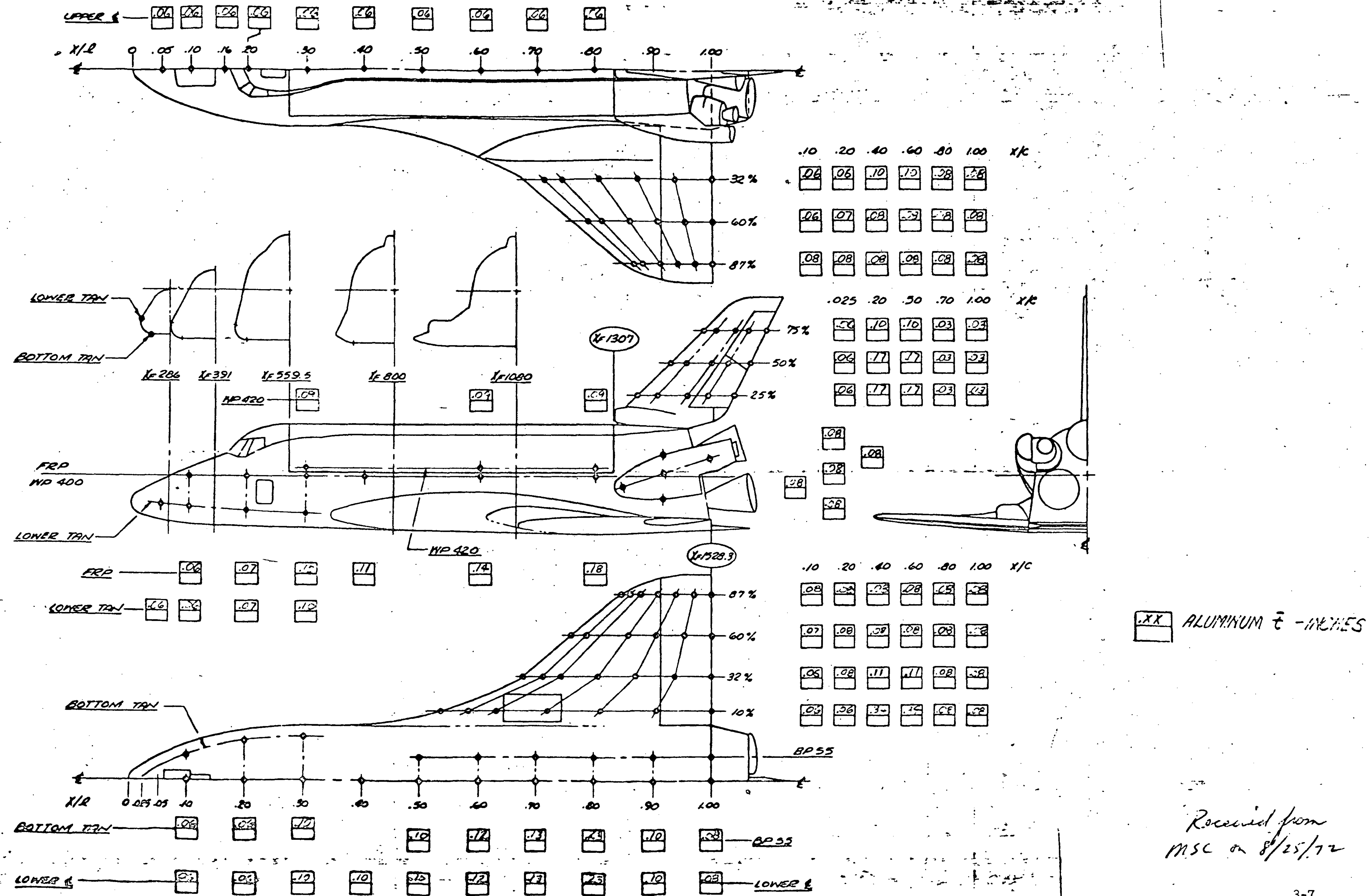

FIG. 3-5 REVISED EQUIVALENT $\bar{t}$ VALUES

Received from MSC on 8/25/72

Section 4

THERMAL ANALYSIS

4.1 Thermal Analysis Requirements

During the initial phase of the program, a plan was developed for the analytical studies to determine the required complexity of the TMM in terms of the support it would provide to: (1) assessment of the feasibility and usefulness of a thermal scale model, (2) performance of trade studies, (3) design of the thermal scale model, and (4) definition of test facilities and source simulation requirements. The requirements for developing the TMM included definition of the nodal network for all internal and external sections of the vehicle and definition of the orbital cases to study for determination of external heat rates. Requirements were also defined for including a typical payload configuration in the TMM.

A thermal analyzer model is considered as one of the basic tools in design of a TSM. Development of the preliminary TMM was initiated as soon as possible after contract go-ahead and was essentially completed during Task 1. Early development of the TMM was considered necessary in order to complete a satisfactory design for the scale model within the specified time schedule.

In developing the TMM, the approach followed was to construct a math model only in sufficient detail to support this study contract. Thus, a coarse nodal network was constructed for purposes of defining important heat flow paths and temperature fields throughout the vehicle. A detailed TMM, typical of that required for vehicle thermal design, was not required for this study. The TMM was set up to include the three dimensional conduction, radiation, and convective networks for each major area of the Orbiter. Provisions were made for incorporating a typical payload configuration in the TMM. Temperature performance of the vehicle was obtained for the extreme hot, cold, and worst transient orbital cases as described in Section 4.4.

To aid in TSM design and to provide a means of comparing scale model results with prototype results, a requirement for developing a TMM of the thermal scale model was established. Computer analysis were planned as the TSM design progressed to support decisions regarding any necessary deviations from exact thermal scale modeling.

4.2 Thermal Math Model Development

Development of the TMM was accomplished in two stages. The first stage consisted of a preliminary TMM developed during Task 1, and included 450 internal and external thermal nodes connected by approximately 700 conduction resistors and 1500 radiation resistors. The second stage incorporated a typical payload configuration and various design refinements that were defined as a result of Task 1 studies. This math model consisted of 550 thermal nodes connected by approximately 800 conduction resistors and 1100 radiation resistors. The revised TMM was about 90% complete when the program was terminated.

Figures 4-1 through 4-8 show the nodal breakdown for various sections of the vehicle. The TMM is symmetrical about a plane passing vertically through the logitudinal centerline of the vehicle. All nodes on the pilot's left are odd numbered, while corresponding nodes on the right are even numbered. The right side value of each corresponding node is one unit greater than those on the left side. Figures 4-9 and 4-10 give the surface and underlying structure node numbers, the node area, and the average aluminum skin thickness assumed.

4.2.1 Basic Assumptions

Various assumptions were made during construction of the TMM to simplify the model and speed its development. These assumptions were divided into two general categories:

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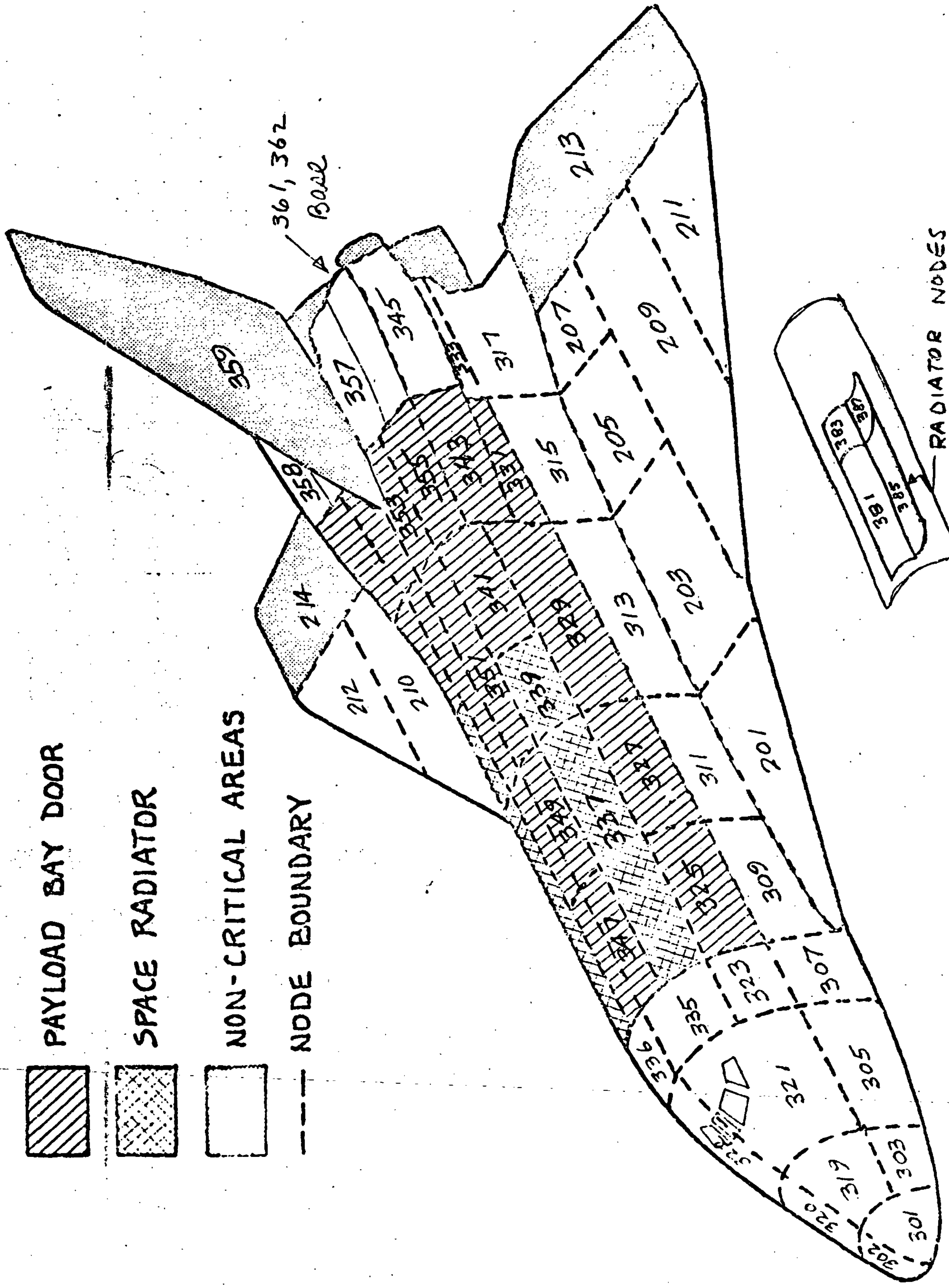

FIGURE 4.1 TAM EXTERNAL SURFACE NODES (PRELIMINARY)

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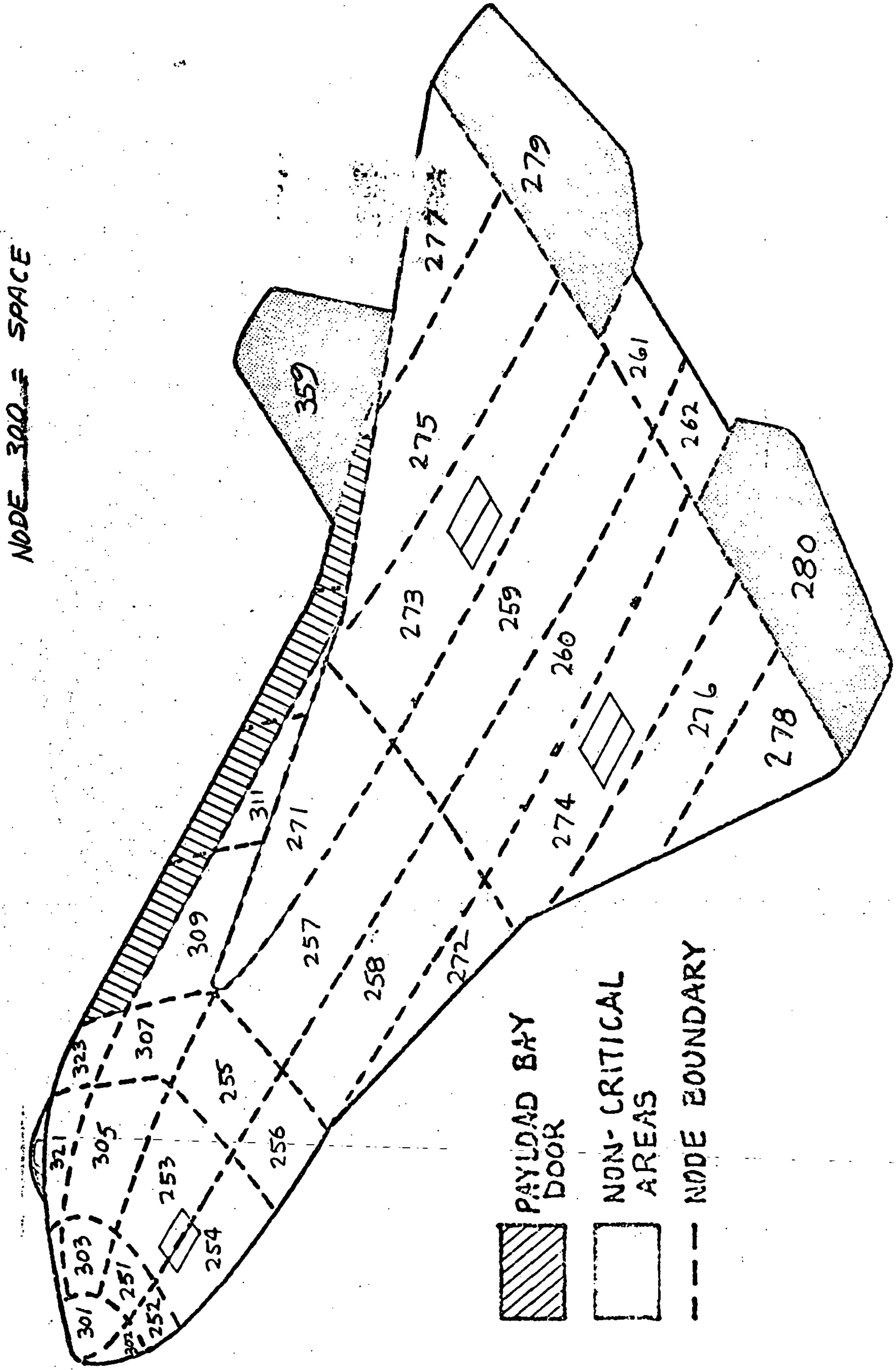

FIGURE 4.2. TMM EXTERNAL SURFACE NODES (PRELIMINARY)

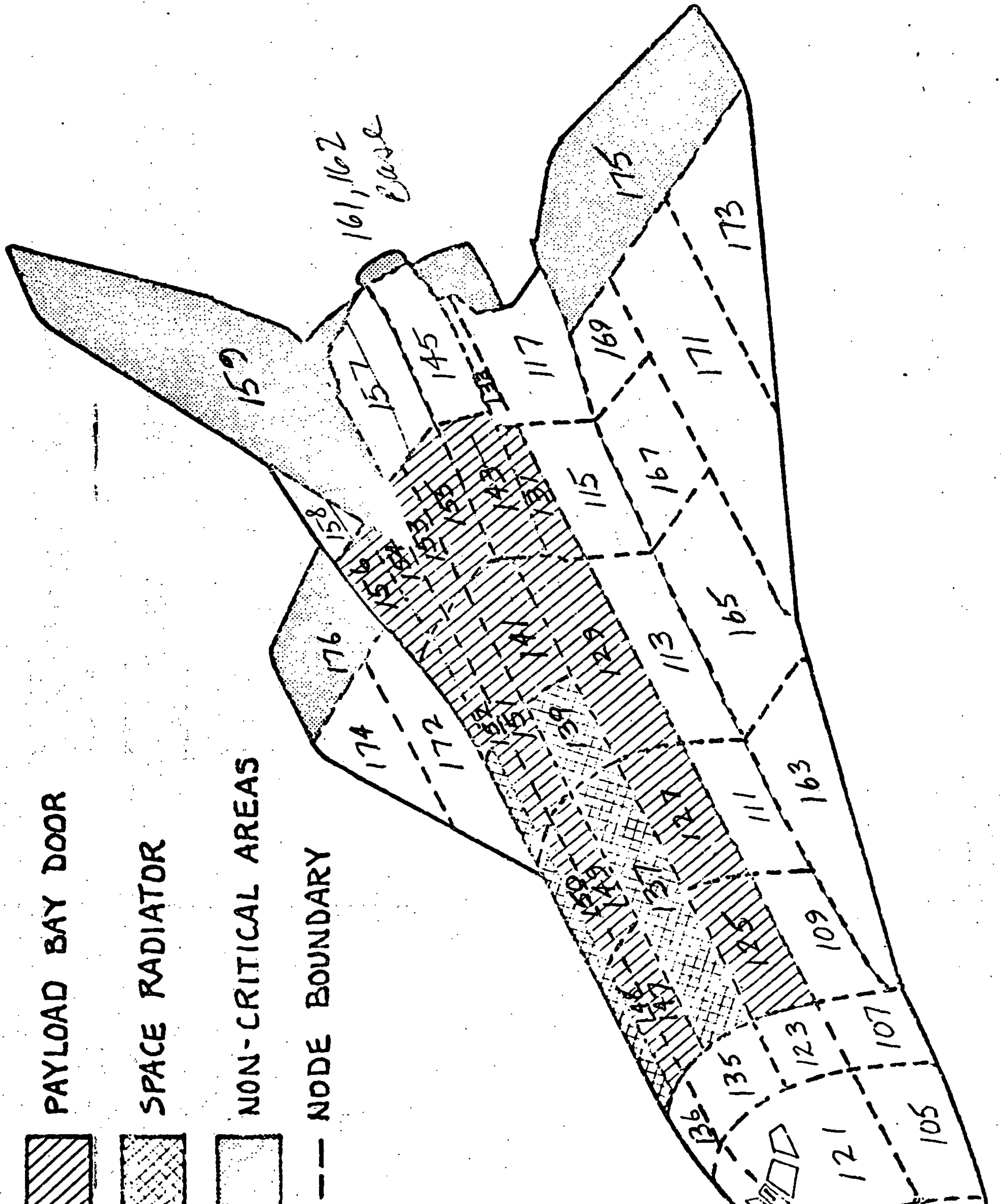

FIGURE 4.3. TMM INTERNAL LI-1500 NODES (PRELIMINARY)

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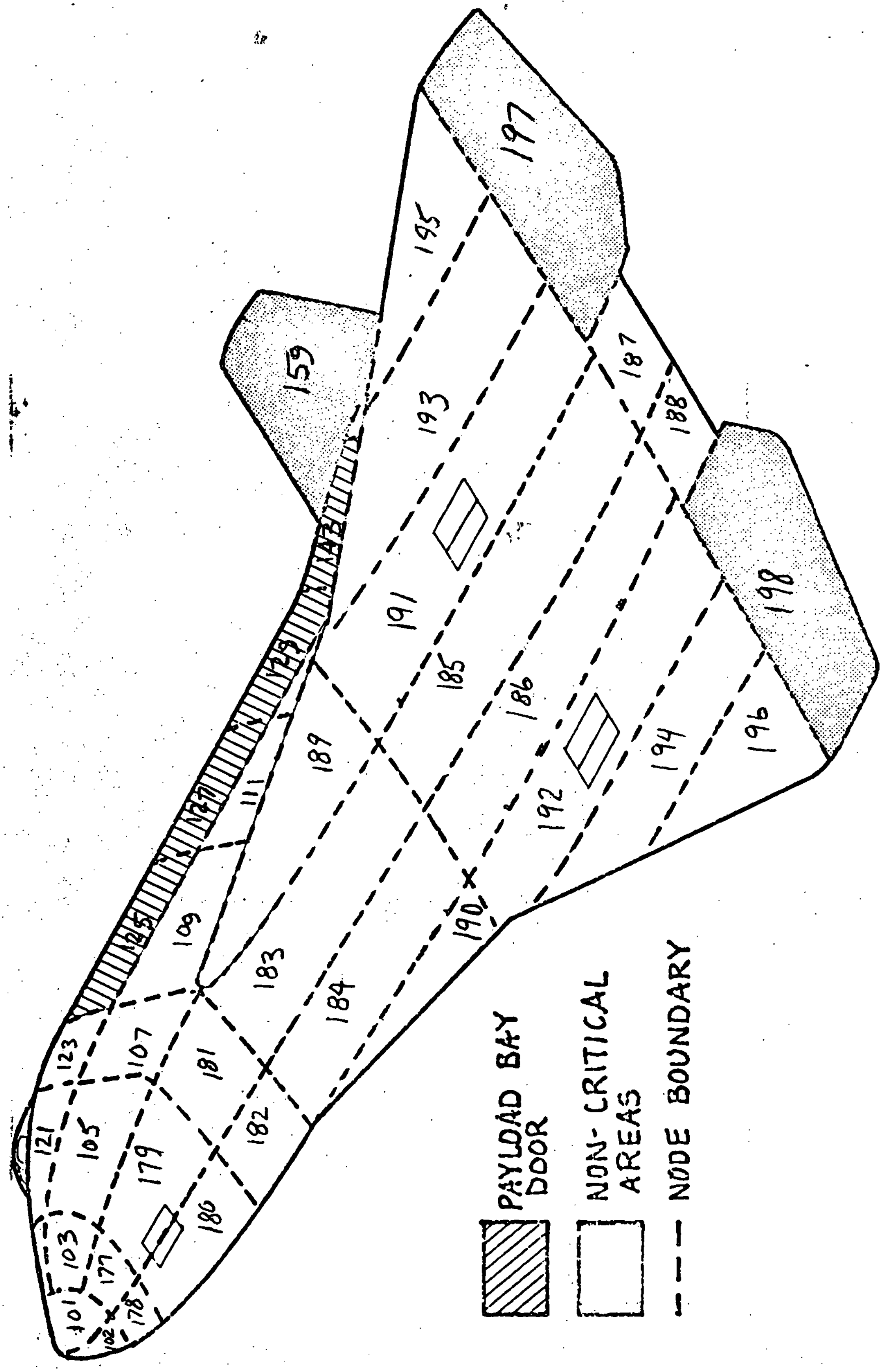

FIGURE 4.4 TMM INTERNAL LI-1500 NODES (PRELIMINARY)

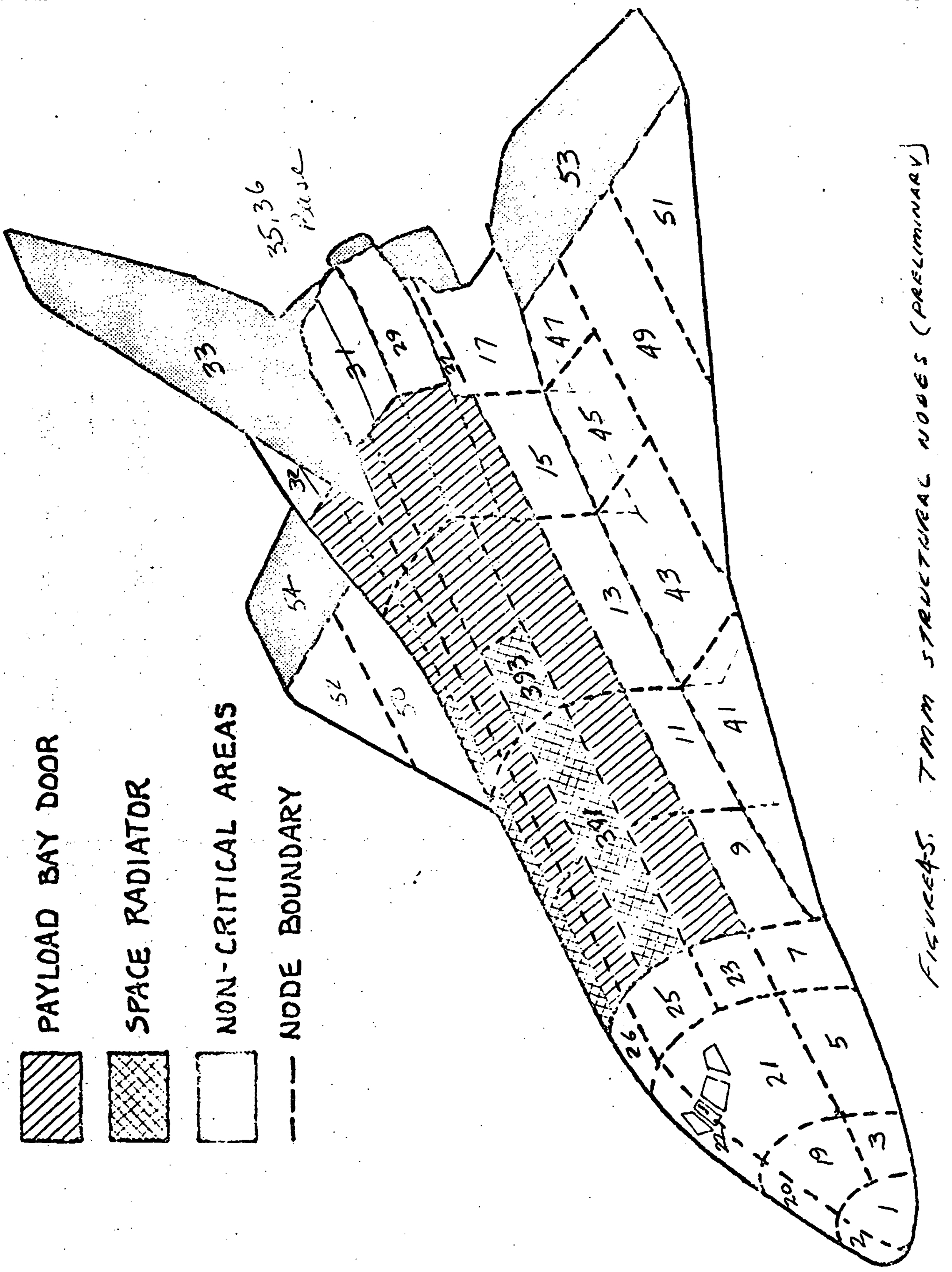

FIGURE 45. TMM STRUCTURAL NODES (PRELIMINARY)

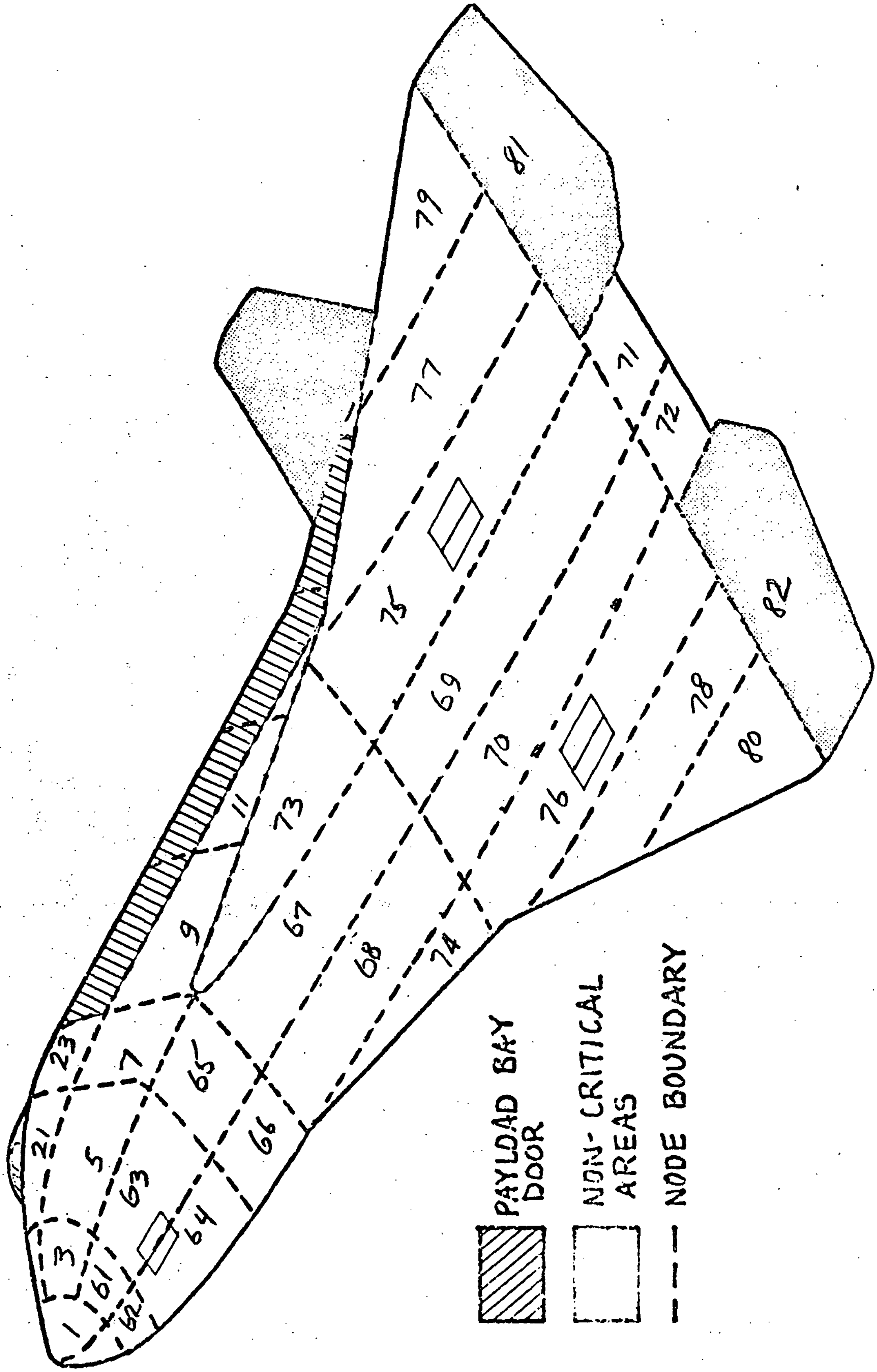

FIGURE 46. TMM STRUCTURAL NODES (PRELIMINARY)

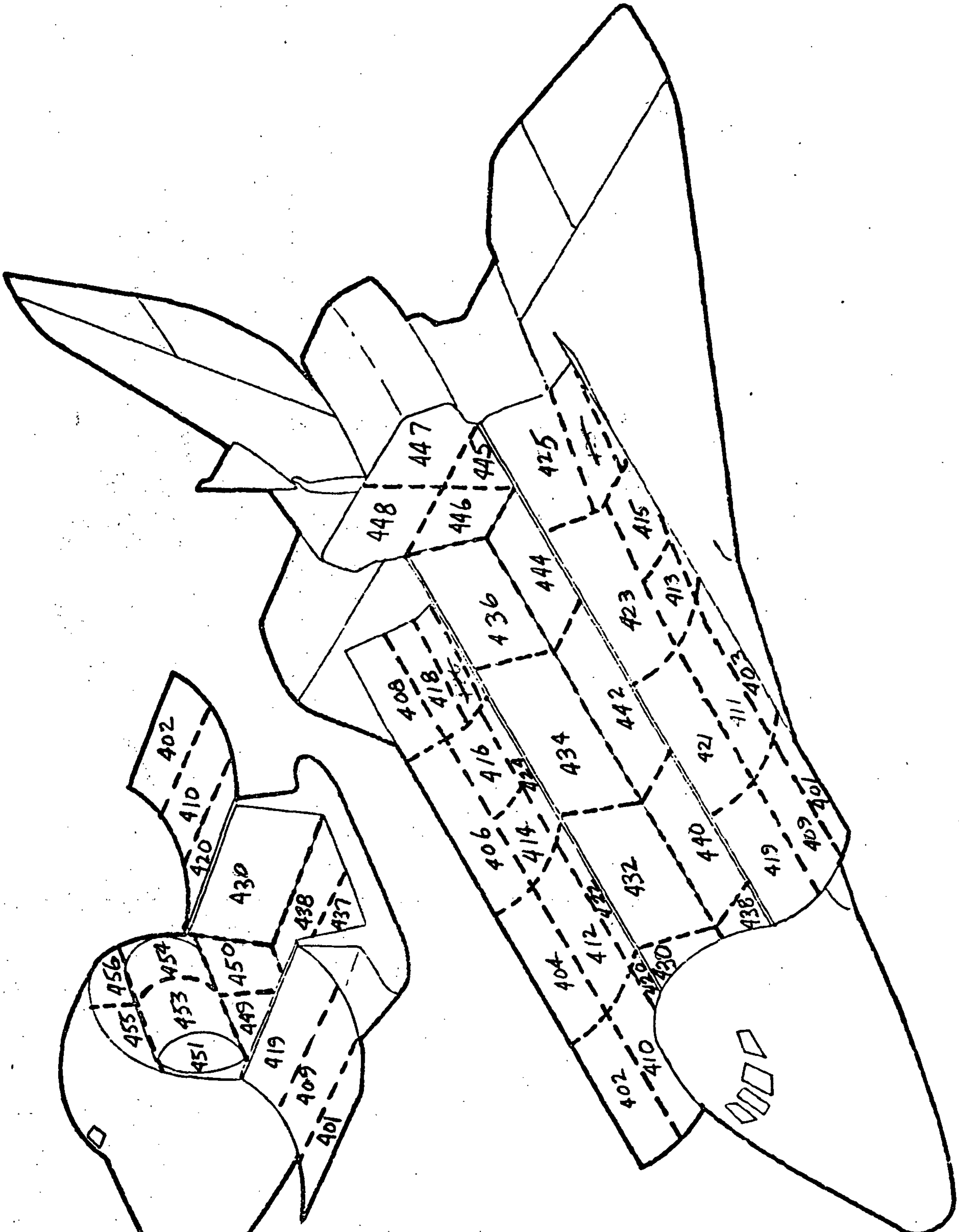

FIGURE 4-7. TMM PAYLOAD BAY NODES (PRELIMINARY)

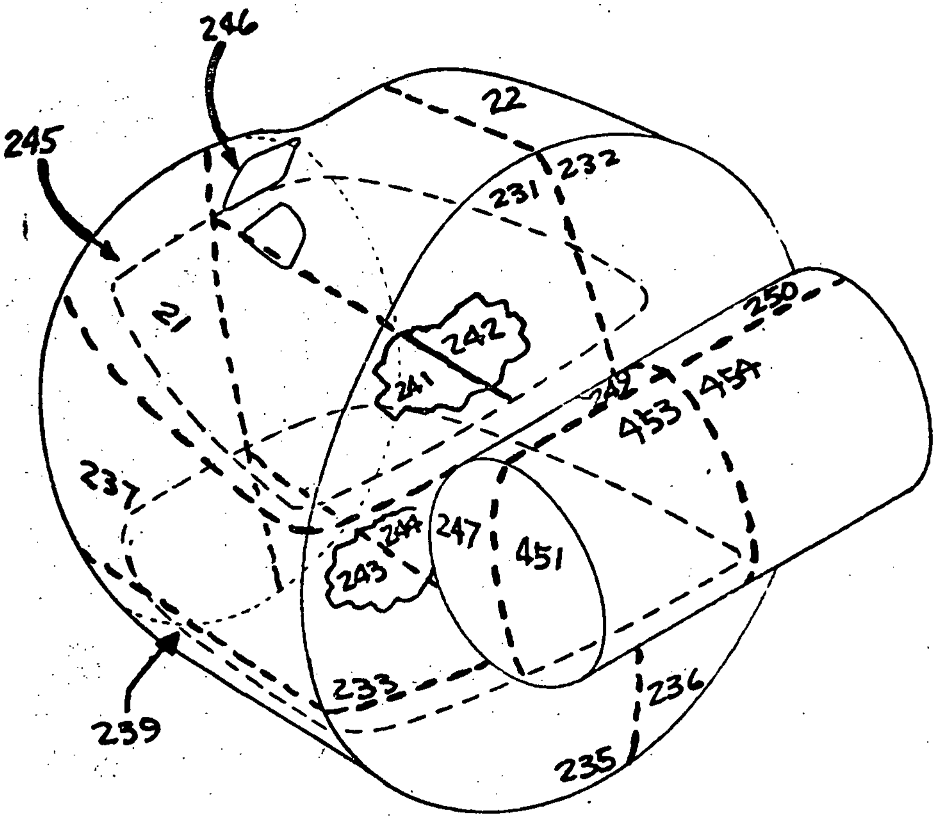

FIGURE 4-8. TMM CREW CABIN SECTION NODES (PRELIMINARY)

21, 22	TOP STRUCTURE (SIDE)
231, 232	TOP AFT BULKHEAD
233, 234	MIDDLE AFT BULKHEAD
235, 236	LOWER AFT BULKHEAD
237, 238	MIDDLE SIDE
239, 240	LOWER SIDE
241, 242	UPPER FLOOR
243, 244	LOWER FLOOR
245, 246	FWD BULKHEAD
247, 248	FWD ½ END OF PAYLOAD HANDLER STATION
249, 250	CABIN SIDE " " "
451, 452	AFT ½ END OF " " "

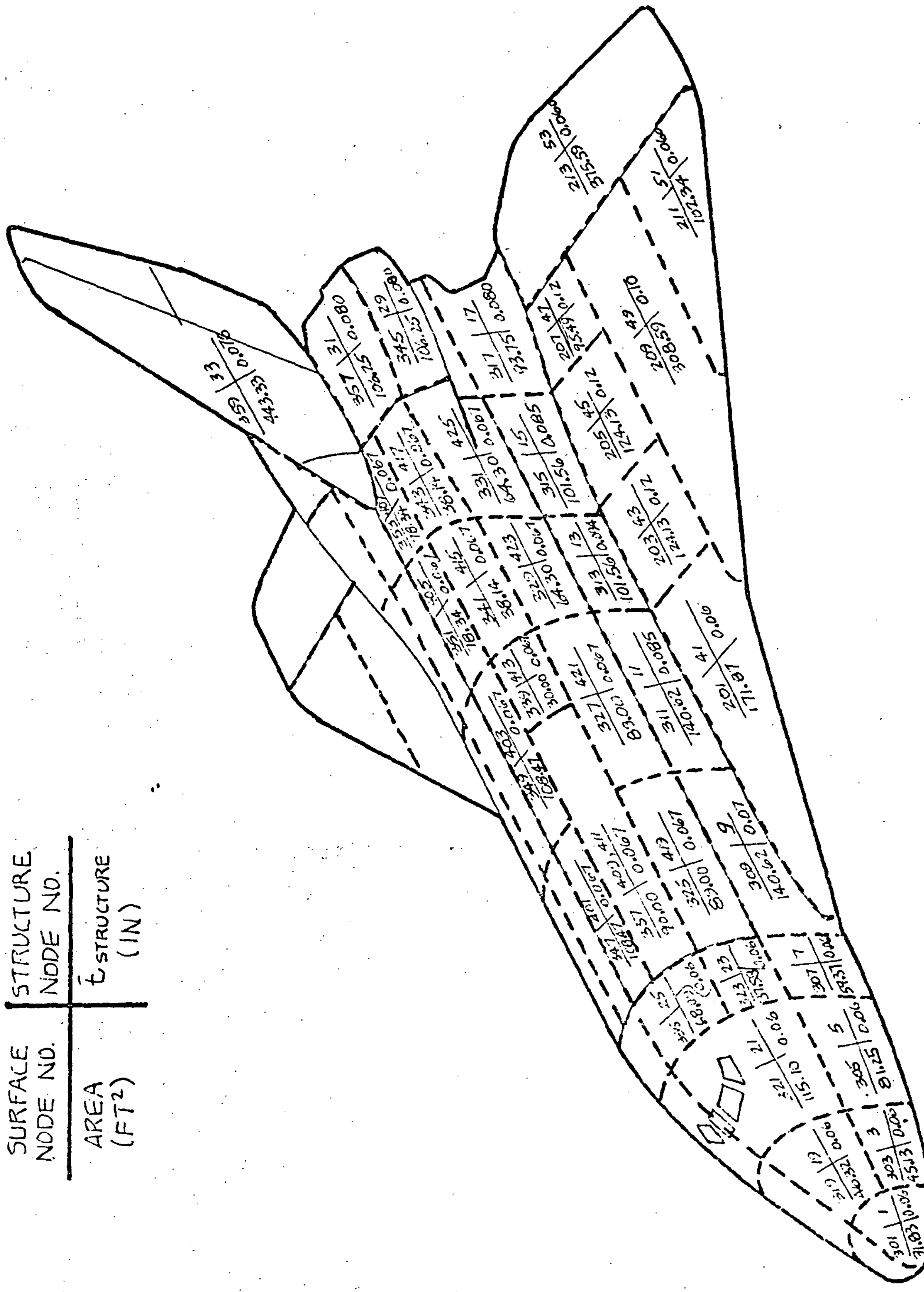

FIG. 4-9 TMM NODAL DESCRIPTION

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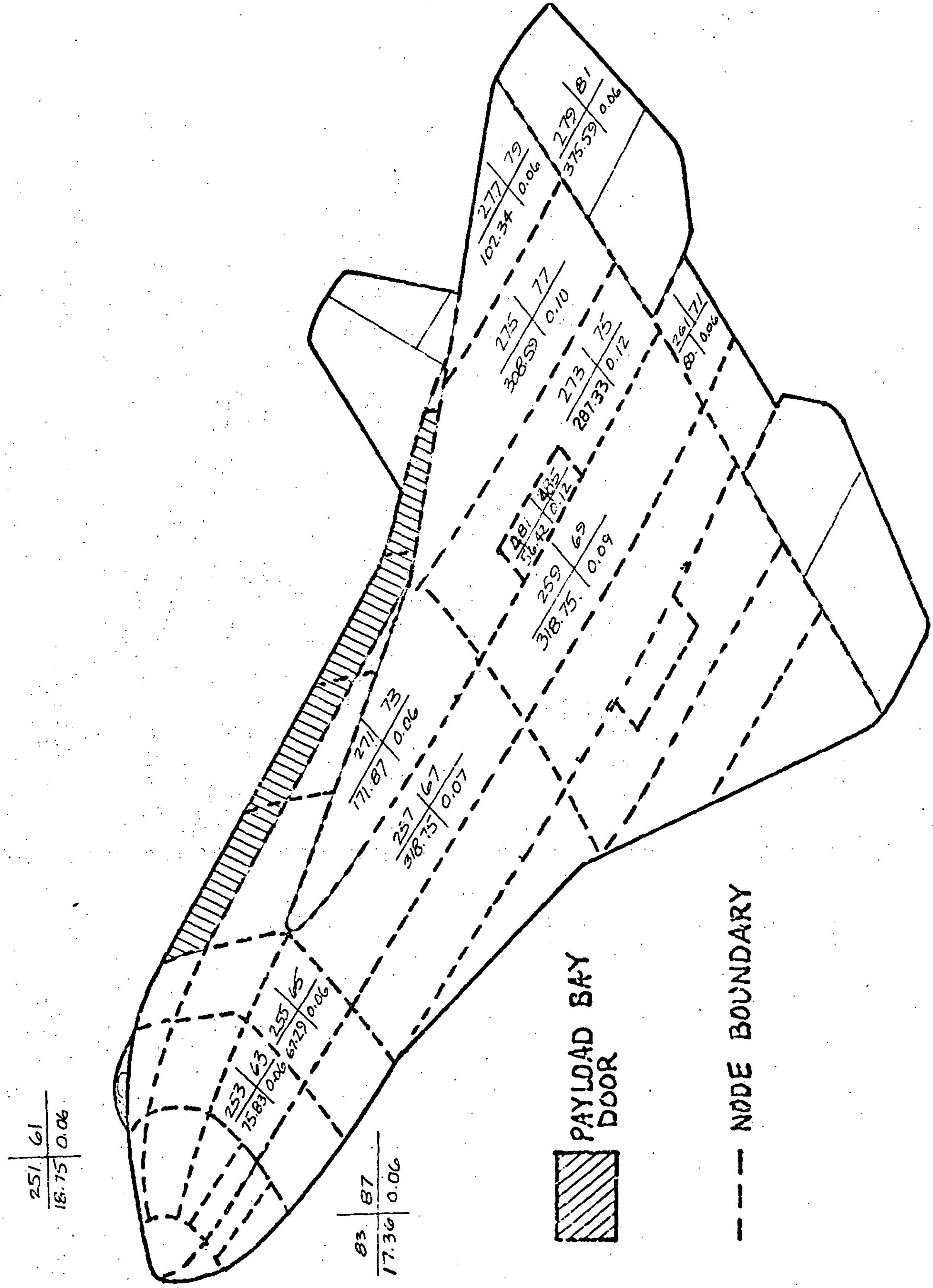


FIG. 4-10 TMM NODAL DESCRIPTION

1) structural, in which the physical shape of the vehicle was altered or distorted, and
2) thermodynamic, in which the overall vehicle heat transfer was altered without affecting the orbital temperatures.

Structural Assumptions

The general shape and physical relationship between surfaces and components of the Shuttle Orbiter were maintained. Some of the exterior surface contours were altered as follows:

1. The region in front of the crew cabin was changed to a conical surface.
2. The vertical stabilizer was assumed to be a trapezoid rather than in its true shape.
3. The body cross section aft of the payload bay was changed to a rectangular shape.
4. The wing upper and lower surfaces were assumed to be flat, with the upper surface canted down 4° to account for the taper of the wing.
5. The base area where the orbiter main engines are located was modeled as a flat plate and modified to account for the engine cutouts. The modifications included a thinner aluminum structure and reduced surface infrared emittance and absorbtance to account for shading by the engines.

The interior structure was assumed to be one of two types, either shear web or trusswork construction. Inside the wing and the fuselage lower surface, 0.032 inch thick aluminum shear webs spaced on 30 inch centers were assumed. Between the fuselage outer surface and payload bay inner surface, trusswork structure was assumed. Figure 4-11shows the assumed construction. The trusswork was assumed to be at the same stations as the shear webs in the body lower surface and consisted of 1.5 inch O.D. tubes with 0.030 inch wall thickness.

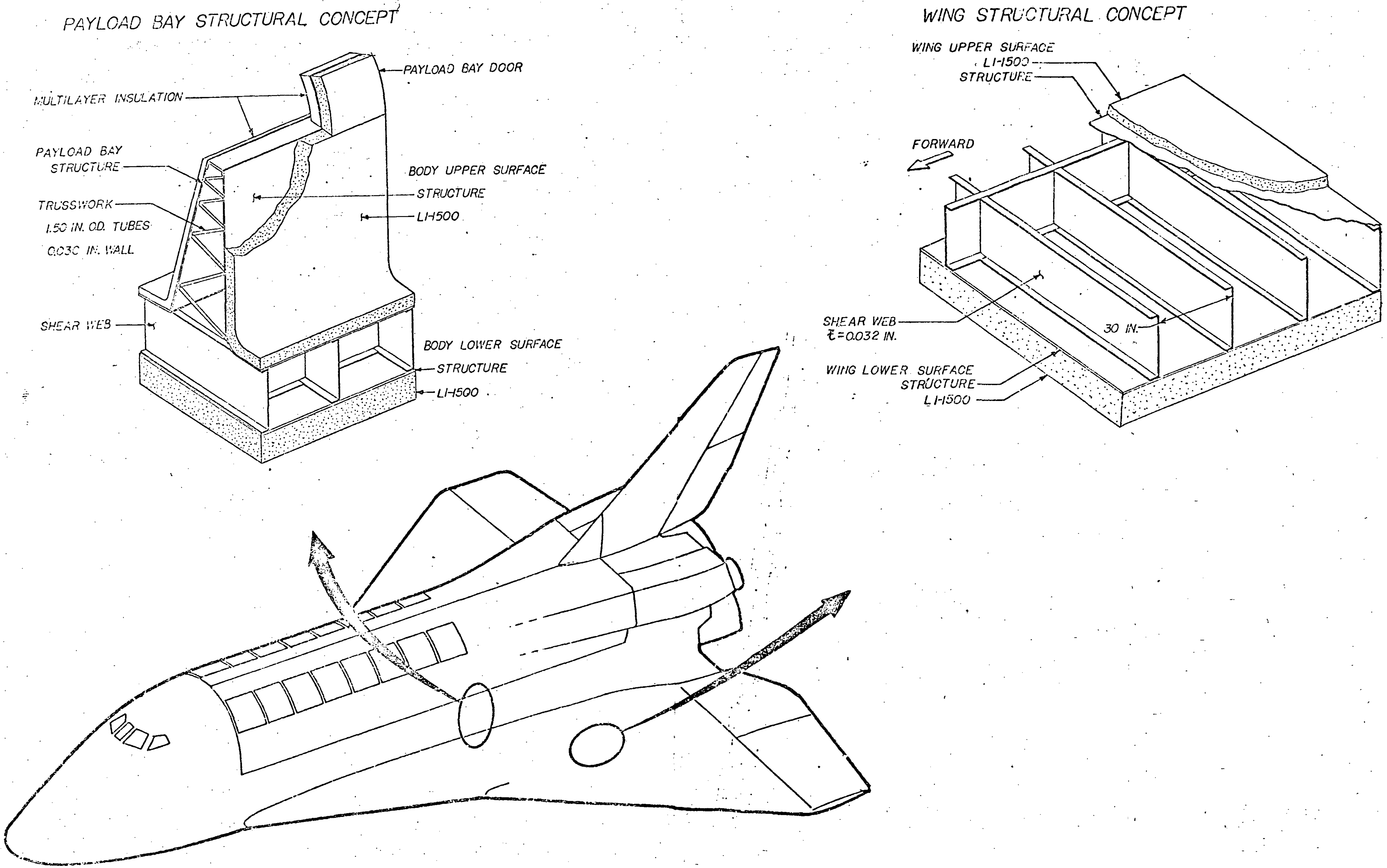


FIGURE 4-11 - SPACE SHUTTLE STRUCTURAL DETAIL

All structural material used in the TMM was assumed to be 2024-T3 aluminum throughout, except for the landing gear bogeys where 6 Al-4V Titanium was used.

By NASA directive, a solid aluminum skin structure was assumed throughout the vehicle in place of the honeycomb structure proposed by LMSCI in Ref. 2. Equivalent skin and structural thicknesses ($\bar{t}$) as furnished by NASA (Figure 3-14) were used in the TMM development.

The landing gear thermal model was based on the LMSCI Orbiter details. Main gear weight was 3541 lbs. per side, including three wheels per side at 210 lbs. each. For the nose gear, 183 lbs. was allowed for the two wheels and 875 lbs. for the bogey. Titanium was assumed for the landing gear bogeys and rubber (C_p = 0.48 BTU/LB$^{\circ}$F, ρ = 75 lbs/ft^3) for the tires.

Thermodynamic Assumptions

Thermodynamic assumptions used in developing the TMM have been directed toward simplifying the computer model while maintaining the correct thermodynamic relationships and orbital temperature response of the vehicle. The following assumptions were made:

1) Heat conduction through the LI-1500 parallel to the surface is negligible compared to conduction through the aluminum skin structure. This was verified by hand calculations and computer analysis. Hand calculations showed that for the thickest LI-1500 (1.74 in.) on top of the thinnest aluminum skin (0.060"), the ratio of the heat conducted through the LI-1500 to that conducted through the skin was 1 to 30. Therefore, for the same temperature difference between two points, 30 times more energy would be conducted through the skin than through the LI-1500 in a direction parallel to the vehicle surface. Computer analysis showed that for two adjacent LI-1500 nodes on the Orbiter, the temperatures were +37.3°F and -67.6°F with or without the lateral LI-1500 conductance.

2) The payload bay doors and landing gear doors were assumed not to be connected to the structure by any heat conduction paths. Although these doors are secured by hinges and latches, these items were assumed to have high thermal resistance. Before any other assumption regarding hinge thermal resistance can be made for the thermal model, it will be necessary to obtain data on the resistance across the actual hinges proposed for the prototype.

3) No ascent aerodynamic heating effects are included since these effects are assumed to be non-critical in determining the orbital temperature response of the Space Shuttle for the modeling program.

4) The crew cabin with its interior equipment was input to the TMM as an insulated shell with an interior convective environment. The cabin insulation was assumed to be 0.75 inch thick polyurethane foam with an internal convective heat transfer coefficient of 0.25 BTU/Ft^2 Hr. No windows were added to the cabin since they have their own cooling system and movable shades. The structural concept consisted of a single pressure shell on the upper surface where the inside cabin contour matches the body contour and a pressure shell and aerodynamic shell on the sides and lower surface where the two contours do not coincide. The internal atmosphere temperature was set at 70°F.

5) Optical properties of the various materials used on the Space Shuttle are shown in Table 1. On the interior structural surfaces, such as the wing upper and lower surfaces, the emittance (ϵ) was assumed to be 0.8, which is characteristic of a surface painted for corrosion control. The payload bay interior surfaces were assumed to have an ϵ of 0.9, as directed by NASA. Radiation coupling between various nodes in the TMM was assumed, not only between exterior surfaces, but also between interior structural nodes such as wing upper and lower surfaces, vertical stabilizer sides, and body structure and payload bay sides. This radiation heat transfer was found to be the primary mode of heat transfer in these areas.

TABLE 1

OPTICAL PROPERTIES OF MATERIALS USED IN THE SPACE SHUTTLE

Material Location	α_s	ϵ	α/ϵ
LI-1500	80	.84	.952
Payload Bay Interior Surfaces	0.9	0.9	1.0
Structure Interior Surfaces	--	0.8	--
Carbon/Carbon Nose Cap	0.9	0.9	1.0

TABLE 2

LI-1500 THICKNESSES

Body Lower Surface		Body Upper Surface	
STA 250 to 575	1.56"	STA 250 - 575	0.63"
375 to 1025	1.66"	575 to 1500	0.40"
1025 to 1500	1.62"	Vertical Stabilizer	0.4"
Wing Lower Surface -	1.74"	Wing Upper Surface	0.4"
Elevon Lower Surface -	2.00"	Elevon Upper Surface	0.4"
		Base	1.6"

6) The only source of power dissipation included in the model to date of program termination were the radiators which dissipate a total of 67000 BTU/Hr to space. Other heat sources which were to be included at a later date include the Reaction Control System propellant, storage areas, fuel cell units, and avionics packages not actively cooled. The 67000 BTU/Hr heat dissipation for the radiators is the maximum value, and any additional heat load would be handled by waste water sublimators. The minimum radiator heat load was assumed to be 0 BTU/Hr for a docked condition with no personnel aboard.

7) LI-1500 thicknesses used for the TMM are shown in Table 2, with the Orbiter stations shown in Figure 4-12. The thickness ranges from 0.4 inch on the upper vehicle surface to 2.0 in. on the elevon lower surface.

4.2.1 Payload Configuration

Per agreement between NASA and LMSC, a typical payload was defined for inclusion in the thermal analytical studies. The configuration and properties of the payload are as follows:

- o Cylindrical shaped payload 60 ft. long by 15 ft. O.D.
- o 2024 aluminum skin with a thickness of 0.03 inch.
- o Emittance of payload external surface of 0.9.
- o Emittance of internal payload bay surfaces of 0.9.
- o Passive payload with no internal energy generator.
- o Assume only radiation coupling between payload and orbiter
- o Assume multilayer insulation on the interior of the payload bay as follows (Refer to Fig. 4-13 for stations):

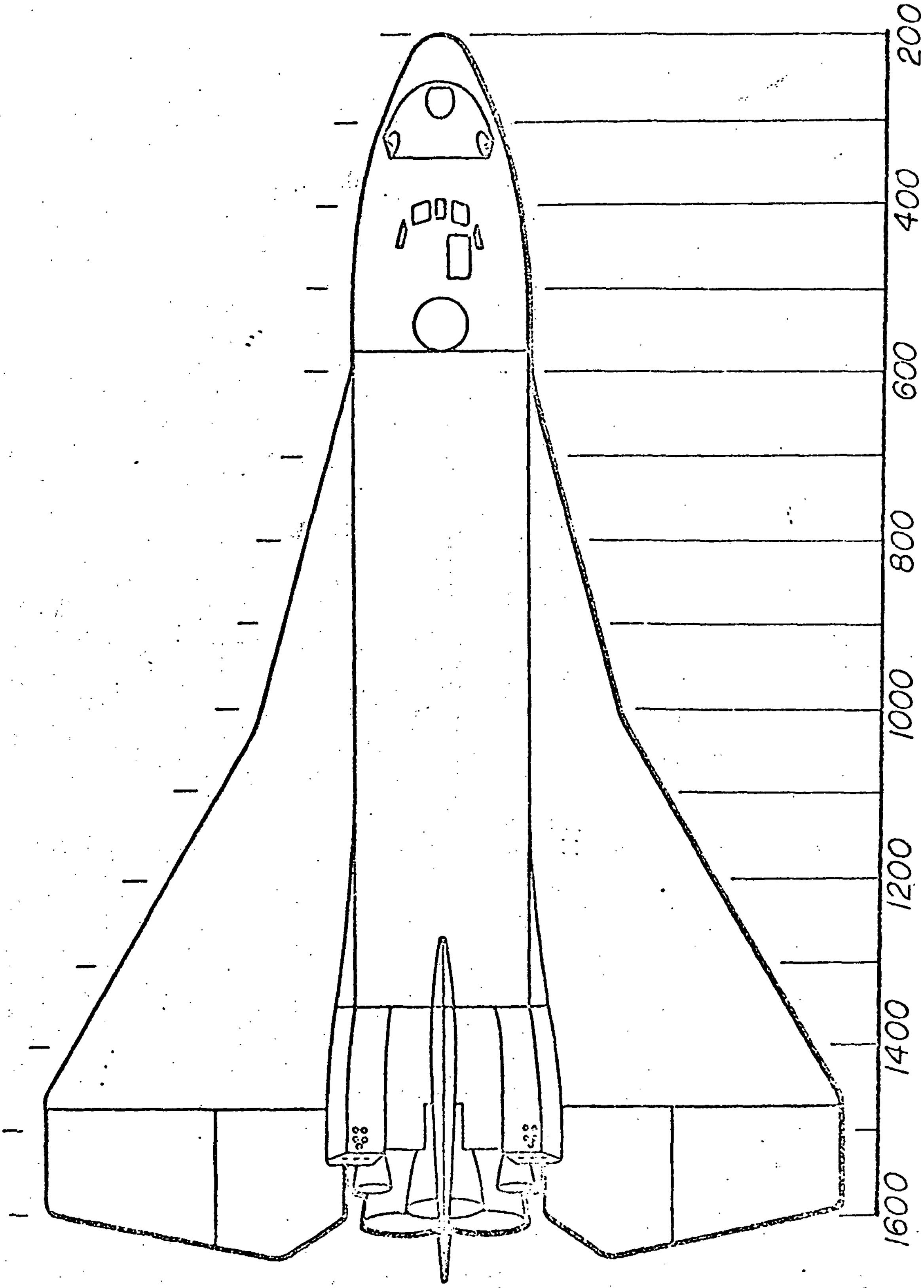


FIGURE 4-12 ORBITER FLIGHT STATIONS

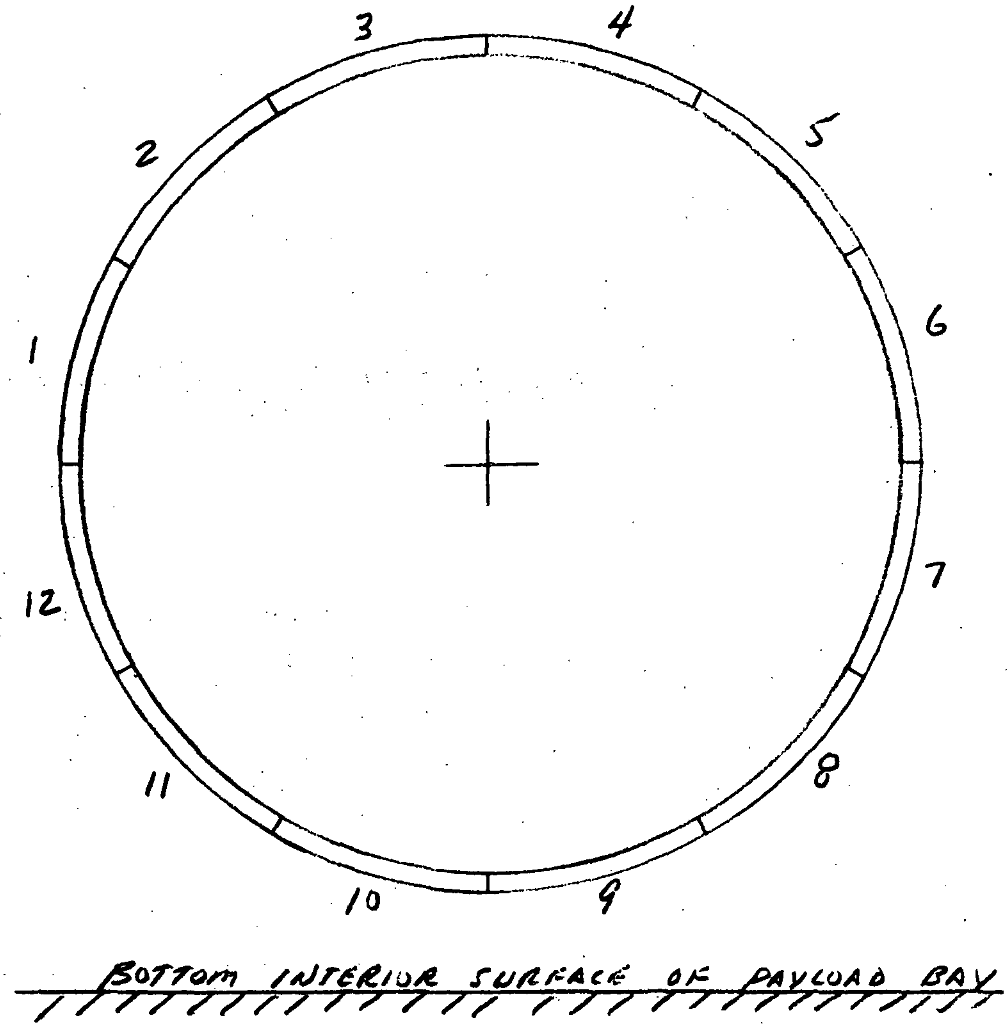


FIG. 4-13 STATION NUMBERS FOR TYPICAL PAYLOAD CONFIGURATION

Surface No.	MLI Thickness (in.)
1	0.85
2	0.58
3	0.58
4	0.58
5	0.58
6	0.58
7	1.98
8	1.98
9	2.41
10	2.41
11	1.98
12	1.98

o Assume properties for the multilayer insulation

$$C_p \text{ @ } 70^\circ F = .315 \text{ BTU/LB } ^\circ F$$

$$\rho = 1.58 \text{ LBS/FT}^3 \text{ (70 layers/in.)}$$

Temp ($^\circ$F)	$C_p(T)/C_p(70^\circ F)$
-360	0.16
-310	0.23
-210	0.35
70	1.0

K @ 0 atm. @ $70^\circ F = 0.7 \times 10^{-3}$ BTU/FT-HR-$^\circ$F

Temp ($^\circ$F)	K(T,P)/K(70°F, 0 ATM)
-300	.338
-200	.375
-125	.438
- 50	.588
0	.725
70	1.0
150	1.312
225	1.75
350	2.475

4.3 Orbital Conditions

Heat rates and vehicle temperatures were calculated using the TMM for the extreme hot, cold, and worst case transient orbital conditions. The three orbital cases are as follows:

Hot Case: $\beta = 90^{\circ}$, solar oriented with the vehicle rolled 45° to provide the maximum projected area of vehicle top side toward the sun, payload doors closed, radiators open.

Cold Case: $\beta = 90^{\circ}$, bottom of vehicle earth oriented, tail facing sun, payload bay doors open.

Worst Transient: Equitorial orbit, payload bay doors open and facing sun in solar orientation; vehicle to pitch or roll 90°, depending on initial orientation, as it enters earth's shadow and to remain earth oriented with bottom facing the earth while in shadow, then pitch or roll 90° to repeat solar exposure of open payload region.

4.4 TMM Results

Steady state temperature results for the hot and cold orbital cases are presented in Figs. 4-14 through 4-20. For the hot case, the aluminum skin temperatures ranged from approximately $190^{\circ}F$ on the wing upper surface to $-140^{\circ}F$ on the fuselage lower surface. For the cold case, all structural temperatures are in the range of $-200^{\circ}F$ to $0^{\circ}F$, except for the sun illuminated aft section and radiator area.

Results were obtained for a short transient run which demonstrated that the computer program was operating correctly. The data was not reduced prior to contract termination, and therefore, will not be presented in this report.

STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
UPPER SURFACE, SUN SIDE

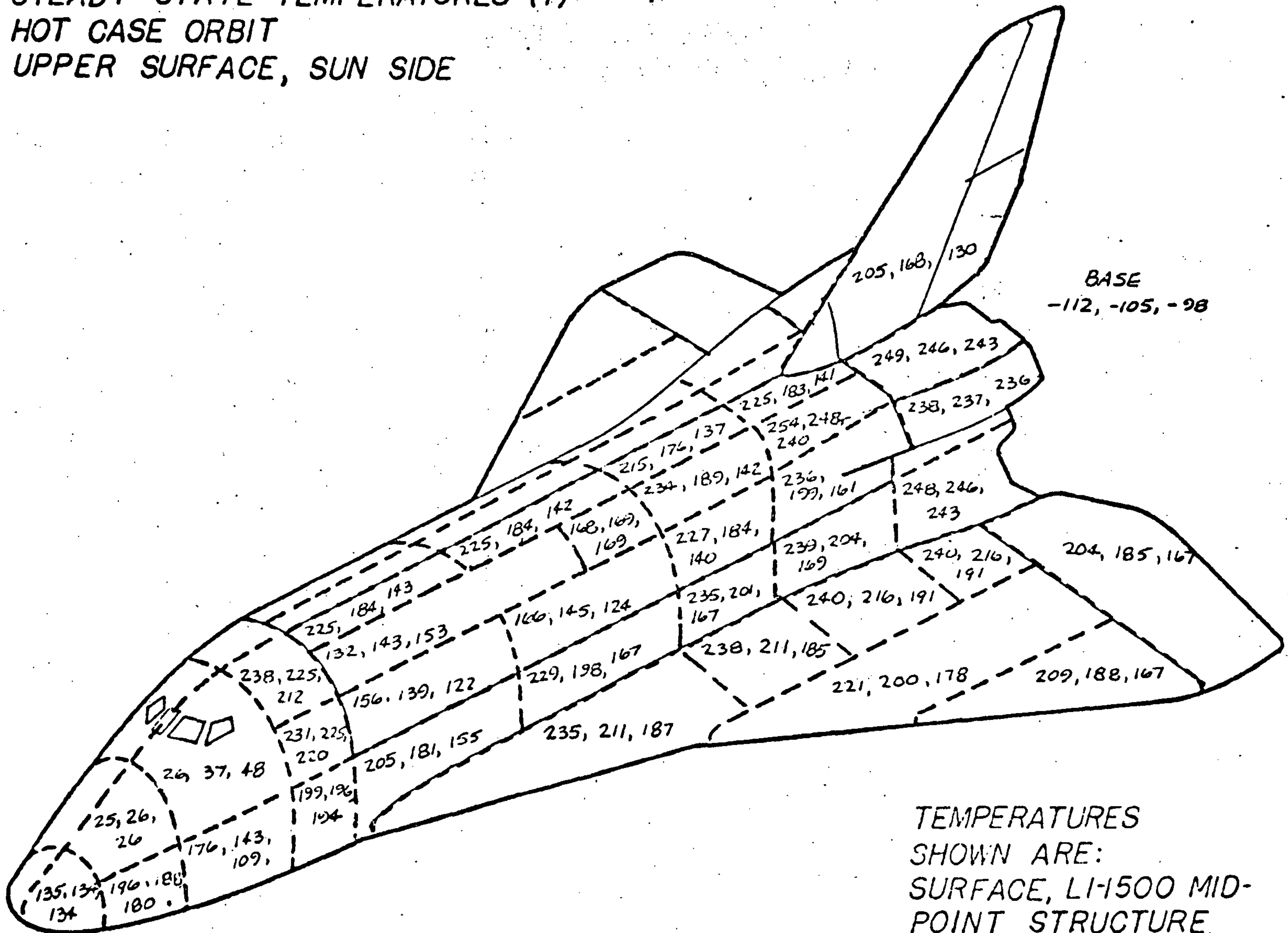


FIG. 4-14

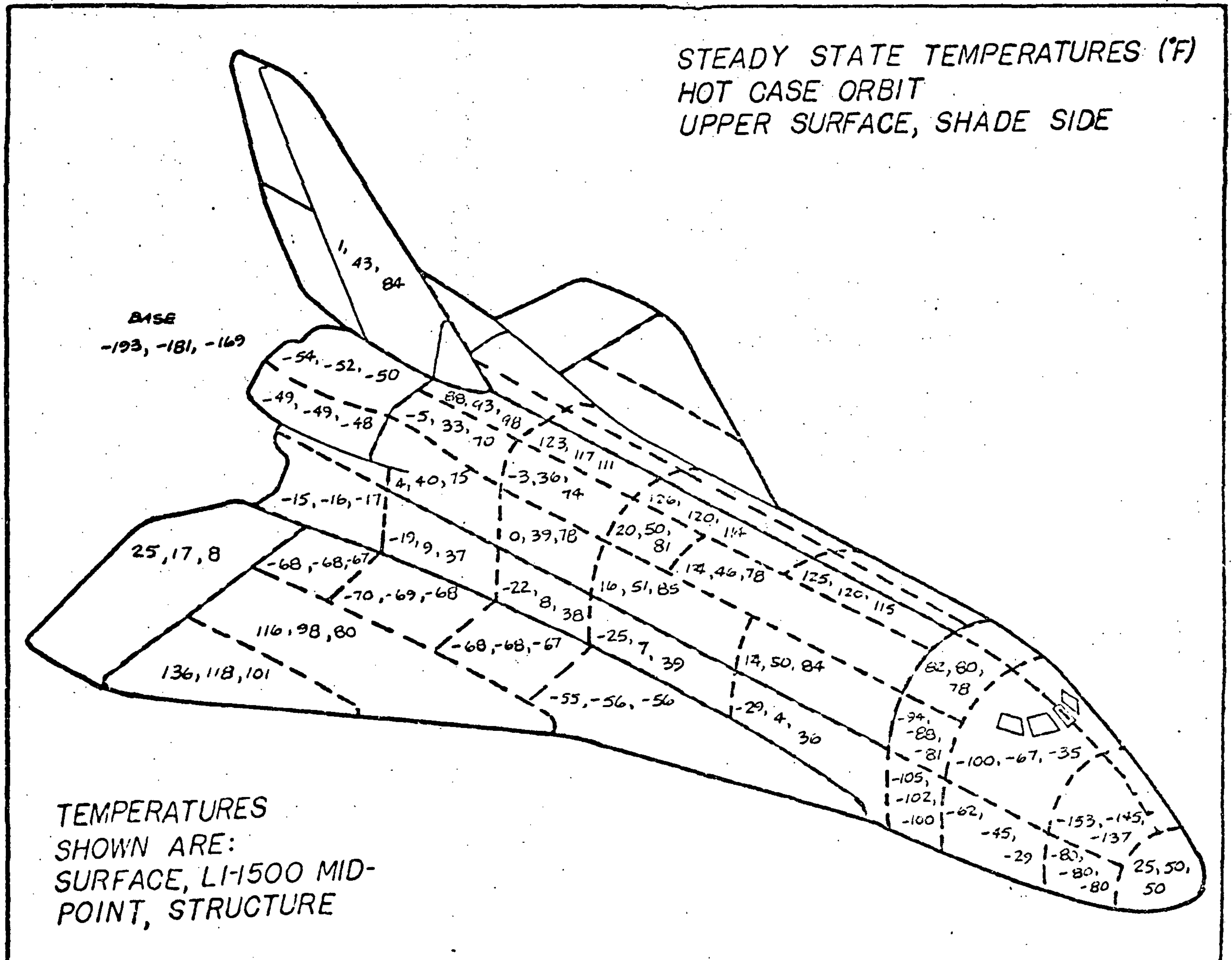

STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
UPPER SURFACE, SHADE SIDE
1, 43, 84
BASE
-193, -181, -169
-54, -52, -50
-49, -49, -48
88, 43, 98
-5, 33, 70
123, 117 111
4, 40, 75
-3, 36, 74
-15, -16, -17
-19, 9, 37
0, 39, 78
126, 120, 114
20, 50, 81
25, 17, 8
-68, -68, -67
-70, -69, -68
-22, 8, 38
16, 51, 85
14, 46, 78
125, 120, 115
116, 98, 80
-68, -68, -67
-25, 7, 39
14, 50, 84
136, 118, 101
-55, -56, -56
-29, 4, 36
82, 80, 78
-94, -88, -81
-100, -67, -35
-105, -102, -100
-62, -45, -29
-153, -145, -137
-80, -80, -80
25, 50, 50
TEMPERATURES
SHOWN ARE:
SURFACE, LI-1500 MID-
POINT, STRUCTURE

FIG. 4-15

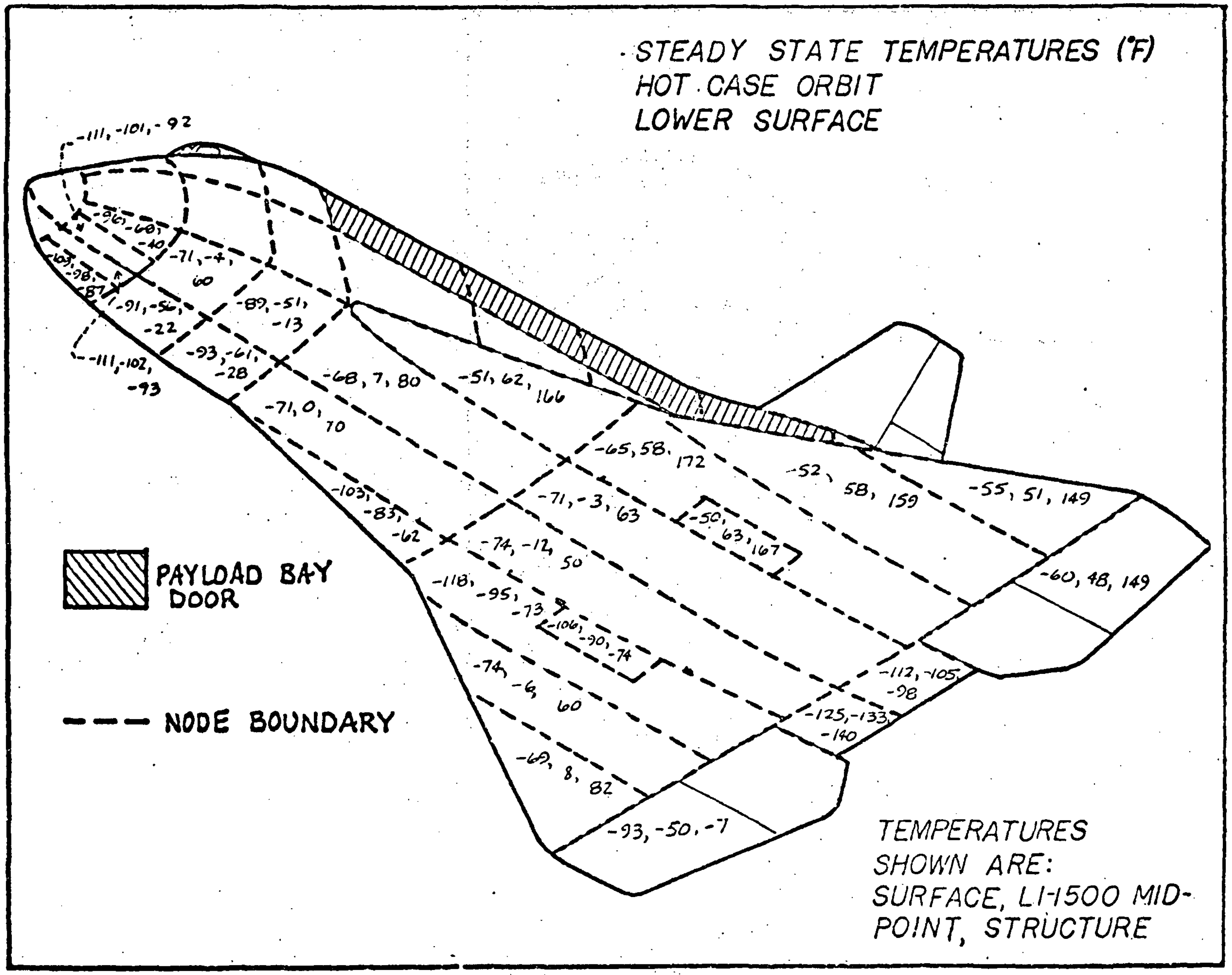
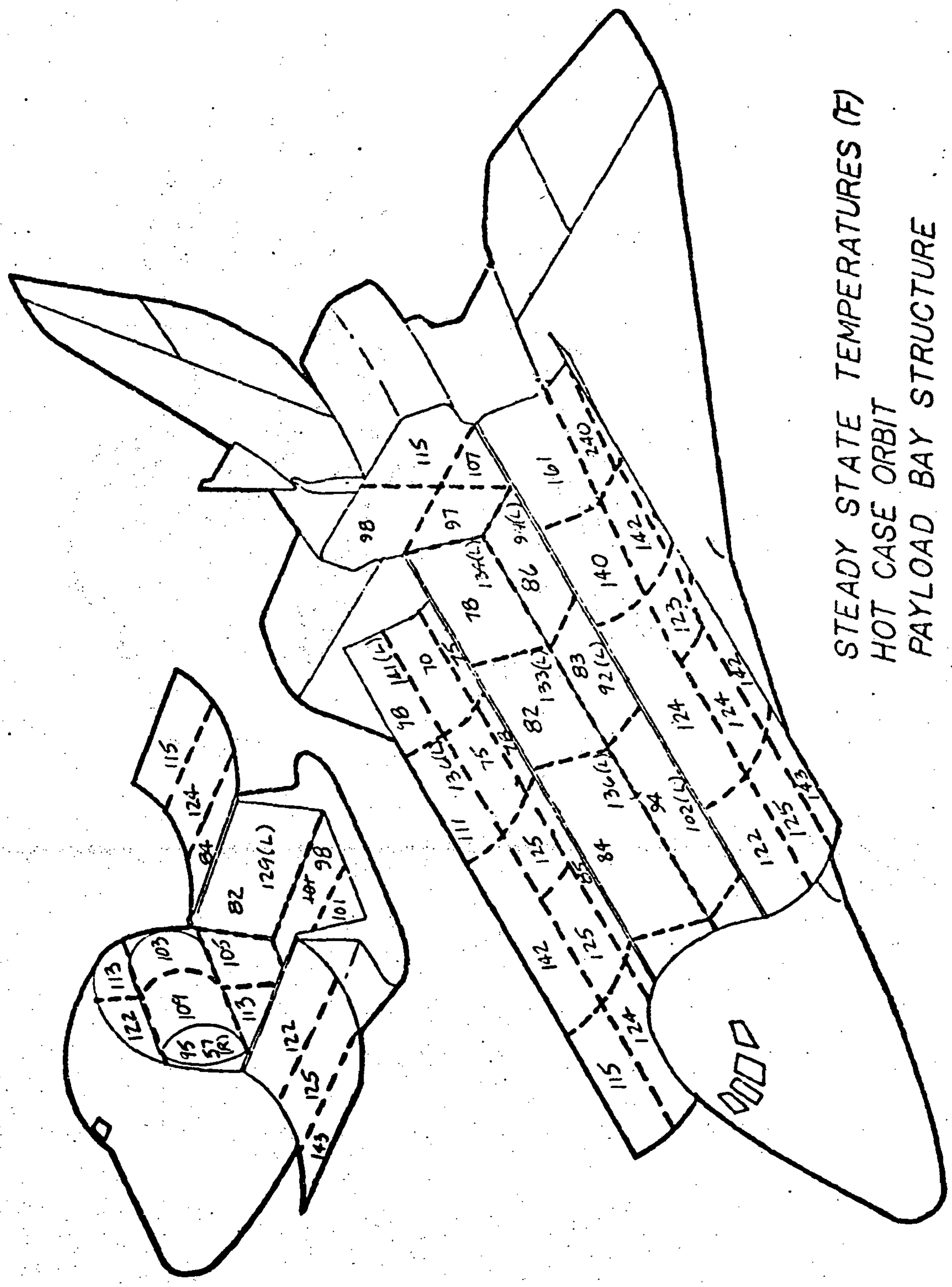

STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
LOWER SURFACE
-111, -101, -92
-71, -4, 60
-89, -51, -13
-91, -56, -22
-93, -61, -28
-111, -102, -93
-68, 7, 80
-51, 62, 166
-71, 0, 70
-65, 58, 172
-52, 58, 159
-55, 51, 149
-103, -83, -62
-71, -3, 63
-50, 63, 167
-74, -12, 50
-60, 48, 149
-118, -95, -73
-106, -90, -74
-112, -105, -98
-74, -6, 60
-125, -133, -140
-69, 8, 82
-93, -50, -7
PAYLOAD BAY DOOR
NODE BOUNDARY
TEMPERATURES SHOWN ARE: SURFACE, LI-1500 MID-POINT, STRUCTURE

FIG. 4-16



STEADY STATE TEMPERATURES (F)
HOT CASE ORBIT
PAYLOAD BAY STRUCTURE

FIG. 4-17

STEADY STATE TEMPERATURES (°F)
COLD CASE ORBIT

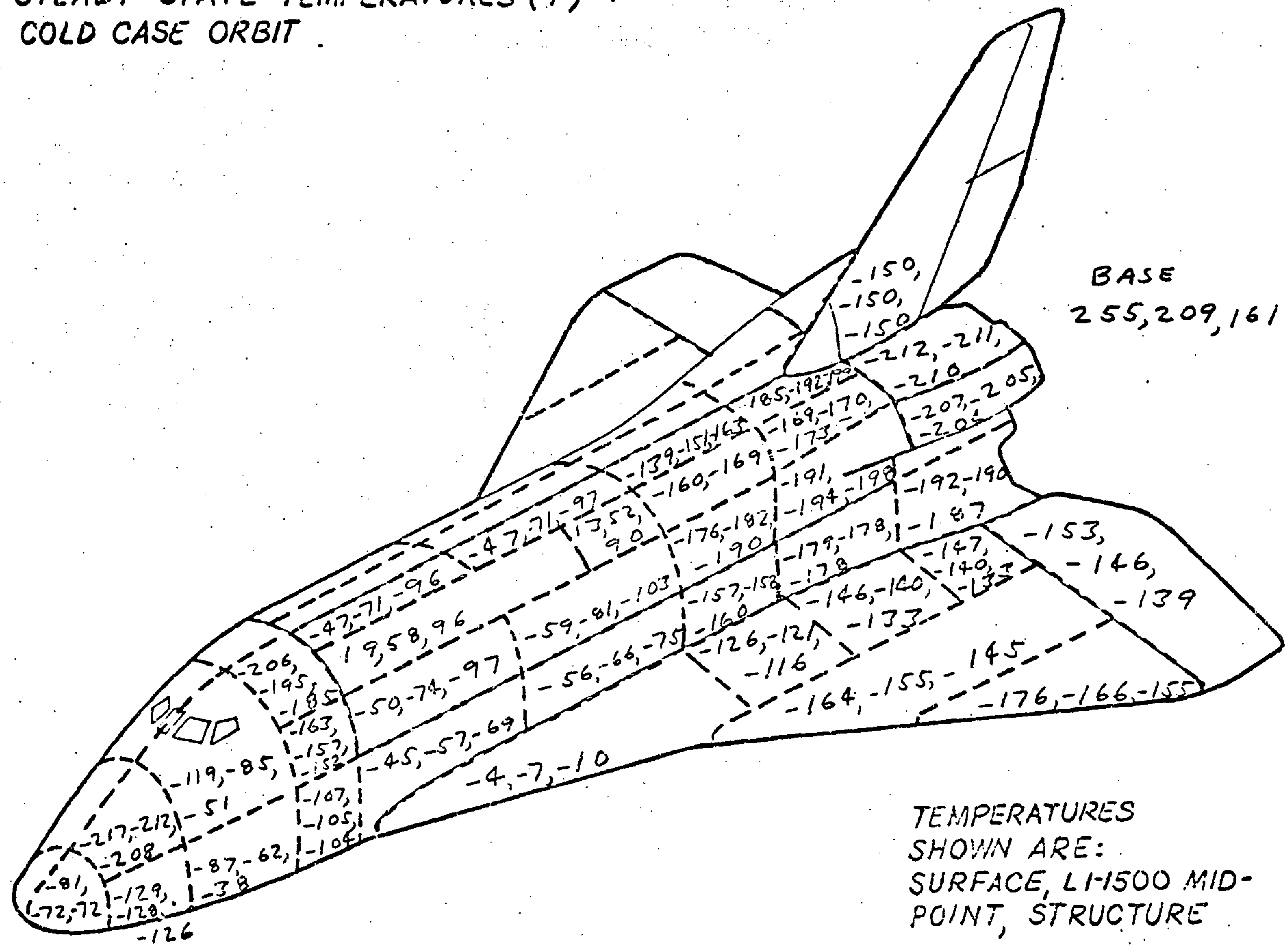


FIG. 4-18

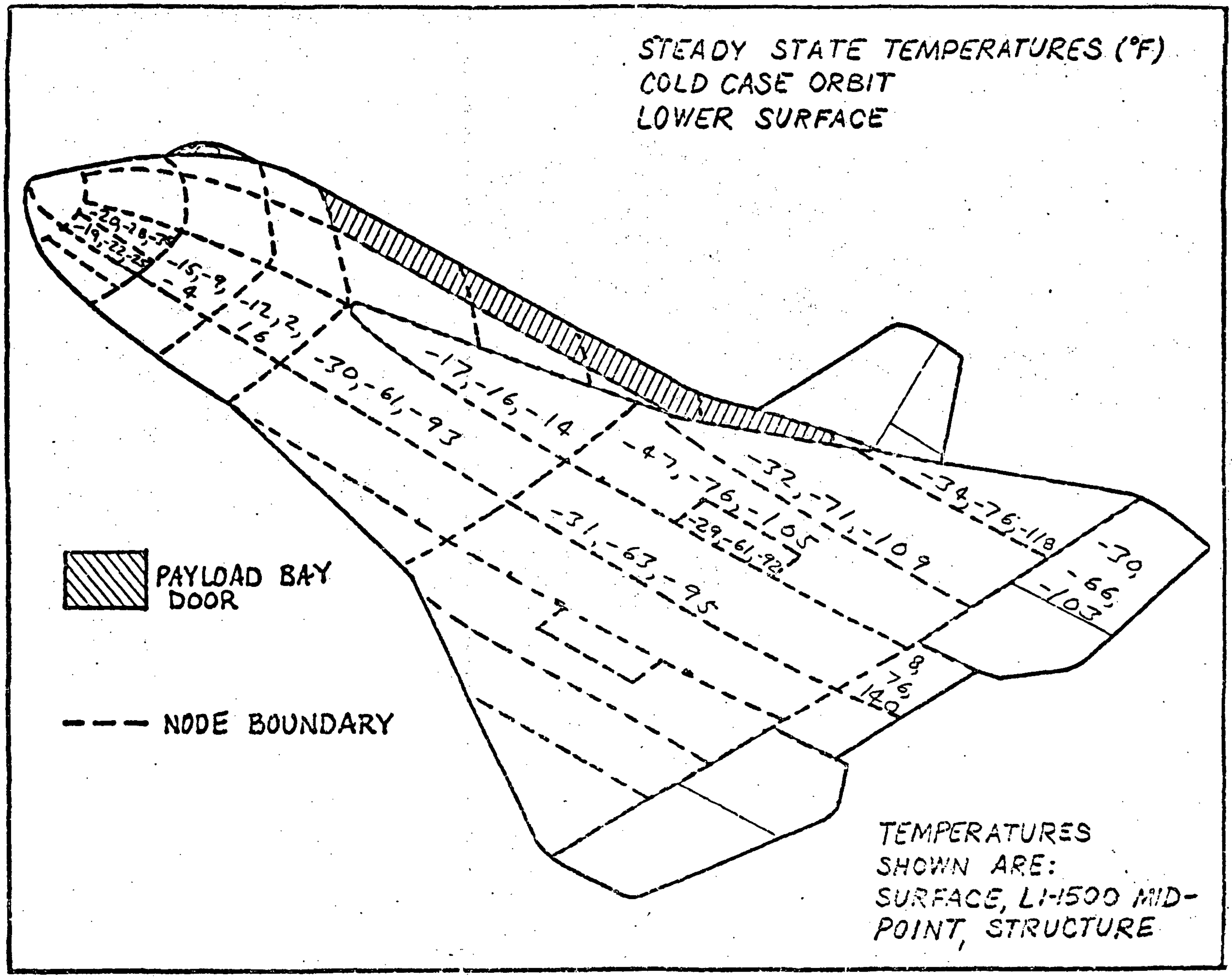
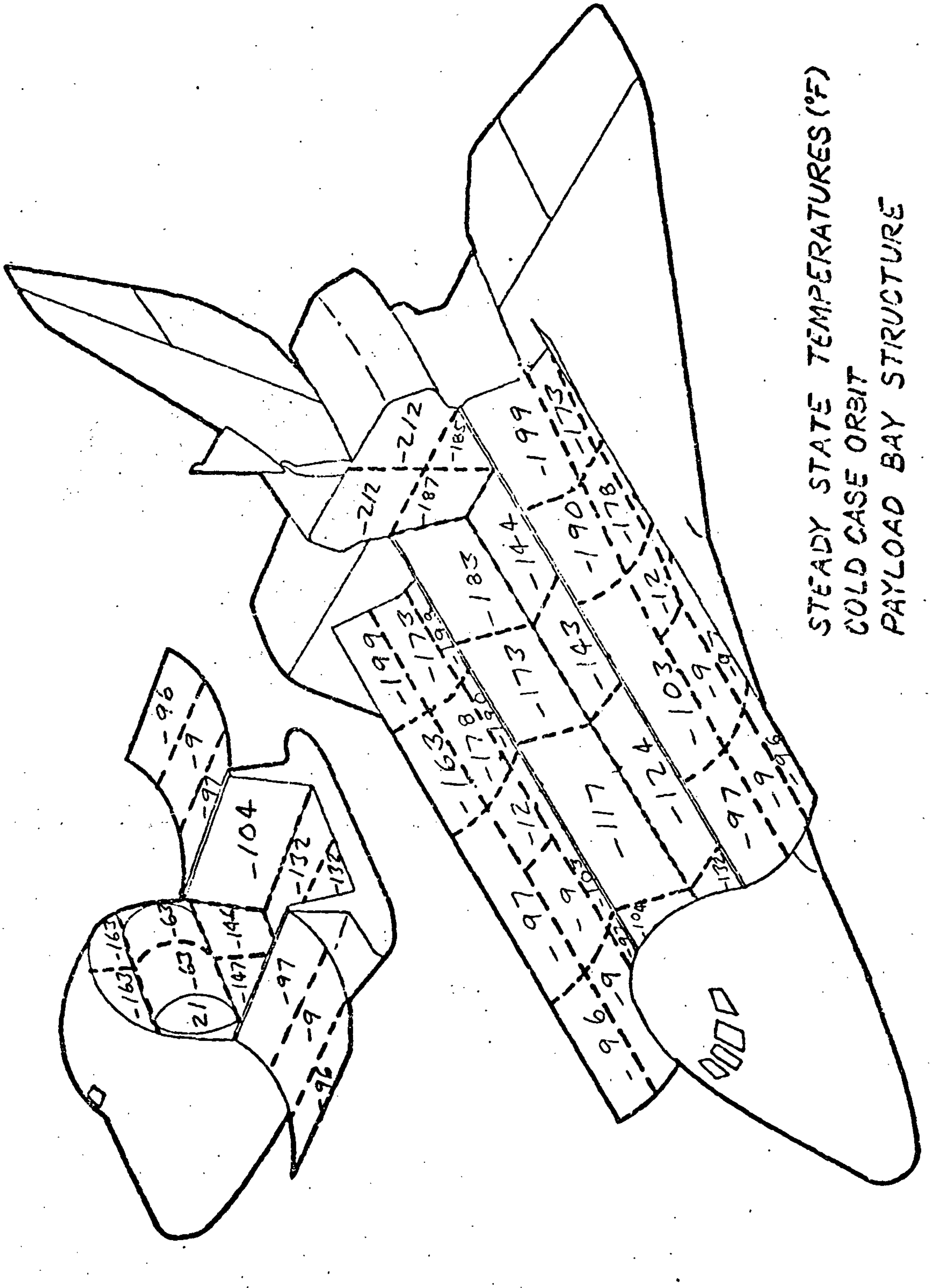

STEADY STATE TEMPERATURES (°F)
COLD CASE ORBIT
LOWER SURFACE
-15,-9,
-4
-12,2,
16
-30,-61,-93
-17,-16,-14
-47,-76,-105
-29,-61,-92
-31,-63,-95
-32,-71,-109
-34,-76,-118
-30,
-66,
-103
8,
76,
140
PAYLOAD BAY DOOR
NODE BOUNDARY
TEMPERATURES SHOWN ARE:
SURFACE, LI-1500 MID-POINT, STRUCTURE

FIG. 4-19



STEADY STATE TEMPERATURES (°F)
COLD CASE ORBIT
PAYLOAD BAY STRUCTURE

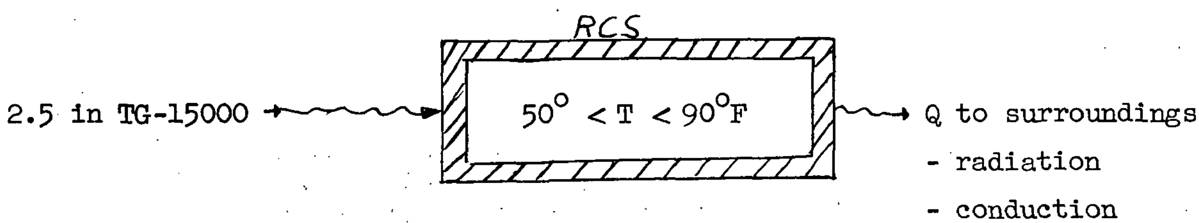
FIG. 4-20

4.5 Internal Power Sources

As discussed in Section 4.2, the initial TMM assumed only power dissipation due to the crew cabin and its associated equipment. Prior to contract termination, various other sources of internal power generation that may have affected internal temperatures were identified. These would have been included in the revised TMM had the program continued. A discussion of these are in the following sections.

4.5.1 RCS and OMS

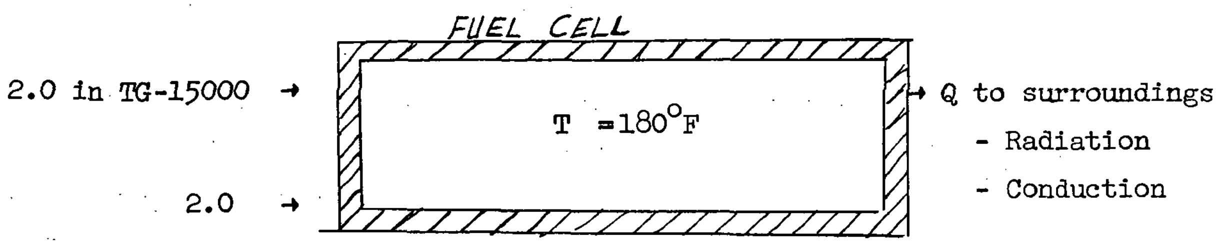
The Reaction Control System (RCS) and Orbital Maneuvering System (OMS) propellants have an allowable temperature range during storage of +50°F to +90°F. For the TMM, a thermal node insulated from its surroundings with 2.5 in. of TG-15000 was assumed. A heat source (or sink) was provided such that when the temperature exceeded the set limits, enough heat would be provided or allowed to dissipate to keep it within the stated limits. Although the RCS thrusters have a catalyst bed which is maintained above 150°F, the heat leak effect was neglected since the thrusters are well insulated and any heat leak from this source can be assumed to keep the propellants warm. A schematic of the thermal model would look like this



Two of these systems will be placed in the aft structural compartment and one forward of the crew cabin. The two located aft would encompass both the RCS and OMS on one side of the vehicle into one node.

4.5.2 Fuel Cells

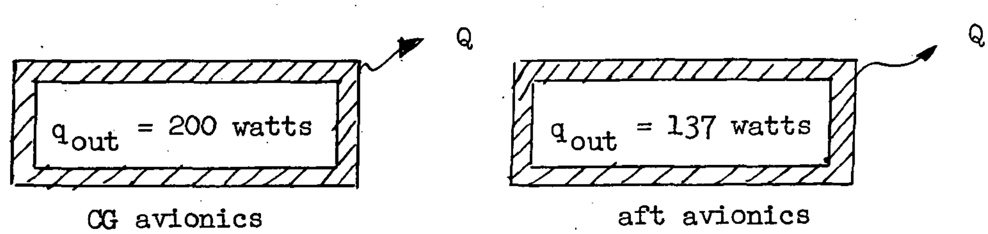
The fuel cell plants, operated continuously in orbit, are actively cooled with most of their excess heat being rejected by the space radiators. Some excess heat will escape in the fuel cell area, however, since they operate at temperatures around 180°F. For the TMM, the recommended approach was to select a thermal node that is maintained at the fuel cell operating temperature of 180°F and insulated from its surroundings as shown below.



One of these units would be placed between the crew cabin and payload bay.

4.5.3 Avionics

Avionics in the space shuttle concept studied during this program are located in three areas, 1) the crew cabin, 2) clustered around the orbiter CG, and 3) in the area behind the payload bay. The crew cabin avionics are actively cooled by the cabin air and are not included as a separate node, in the TMM. Passive cooling is assumed for the other two avionics packages, with all heat produced being absorbed by the surroundings. Power dissipation for the aft electronics is 137 watts. For the CG avionics, the power dissipation is estimated as 200 watts. The recommended setup for the TMM is shown below:



4.5.4 Hydraulics

The hydraulic system fluid has a lower temperature limit of $-40^{\circ}F$. When the fluid temperature reaches this point, pumps are started to circulate the fluid until the temperature goes above $-20^{\circ}F$, at which time the pumps are shut down. Due to the small size of actuators and lines inside the vehicle extremities, these items were not planned for inclusion in the TMM. The pumps were assumed to be located in the region behind the payload bay where other heat producing items are located. This would provide the pumps with enough heating such that they would need no active thermal control of their own. Due to the fact that their primary duty cycle is during ascent and entry only, the heat dissipated during their orbital operation was assumed to be small enough such that it can be neglected.

Section 5

TSM DESIGN

Detailed design of the scale model was to be accomplished during the second phase of the program. This effort had just started when notification was received of contract termination. However, enough progress had been made to select a design scale ratio of 1/3 and some of the materials for modeling the conduction heat paths. Also, relatively firm requirements were established for modeling the crew cabin atmosphere and candidate gases were selected for convective modeling.

TSM design was to be accomplished in accordance with the standard modeling laws developed for the "temperature preservation" approach. The general criteria for thermal scale modeling in the space environment has been established and reported by numerous authors, and thus, will not be repeated in great detail in this report. The basic approach to be followed in design of the TSM was to select a scale ratio that appeared to satisfy the majority of the model's requirements and then proceed to model all areas of the vehicle to meet that scale ratio. It was anticipated that various compromises in modeling would be required for some areas of the vehicle that could not be scaled exactly. It is also a possibility that the scale ratio of 1/3 would have to be adjusted $\pm$ 10% as the process of selecting model materials progressed.

Had the program been completed, the end result of the model design activity would have been the specification of materials, size, shape, weight, internal power sources, external heat flux requirements, details of critical thermal conduction joints, details for modeling the crew cabin atmosphere, and details of modeled components and their location relative to each other. A complete set of specifications and engineering drawings was planned to document details of the TSM design.

5.1 Design Requirements

During Task 1, preliminary TSM design requirements were established. This effort involved consideration of which modeling approach to follow (i.e. temperature preservation, materials preservation, etc.), scale ratio, materials, modeling of convective heat flow, TSM size in relation to that of available test facilities, important conduction and radiation heat flow paths, level of detail for modeling various systems and components, and the potential applications of the TSM. The design criteria established during this initial phase are listed in Appendix A, Section A.6.

5.2 Conduction - Radiation Modeling

From Ref. 3, the criteria for modeling the conduction and radiation temperature fields are as follows:

$$\frac{\rho^* V^* C^* T^*}{\theta^*} = A_I^* I^* = Q^* = \frac{K_n^* A_n^* T^*}{L^*} = A_i^* T^{*4} \tag{1}$$

For temperature preservation, $T^* = 1$, and Eq. 1 becomes

$$\frac{\rho^* V^* C^*}{\theta^*} = A_I^* = Q^* = \frac{K_n^* A_n^*}{L^*} = A_i^* \qquad (2)$$

For the model to represent prototype behavior, Eq. 2 must be satisfied for all conditions and for all heat flow paths. This would be an easy task if an infinite number of materials were available with thermophysical properties that spanned the entire range. However, in reality this is not the case, therefore, compromises must be made in the scale criteria. These compromises generally are involved with distortion of the conduction heat flow path by adjusting the cross sectional areas or the lengths of heat flow paths to properly scale the amount of energy conducted. Such distortions must be done in such a manner so that the important radiative heat transfer areas are not adversely affected.

Approximately 98% of the Shuttle Orbiter's external surface is covered with insulation termed the thermal protection system (TPS) that protects the vehicle structure and internal components from the severe heating during the ascent and entry stages of flight. For purposes of this study, the LMSC material LI-1500 was assumed for the TPS. The thermal conductivity for this material is 0.0142 BTU/Hr Ft $^{\circ}$R at 0°F and is essentially isotropic. The density is approximately 15 lbs/ft^3, and the thickness ranges from a minimum of 0.4 inch to 2 inches over the Shuttle surface.

Because of the wide usage of this material on the Orbiter, particular attention was paid to modeling the LI-1500 in combination with the underlying aluminum substrate. For purposes of this study, 2024-T3 aluminum was assumed for the skin and much of the other structural components. This aluminum alloy has a thermal conductivity of 70 BTU/Hr Ft $^{\circ}$R at 75°F. In modeling conduction through the LI-1500 and aluminum skin, certain assumptions have to be made due to this combination of materials and due to the materials available for use on a thermal model. The required assumptions result in distortion of the thickness scale ratio. Based on results from the TMM analysis, the assumption can be made that heat flow in the LI-1500 is one-dimensional (i.e. perpendicular to the skin surface) and that heat flow in the aluminum skin is two dimensional (i.e. circumferential and longitudinal in the aluminum skin).

Designating LI-1500 as material No. 1 and the aluminum skin as material No. 2, and equating the various dimensionless groups in Eq. 2, the following equations are obtained:

For the case of heat flow perpendicular to the vehicle surface $A^*_{n_1} = L^{*2}_1$, and L^*_1 in Eq. 2 becomes the thickness ratio t^*_1.

Thus,

$$K^*_{n_1} = t^*_1$$

$$\theta^* = \rho^*_1 \, C^*_1 \, K^*_1$$

$$I^* = 1$$

$$Q^*_1 = L^{*2}_1$$

For the case of heat flow parallel to the vehicle surface, the aluminum skin is the predominate mode of heat transfer and the following identities result:

$$K^*_{n_2} = \frac{L^{*2}_2}{t^*_2}$$

$$\theta^* = \rho^*_2\, C^*_2\, t^*_2$$

$$I^* = 1$$

$$Q^*_2 = L^{*2}_2$$

Q^* represents the heat flux by conduction, internal radiation, or any internally generated heat flux within the vehicle

Therefore, to be more general

$$Q^* = L^{*2}$$

where L^* is the overall length scale ratio which for the Shuttle TSM is equal to 1/3.

In attempting to model the LI-1500, it was found that very few materials are available which have a conductivity lower than LI-1500. Multilayer insulation does, however the conductivity of this material is not isotropic. One candidate material for modeling LI-1500 is a Johns-Manville insulation called MIN-K. This material has a thermal conductivity under vacuum conditions that is approximately 1/2 that of LI-1500 at room temperature. Therefore, from the equation $K^*_{n_1} = t^*_1$, we find that the thickness ratio t^*_1 is 0.5.

This means that the thickness of insulation on the thermal model would be 1/2 that of the prototype.

For modeling the aluminum skin structure it was found that some of the stainless steels can be used; however, in using stainless steel for this application, distortion of the skin thickness ratio is required to model the heat flow because the conductivity of stainless stell is not exactly 1/3 that of aluminum.

Additional problems occur when trying to model the transient response. The full impact of compromises that may have to be made to predict transient behavior of the spacecraft were not determined prior to contract termination.

5.3 Convective Modeling

The important criteria for convective modeling are derived in Ref. 4. There it is shown that the criteria can be determined from the general equation:

$$h^* A_n^* T^* = Q^* = A_i^* T^{*4} = \frac{\rho^* V^* C^* T^*}{\theta^*} = A_I^* I^* = \frac{K^* A_n^* T^*}{L^*}$$

From this it is found that the similitude criteria are

$$h^* = K^* (1/L^*)$$

In order to meet the above criteria the Reynolds, Grashof and Prandtl numbers must remain invariant. If this is done then the exact temperature preservation criteria becomes

$$Kg^* = L^*$$

However, in order to meet these criteria a suitable gas must be found which will preserve the Prandtl number while at the same time have a conductivity ratio equal to L^*. The Reynolds number can usually be preserved by adjusting the flow velocity while the Grasholf number can be preserved by adjusting the gas pressure according to

$$P^* = (\mu^*/M^*)\ (1/g^*\beta^*)^{1/2}\ (1/L^*)^{3/2}$$

Generally it is difficult to find a suitable gas which meets the above criteria. To overcome this problem a system using "Scaling Compromises" was derived by Shannon (Ref. 5). These compromises involve the so-called "heat transfer coeffocient preservation" method and the "mass flux preservation" method.

These two methods use the same gas in the prototype and model and adjust to flow conditions in order to attempt to maintain similitude. It was shown in Ref. 5 that if either mass flux or heat transfer coefficient were preserved then reasonable similitude could be obtained.

Both temperature preservation techniques and scaling compromises techniques were investigated during Task I of the program. The results of these studies are given in Appendix B and will be reviewed briefly here.

The preliminary analysis of the prototype heat transfer coefficients showed that in a 1-g 1-atm test the free convection heat transfer coefficient would be $\bar{h}_c = 0.43$ BTU/Hr Ft^2 ^{o}F while the forced convection coefficient would only be $\bar{h}_c{}^p{}_p = 0.25$ BTU/Hr Ft^2 ^{o}F. That is free convection would be dominant in both prototype and model.

The "Scaling Compromise" method of heat transfer coefficient preservation could be used with reasonable results in a 1-g test for scale ratios of $L^* = 1/3$ (Figure 5.1) and Reynolds greater than 10^3. Likewise the mass flux preservation would also give good results at any Reynolds number. Therefore, either method could be used to model the prototype in a 1-g test.

In order to simulate zero-g conditions the pressure of the gas in the model would have to be lowered. If the pressure is lowered so that free convection is eliminated then very good results could be obtained using scaling compromises (Fig. 5.2).

NON-DIMENSIONAL TEMPERATURE DIFFERENCE AS A FUNCTION OF REYNOLDS NO. FOR ONE G FIELD

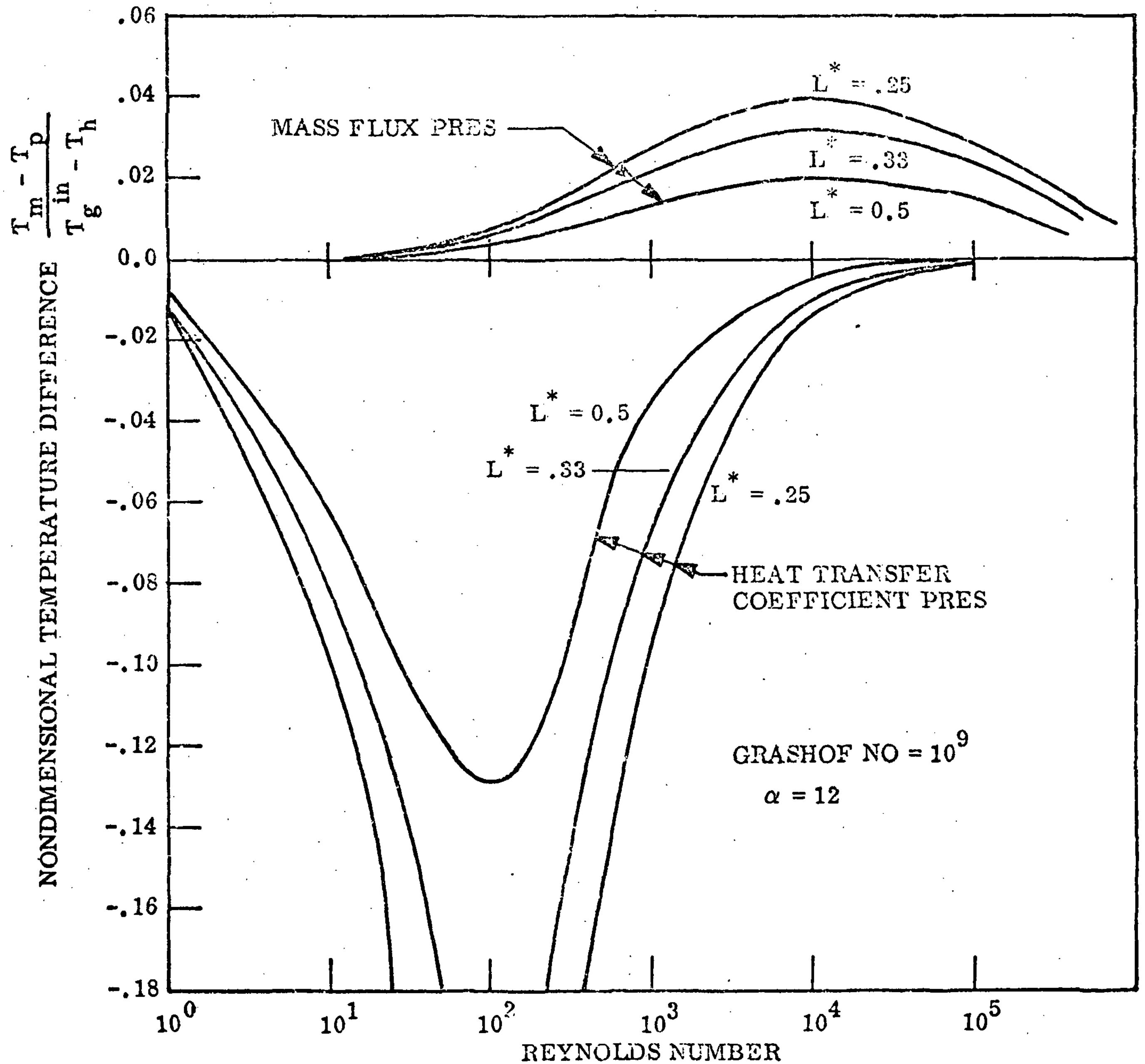


Fig. 5.1

NON-DIMENSIONAL TEMPERATURE DIFFERENCE AS A FUNCTION OF REYNOLDS NO. FOR ZERO G SIMULATION

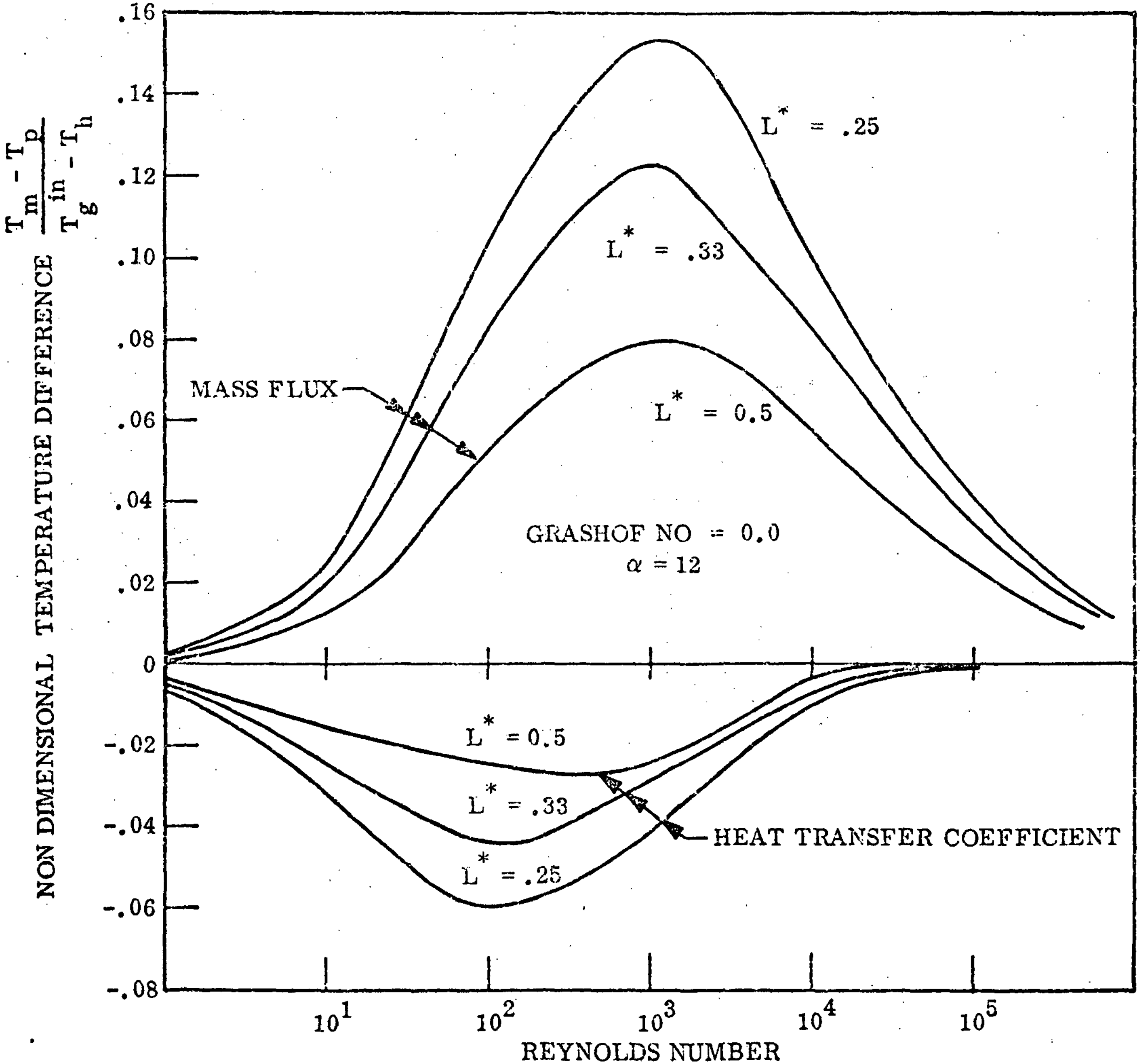


Fig. 5.2

However, if the pressure is lowered then flow velocities must be increased to maintain the correct Reynold's ratio for thermal similitude. This creates a problem as shown on Figure 5.3.

For a scale ratio of $L^* = 0.3$ it is found that the velocity of the gas in the model must be increased so that $V^* \approx 18$. Since the anticipated prototype flow rates are around 30 fps this would give a model flow rate approaching Mach 0.5. At these velocities compressibility comes into play, and the similitude criteria do not hold

Therefore, we find that Scaling Compromises can give good results for demonstrating 1-g performance, however, it is impractical to use them to demonstrate anticipated 0-g performance in the Space Shuttle.

A different situation arises when strict "temperature preservation" method is used. With this method the main problem is to find a suitable gas which will meet the criteria

$$Kg^* = L^* \text{ and } Pr^* = 1$$

Once this is done the Reynolds and Grashof numbers can be adjusted by varying the flow velocities and pressures.

During the study it was found that sulfur dioxide (SO_2) could be used if the scale ratio was $L^* = 1/2.5$. With this scale ratio and sulfur dioxide strict temperature preservation could be maintained in a one-g test if the velocity ratio was reduced to $V^* = 0.575$ and the pressure ratio increased to $P^* = 1.65$, both obtainable values.

In addition zero-gravity performance could be simulated by reducing the model pressure so that $P^* = 0.21$ and increasing the velocity ratio to $V^* = 2.8$.

Similar conditions can be found using Freon 11 and a scale ratio of $L^* = 1/3$. However, Freon 11 has some problems associated with its use.

VELOCITY SCALE RATIO VERSUS DIMENSIONAL SCALE RATIO FOR ZERO g SIMULATION

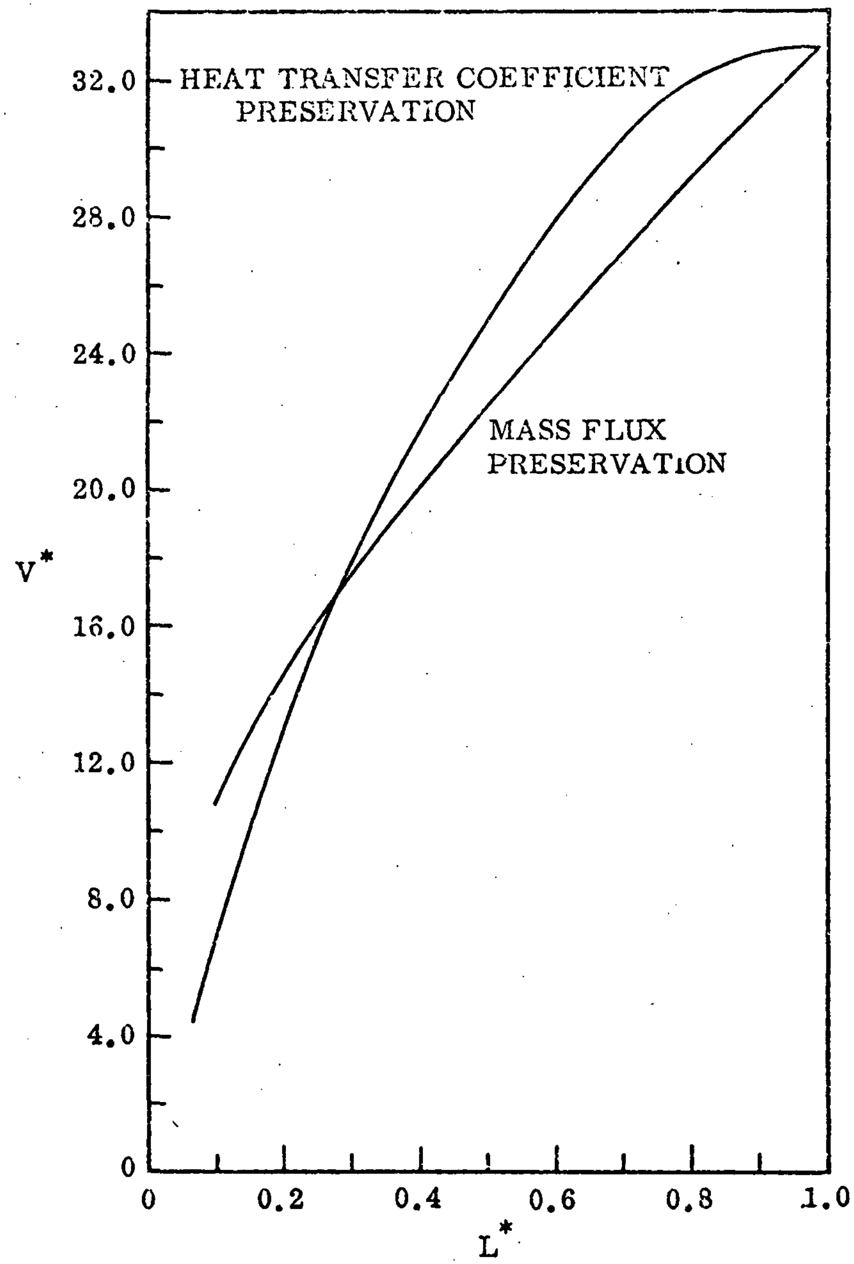


Fig. 5.3

In conclusion the convective studies showed that all three methods "Mass Flux", "Heat Transfer Coefficient" and "Temperature" preservation could be used with good results to simulate a 1-g combined forced and free convection prototype. However, only the Temperature Preservation method would be practicable for 0-g simulation.

Section 6

TSM TEST REQUIREMENTS

An important objective of the study was to define the requirements for testing the thermal scale model once the design had been completed. Preliminary test requirements were established and are presented in Appendix B.

6.1 Solar Simulation

At the beginning of this program, the thermal control surfaces proposed for a major portion of the Shuttle Orbiter's external surfaces had values of solar absorptance and infrared emittance that gave an α_s/ϵ of near unity. However, the crew compartment windows and the deployable radiators have α_s/ϵ values that are different from the other surfaces and are not unity, and in addition, it is desirable to test the model under conditions where the payload bay doors are both open and closed. Considering these factors and the possibility that a change to a low α_s/ϵ thermal control surface for the TPS may be made as the Shuttle design progresses, it is highly desirable that the incident solar heat flux on the model duplicate the solar energy spectrum.

The use of infrared lamps for simulating the external incident heat flux was considered but not thoroughly investigated. Applying this technique requires either duplication of adsorbed energy into the surfaces using analytical methods as a basis for determining the heat flux required, or it requires that the external surface temperatures be controlled to analytically predicted levels during the test. Both of these methods present limitations in testing of the thermal model. The use of infrared lamps for this application should not be entirely ruled out, however, further study is required to completely evaluate the problems associated with their use.

6.2 Test Facility

Due to the size of the Shuttle vehicle, the desirability to design a thermal model with a scale ratio of near 1/3, and the potential necessity for testing the model using exact solar simulation, NASA/MSC Chamber A test facility would be required for the test program. The present solar simulator in Chamber A is too small for testing a 1/3 scale model, and thus, would have to be modified to meet the requirements of this program. There are two possibilities that could be considered for modification of the solar simulator. The first would be to increase the total number of lamps to provide a side sun of 40 ft. by 26 ft. and a top sun of 26 ft. in diameter. The second approach would be to re-arrange the present lamps bank system in a modified triangular shape as shown on pages B-32 and B-33 in Appendix B. This shape corresponds to the general shape of the shuttle orbiter.

6.3 Test Requirements

A list of preliminary test requirements for the TSM program is presented on page B-34 of Appendix B.

Section 7

POTENTIAL APPLICATIONS OF THE TSM

The primary advantage of using a physical scale model as described herein is that it would provide test data on both a system and subsystem level during the early design stages in order to guide the thermal design and to assure that an optimum design was accomplished for the entire shuttle orbiter. In addition, final verification of the thermal design could be achieved on a complete system level. A list of potential applications for the TSM is presented on page B-36 of Appendix B.

Section 8

CONCLUSIONS

Due to termination of the contract before its completion, all the objectives could not be achieved. However, from the results obtained during the initial studies that were completed, the following can be concluded:

1. A 1/3 scale model can be designed that models the conduction, radiation, and convective heat flow fields for steady state conditions.
2. The basic modeling approach to be followed would be that of temperature preservation. Some distortions in the scale ratio would be required due to limitations in available materials, however, the major heat flow paths and important temperature fields would be preserved.
3. The thermal math model proved to be a useful tool in establishing the criteria for modeling the conduction heat flow paths. It was concluded that for steady state conditions, one dimensional heat flow perpendicular to the vehicle surface can be assumed in modeling the low conductivity TPS while neglecting the heat flow in the aluminum substrate in this direction. Results from the TMM indicated that the circumferential and longitudinal heat flow is predominately due to the aluminum substrate, and the heat flow in the TPS in these directions can be neglected.
4. Materials are available for modeling the conduction heat paths.
5. The radiation paths would be modeled by preserving geometrical identity of the major surface areas and by using identical thermal control surfaces.
6. Gases are available for modeling the convective fields in the crew compartment area.

7. There would be some problems associated with modeling the transient conditions, however, the extent and ramifications of these problems were not fully explored due to the contract termination.
8. The preliminary studies indicated that it would be desirable to test the model using a solar simulator which closely approximates the solar spectrum. For a 1/3 scale model, the solar simulator in NASA/MSC Chamber A test racility would have to be modified to accommodate a model of this size. The trade off in using other means of applying the external incident heat flux, such as tungsten lamps, was not determined.
9. Several potential applications of the TSM, beneficial to achieving an optimum thermal design for the shuttle, were identified.

Section 9
REFERENCES

1. K. N. Marshall ; W. G. Foster; "Space Shuttle Thermal Scale Modeling Application Study Program Plan" LMSC #D309067, Lockheed Palo Alto Research Laboratory, Palo Alto, California, Aug. 11, 1972.

2. "Space Shuttle Technical Proposal, Volume III," LMSC/D157364, May 12, 1972.

3. R. E. Rolling, "Thermal Scale Modeling in a Simulated Space Environment," N-05-66-1, Final Report for NASA/MSFC Contract NAS 8-1152, Lockheed Palo Alto Research Laboratory, Palo Alto, June 1966.

4. "Space Shuttle Thermal Scale Modeling Application Study Proposal, Volume I" LMSC/D082269, May 3, 1972.

5. R. L. Shannon; "A Thermal Scale Modeling Study for Apollo and Apollo Applications," D180-15048-1, Boeing Co., Seattle, Washington, June 1972.

APPENDIX A

ANALYSIS AND STUDY PLAN

The Analysis and Study Plan was prepared to describe the approach to be used in accomplishing Task 2. It was in the process of being prepared when the Space Shuttle TSM contract was terminated. Since it was not submitted, it is included here as Appendix A.

ANALYSIS AND STUDY PLAN

NAS 9-12991

1.0 INTRODUCTION AND SUMMARY

The overall objective of the scale modeling application study is to assess the capabilities and sensitivities of a Space Shuttle thermal scale model (TSM) to support the integrated thermal control design, define a test article and a preliminary test plan, and to provide cost and scheduling information for final design and fabrication of the test article. The study consists of five major tasks as follows:

Task 1: Shuttle Vehicle Design Review and Study Plan
Task 2: Thermal Scale Model Design
Task 3: Preliminary Test Plans
Task 4: Shuttle TSM Cost and Schedule
Task 5: Study Reports

The primary activity on the program to date has been on Task 1, which is now complete with the issuance of this document. Task 1 was involved with preliminary studies to define the role of a TSM in the overall Shuttle integrated thermal design and verification plan. The activities under Task 1 consisted of: 1) a complete review of the Phase B Shuttle Orbiter configuration and systems; 2) development of a thermal mathematical model (TMM) for the prototype; 3) view factor and heat rate calculations; 4) development of preliminary TSM design and test requirements; and 5) preparation of an analysis and study plan for use in Task 2.

The analysis and study plan presented herein describes the approach to be used in accomplishing Task 2 which is the detailed design of the TSM. Reference 1, which presented the program plan for the entire contractual effort, and provides an overall schedule and WBS to be used for all Tasks including Task 2. The present document deals only with Task 2 and describes in detail each of the subtasks to be accomplished for Task 2. In addition, the major results of Task 1 studies are presented. These include 1) a listing of the important thermal control areas and their temperature requirements, 2) TSM design criteria 3) preliminary TSM test requirements, 4) Level of detail for modeling the various subsystems, 5) a tentative list of trade studies to be performed, and 6) a list of potential applications for the TSM.

2.0 TASK 2 WORK BREAKDOWN STRUCTURE AND SCHEDULE

The work breakdown structure for Task 2, showing the major subtasks and their interrelationships, is presented in Fig. A-1. A schedule for Task 2 activity is presented in Fig. A-2. Each subtask is described in detail in the following section of this document.

3.0 THERMAL MATH MODEL (TMM) REFINEMENT AND CALCULATIONS

A TMM of the prototype Shuttle Orbiter was developed during Task 1. In developing the TMM, the approach followed was to construct a TMM in sufficient detail to support this study. Thus, a course nodel network was constructed for purposes of defining important heat flow paths and temperature fields throughout the vehicle. A detailed TMM, typical of that required for vehicle thermal design, is not required for this study and is certainly beyond the scope of this contractual effort. Background information and details of the TMM development for this program may be found in Reference 1.

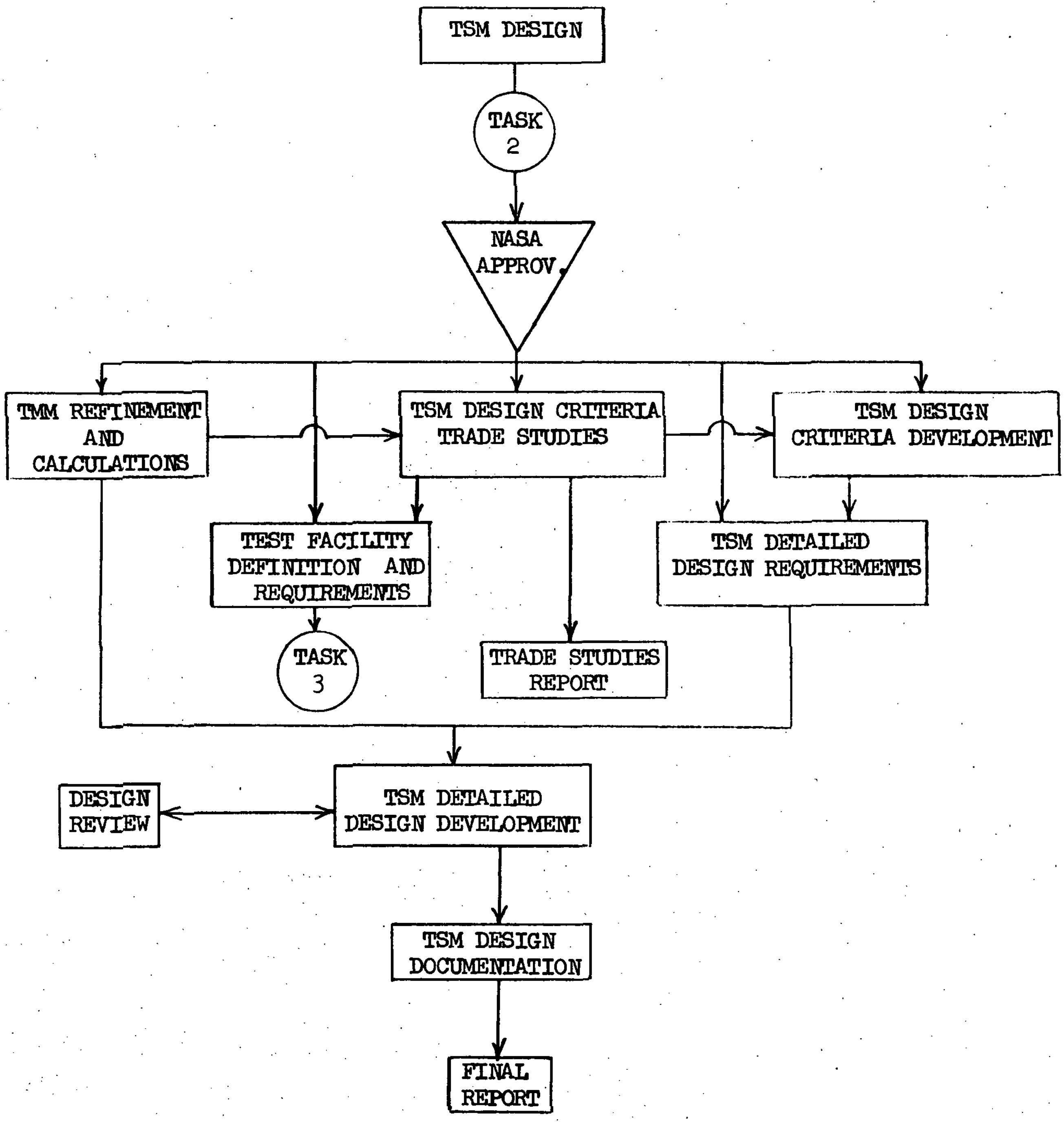


Figure A-1 Task 2 Work Breakdown Structure

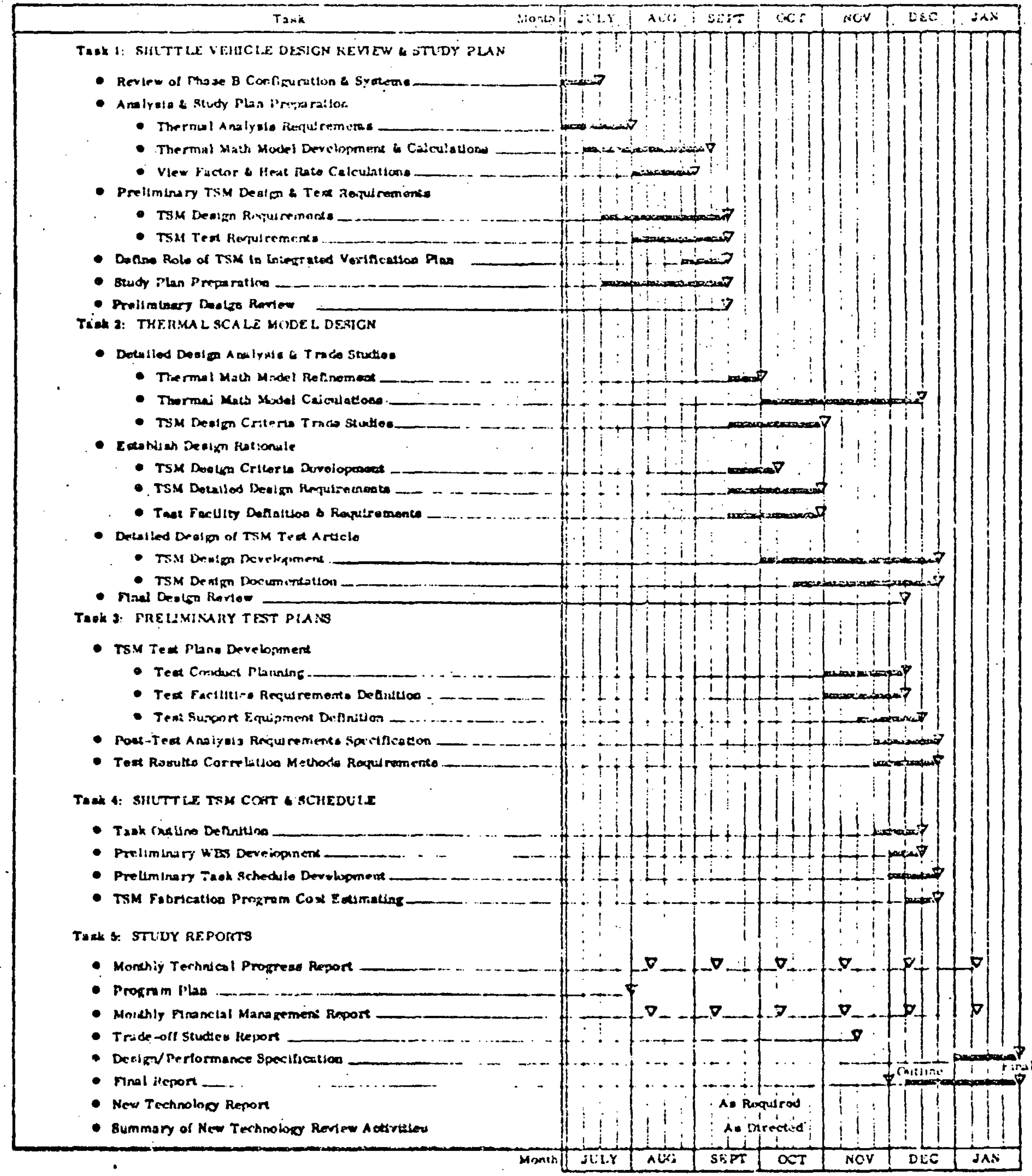


Fig. A.2 Program Schedule

During the initial stages of Task 2, the TMM will be refined to take into consideration further details of the Space Shuttle thermal design and changes resulting from the Task 1 studies. Computer calculations will be performed using the up-dated TMM to study steady-state and transient vehicle response for the extreme hot, cold, and worst transient orbital cases as defined in Ref. 1. These analyses will form the basis for detailed TSM design. Specific refinements in the TMM will include,but not be limited to the following:

a) The revised set of $\bar{t}$ for the aluminum skin structure received from NASA/MSC on August 25, 1972, will be incorporated into the TMM provded that hand calculations show these changes may have significant affects.

b) Additions to the TMM in the crew cabin region, with specific considerations given to modeling window effects, will be added as required to accomplish TSM design.

c) Heat generated from internal sources will be incorporated in the TMM.

d) A cylindrical shaped payload with properties as agreed upon by NASA and LMSC will be incorporated into the TMM.

e) Changes will be made to include multilayer insulation on the interior walls of the payload bay region.

To aid in TSM design and to provide a means of comparing scale model results with prototype results, a TMM of the TSM will be developed. Computer analysis will be conducted as the TSM design progresses to support decisions regarding any deviations from exact scale modeling that may become necessary.

4.0 IMPORTANT THERMAL CONTROL AREAS AND TEMPERATURE REQUIREMENTS

Various important thermal control areas for the Shuttle Orbiter were identified during Task 1. These areas are listed along with their temperature requirements as follows:

1) Crew compartment ($T_{DEW\ Max}$ to $113^{o}F$)
2) Payload bay ($-100^{o}F$ to $150^{o}F$)
3) Landing gear and tires ($-65^{o}F$ to $350^{o}F$)
4) Avionics ($-65^{o}F$ to $160^{o}F$)
5) Elevon and rudder actuator system hydraulics ($-65^{o}F$ to $250^{o}F$)
6) Propellants ($50^{o}F$ to $90^{o}F$)
7) Payload/Orbiter thermal exchange
8) Radiators
9) External insulation bond line ($-250^{o}F$ to $350^{o}F$)
10) Structure temperature prior to entry ($100^{o}F$ max)
11) Auxiliary power units (APU)
12) Fuel cells
13) Transients for various subsystems
14) Cryogenic tanks
15) RCS thrusters

5.0 TRADE STUDIES

A trade study as defined for this program is any study in which one approach is evaluated against another for purposes of establishing an optimum thermal scale model for utilization in Shuttle thermal design. Trade studies will be performed to establish the degree of complexity required for the model design, test facility requirements, and a testing program that best meets the needs of the overall shuttle thermal design requirements. The objectives of the trade studies are threefold: (1) to provide a guide to the model designer in terms of final selection of scale ratio, materials, scaling sensitivities for various thermodynamic features, level of detail for scaling systems and subsystems, and defining the degree to which distortion in temperature fields and compromises in eact modeling can be tolerated; (2) to determine whether the model can be used as a thermal testbed for evaluating certain materials and thermal-control systems proposed for use on the prototype; and(3) to determine the proposed complexity of the model design and test program in terms of the effect on manufacturing difficulties and costs involved, as well as the impact on the overall program schedule.

The TMM will be utilized during the trade studies largely to determine the impact of distortion of various heat flow fields. The TMM will show which thermal paths are critical and which can be distorted without affecting vehicle performance.

6.0 TSM DESIGN CRITERIA DEVELOPMENT

Before starting final TSM design, definitive design criteria will be established to provide firm guidelines for the model designers. The criteria will be developed from the analyses and studies performed during Task 1 and from initial results of the trade studies. The criteria will include definition of overall modeling approach (e.g., temperature preservation, materials preservation, etc.) overall scale ratio, level of detail for modeling systems and subsystems,and the approach to use in modeling the various radiation, conduction, and convection heat flow paths. Design criteria established during Task 1 are as follows:

1) Use "temperature preservation" approach, $T^* = 1$, for conduction-radiation modeling.
2) Study both pure "temperature preservation" and "scaling compromises" approaches for convection modeling.
3) Overall length scale ratio to be between 1/2.5 and 1/4.
4) Assume one-dimensional modeling for external insulation and internal MLI wherever possible.
5) Assume two-dimensional modeling for the aluminum skin structure.
6) Use identical optical properties between model and prototype for both internal and external surfaces.

7) Geometrical identity between model and prototype for all major external surfaces.

8) Heat exchangers and fluid loops will not be modeled.

9) Radiators to be modeled with simulated heat load applied.

10) Wings and vertical stabilizer extremities will be deleted or heated with infrared energy as required so that model will fit within solar simulation capability.

11) Cylindrically shaped payload will be included in design studies per NASA/LMSC agreement.

12) With the exception of the crew cabin, active thermal control systems will not be modeled, however, their effects on various temperature fields will be considered in TSM design.

7.0 TSM DETAILED DESIGN REQUIREMENTS

Throughout the initial stages of Tank 2, specific requirements will be established for design of the TSM. These will include such items as requirements for structural integrity of the pressure shell and other structural areas and requirements for modeling crew cabin windows and other discrete sections of the vehicle.

8.0 TSM DETAILED DESIGN DEVELOPMENT

Detailed design of the TSM will be accomplished in accordance with the design criteria outlined in Section 6 and in accordance with the standard modling laws developed for the temperature preservation approach. During this design effort, modeling of the three basic modes of heat transfer--conduction, radiation, and convection -- will be considered. Deviation from strict adherence to the exact modeling laws for convection and for certain conduction paths may be necessary and is discussed in 8.1 and 8.2.

The basic approach to be followed in design of the TSM is to select a scale ratio which appears to satisfy the majority of the model's requirements and then proceed to model all areas of the vehicle to that scale ratio. Scaling compromises will be made for those areas of the vehicle that cannot be modeled exactly.

Results from computer runs of the prototype TMM will be used to guide the model design. In addition, a TMM of the scale model will be developed for purposes of evaluating the affects of scaling compromises and for comparing TSM and prototype temperatures. Comparison of results from the two math models will provide the means by which the model design is judged on its ability to predict thermal behavior of the prototype.

The end result of the model design activity will be the specification of materials, size, shape, weight, internal power sources, external heat flux requirements, details of critical thermal conduction joints, details for modeling the crew cabin atmosphere, and details of modeled components and their location relative to each other. A complete set of specifications and engineering drawings will be produced to document details of the TSM design. These drawings will conform to Category A, Form 3 of Specification MIL D-1000 and MIL STD-100.

8.1 Conduction-Radiation Modeling

In modeling the conduction and radiation heat flow paths, adherence to the modeling criteria of $K^* = L^*$, $I^* = 1$, and $\theta^* = \rho L^* C^*$ will be complied with wherever possible. Due to the limitations in available materials, it is anticipated that in modeling certain conduction paths, distortion of the thickness ratio may be required to model the overall heat flux. Where distortion of the conductive path is required, it will be done in such a manner that the radiant exchange between components is not affected.

In modeling conduction through the external insulation (i.e. LI-1500) and aluminum skin, certain assumptions have to be made due to the combination of materials involved which distort the thickness scale ratio. Based on results derived from the TMM during Task 1, the assumption can be made that heat flow in the LI-1500 is one-dimensional (i.e., perpendicular to the skin surface) and that heat flow in the aluminum substrate is two dimensional (i.e. circumferential and longitudinal in the aluminum skin). During the detailed design phase, additional analysis will be made to determine the validity of this assumption for all external surface areas of the vehicle.

In modeling the interior portions of the Orbiter, it is important for the TSM to have the ability to demonstrate the temperature environment in critical thermal control areas and around temperature critical systems such as fuel lines, fuel tanks, and hydraulic lines.

8.2 Convective Modeling

During Task 1 it was shown that thermal modeling of the pressurized sections of the space shuttle is feasible from a theoretical point of view. It will be possible to obtain good correlation between the one-g test of the mock-up and the one-g test of the model. Three methods can be used in the model test, each of which has certain advantages and disadvantages. The three methods are: 1) scaling compromise using heat transfer coefficient preservation, 2) scaling compromise using mass flux preservation, 3) strict temperature preservation.

The first two of these methods permits the use of any scale ratio for the model. In addition, both heat transfer coefficient preservation and mass flux preservation can give accurate results for a combined forced and free convection test such as will be done on the full size mockup. The third method requires selection of a suitable gas and scale ratio combination. However, if such a combination can be found then not only can the one-g performance be compared with the full size mockup tests, but in addition zero-g simulation test will be possible. This will give accurate data as to the thermal performance which can be expected in orbit. Two possible gas-scale-ratio combinations are sulfer dioxide with a $L^* = 1/2.5$ and Freon 11 with a $L^* = 1/3$.

The conclusions reached during Task 1 are that accurate scale modeling can be accomplished on the crew section of the space shuttle. If the right scale ratio is chosen three different approaches can be tested making the model an extremely useful tool as a TMM verifier and as a zero-g performance demonstrator. The one question which remains is whether or not the modeling of the pressurized areas can be accomplished with enough accuracy and with reasonable costs.

During Task 2 two approaches formodeling the convective systems will be evaluated. The first will be to evaluate the feasibility of modeling the pressure shell. This feasibility study will include scale ratios and materials and compromises where needed. It may well be that the scale ratio chosen for the overall model may not suit the modeling of the pressure shell. Such compromises must be evaluated in order to determine the feasibility of the model.

The second approach will be to determine how the model can be constructed to act as a thermal systems tool. This will include access to the interior and the level of detail of the interior components such as hard mounts and equipment racks.

During Task 2 the convective system will be modeled but the detail and effort will depend on the trade studies outlined above which will be conducted simultaneously.

8.3 Level of Detail for Modeling Systems

During Task 1 an assessment was made regarding the level of detail required for modeling the various systems and subsystems within the Shuttle Orbiter. As design of the model progresses and an analysis provides better definition of thermal behavior, it may be necessary to change the level of detail for some of the components. The following list presents the major systems and a brief statement about the level of detail to which they will be modeled as presently defined:

1) Crew compartment: The cabin atmosphere and internal equipment will be modeled to the extent discussed in Section 8.2.
2) Avionics: The avionics in the forward section are convectively cooled by their own closed cycle air supply and as such, the details of this section will not be modeled. The heat that the avionics may supply to surrounding structure and to other areas of the vehicle will be scaled in the model.

3) <u>Landing gear and tires</u>: This system will be modeled in detail to provide in the model a means of duplicating the thermal environment of the prototype.

4) <u>RCS and OMS</u>: These systems will not be modeled in detail, however, their mass specific heat and the heat that they provide to their surroundings will be scaled in the model.

5) <u>Payload bay</u>: This section will be modeled in detail.

6) <u>External and internal insulation</u>: One-dimensional effects will be modeled very closely. Two-dimensional effects will be studied and scaled where important.

7) <u>Skin structure and supporting structure</u>: These members will be modeled in detail, however geometric similarity may not be possible to achieve due to material limitations.

8) <u>External and internal thermal control surfaces</u>: The optical properties between model and prototype will be preserved.

9) <u>Wings and vertical stabilizer</u>: The shape and major thermal control characteristics of these sections will be modeled as closely as possible.

10) <u>Hydraulic pumps and lines</u>: These will not be modeled, however the model will be designed such that the temperature environment in which these items are located will be duplicated from prototype to model.

11) <u>Miscellaneous internal power sources</u>: The thermal energy dissipated from these sources will be scaled in the model.

12) <u>Fluid loops, heat exchangers, and radiators</u>: The fluid loops and heat exchangers will not be modeled. The radiators will be modeled in detail, and their heat input will be supplied by attached heating elements.

13) <u>Connecting joints and hinges</u>: Due to lack of information about actual thermal conductance across joints and hinges for the Shuttle, the model will be designed assuming infinite conductance for solid connections and zero conductance for hinged connections.

14) Cryogenic tanks: The extent to which these will be modeled has not yet been decided.

9.0 POTENTIAL APPLICATIONS OF TSM FOR USE IN SHUTTLE THERMAL DESIGN

The most important application of the TSM is that of providing a tool for preliminary and final thermal design studies, however, there are other potential applications that must be considered for optimum utilization of a thermal scale model. During Task 1 studies, a list was compiled of potential applications for the TSM. These and other applications will be studied throughout the remainder of the program, with a major goal of defining the role of the TSM in the overall Shuttle integrated thermal design program. Potential applications of the scale model are as follows:

1) To provide experimental data on both a system and subsystem level during the Shuttle's initial design stages for use in optimizing the orbiters thermal design.
2) Provide information for refinement of the prototype TMM.
3) To aid in design of the crew cabin convective cooling system.
4) To study the performance of the convective cooling in a simulated zero g environment.
5) To determine the effects of external thermal control surface specularity on vehicle temperatures.
6) To study Shuttle/Payload thermal interaction, and to evaluate the adequacy of payload thermal control.
7) To study radiator performance in a total system environment.
8) Test vehicle for external insulation to study performance of the thermal/mechanical design.
9) Multipurpose test object serving as a test bed for subsystem testing, including those systems with active thermal control.
10) Thermal test vehicle for evaluating performance of thermal design changes and for confirmation of overall system final thermal design.
11) Test vehicle for verification of launch pad thermal protection system.

10.0 TEST FACILITY DEFINITION AND REQUIREMENTS

During Task 2, studies will be undertaken to define the test facility requirements for TSM testing. In defining these requirements, the role that the model will play in the overall Shuttle program must be kept in mind. Such items as the use of solar simulation or infrared lamps for external heat flux irradiation will be studied.

APPENDIX B

CONCEPT REVIEW HANDOUT

At the conclusion of Task I a program review of the work accomplished to date was held at NASA/MSC Houston. A handout was prepared and presented at the review. This handout which is included as an appendix gives an outline of the work accomplished, potential model applications, test requirements, and program approach.

APPENDIX B

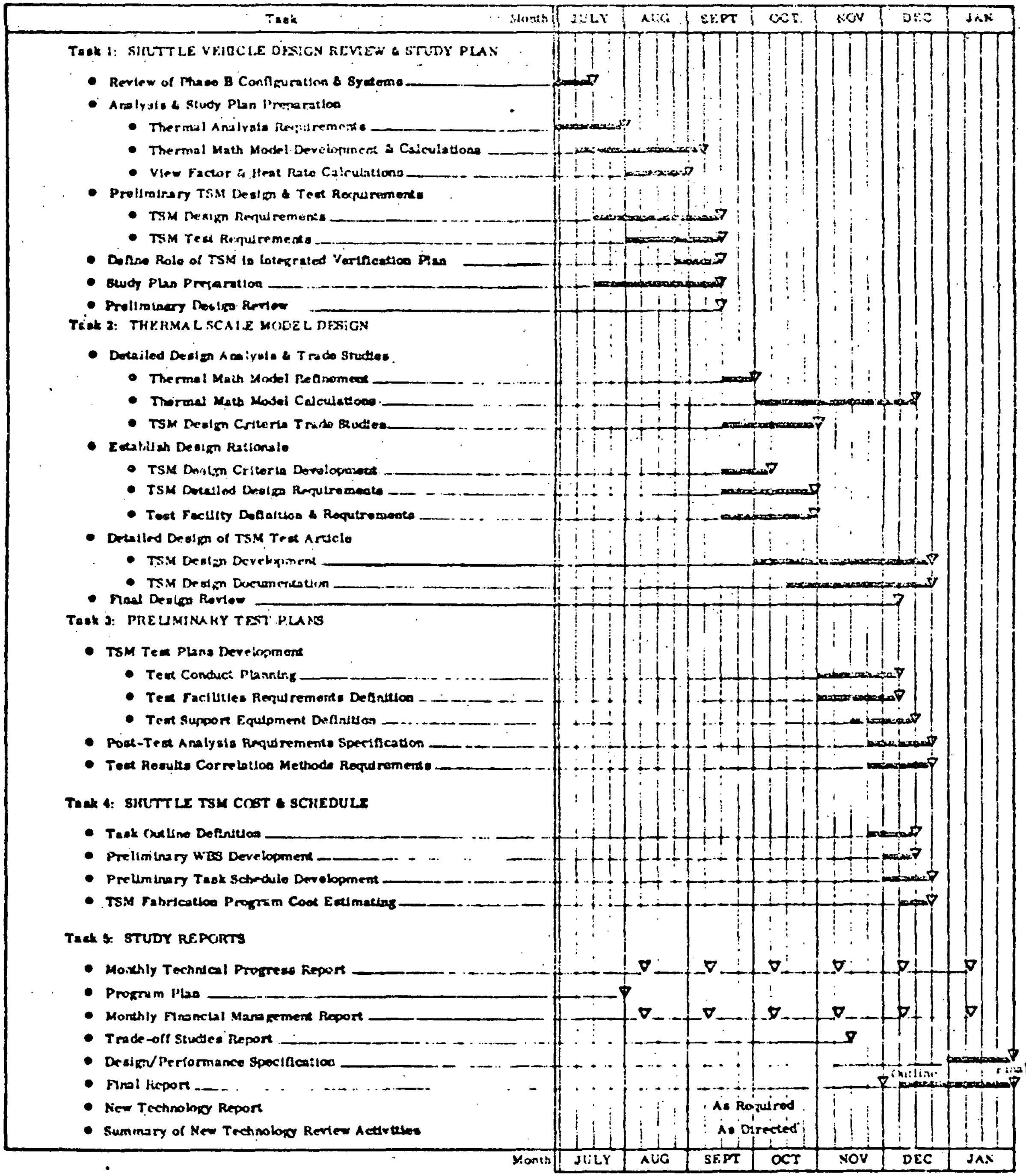
CONCEPT REVIEW
HANDOUT

Presented Under
NASA/MSC CONTRACT NAS 9-12991
SPACE SHUTTLE THERMAL SCALE MODELING
APPLICATION STUDY

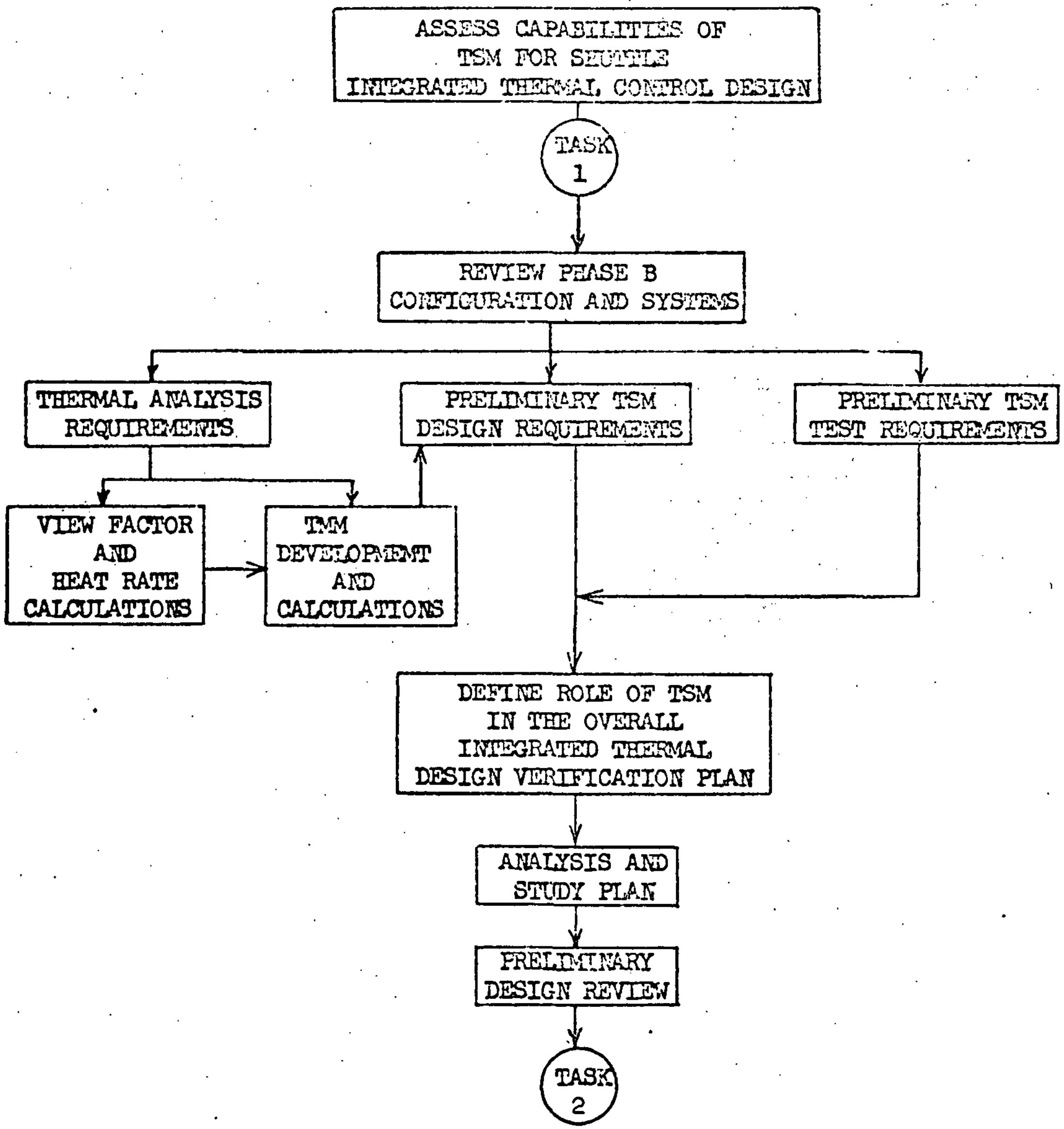
By

LOCKHEED PALO ALTO RESEARCH LABORATORY
LOCKHEED MISSILES & SPACE COMPANY, INC.
PALO ALTO, CALIFORNIA

Program Schedule



TASK 1 Work Breakdown Structure



ORBITAL CASES

Hot Case:

$\beta = 90^{\circ}$ solar oriented with maximum projected area of vehicle top surface toward sun, payload doors closed.

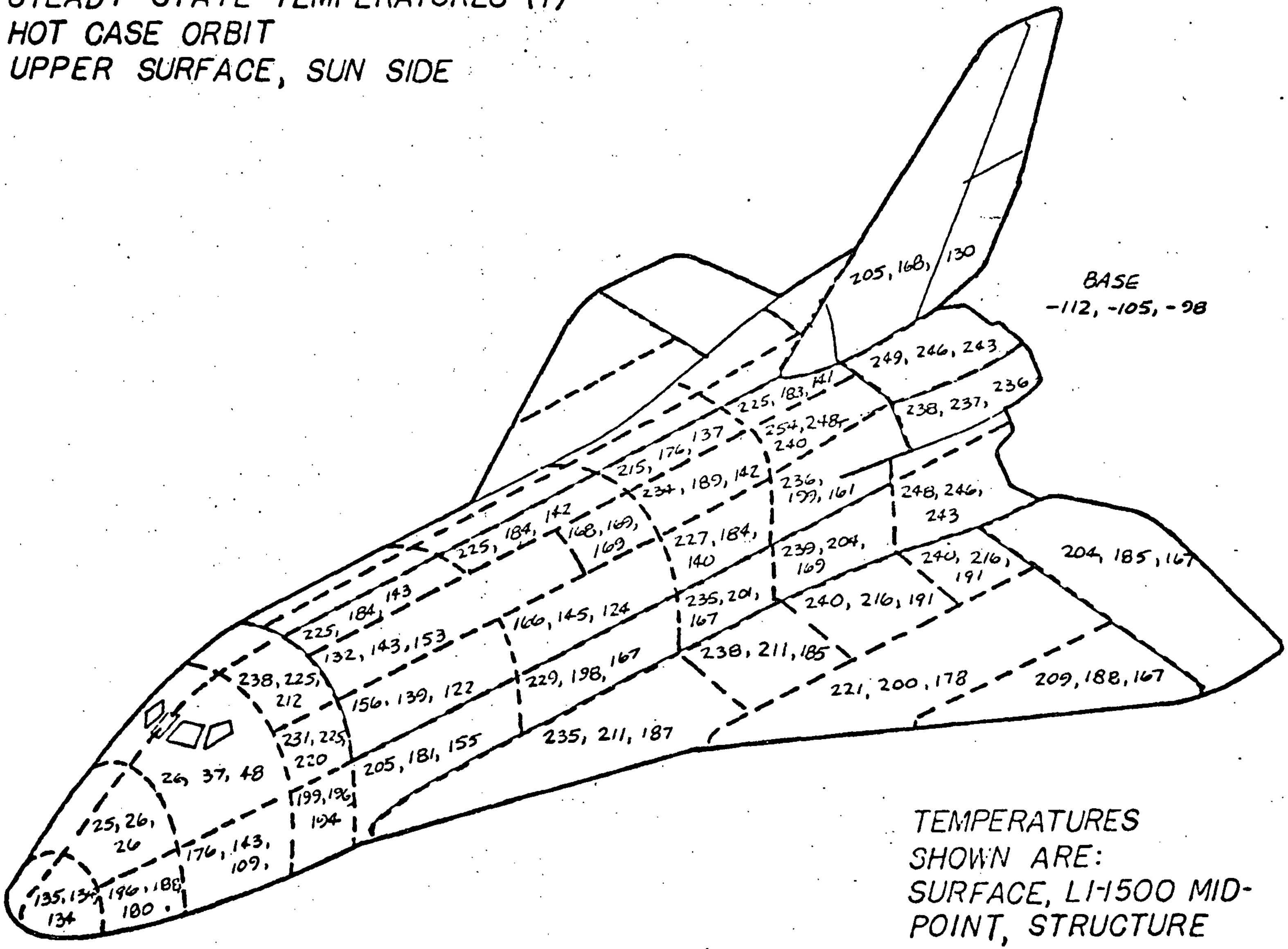
Cold Case:

$\beta = 90^{\circ}$, bottom of vehicle earth oriented, tail facing sun, payload bay doors open.

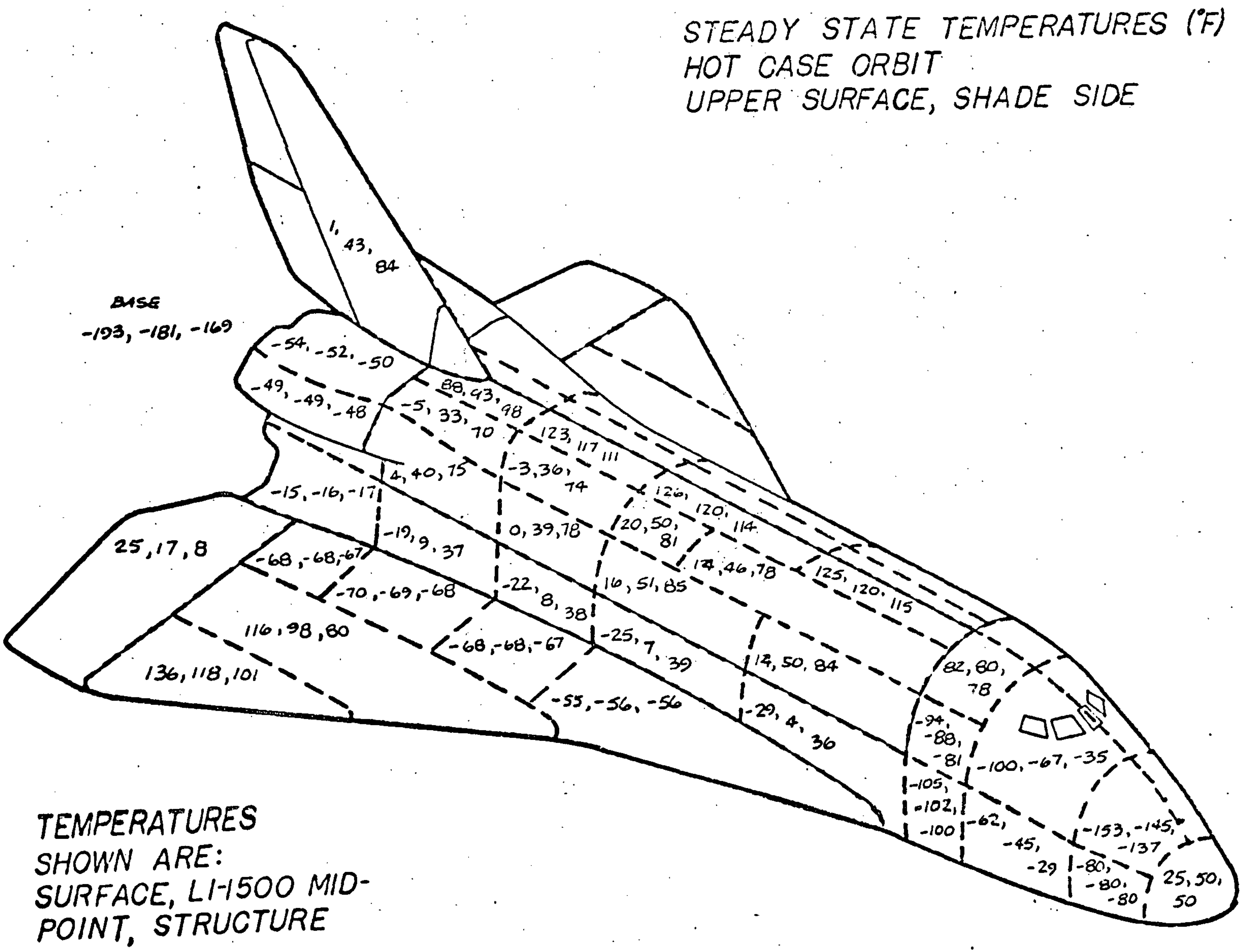
Worst Transient:

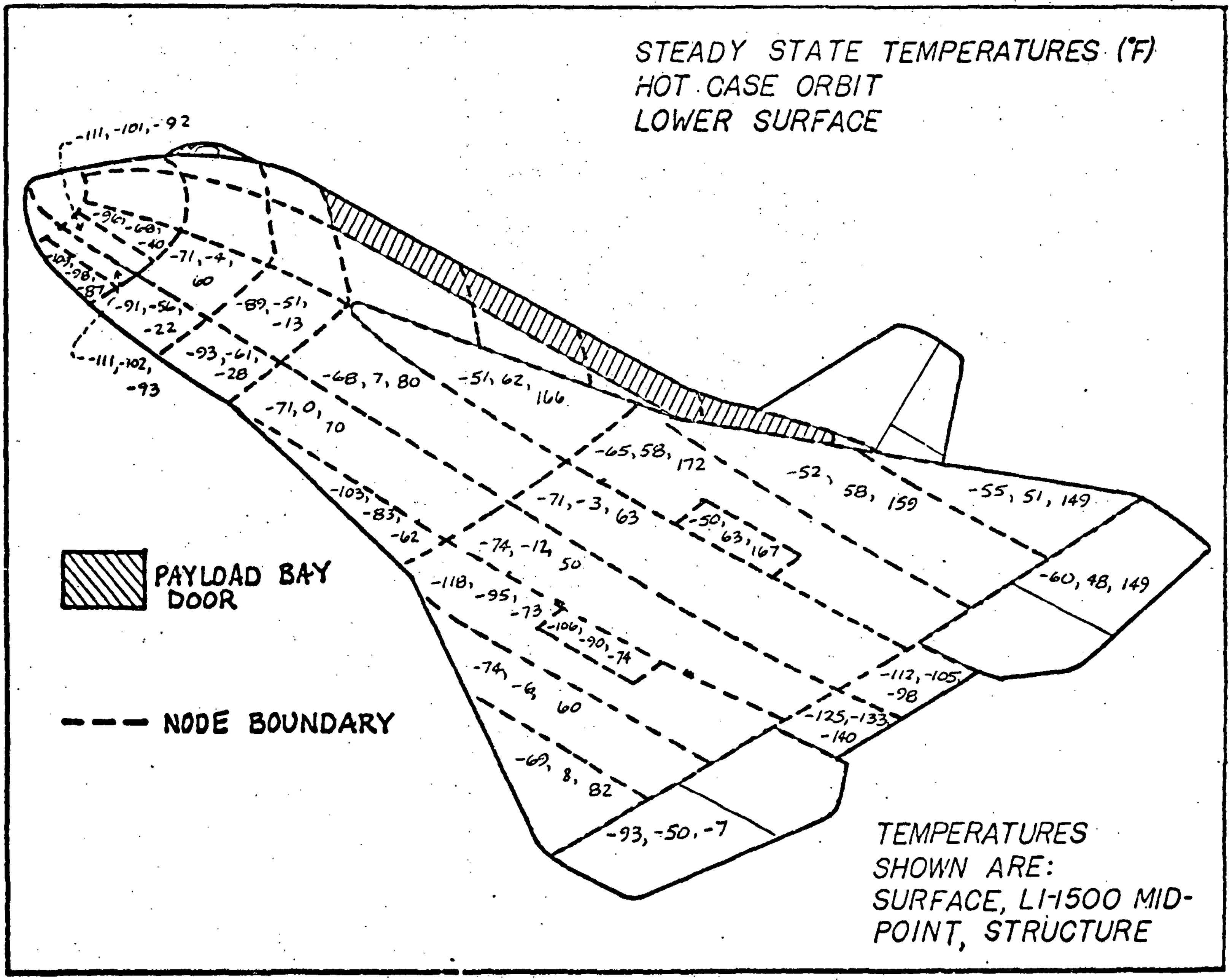
Equitorial orbit, payload bay doors open and facing sun in solar orientation, vehicle to pitch or roll 90°, depending on initial orientation, as it enters earth's shadow and to remain earth oriented with bottom facing the earth while in shadow, then pitch or roll 90° to repeat solar exposure of open payload region.

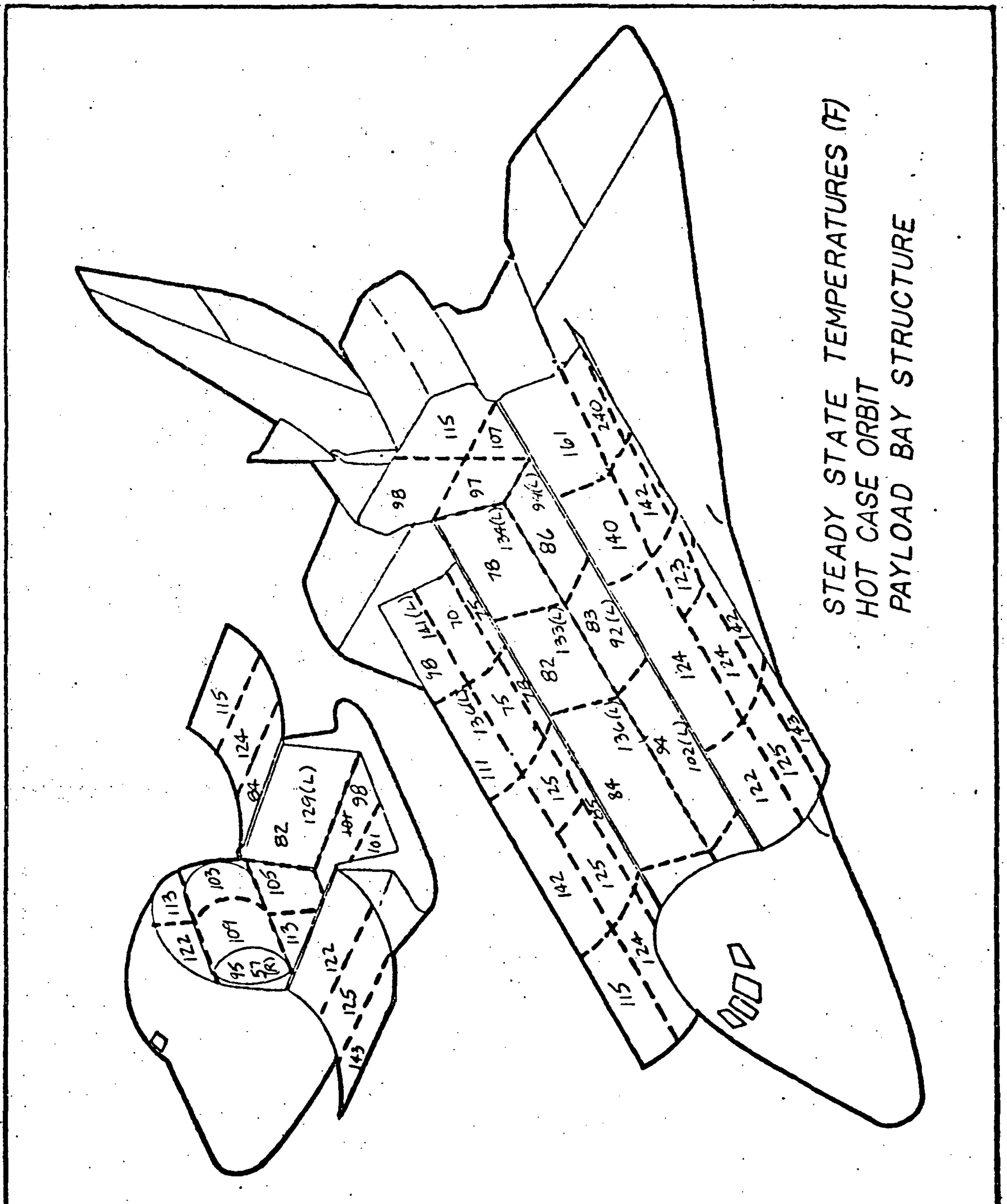
STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
UPPER SURFACE, SUN SIDE



TEMPERATURES
SHOWN ARE:
SURFACE, LI-1500 MID-
POINT, STRUCTURE

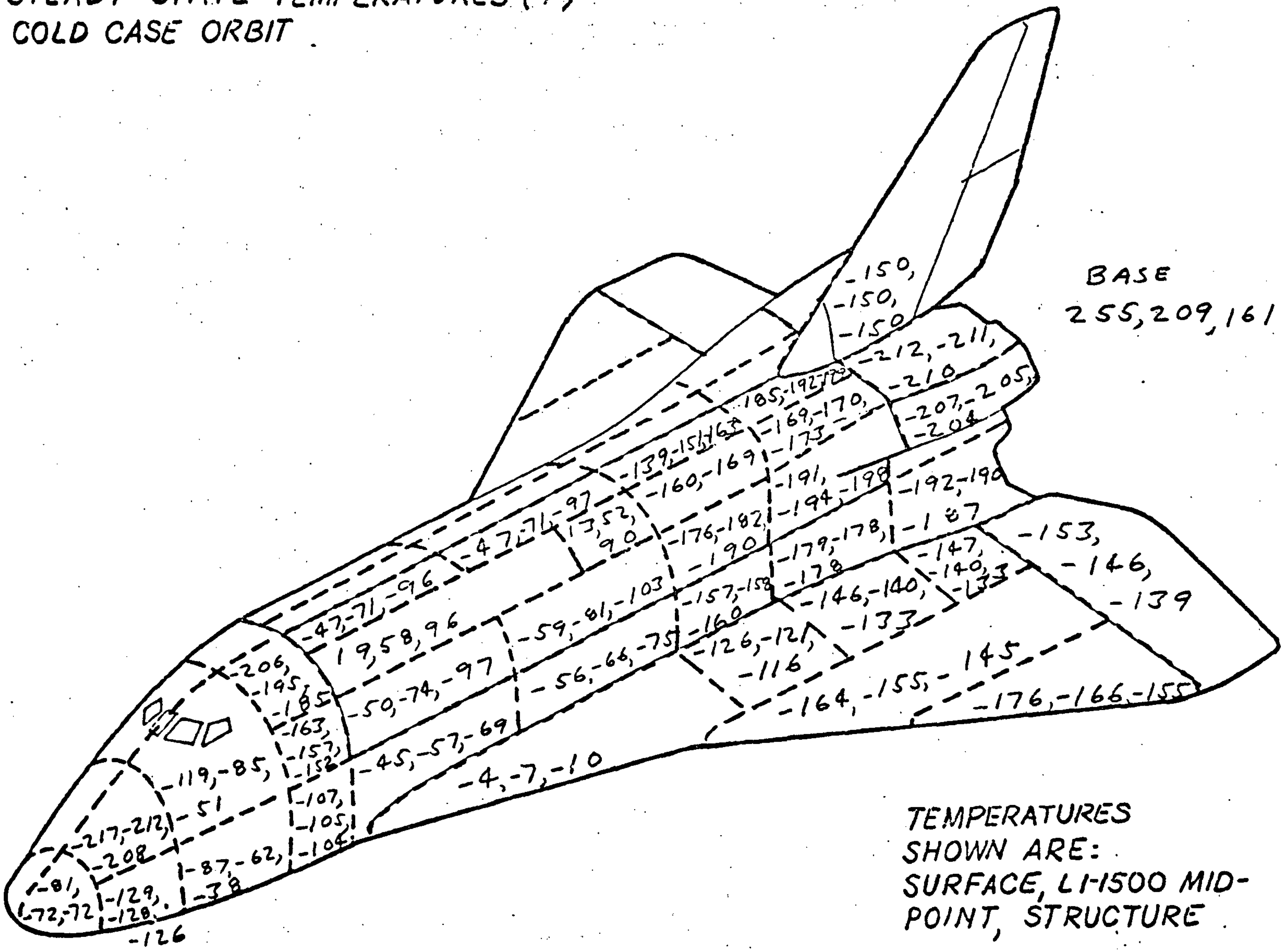

STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
UPPER SURFACE, SHADE SIDE
1, 43, 84
BASE
-193, -181, -169
-54, -52, -50
-49, -49, -48
88, 93, 98
-5, 33, 70
123, 117 111
4, 40, 75
-3, 36, 74
-15, -16, -17
126, 120, 114
20, 50, 81
0, 39, 78
-19, 9, 37
25, 17, 8
-68, -68, -67
-70, -69, -68
-22, 8, 38
16, 51, 85
17, 46, 78
125, 120, 115
116, 98, 80
-68, -68, -67
-25, 7, 39
14, 50, 84
136, 118, 101
-55, -56, -56
-29, 4, 36
82, 80, 78
-94, -88, -81
-100, -67, -35
-105, -102, -100
-62, -45, -29
-153, -145, -137
-80, -80, -80
25, 50, 50
TEMPERATURES
SHOWN ARE:
SURFACE, LI-1500 MID-
POINT, STRUCTURE


STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
LOWER SURFACE
-111, -101, -92
-71, -4, 60
-89, -51, -13
-91, -56, -22
-93, -61, -28
-111, -102, -93
-68, 7, 80
-51, 62, 166
-71, 0, 10
-65, 58, 172
-52, 58, 159
-55, 51, 149
-103, -83, -62
-71, -3, 63
-50, 63, 167
-74, -12, 50
-60, 48, 149
-118, -95, -73
-106, -90, -74
-112, -105, -98
-74, -6, 60
-125, -133, -140
-69, 8, 82
-93, -50, -7
PAYLOAD BAY DOOR
NODE BOUNDARY
TEMPERATURES SHOWN ARE: SURFACE, LI-1500 MID-POINT, STRUCTURE

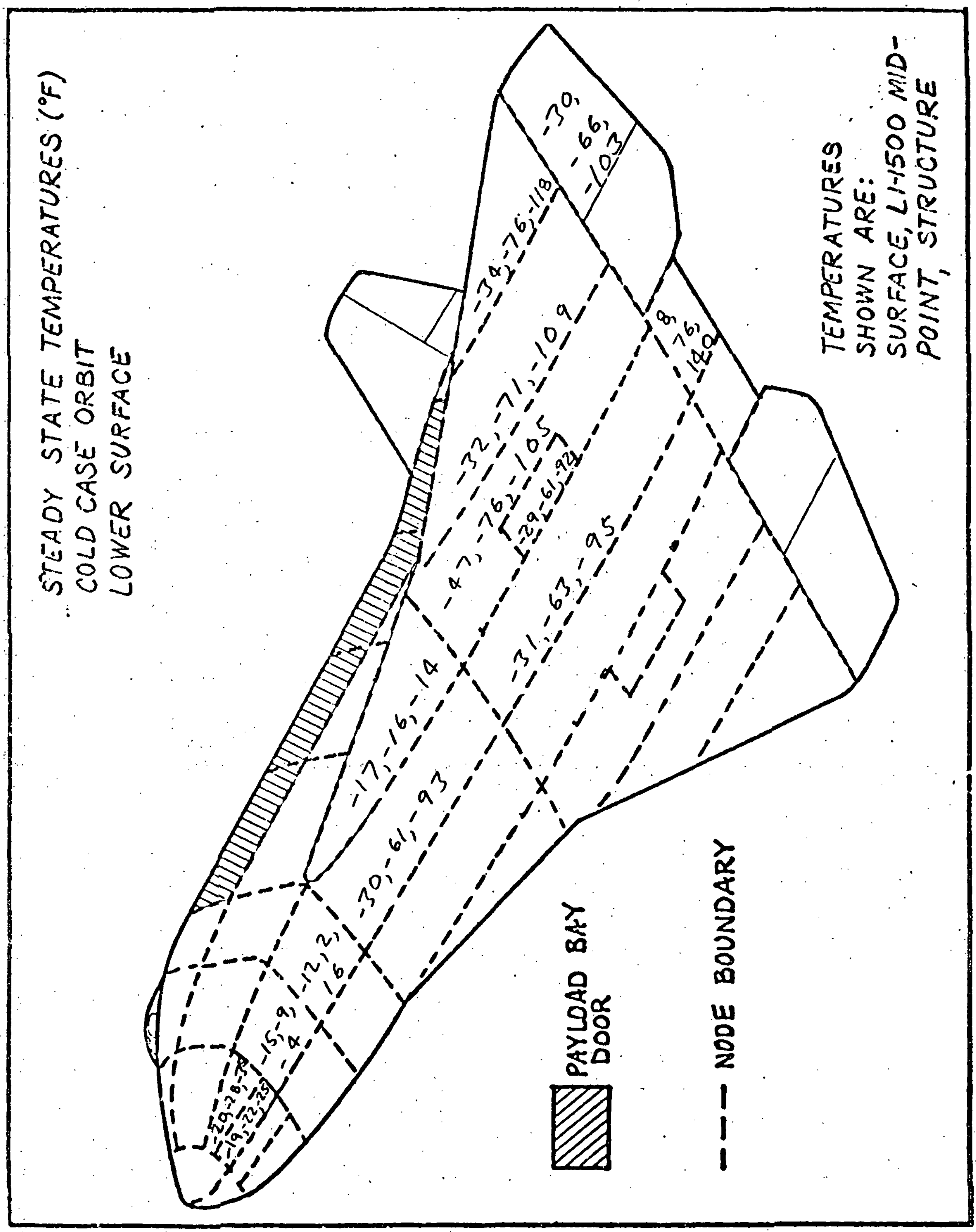


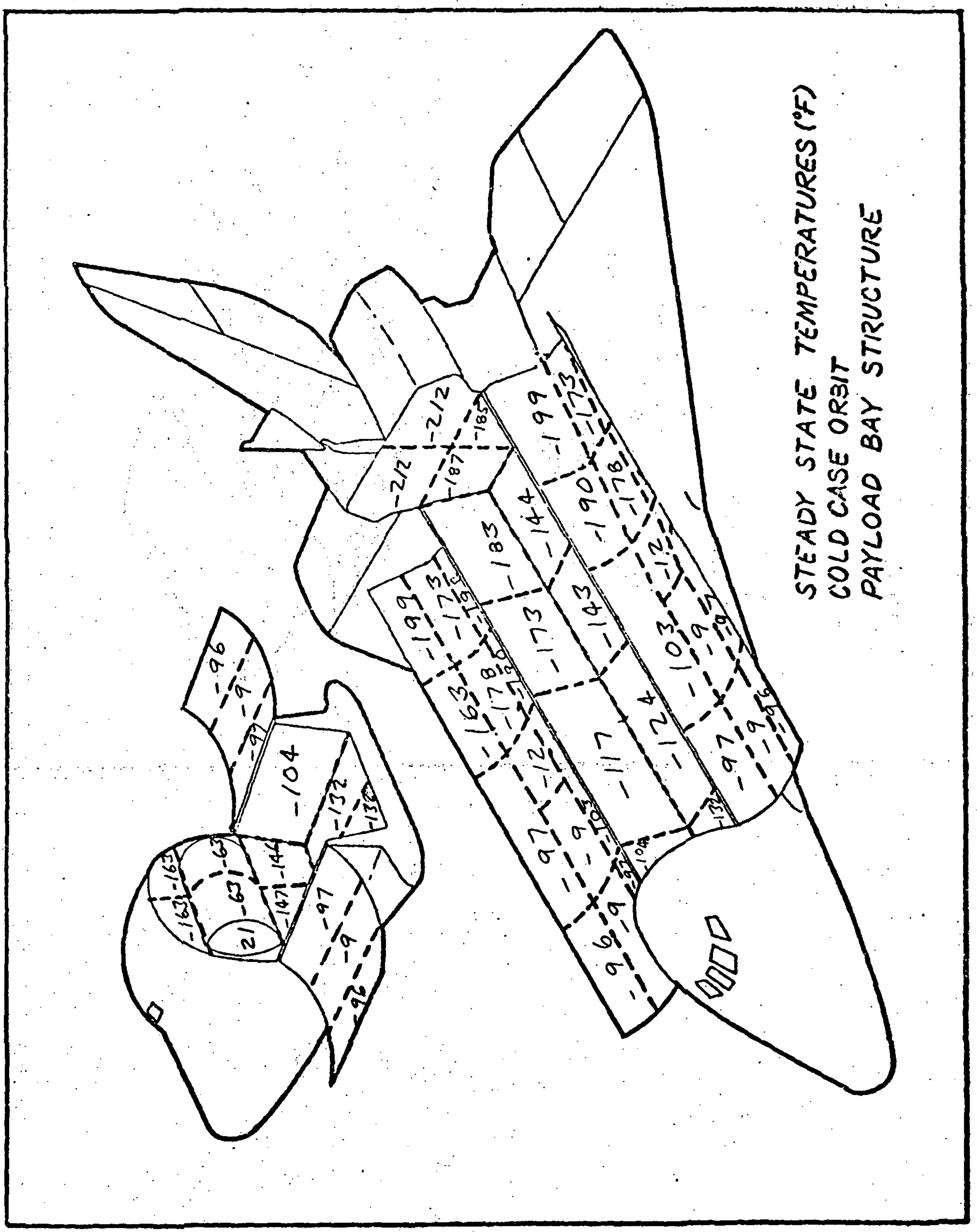
STEADY STATE TEMPERATURES (°F)
HOT CASE ORBIT
PAYLOAD BAY STRUCTURE

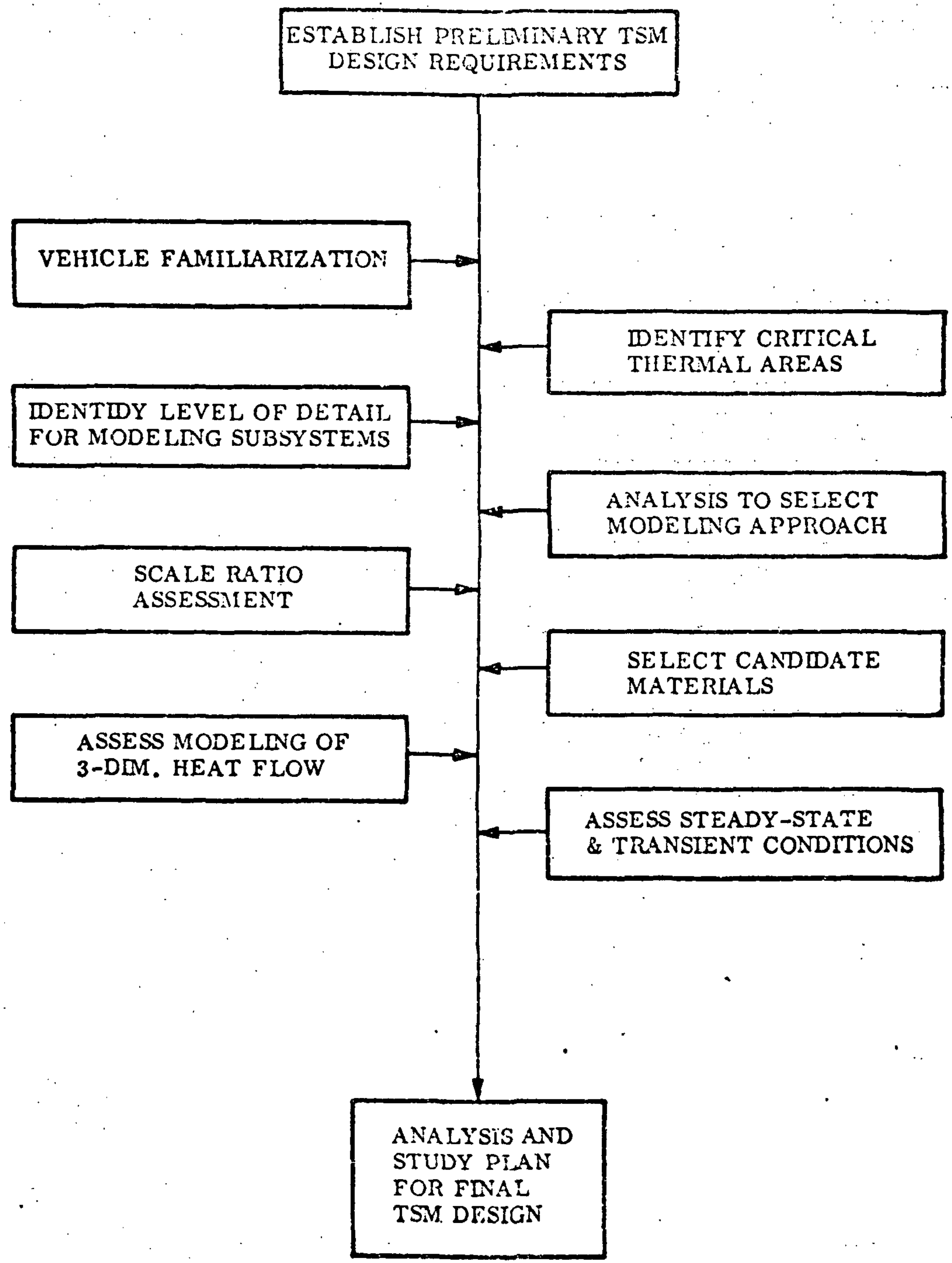
STEADY STATE TEMPERATURES (°F)
COLD CASE ORBIT



TEMPERATURES
SHOWN ARE:
SURFACE, LI-1500 MID-
POINT, STRUCTURE


STEADY STATE TEMPERATURES (°F)
COLD CASE ORBIT
LOWER SURFACE
TEMPERATURES SHOWN ARE:
SURFACE, LI-1500 MID-POINT, STRUCTURE
-30, -66, -103
-34,-76,-118
-32,-71,-109
8, 76, 140
-47,-76,-105
-29,-61,-92
-31,-63,-95
-17,-16,-14
-30,-61,-93
-12,2, 16
-15,-9, -4
PAYLOAD BAY DOOR
NODE BOUNDARY


STEADY STATE TEMPERATURES (°F)
COLD CASE ORBIT
PAYLOAD BAY STRUCTURE


ESTABLISH PRELIMINARY TSM DESIGN REQUIREMENTS
VEHICLE FAMILIARIZATION
IDENTIFY CRITICAL THERMAL AREAS
IDENTIDY LEVEL OF DETAIL FOR MODELING SUBSYSTEMS
ANALYSIS TO SELECT MODELING APPROACH
SCALE RATIO ASSESSMENT
SELECT CANDIDATE MATERIALS
ASSESS MODELING OF 3-DIM. HEAT FLOW
ASSESS STEADY-STATE & TRANSIENT CONDITIONS
ANALYSIS AND STUDY PLAN FOR FINAL TSM DESIGN

IMPORTANT THERMAL CONTROL AREAS AND REQUIREMENTS

1. Crew Compartment ($T_{Dew\ Max}$ to $113^{\circ}F$).
2. Payload Bay ($-100^{\circ}F$ to $150^{\circ}F$).
3. Landing Gear and Tires ($-65^{\circ}F$ to $350^{\circ}F$).
4. Avionics ($-65^{\circ}F$ to $160^{\circ}F$).
5. Elevon and Rudder Actuator System Hydraulics ($-65^{\circ}F$ to $250^{\circ}F$).
6. Propellants ($50^{\circ}F$ to $90^{\circ}F$).
7. Payload/Orbiter Thermal Exchange.
8. Radiators.
9. External Insulation Bond Line ($-250^{\circ}F$ to $350^{\circ}F$)
10. Structure Temperature Prior to Entry ($100^{\circ}F$ max)
11. Radiant Exchange Between Radiators and Other External Portions of Vehicle.
12. Overall Vehicle Heat Balance and Temperature Control.

CONVECTIVE SCALE MODELING FOR SPACE SHUTTLE

Four Sections:

1. Flight Deck (Volume $\approx$ 600 cu. ft.)
2. Crew Compartment (Volume X 1450 cu. ft.)
3. Payload Handling Station (Volume $\approx$ 140 cu. ft.)
4. Airlock(Volume X 230 cu. ft.)

PRESSURIZED COMPARTMENT EQUIPMENT

Flight Deck	1) Flight and Payload Monitor Consoles
	2) Avionics Racks
	3) Seats
Crew Cabin	1) Food Preparation Station
	2) Hygiene Station
	3) Cabinets
	4) Seats
	5) Avionics Racks
Payload Handling and Airlock	1) Storage Racks
	2) Avionics Racks

HEAT LOADS

Avionics $\approx$ 30,000 BTU/Hr

Environment Interchange $\approx$ 6000 BTU/Hr
(Depends on Orientation)

Metabolic $\approx$ 400 BTU/Hr/Man

THERMAL CONTROL

Cabin Air Temperature	65 - 80°F
Cabin Wall Temperature	Above Dew Point to 113°F
Cabin Ventilation	Nominal 25 FPM

CONVECTIVE MODELING TECHNIQUES

1. Scaling Compromises

   a) Heat Transfer Coefficient Preservation

   b) Mass Flux Preservation

2. Temperature Preservation

MODELING CRITERIA

<u>Heat Transfer Coefficient Preservation</u>

$$h_c^* = 1$$

$$Re^*_{L/T} = L^* C^* Pr^*_{L/T}$$

$$Gr^*_{L/T} = L^* C^* Pr^*_{L/T}$$

<u>Mass Flux Preservation</u>

$$(\rho V)^* = 1$$

$$Re^* = L^*$$

<u>Temperature Preservation</u>

$$T^* = 1$$

$$Kg^* = L^*$$

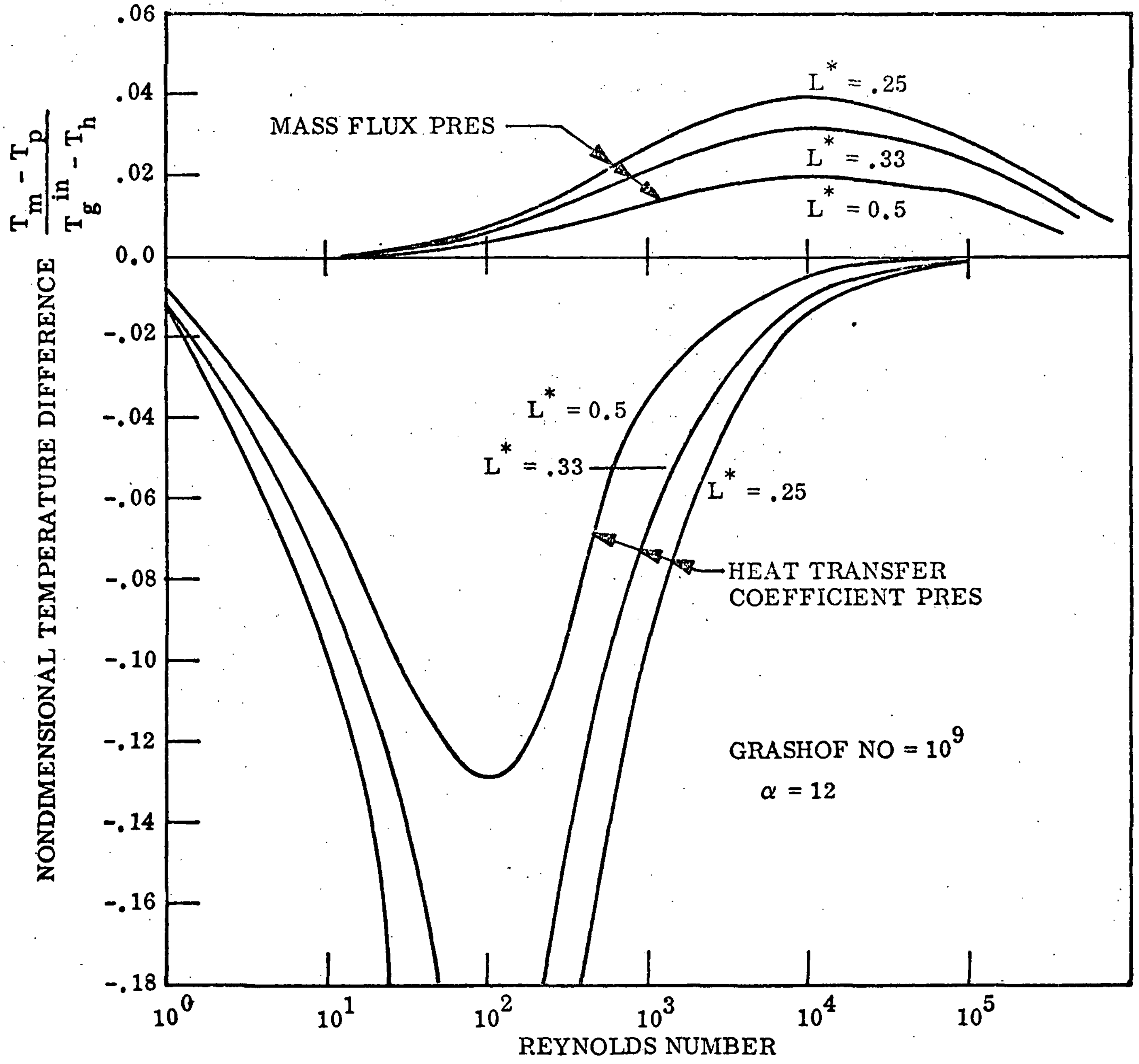
$$Re^* = 1$$

$$Gr^* = 1$$

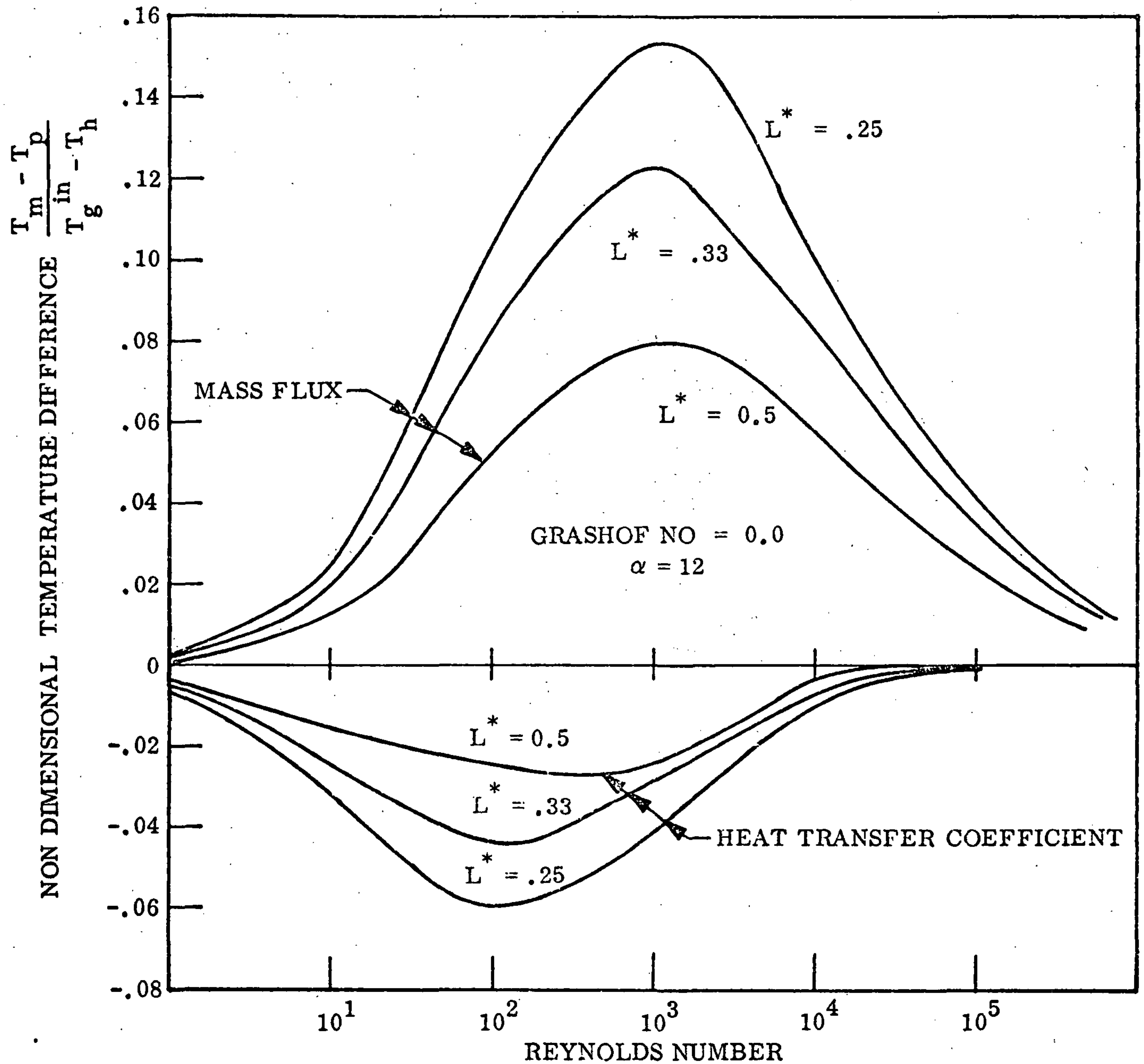
$$Pr^* = 1$$

requires $P^* = \frac{\mu^*}{M^*} \left(\frac{1}{g^* \beta^*}\right)^{1/2} \left(\frac{1}{L^*}\right)^{3/2}$

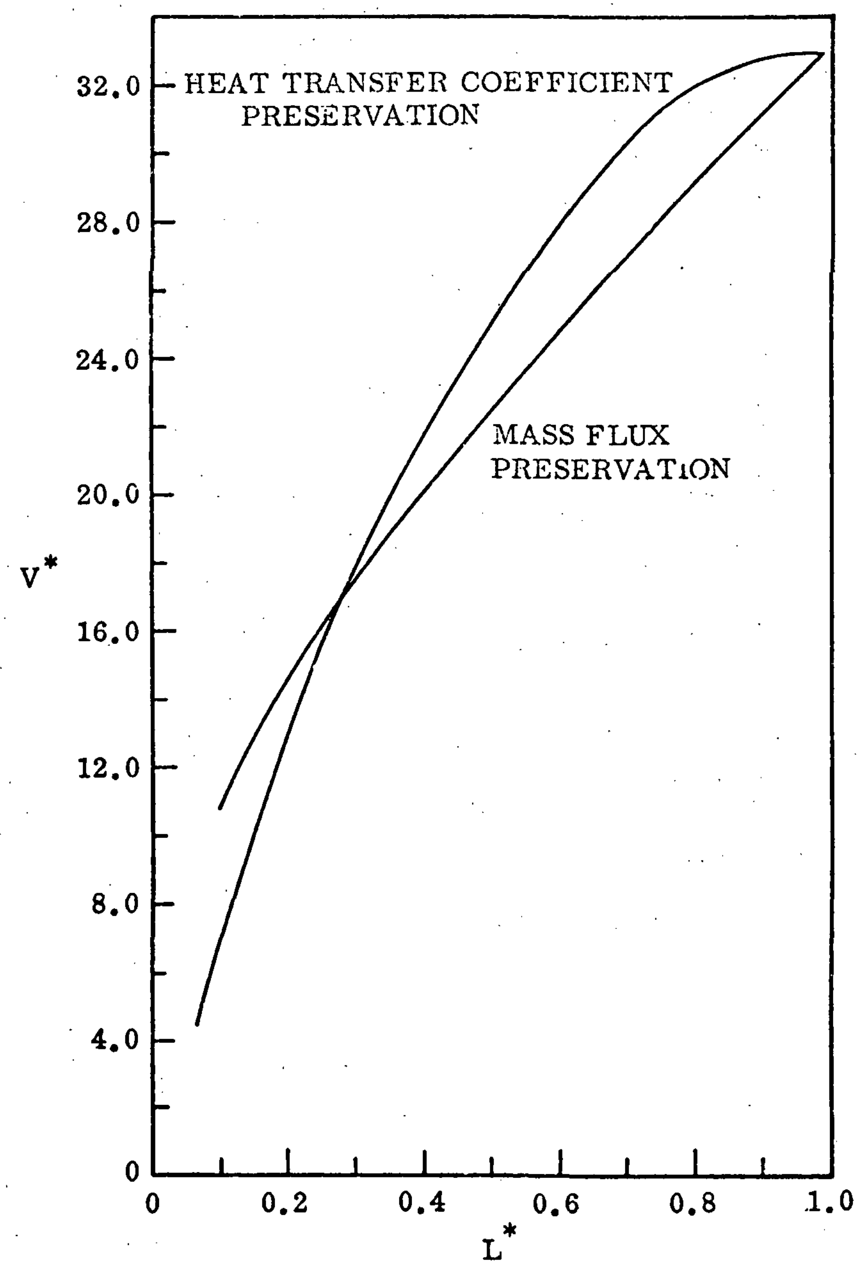
NON-DIMENSIONAL TEMPERATURE DIFFERENCE AS A FUNCTION OF REYNOLDS NO. FOR ONE G FIELD



NON-DIMENSIONAL TEMPERATURE DIFFERENCE AS A FUNCTION OF REYNOLDS NO. FOR ZERO G SIMULATION



VELOCITY SCALE RATIO VERSUS DIMENSIONAL SCALE RATIO FOR ZERO g SIMULATION



VALUES OF HEAT TRANSFER COEFFICIENT FOR ONE G FIELD

PROTOTYPE

Forced $h_c \approx 0.3$ BTU/Ft^2 Hr ^oR

Free $h_c \approx 0.2$ to 0.4 BTU/Ft^2 Hr ^oR

MODEL : $L^* = 1/3$

Heat Transfer Coefficient Preservation

Forced $h_c \approx 0.3$ BTU/Ft^2 Hr ^oR

Free $h_c \approx 0.17$ to 0.35 BTU/Ft^2 Hr ^oR

Mass Flux Preservation

Forced $h_c \approx 0.18$ BTU/Ft^2 Hr ^oR

Free $h_c \approx 0.17$ to 0.35 BTU/Ft^2 Hr ^oR

Temperature Preservation

Forced $h_c \approx 0.3$ BTU/Ft^2 Hr ^oR

Free $h_c \approx .17$ to $.35$ BTU/Ft^2 Hr ^oR

(Requires $L^* = 1/2.5$
and uses SO_2 atmosphere)

ZERO G SIMULATION

Heat Transfer Coefficient Preservation ($L^* = 1/3$)	- Requires $P^* = 0.019$ Results in $V^* = 17$ ($M \approx 0.5$)
Mass Flux Preservation ($L^* = 1/3$)	- Requires $P^* = 0.05$ Results in $V^* = 18$ ($M \approx 0.5$)
Temperature Preservation ($L^* = 1/2.5$) Gas = SO_2	- Requires $P^* = 0.21$ $V^* = 0.575$

MODELING OF CREW CABIN

Advantages

1. Provide a means with which to determine the interaction of the crew section with the rest of the vehicle thermal control and systems.
2. Possible means of reducing the affects of natural convection in order to simulate 0-g condition.
3. Study effects of moving equipment around inside the cabin.
4. Use as a test bed to determine affect of design changes on system.

Disadvantages

1. Cost of constructing the Model.
2. Question as to whether accurate simulation of flow paths can be accomplished in Model.
3. Cost of obtaining the necessary support equipment i.e.: pump and gas for temperature preservation.

CONDUCTION - RADIATION MODELING CRITERIA

For $T^* = 1$

Insulation Thickness:	$t^*_{ins.} = K^*_{ins.}$
Time:	$\theta^* = t^{*2}_{ins.} \ (\rho^* C^* / K^*)_{ins.}$
Length:	$L^{*2} = \theta^* \ (K^* / \rho^* C^*)_{sub.}$
Substrate Thickness:	$t^*_{sub.} = (K^*_{ins.} / t^*_{ins.}) \ (L^{*2} / K^*_{sub.})$

For one potential candidate insulation material for use on the Model,

$t^*_{ins.} \approx 0.5$ and $\theta^* \approx 1$

Then L^* and $t^*_{sub.}$ can be controlled by the choice of materials for modeling the aluminum substrate.

EXAMPLE:

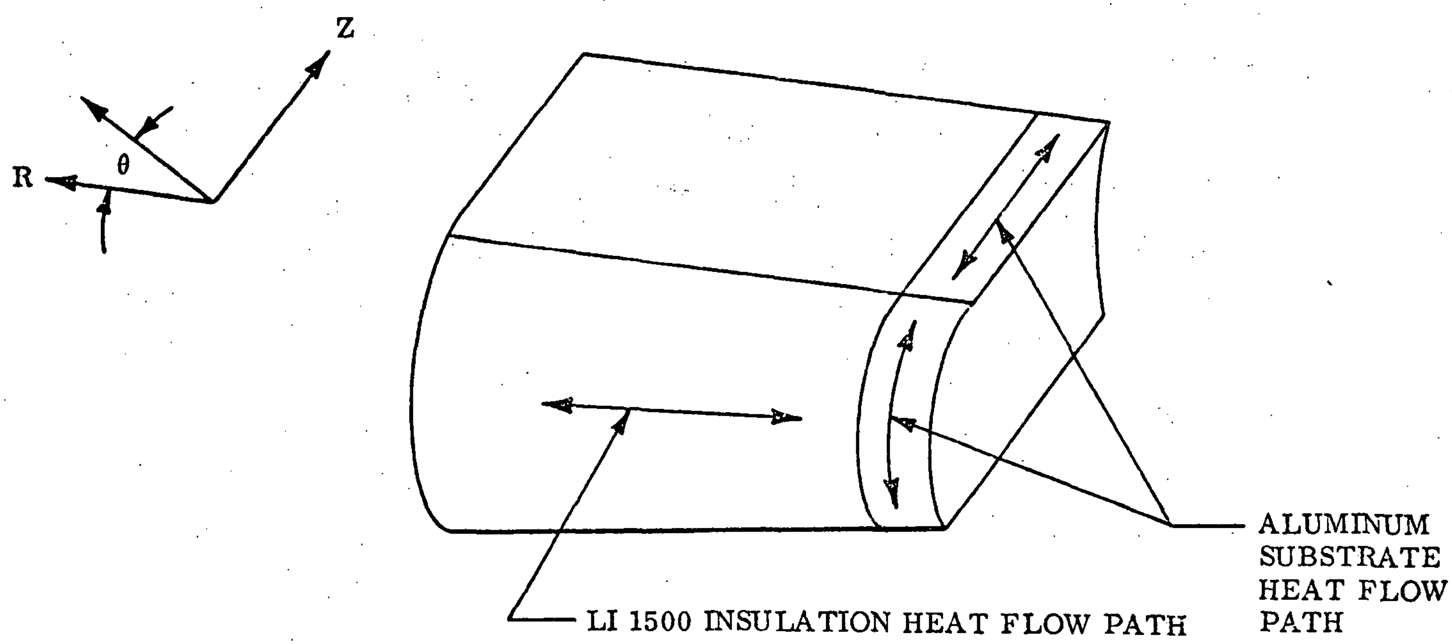
Assuming 6061 Alum. for Prototype
Selecting 430 S.S. for Model

$K^*_{sub.} = 0.15$ and $(\rho C)^*_{sub.} = 1.4$

$L^* = 0.33 \approx 1/3$

$t^*_{sub} = 0.40$

CONDUCTION PATHS THROUGH
EXTERNAL INSULATION/ALUM.
SKIN



BASED ON COMPUTER RESULTS, THE FOLLOWING ASSUMPTIONS ARE VALID:

LI 1500 HEAT PATH: 1 DIM

SUBSTRATE HEAT PATH: 2 DIM

LEVEL OF DETAIL FOR MODELING SYSTEMS

1. Forward Fuselage

    - o Crew Compartment
    - o Avionics
    - o Forward RCS Module
    - o Air Refueling Module
    - o Nose Landing Gear Assembly
    - o Docking Mechanism
    - o Misc. Electronic Equipment

2. Fluid Loops, Heat Exchangers, and Radiators

3. Payload Bay

4. Typical Payload

5. Wings

    - o Elevon Hydraulics
    - o Main Landing Gear Assembly

6. Vertical Stabilizer and Rudder

7. External and Internal Thermal Control Surfaces

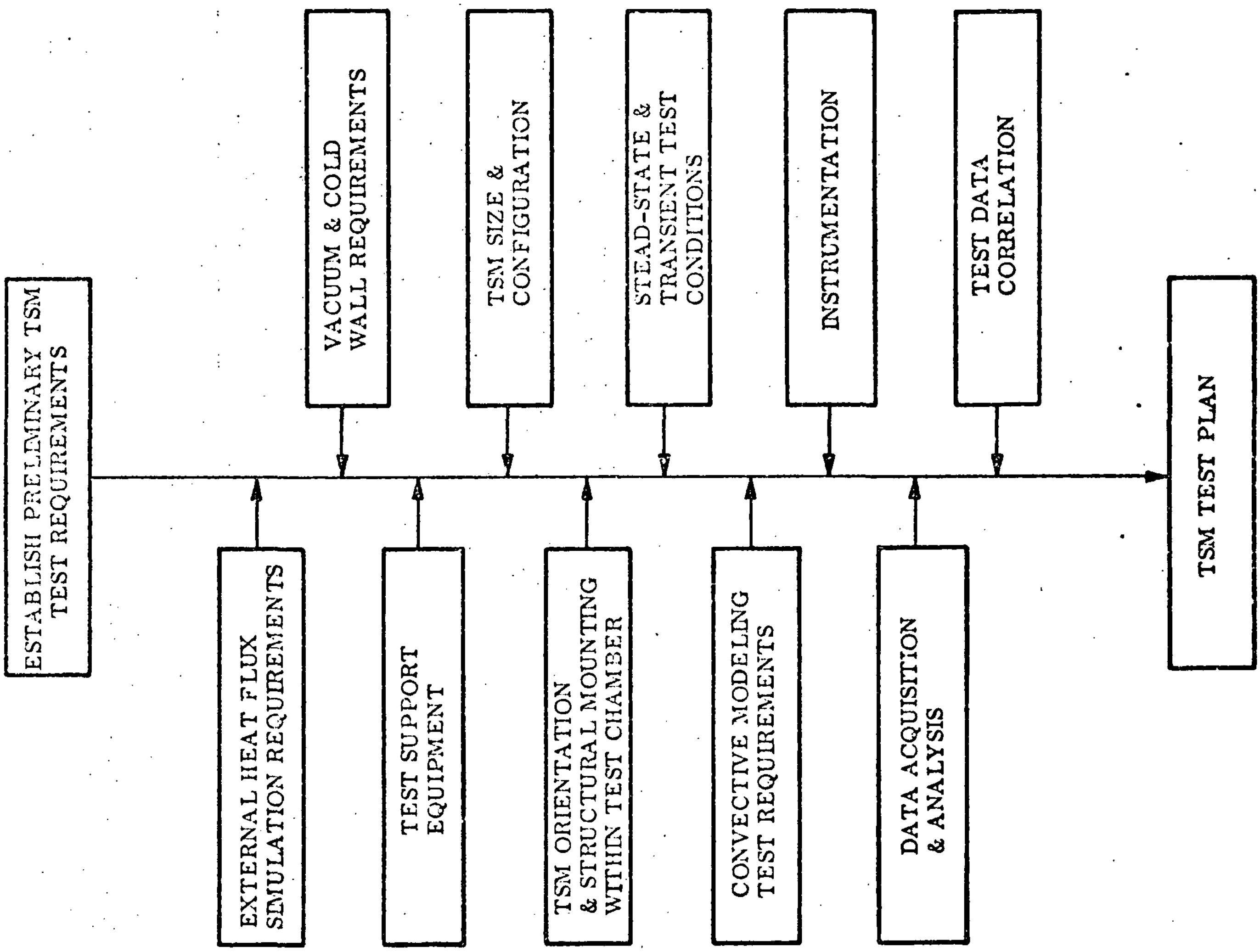
8. External and Internal Insulation

9. Skin Structure

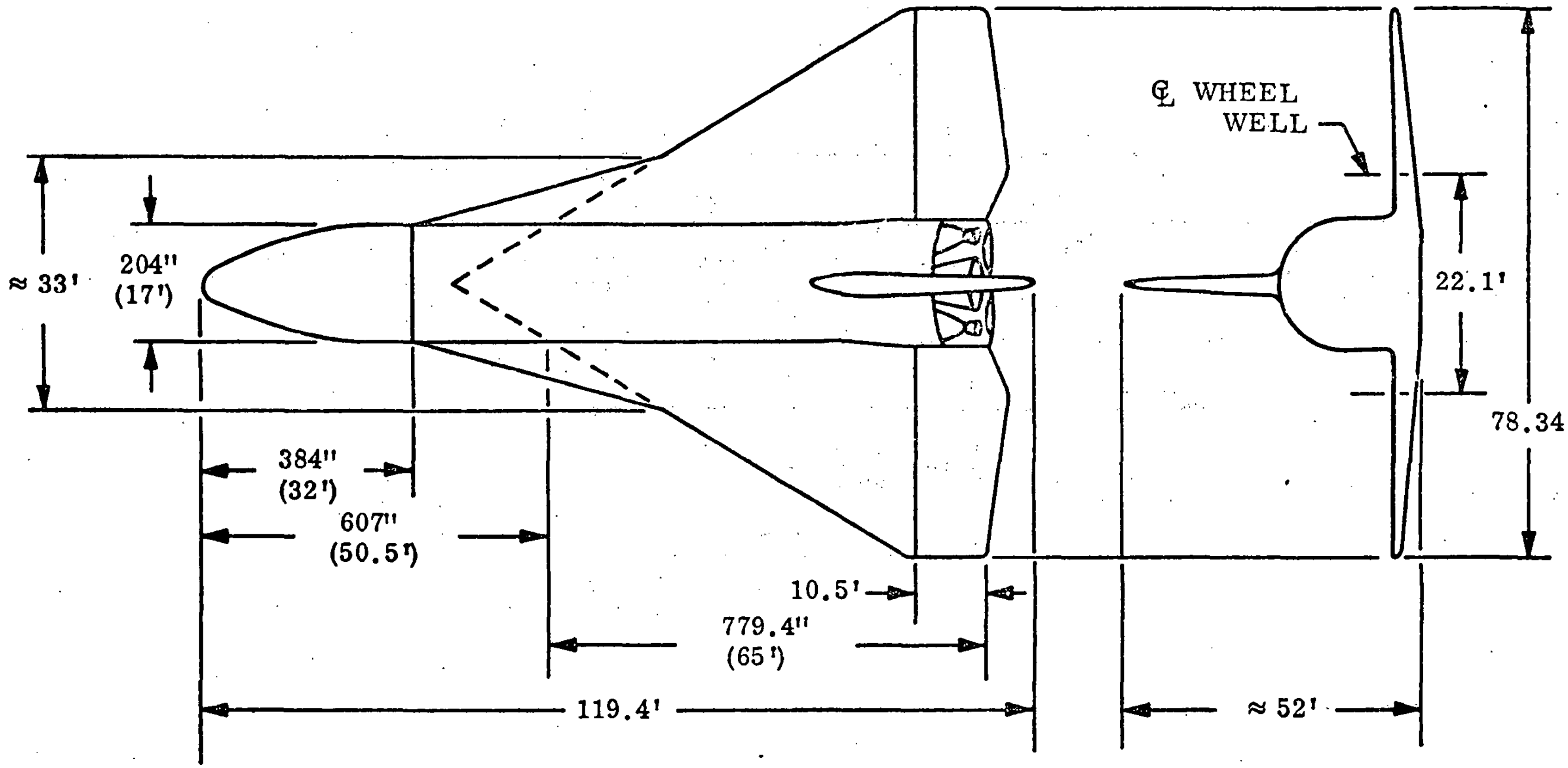
10. Aft Fuselage

    - o RCS Modules
    - o OMPS Modules
    - o Main Engines
    - o Propellant Tanks and Lines
    - o Structure

11. Connecting Joints and Hinges


ESTABLISH PRELIMINARY TSM TEST REQUIREMENTS
VACUUM & COLD WALL REQUIREMENTS
TSM SIZE & CONFIGURATION
STEAD-STATE & TRANSIENT TEST CONDITIONS
INSTRUMENTATION
TEST DATA CORRELATION
EXTERNAL HEAT FLUX SIMULATION REQUIREMENTS
TEST SUPPORT EQUIPMENT
TSM ORIENTATION & STRUCTURAL MOUNTING WITHIN TEST CHAMBER
CONVECTIVE MODELING TEST REQUIREMENTS
DATA ACQUISITION & ANALYSIS
TSM TEST PLAN

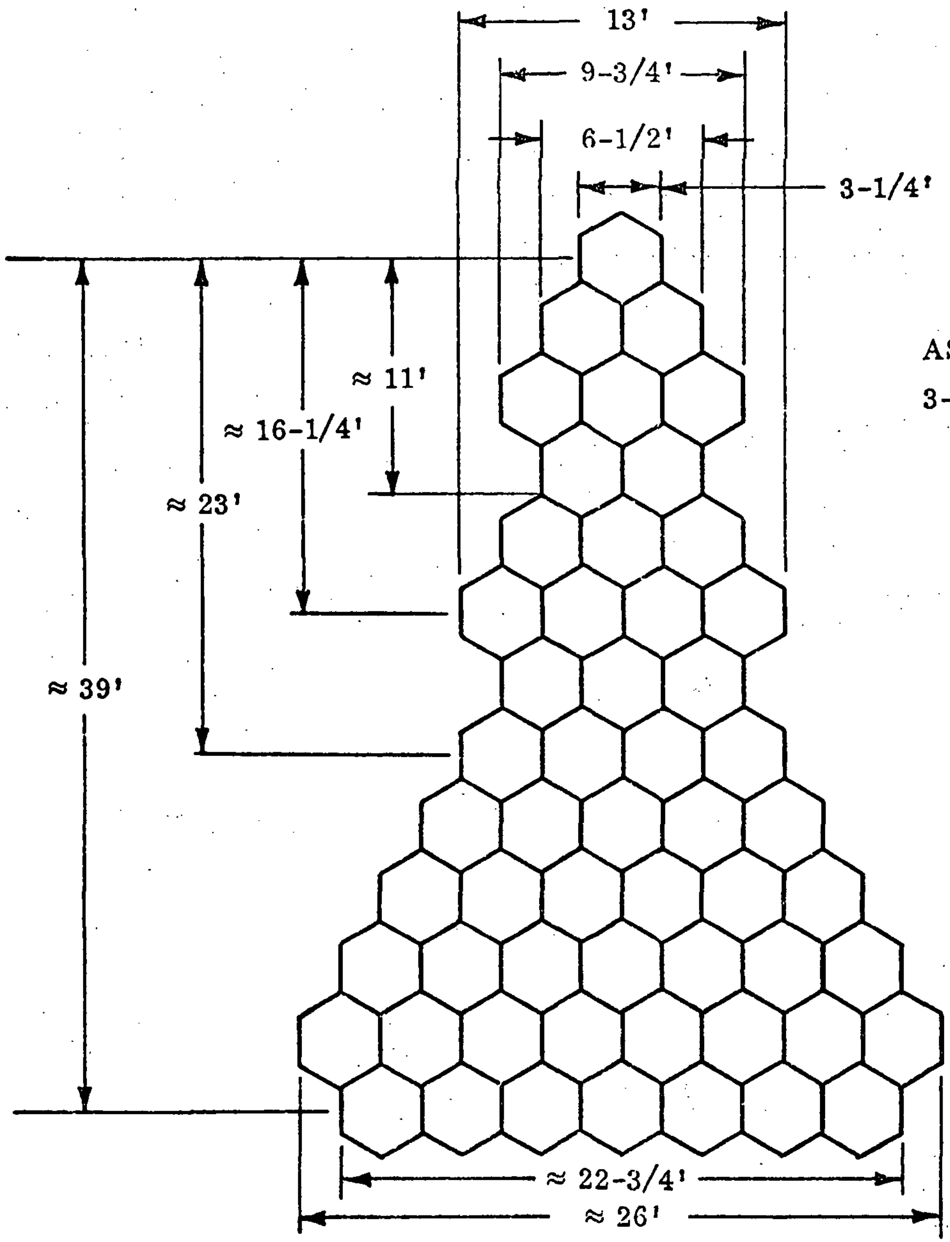
SHUTTLE OVERALL DIMENSIONS
(LMSC CONCEPT)



MODEL SIZE VS. SCALE RATIO

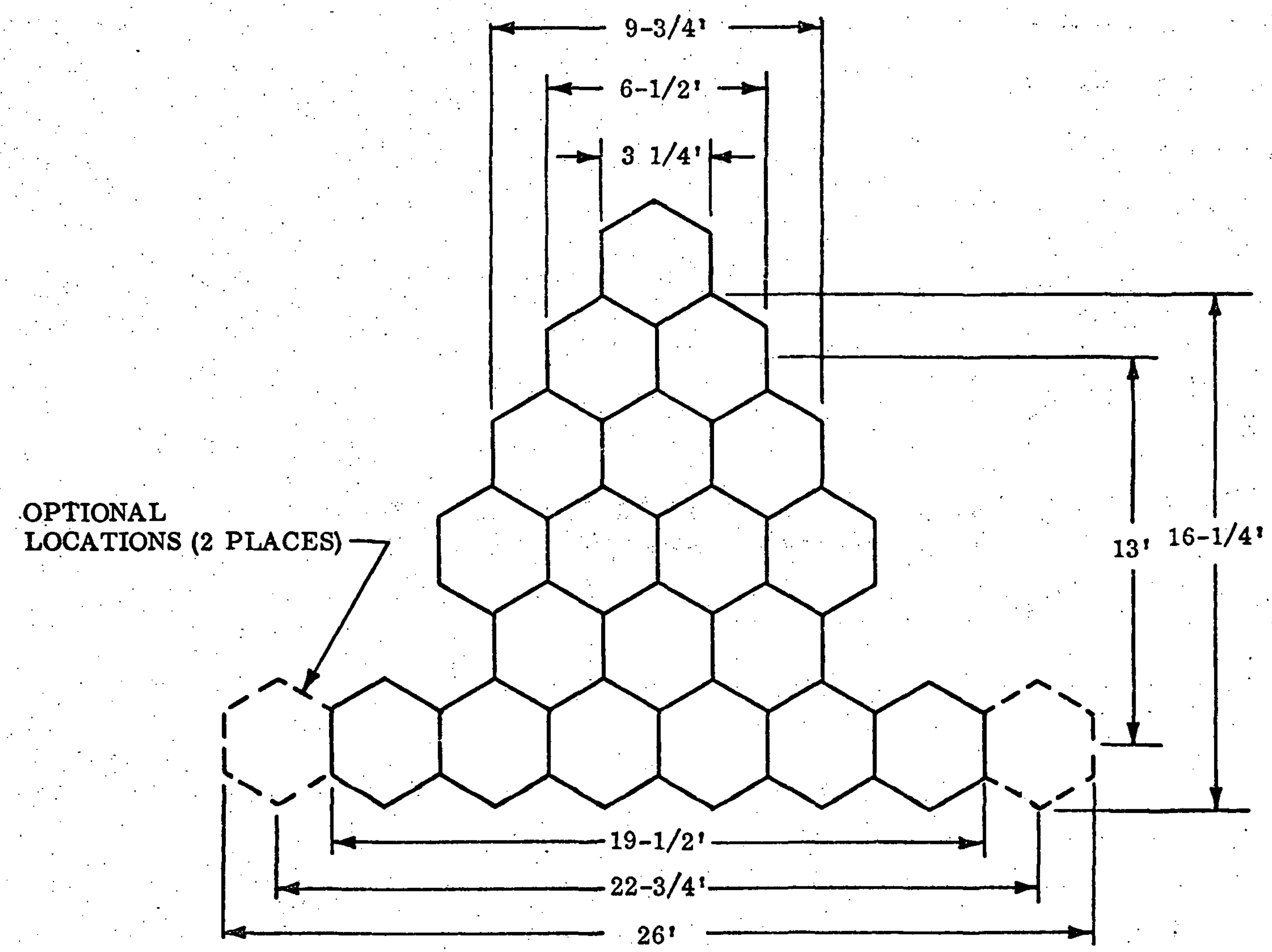
Scale Ratio	Principal Dimensions L (Ft.)	W (Ft.)	H (Ft.)
Full	119.4	78.34	52
1/2	59.7	39.17	26
1/2.5	47.8	31.3	20.8
1/3	39.8	26	17.3
1/3.5	34.2	22.4	14.85
1/4	29.8	19.5	13

REARRANGEMENT OF MSC CHAMBER-A SIDE SUN TO CONFORM TO SHUTTLE SHAPE (55 LAMP ARRAY)



ASSUMING:
3-1/4' PER LAMP DIA.

REARRANGMENT OF CHAMBER-A TOP SUN TO CONFORM TO SHUTTLE SHAPE – 19 LAMP ARRAY



TSM PRELIMINARY TEST REQUIREMENTS

1. External Heat Flux Simulation
   - o Solar Simulation; top and side sun
   - o Earth Emission
2. LN_2 Cold Wall
3. Vacuum: Pressure $< 10^{-5}$ Torr
4. TSM Internal Convection Simulation
5. TSM Vertical Orientation in Test Chamber
6. Support Structure Required for TSM
7. Instrumentation
   - o Temperature (approx. 600 thermocouples)
   - o Crew Cabin Pressure
   - o Gas Flow Rate
   - o Power Input
   - o Standard Test Facility Monitors
8. Data Acquisition
   - o Continuous Monitoring and Strip Chart Recording of Selected Thermocouples for All Tests
   - o All Thermocouple Data Recorded for Steady State Conditions
   - o All Thermocouple Data Recorded Periodically Throughout Transient Tests
   - o TSM Internal Pressures Monitored and Recorded Continuously During Test
   - o Test Facility Conditions Monitored as Required
9. Correlation of Test Results
   - o Temperatures in ^{o}F and ^{o}K from Thermocouple Data
   - o Compare TSM Temperatures with TMM Temperatures

TSM DETAILED DESIGN CRITERIA

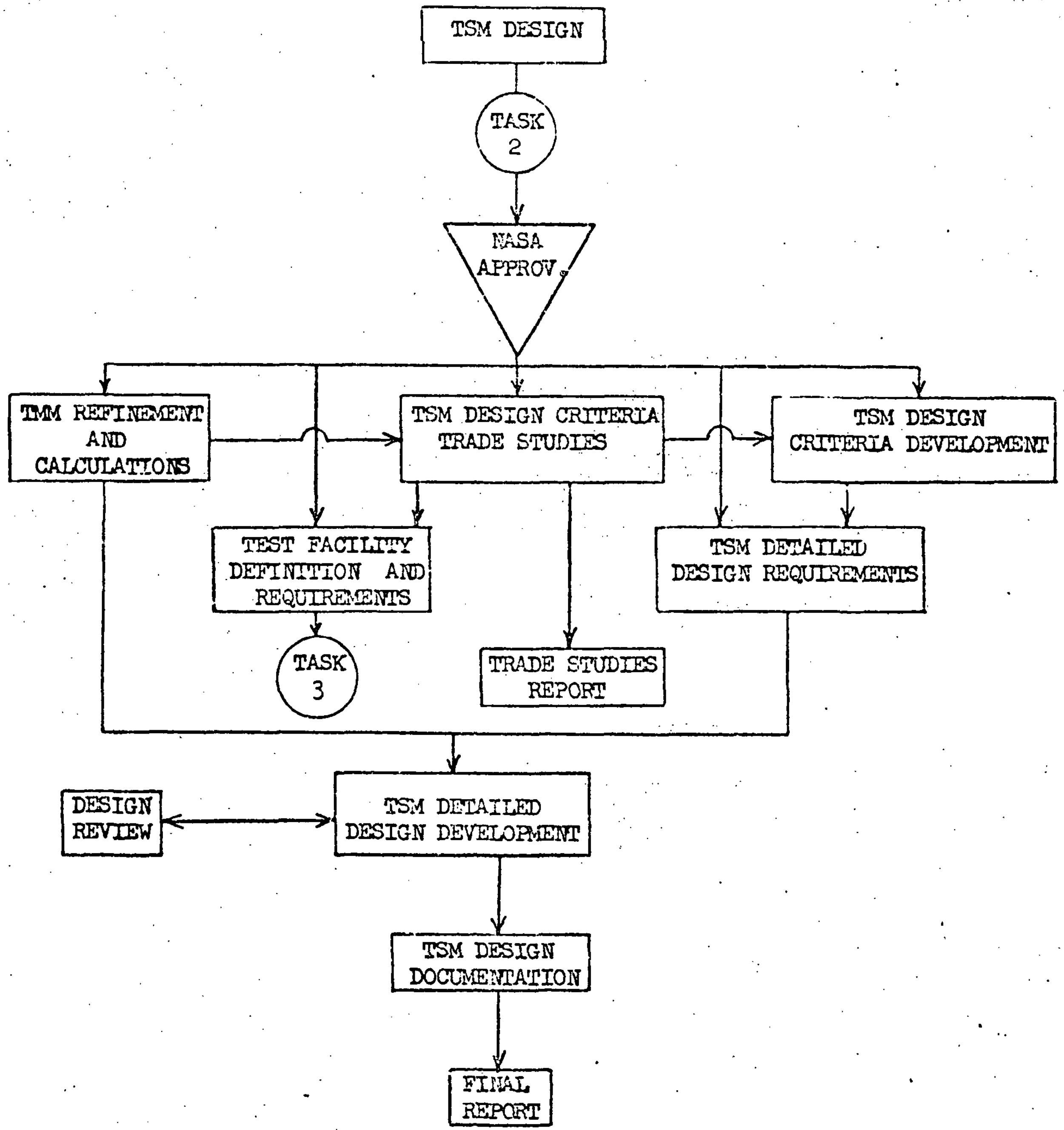
1. Use "Temperature Preservation" Approach, $T^* = 1$, for Conduction-Radiation Modeling.
2. Study Both Pure "Temperature Preservation" and "Scale Compromises" Approaches for Convection Modeling.
3. Overall Length Scale Ratio = 1/2.5 to 1/3.
4. Assume One-Dimensional Modeling for External Insulation and Internal MLI.
5. Assume Two-Dimensional Modeling for the Aluminum Skin Structure.
6. Identical Optical Properties Between Model and Prototype for Both Internal and External Surfaces.
7. Heat Exchangers and Fluid Loops not to be Modeled.
8. Radiators to be modeled with Simulated Heat Load Applied.
9. Wings and Vertical Stabilizer Extremities to be Deleted as Required to Fit Within Solar Simulation Capability.
10. Cylindrically Shaped Payload to be Included in Design Studies.
11. Active Thermal Controlled Systems to Receive Little Scale Modeling Attention.
12. Propulsion System Components to be Modeled Only to the Extent that They Affect Other Areas of Vehicle.

POTENTIAL APPLICATIONS OF TSM FOR USE IN SHUTTLE THERMAL DESIGN VERIFICATION

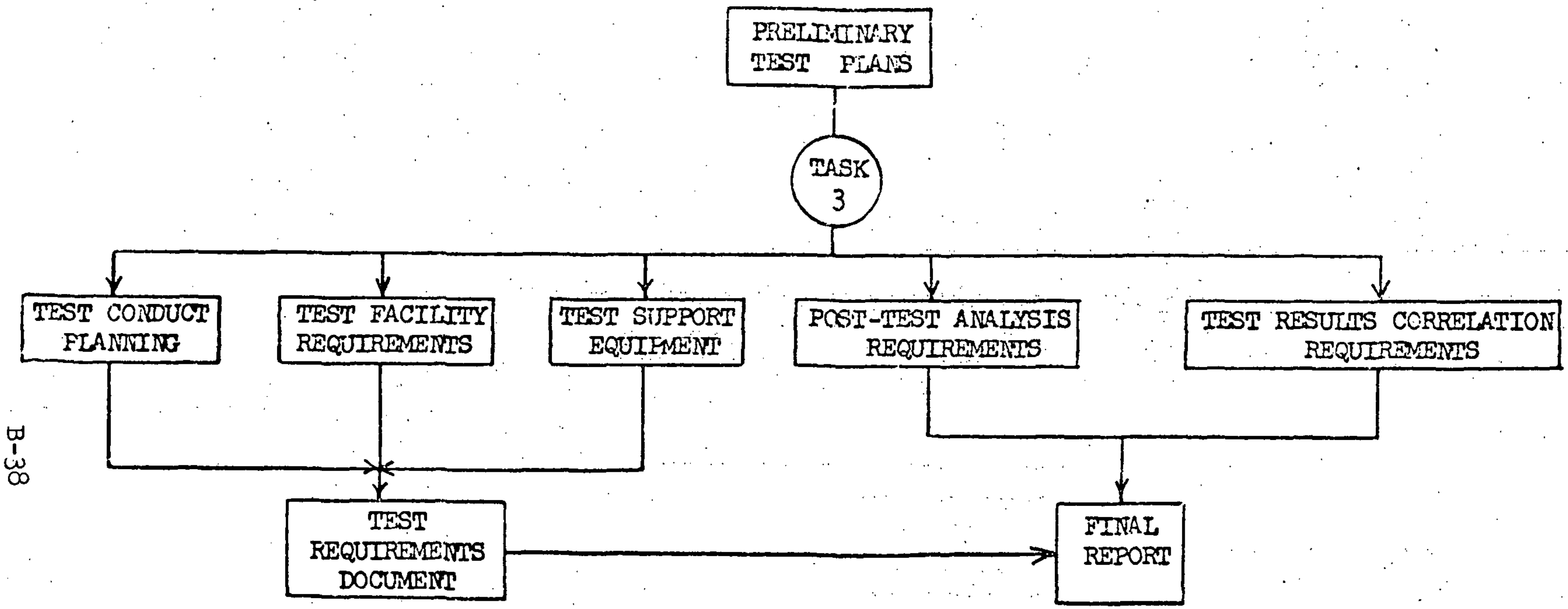
1. Refinement of Shuttle Orbiter TMM.
2. Thermal Test Verification of Vehicle Thermal Design.
3. Evaluation of Convective Cooling System.
4. Evaluation of Payload/Shuttle Thermal Interaction.
5. Evaluation of TPS for Various Payloads.
6. Evaluation of the Effects of Specularity of Thermal Control Surfaces.
7. Evaluation of Performance of Radiators and Interaction with External Vehicle Surfaces.
8. Evaluation of Performance of Certain Active Thermal Control Systems.
9. Multipurpose Thermal Scale Model to Serve as a Test Bed for Subsystem Testing.
10. Test Vehicle for External Insulation (i.e. LI-1500) to Establish Adequacy of Mechanical and Thermal Design.

<u>Primary Advantage of TSM</u>

To provide test data on both a system and subsystem level during the early design stages in order to assure an optimum thermal design for the entire Shuttle Orbiter.


TSM DESIGN
TASK 2
NASA APPROV.
TMM REFINEMENT AND CALCULATIONS
TSM DESIGN CRITERIA TRADE STUDIES
TSM DESIGN CRITERIA DEVELOPMENT
TEST FACILITY DEFINITION AND REQUIREMENTS
TASK 3
TRADE STUDIES REPORT
TSM DETAILED DESIGN REQUIREMENTS
DESIGN REVIEW
TSM DETAILED DESIGN DEVELOPMENT
TSM DESIGN DOCUMENTATION
FINAL REPORT

TASK 3 Work Breakdown Structure



TASK 4 Work Breakdown Structure