AD NUMBER
AD390196
CLASSIFICATION CHANGES
TO: unclassified
FROM: confidential
LIMITATION CHANGES
TO: Approved for public release, distribution unlimited
FROM: Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; MAR 1968. Other requests shall be referred to Air Force Flight Dynamics Lab., AFSC, Wright-Patterson AFB, OH 45433.
AUTHORITY
31 Mar 1980, DoDD 5200.10; AFWAL ltr, 12 Dec 1986

AD 390 196

AUTHORITY:
AFWAL etc.
12 Dec 86

GENERAL DECLASSIFICATION SCHEDULE

IN ACCORDANCE WITH

DOD 5200.1-R & EXECUTIVE ORDER 11652

SECURITY MARKING

The classified or limited status of this report applies to each page, unless otherwise marked.
Separate page printouts MUST be marked accordingly.

CONFIDENTIAL

AFFDL-TR-68-24
Part V

AD390196

(Unclassified)

PRELIMINARY DESIGN AND EXPERIMENTAL INVESTIGATION OF THE FDL-5A UNMANNED HIGH L/D SPACECRAFT

Part V Vehicle Design

P. P. Plank, I. F. Sakata, and M. Verhaegh
Lockheed-California Company

TECHNICAL REPORT AFFDL-TR-68-24, PART V

March, 1968

Air Force Flight Dynamics Laboratory
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

CONFIDENTIAL

Best
Available
Copy

NOTICES

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

This report is subject to special export controls and each transmittal to foreign governments or foreign nationals may be made only with prior approval of the Air Force Flight Dynamics Laboratory (FDMS), Wright-Patterson Air Force Base, Ohio 45433.

This classified report contains information on the design aspects of potential future weapons systems.

Copies of this report should not be returned unless return is required by security considerations, contractual obligations, or notice on a specific document.

(Unclassified)

PRELIMINARY DESIGN AND EXPERIMENTAL INVESTIGATION OF THE FDL-5A UNMANNED HIGH L/D SPACECRAFT

Part V – Vehicle Design

P. P. Plank, I. F. Sakata, and M. Verhaegh

(U) FOREWORD

(U) This is the final report of work performed under Contract No. AF33(615)-5241, "Preliminary Design of Two Volumetrically Efficient High L/D Unmanned Flight Test Vehicles". This report was prepared under Project 1366, "Aerodynamics and Flight Mechanics", Task 136616, "Synthesis of Hypersonic Vehicles".

(U) The work was sponsored by the Aerospace Vehicle Branch, Flight Mechanics Division, Air Force Flight Dynamics Laboratory. The research investigation was performed under the direction of the Air Force Project Engineer Mr. Thomas R. Sieron. Mr. C. J. Cosenza and Mr. A. C. Draper of AFFDL provided overall technical guidance.

(U) The work was accomplished by the Lockheed-California Company, Burbank, California and the report is also identified as LR 21204.

(U) This is Part V of a five part report:

Part I	Summary
Part II	Parametric Configuration Development and Evolution
Part III	Aerodynamics
Part IV	Aerothermodynamics
Part V	Vehicle Design

(U) This manuscript was released by the authors for publication in January 1968.

(U) The contributions of the following individuals to this report are gratefully acknowledged:

T. J. Beasley	J. P. McCabe
W. C. Grove	R. D. Mijares
R. S. Hassan	R. W. Stump
P. S. Hirasawa	B. C. Wollner
J. W. Lewis	J. C. Wczniak

(U) This technical report has been reviewed and is approved.

Philip P. Antonatos

Philip P. Antonatos
Chief, Flight Mechanics Division
Air Force Flight Dynamics Laboratory

(U) ABSTRACT

(U) A parametric analysis of a broad spectrum of thermostructural concepts is presented. A complete structural concept for the 35-foot entry test vehicle using the FDL-5 configuration is defined. Subsystems for the entry test vehicle are selected. A weight breakdown and a weight summary for the test vehicle are presented.

TABLE OF CONTENTS

TABLE OF CONTENTS (Concluded)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS CONTINUED

LIST OF ILLUSTRATIONS (Concluded)

LIST OF TABLES

SECTION 1

(U) INTRODUCTION

(U) Design of the unmanned flight test vehicle was accomplished under the following guidelines:

- (U) Maintain basic FDL-5 configuration mold lines
- (U) Provide L/D = 3.0 at hypersonic speeds (H = 200,000 ft, V = 20,000 fps)
- (U) Maintain high volumetric efficiency
- (U) Design to be stable and controllable over the entire Mach number range.
- (U) Design to be capable of performing a tangential landing.
- (U) Select minimum size vehicle with maximum internal volume.
- (U) Design to be capable of flying the two reference trajectories.
- (U) Use past experience and existing information as the basis for structural concept selection, subsystem selection, and weight determination.

These guidelines were generally compatible, and provided a reasonable set of goals for the study. The broad objectives of the design effort include:

- (U) Selection of the structural concept for the unmanned flight test vehicle
- (U) Selection of the subsystems
- (U) Provision of a weight breakdown.

(U) Selection of the structural concept was affected most by the guidelines for (1) trajectories, (2) for high volumetric efficiency and (3) for maximum internal volume. Other than the consideration given to material limitations, structural considerations were not used to modify the FDL-5 configuration lines from those determined to meet the aerodynamic and geometric requirements.

(U) A comprehensive parametric study of candidate structural concepts, performed as a part of the Lockheed in-house programs, was used extensively. Significant findings from that study are repeated in this volume to support the structural concept selection.

(U) Subsystems were selected with emphasis on the use of existing equipment. Information from previous test vehicle design studies and from the X-20, ASSET, and PRIME programs was reviewed and also used in subsystem selection.

(U) The weight of the flight test vehicle and the associated weight breakdown were determined from a detailed evaluation of the structural, thermal and systems analyses.

SECTION 2

STRUCTURAL CONCEPT SELECTION

(U) This section contains the material leading to the selection of the structural concept for the high L/D unmanned flight test vehicles. The parametric analyses are discussed and the selected design concept described. The parametric investigation applies to radiative structures only.

2.1 (U) STRUCTURAL PARAMETRIC INVESTIGATION

(U) The candidate structural concepts considered for the parametric study included those in which the primary structure is:

- (U) Insulated
- (U) Insulated and actively cooled
- (U) Radiation cooled (hot)

(U) Structural panel configurations included:

- (U) Single-corrugation
- (U) Honeycomb-sandwich
- (U) Corrugation-stiffened
- (U) Integrally stiffened
- (U) Zee-stiffened

(U) Materials included:

- (U) Aluminum alloy
- (U) Beryllium alloy
- (U) Cobalt-based alloy
- (U) Nickel-based alloy

- (U) Columbium alloy
- (U) Tantalum alloy

(U) Fibrous composites were not considered because of low loads and minimum gage restraints, as well as technology considerations. Dispersion strengthened materials should be considered for future analyses.

(U) Both fibrous and hard insulation were included.

2.1.1 (U) Rationale For Structures Parametric Evaluation

(U) The initial parametric study effort encompassed a review of work completed previously, and included the determination of factors essential to selection of a structural concept for an unmanned flight test vehicle. The basic selection criteria included:

- (U) Structural concept evaluation
- (U) Structural panel configuration evaluation
- (U) Materials evaluation
- (U) Basic structural weights estimation
- (U) Thermal protection system requirements
- (U) Nonoptimum considerations
- (U) Vehicle applications (research)
- (U) Number of missions
- (U) Vehicle-booster configuration compatibility
- (U) Internal temperature constraints
- (U) Structural arrangement constraints
- (U) Considerations of fabricability, inspectability, and maintainability.

In addition, the specific study guidelines discussed below were established to provide a basis for the thermostructural trades.

2.1.1.1 (U) Vehicle Configuration. The parametric effort utilized the high-volume, high L/D F-5 vehicle (Figure 1) configured under Contract AF 33(615)1884. This vehicle configuration was established as the baseline vehicle during the test vehicle configuration evolution, and the parametric structural study effort was directed towards defining the structural system for this typical vehicle. The various thermostructural trades conducted for this vehicle were in sufficient depth to substantiate the structural weight estimates, and to provide parametric data to determine the structural concept for the flight test vehicle.

(U) The selected FDL-5 configuration was used for the structural design effort to determine the applicable design concept for the vehicle configuration. It was also used for the stress analysis of various structural components to substantiate design weight estimates.

2.1.1.2 (U) Booster and Trajectory Data. To establish typical external loads for the structural parametric investigation, the Atlas booster and trajectory data used in Ref. 1 were applied. Upon selection of the Titan III booster for the final structural design, the external boost loads were reevaluated. The resulting load conditions are discussed in Section 2.1.2.

2.1.1.3 (U) Temperature Data. The F-5 vehicle isotherms and heating data are representative of those of the general class of high-volume, high L/D configurations and have been used in the thermostructural trades. Peak heating temperature distribution on the FDL-5 configuration is described in Part IV of this report.

2.1.1.4 (U) Vehicle Application. Specific work accomplished under this parametric investigation was directed toward selection of an unmanned flight test vehicle. However, structural design criteria were formulated for possible manned applications of these systems and these effects are included in the parametric data.

2.1.1.5 (U) Number of Missions. Parametric evaluation of the vehicle structure is based on a maximum of five flights.

2.1.1.6 (U) Internal Temperature Constraint. To determine the structural concept best suited for the various possible applications, two temperature constraints were considered:

- (U) Maximum usable structural temperature
- (U) 70°F backface temperature

2.1.1.7 (U) Maximum Usable Internal Volume Constraint. The possibility of manned and other applications requires that the internal structural arrangement provide maximum usable internal volume. This is accomplished by design of frames and shell structure to absorb minimum internal volume.

2.1.2 (U) Structural Design Criteria and Loads

(U) The design criteria formulated in Ref. 1 were utilized to establish the design loads presented below. Specific guidelines for structural analysis include:

- (U) Preliminary Boost Loads: F-5 Flight Test Vehicle - Atlas Launch Vehicle, $(\alpha q)_{max}$ = 3880 deg PSF.
- (U) Final Boost Loads: F-5 Flight Test Vehicle - Titan III Launch Vehicle, $(\alpha q)_{max}$ = 4500 deg PSF.

- (U) Factor of Safety (Load-Stress)

  Ultimate Factor = 1.33

  Limit Factor = 1.00

- (U) Combined Load-Temperature Factor

  Ultimate Factor on Thermal Strain = 1.25
  (Combine with the Ultimate Factor above)

- (U) Dynamic Magnification Factor - Exit Trajectory

  γ, dynamic factor = 1.2

2.1.2.1 (U) F-5 Flight Test Vehicle/Atlas Launch Vehicle Exit Loads. Exit loads for the F-5 vehicle on the Atlas booster have been calculated at the maximum αq condition. The wind profile has a peak velocity of 250 fps at the altitude for maximum dynamic pressure. Maximum αq with head wind is 3880 deg-psf and -3260 deg-psf for tail wind. (Figure 2)

(U) Limit loads (bending moments, axial loads, and shears) for an αq value of 3880 deg-psf are presented in Figure 3.

(U) Limit surface pressures calculated at the maximum αq conditions are presented in Figure 4.

2.1.2.2 (U) F-5 Flight Test Vehicle/Titan III Launch Vehicle Exit Loads. Design pressures and loads were calculated at the maximum αq condition resulting from booster response to severe wind. The F-5/Titan III Flight Test Vehicle-Launch Vehicle combination was utilized and a wind profile with a peak velocity of 250 fps at the altitude for maximum dynamic pressure was selected. Maximum αq is equal to 4500 deg-psf and occurs at an altitude of 37,000 feet. Net limit shears, bending moments and axial loads from this condition are presented in Figure 5.

(U) Figure 4 shows the limit design pressures for the nose, leading edge, upper and lower surfaces. These values envelope all design conditions under the assumed criteria.

2.1.2.3 (U) Entry Loads. Shears and bending moments during entry are shown in Figure 6 and are based on the trajectory shown in Figure 7.

(U) Figure 8 presents leading edge, lower, and upper surface limit pressures for an angle of attack equal to 18 degrees and a bank angle of zero.

(U) Limit shear and bending moments for the F-5 outboard fin during entry are presented in Figure 9. For design, these loads are applied simultaneously with the temperatures associated with the peak heating portion of entry. Fin loads due to gust during the terminal phase are less critical than those encountered during launch. Accordingly, loads estimated during the launch phase are sufficient to account for this condition.

2.1.2.4 (U) Landing Loads. Limit shears and bending moments for the landing condition are presented in Figure 10. The coefficients of friction used are 0.10 for the nose gear and 0.40 for the main gear.

2.1.2.5 (U) Titan Structural Capability. Structural load capability for the Titan III booster with 30 feet and 35 feet F-5 payloads has been calculated and is presented in Figure 11. Load comparison is on the basis of equivalent axial load which is composed of axial load due to acceleration and drag plus that due to bending moment. Comparison is made at the point of αq (4500 deg-psf). Maximum αq is calculated from the booster response to wind. Peak winds equal to 250 fps at the altitude of maximum dynamic pressure were used. The booster is submarginal for the 35 foot vehicle with the values of factor of safety and dynamic response factor shown. It is satisfactory if the dynamic response factor is reduced to 1.27.

(U) A comparison of exit parameters for the F-5/Atlas and F-5/Titan configuration are presented in Figure 2. Maximum αq is listed for head and tail winds of 250 fps occurring at the altitude of maximum dynamic pressure. A head wind produces up-bending and a tail wind down-bending.

2.1.3 (U) Materials and Coating Review and Selection

(U) Selection of candidate metallic materials for the flight test vehicle (Table 1) was based on application for primary load-carrying, heat-shield, and leading-edge designs. Factors considered include:

- (U) Thermal-physical properties
- (U) Mechanical properties
- (U) Creep resistance
- (U) Formability
- (U) Weldability
- (U) Oxidation resistance
- (U) Other joining techniques
- (U) Elastic weight efficiency

Minimum gage criteria were formulated (Table 2) for the candidate structural panel configurations and applicable material system. These criteria are necessary, as the gages established by stress analysis are often less than those which can be economically fabricated and assembled by normal manufacturing processes.

TABLE 1

(U) METALLIC MATERIALS EVALUATED FOR
FLIGHT TEST VEHICLE APPLICATION

Material		Maximum Temperature Utilization (°F)	Area of Application
Alloy	Designation		
Aluminum	2219T81	300	Internal load-carrying structure
Beryllium	Be-38Al	600	Internal load-carrying structure
Beryllium	AMS 7902	900	Internal load-carrying structure
Titanium	Ti 8Al-1Mo-1V	900	Internal load-carrying structure
Nickel base	Inconel 718	1400	Internal and external load-carrying structure. Heat shields (1800°F)
Nickel base	Inconel 625	1500	Internal and external load-carrying structure. Heat shields (1800°F)
Cobalt base	Haynes 25	1600	Internal and external load-carrying structure. Heat shields (1800°F)
Columbium	Cb 752	2500	External load-carrying structure. Heat shields, leading edges, and associated support.
Tantalum	Ta-10W, T-222	3500	Heat shields, leading edges, and associated support.

TABLE 2

(U) MINIMUM GAGE FOR METALLIC MATERIALS (INCHES)

MATERIAL	STRUCTURAL CONFIGURATION						FRAME BULKHEAD APPLI-CATION
Alloy designation	SINGLE CORRU-GATION	CORRUGATION-STIFFENED Skin	CORRUGATION-STIFFENED Corrug.	INTEGRALLY STIFFENED	HONEYCOMB SANDWICH Skin	HONEYCOMB SANDWICH Core	
Aluminum 2219T81	.016	.016	.012	.016(1)	-	-	.020
Beryllium Be-38Al	.016	-	-	.016	-	-	.020
Beryllium AMS7902	.016	-	-	.016	-	-	.020
Titanium Ti 8Al-1Mo-1V	.016	.016	.012	.016	-	-	.020
Nickel base Inconel 718	.016(3)	.016(3)	.012(3)	(2)	.012(3)	.002	.020
Nickel base Inconel 625	.010	.010	.010	(2)	.008	.002	.012
Cobalt base Haynes 25	.010	.010	.010	(2)	.008	.002	.012
Columbium Cb 752-R512E	-	.012(4)	.012(4)	.012	.010(.008)(5)	.002(6)	.016
Tantalum(7) 90Ta-10W	-	-	-	.012	-	-	.016

NOTES

(1) Minimum selected based on manufacturing consideration.
(2) Not considered because of serious mfg. problems (warpage, distortion, extremely difficult chemically to mill)
(3) Gages selected because of distortion due to heat treatment of thinner gages.
(4) Poor structural resistant welds: projected application of solid state roll diffusion bonded technique.
(5) ( ) value indicates heat shield application.
(6) Core material: Columbium alloy D-36 (ρ = .286 lb/in.^{3})
(7) For strength requirements consider tantalum alloy T-222.

2.1.4 (U) Parametric Structural Concept Evaluation

2.1.4.1 (U) Candidate Structural Concepts. The basic structural concepts considered for the parametric evaluation included:

- (U) Insulated - Monolithic heat shield of hard fused silica insulation (Figure 12)
- (U) Insulated - Metallic heat shield with external or external-internal dynaquartz insulation (Figure 13)
- (U) Insulated and Cooled - Metallic heat shield with external or external-internal insulation and indirect active cooling using expendable water (Figure 14)
- (U) Hot Load Carrying - with/without internal insulation or internal insulation with indirect active cooling using expendable water (Figure 15)

(U) Figures 16 through 20 define the scope of structural concepts considered in this analysis. Initially, application of the basic concepts was made to the basic body structure of the vehicle only. The hot-load-carrying concept was used in specific areas (i.e., nose section, fins, control surfaces, leading edge chine) where geometric constraints and temperatures dictate its use.

2.1.4.2 (U) Structural-Material Concept Analysis. The structural requirements related to vehicles of the F-5 class were determined and the selection of panel configurations (Figure 21) and materials (Table 1) for primary load-carrying and heat-shield application was made. The main criteria for selection were:

- (U) Structural Efficiency
- (U) Manufacturing Capabilities
- (U) Minimum Gage Criteria

(U) Structural Efficiency of Primary Load Carrying Panels - Optimum structural proportions for the panel configuration were based on a design of simultaneous general and local instability failure. The equations for these modes of failures were combined and by algebraic manipulation the geometric and material property terms were separated and grouped as parameters. The material geometry parameters (i.e., stiffener thickness/skin thickness ratio for integrally stiffened configuration) plotted separately result in a geometry maximum efficiency (ε) plot for each different structural configuration being considered. The efficiency factor was then adapted in a general closed-form equation of the type presented below:

$$\frac{N_x}{LE_c\eta} = \varepsilon \left(\frac{\bar{t}}{L}\right)^n$$

where:

N_x = Axial distributed load (lb/in.

L = Frame spacing (in.)

E_c = Compression modulus (psi)

η = Plasticity correction factor

ε = Efficiency factor

$\bar{t}$ = Effective structural thickness (in.)

n = Exponent

(U) Wide column design charts were formulated (Figures 22 through 30) based on the preceding closed-form equation to permit an initial evaluation of the candidate panel configuration for the selected material system.

(U) Manufacturing Capabilities - The state of the art of joining and forming technology and available equipment for manufacturing components from thin-gage refractory metals (requiring oxidation-resistant coating) and superalloys had a pronounced impact on the selection of panel configurations.

(U) Both the primary load-carrying structure and heat shields involve a considerable amount of forming operations. The use of thin gages for structural efficiency requires small bend radii for stiffened panel configurations. Both types of panels require a considerable amount of joining without degrading the mechanical properties from those of the parent material, and with adequate surface smoothness. Brazing or an equivalent method is considered essential for structural configurations such as honeycomb or truss core sandwich panels. Heat shield panel designs involving thin core depth sections to minimize the thermal stress effect require less stringent high temperature strength.

(U) Minimum Gage Criteria - Minimum gage for fabrication of acceptable structural elements, sheet thickness availability, and sheet thickness variations, were considered in the panel configuration selection. Minimum gage criteria (Table 2) were established for each candidate panel configuration with consideration given to joining and fabrication techniques. A comparison of aluminum, titanium, beryllium, superalloy and refractory metal for axial load-carrying panels of minimum gage is presented in Figures 31 through 33. Nonoptimum factors (Table 3) to account for attachments and close-outs were determined on the basis of experience and detail design and analysis, and are included in the weight comparison. Figure 34 summarizes the stability allowable ($N_x/\bar{t}_s$) and panel weights for minimum gage structural panels considered for this study.

2.1.4.3 (U) Heat Shield and Support Configurations. Heat shield and support weights are established for various configurations and combinations. Heat

TABLE 3

(U) NONOPTIMUM FACTOR (NOF) CONSIDERED FOR STRUCTURAL PANEL EVALUATION

Structural Concept	Structural Panel Configuration	NOF
INSULATED	Single Corrugation	1.10
INSULATED & ACTIVELY COOLED	Single Corrugation	1.15
	Skin Corrugation	1.20
HOT MONOCOQUE	Honeycomb Sandwich	1.30
	Unflanged Integrally Stiffened	1.25
	Integral Zee	1.25
	Skin Corrugation	1.25
HOT LOAD CARRYING (shear panels)	Honeycomb Sandwich	1.25
	Unflanged Integrally Stiffened	1.20
	Integral Zee	1.20
	Skin Corrugation	1.20

UNCLASSIFIED

shields are sized for strength, stiffness requirements and minimum gage requirements. Table 4 presents typical heat shield and support weights of both columbium alloy and superalloy panels for 15" x 15" panels. Table 5 presents effective gages of the weight contributing elements. Weight data for fused silica heat shields are included in Table 6.

2.1.4.4 (U) Primary Load Carrying Structure. The structural arrangement was established for the study vehicle by considering the functional requirements of landing gears and doors; fins and attachments; equipment location and access; and booster attachment.

(U) The design of primary load-carrying panels applicable to each thermostructural concept considers such factors as 1) external applied loads, 2) panel flutter, 3) minimum gage, 4) panel stability and strength, and 5) nonoptimum consideration for the edge conditions related to each thermostructural concept. The effect of these factors on the design of a typical lower surface panel is presented in Figure 35. The importance of observing the minimum gage criteria to reflect realistic basic panel weights is shown for design of vehicles of this class which are subjected to low loading intensities. For the first 360 inches, the minimum gage is seen to be greater than the gage requirements to support basic loads.

(U) The longeron configurations are established to satisfy the functional and structural design requirements. Factors for consideration included 1) adequate attachment provision, 2) minimum gage, 3) local and overall stability requirements, and 4) geometric considerations to minimize thermal stresses.

(U) The assessment of frames for strength and stability requirements resulted in the selection of the hat section configuration over the "J" section, channel, and zee shapes, because of its thermostructural efficiency. Maximum usable internal volume is obtained by providing frames to span the width of the vehicle without intermediate posts or truss supports. Adequate bending strength to transmit pressure loads to the basic shell is provided for in the frame design. To reflect a realistic weight, nonoptimum factors (which include the effects of gussets, splices, and clips to attach the frames to the panels and longerons) were also assessed.

2.1.4.5 (U) Thermal Protection System. The long flight time and low-to-moderate heat fluxes associated with vehicles of this class focuses attention upon radiative types of thermal protection systems. From a volumetric efficiency point of view, particular interest is placed upon radiative concepts utilizing active cooling. For this study effort an indirect active cooling system using water and a passive system using insulation only were selected. The indirect active cooling involves a heat transport loop consisting of tubes and a transport medium (water) attached to the structure being cooled. Heat is absorbed and is circulated to a heat exchanger where the heat is removed by radiation or an expendable coolant (water). Pumps, temperature sensors, and flow meters are essential to promote coolant circulation and control.

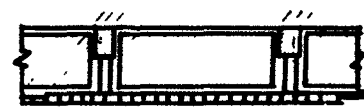
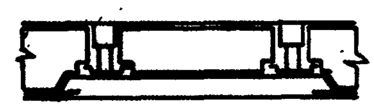
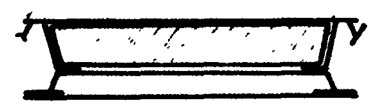
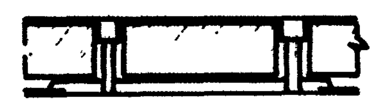
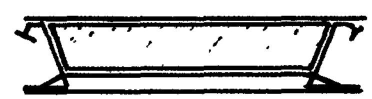
TABLE 4

(U) METALLIC HEAT SHIELD CONCEPTS

EVALUATED FOR

FLIGHT TEST VEHICLE APPLICATION

(Aspect Ratio: 1; Panel Size: 15" x 15")

HEAT SHIELD SYSTEM		GEOMETRY	MATERIAL	WEIGHT (LB/FT^2)		
Config.	Support			Panel	Support (1)	System
Honeycomb Sandwich	Four (4) Post Support		Cb 752-R512 (D36 Core)	1.33	0.40	1.73
			Inconel 625	1.16	0.30	1.46
Corrugation Stiffened	Four (4) Post Support		Cb 752-R512	2.49	0.40	2.89
			Inconel 625	1.85	0.30	2.15
Corrugation Stiffened	Fwd. & Aft Edge Support		Cb 752-R512	2.18	0.50	2.68
			Inconel 625	1.62	0.40	2.02
Unflanged-Integrally Stiffened	Four (4) Post Support		Cb 752-R512	2.00	0.40	2.40
			Inconel 625	(2)	(2)	(2)
Unflanged-Integrally Stiffened	Fwd. & Aft Edge Support		Cb 752-R512	1.66	0.50	2.16
			Inconel 625	(2)	(2)	(2)

UNCLASSIFIED

(1) Average unit weight - varies with support weight

(2) Does not apply

TABLE 5

(U) HEAT SHIELD UNIT WEIGHTS (LB/FT^2)

(Panel Size: 15" x 15")

BASIC MATERIAL:		Columbium Alloy Cb 752-R512 (ρ = .326 lb/in^3)					Nickel Base Alloy Inconel 625 (ρ = .305 lb/in^3)		
PANEL CONFIGURATION:		Honeycomb Sandwich	Skin-Corrug. Stiffened		Unflanged Integr. Stiffened		Honeycomb Sandwich	Skin-Corrug. Stiffened	
SUPPORT CONCEPT:		Post	Post	Edge	Post	Edge	Post	Post	Edge
Basic Panel	$\bar{t}_p$(in)	.0170	.0260	.0286	.0190	.0209	.0170	.0221	.0243
Edge Reinforcement	$\bar{t}_e$(in)	.0008	.0064	.0064	.0043	.0043	.0008	.0060	.0060
Stiffener, Core and/or Braze	$\bar{t}_b$(in)	.0075	.0150	.0060	.0150	.0060	.0074	.0120	.0048
Coating	$\bar{t}_c$(in)	.0016	.0032	.0033	.0024	.0025	---	---	---
Contingency	$\Delta \bar{t}$(in)	.0012	.0025	.0022	.0020	.0017	.0012	.0020	.0018
TOTAL PANEL	$\bar{t}_T$(in)	.0282	.0531	.0465	.0427	.0354	.0264	.0421	.0369
Support	$\bar{t}_s$(in)	.0085	.0085	.0107	.0085	.0107	.0068	.0068	.0091
HEAT SHIELD SYSTEM	$\bar{t}_{HS}$(in)	.0367	.0616	.0572	.0512	.0461	.0332	.0489	.0460
H.S.SYSTEM WEIGHT	W_{HS}(lb/ft^2)	1.73	2.89	2.68	2.40	2.16	1.46	2.15	2.02

UNCLASSIFIED

(REVERSE SIDE IS BLANK)

TABLE 6

THERMO-STRUCTURAL WEIGHT COMPARISON - F·

STRUCTURAL CONCEPT	PRIMARY LOAD-CARRYING STRUCTURE						THERMAL	
	TEMP (°F)	MATERIAL		PANEL CONFIG.	UNIT WT(#/ft²)		T_2 BACK-FACE TEMP.	LOWER SURFACE(S
		Alloy	Desig.		w_{SL}	w_{SU}		Description
1A INSULATED	300	Aluminum	2219T81	Single corrug.	0.94 0.94 0.94	0.97 0.97 0.97	70 150 300	Fused silica Hard insulatio w/phenolic ove
1B INSULATED	600	Beryllium	Be-38A1	Single corrug.	0.75 0.75 0.75	0.60 0.60 0.60	70 600 600	(same as above)
1C INSULATED	900	Titanium	Ti8-1-1	Single corrug.	1.10 1.10 1.10	1.03 1.03 1.03	70 300 900	(same as above)
1D INSULATED	1500	Nickel base	Inconel 625	Single corrug.	1.59 1.59 1.59 1.59	1.42 1.42 1.42 1.42	70 300 900 1500	Dyna-quartz Intern-External Insula. HCPS Heat shield
1E INSULATED	300	Aluminum	2219T81	Single corrug.	0.94 0.94 0.94	0.97 0.97 0.97	70 150 300	Dyna-quartz Internal-exter Insulation HCP heat shield
1F INSULATED	600	Beryllium	Be-38A1	Single corrug.	0.75 0.75 0.75	0.60 0.60 0.60	70 300 600	(same as above)
1G INSULATED	900	Titanium	Ti8-1-1	Single corrug.	1.10 1.10 1.10	1.03 1.03 1.03	70 300 900	(same as above)
2A INSULATED(U) AND ACTIVELY COOLED (L)	70 150 300	Aluminum	2219T81	Single corrug.	0.94 0.94 0.94	0.97 0.97 0.97	70 150 300	(Same as above cept with indi active cooling)
2B INSULATED(U) AND ACTIVELY COOLED (L)	70 150 300	Beryllium	Be-38A1	Single corrug.	0.75 0.75 0.75	0.60 0.60 0.60	70 150 300	(same as above)
2C INSULATED AND ACTIVELY COOLED	70 150 300	Aluminum	2219T81	Single corrug.	0.94 0.94 0.94	0.97 0.97 0.97	70 150 300	(same as above)
2D INSULATED AND ACTIVELY COOLED	70 150 300	Beryllium	Be-38A1	Single corrug.	0.75 0.75 0.75	0.60 0.60 0.60	70 150 300	(same as above)

* Honeycomb sandwich post supported heat shield (HCPS)

TABLE 6

HT COMPARISON - F-5 STUDY VEHICLE

	THERMAL PROTECTION SYSTEM (T.P.S.)				STRUCTURE + T.P.S. WEIGHTS					
	LOWER SURFACE(S_L=159ft^2)		UPPER SURFACE(S_U=190.6ft^2		w_L	w_U	W_L	W_U	W_T	w_T
MP.	Description	w_{TPS}	Description	w_{TPS}	(lb/ft^2)		(lb)		(lb)	(lb/ft^2)
	Fused silica Hard insulation w/phenolic overlay	6.15 5.78 5.22	Dyna-quartz internal-external insulation HCPS* heat shield	3.10 2.66 2.23	7.09 6.72 6.16	4.07 3.63 3.20	1128 1069 980	776 692 610	1904 1761 1590	5.45 5.05 4.55
0 0 0	(same as above)	6.15 5.22 4.28	(same as above)	3.10 2.23 1.97	6.90 5.97 5.03	3.70 2.83 2.57	1098 950 800	706 540 490	1804 1490 1290	5.16 4.26 3.69
0 0 0	(same as above)	6.15 5.22 3.47	(same as above)	3.10 2.23 1.82	7.25 6.32 4.57	4.13 3.26 2.85	1152 1004 726	788 620 543	1940 1624 1269	5.55 4.65 3.63
0 0 0 0	Dyna-quartz Intern-External Insula. HCPS Heat shield	4.69 4.24 3.50 2.80	Dyna-quartz internal insulation w/ modular heat shield	2.45 1.58 1.17 0.81	6.28 5.83 5.09 4.39	3.87 3.00 2.59 2.23	948 928 809 898	738 572 494 425	1686 1500 1303 1123	4.82 4.30 3.73 3.22
0 0 0	Dyna-quartz Internal-external Insulation HCPS heat shield	4.69 4.56 4.24	Dynaquartz internal-external insulation HCPS heat shield	3.10 2.66 2.23	5.63 5.50 5.18	4.07 3.63 3.20	895 874 824	776 692 610	1671 1566 1434	4.79 4.48 4.10
0 0 0	(same as above)	4.69 4.24 3.91	(same as above)	3.10 2.23 1.97	5.44 4.99 4.66	3.70 2.83 2.57	865 794 740	706 540 490	1571 1334 1230	4.50 3.82 3.52
0 0 0	(same as above)	4.69 4.24 3.50	(same as above)	3.10 2.23 1.82	5.79 5.34 4.60	4.13 3.26 2.85	920 849 731	788 620 543	1708 1469 1274	4.89 4.20 3.65
70 50 00	(Same as above except with indirect active cooling)	4.03 3.93 3.84	Dyna-quartz internal-external insul.HCPS*ht shld	3.10 2.66 2.23	4.97 4.87 4.78	4.07 3.63 3.20	790 775 760	776 692 610	1566 1467 1370	4.48 4.20 3.92
70 50 00	(same as above)	4.03 3.93 3.84	(same as above)	3.10 2.66 2.23	4.78 4.68 4.59	3.70 3.26 2.83	760 745 730	706 621 540	1466 1366 1270	4.20 3.91 3.64
70 50 00	(same as above)	4.03 3.93 3.84	(same as above, except with indir. active cooling)	2.65 2.62 2.55	4.97 4.87 4.78	3.62 3.59 3.52	790 775 760	690 685 671	1480 1460 1431	4.23 4.18 4.10
70 50 00	(same as above)	4.03 3.83 3.84	(same as above)	2.65 2.62 2.53	4.78 4.68 4.59	3.25 3.22 3.15	760 745 730	620 614 600	1380 1359 1330	3.95 3.89 3.81

UNCLASSIFIED

(Continued)

(REVERSE SIDE IS BLANK)

TABLE 6 (Continued)

THERMO-STRUCTURAL WEIGHT COMPARISON -

STRUCTURAL CONCEPT	PRIMARY LOAD-CARRYING STRUCTURE						THERMA
	TEMP (^oF)	MATERIAL		PANEL CONFIG.	UNIT WT($\#/ft^2$)		T_2 BACK-FACE TEMP.	LOWER SURFACE
		Alloy	Desig.		W_{SL}	W_{SU}		Descriptio
3A HOT LOAD-CARRYING	70 300 900 1500 2500	Nickel base (U) columbium (L)	Inconel 625 Cb752-R512	Honeycomb sandwich	3.61 3.61 3.61 3.61 3.61	2.69 2.69 2.69 2.69 2.69	70 300 900 1500 2500	Dyna-quartz i ternal insula
3B HOT LOAD CARRYING	70 300 900 1500 2500	Nickel base(U); Columbium (L)	Inconel 625 Cb 752 - R512	Skin-corrug. Unflanged int. stif.	3.54 3.54 3.54 3.54 3.54	2.27 2.27 2.27 2.27 2.27	70 300 900 1500 2500	(same as abov
3C HOT LOAD CARRYING	70 150 300	Nickel base (U) Columbium (L)	Inconel 625 Cb 752-R512	Honeycomb sandwich	3.61 3.61 3.61	2.69 2.69 2.69	70 150 300	Dyna-quartz i ternal insul. w/indirect active coolin
3D HOT LOAD CARRYING	70 150 300	Nickel base (U) Columbium (L)	Inconel 625 Cb752-R512	Skin-cor. Unflang. int. stif.	3.54 3.54 3.54	2.27 2.27 2.27	70 150 300	(same as abov
3E HOT LOAD CARRYING	70 300 900 1500 2500	Nickel base (U); Columbium (L)	Inconel 625 Cb752 - R512	Single corrug. Unflag. Int.stif.	3.54 3.54 3.54 3.54 3.54	1.42 1.42 1.42 1.42 1.42	70 300 900 1500 2500	Dyna-quartz i ternal insula-tion
3F HOT LOAD CARRYING	70 150 300	Nickel base (U) Columbimm (L)	Inconel 625 Cb752-R512	Single corrug. Unflanged Int. stif.	3.54 3.54 3.54	1.42 1.42 1.42	70 150 300	Same as above except with indirect acti cooling.

(U) Upper
(L) Lower

ABLE 6 (Continued)

L WEIGHT COMPARISON - F-5 STUDY VEHICLE

	THERMAL PROTECTION SYSTEM (T.P.S.)				STRUCTURE + T.P.S. WEIGHTS					
BACK-TEMP.	LOWER SURFACE (S_L=159ft^2) Description	w_{TPS}	UPPER SURFACE(S_U=190.6ft^2) Description	w_{TPS}	w_L (lb/ft^2)	w_U (lb/ft^2)	W_L (lb)	W_U (lb)	W_T (lb)	w_T (lb/ft^2)
70	Dyna-quartz internal insulation	2.96	Dyna-quartz internal insulation	1.64	6.57	4.33	1045	825	1870	5.36
300		2.51		0.77	6.12	3.46	973	660	1633	4.68
900		1.77		0.36	5.38	3.05	856	581	1437	4.12
500		1.07		0	4.68	2.69	745	513	1258	3.60
500		0		0	3.61	2.69	575	513	1088	3.12
70	(same as above)	2.96	(same as above)	1.64	6.50	3.91	1032	745	1777	5.09
300		2.51		0.77	6.05	3.04	962	580	1542	4.43
900		1.77		0.36	5.31	2.63	845	501	1346	3.86
500		1.07		0	4.61	2.27	735	432	1167	3.34
500		0		0	3.54	2.27	564	432	996	2.85
70	Dyna-quartz internal insul. w/indirect active cooling	2.30	Dyna-quartz internal insul. w/indirect active cooling	1.19	5.91	3.88	940	740	1680	4.81
150		2.20		1.16	5.81	3.85	924	735	1659	4.75
300		2.11		1.09	5.72	3.78	909	720	1629	4.66
70	(same as above)	2.30	(same as above)	1.19	5.84	3.46	928	660	1588	4.54
150		2.20		1.16	5.74	3.43	913	655	1568	4.49
300		2.11		1.09	5.65	3.36	899	640	1539	4.40
70	Dyna-quartz internal insulation	2.96	Modular heat shield w/Dyna-quartz internal insulation	2.45	6.50	3.87	1032	738	1770	5.07
300		2.51		1.58	6.05	3.00	962	572	1534	4.40
900		1.77		1.17	5.31	2.59	845	494	1339	3.83
500		1.07		0.81	4.61	2.23	735	425	1160	3.32
500		0		0.81	3.54	2.23	564	425	989	2.83
70	Same as above except with indirect active cooling.	2.30	(same as above except with indirect active cooling)	2.00	5.84	3.42	928	651	1579	4.51
150		2.20		1.97	5.74	3.39	913	646	1559	4.46
300		2.11		1.90	5.65	3.32	899	632	1531	4.39

UNCLASSIFIED

(REVERSE SIDE IS BLANK)

(U) The insulation and insulation-cooling requirements for various locations of the vehicle upper and lower surfaces were established for a range of backface temperatures (70°F to 2500°F). Dynaquartz insulation ($\rho = 4.5$ lb/ft^3) was selected for application because of its superior performance to 3000° F. It is lightweight and dimensionally stable at soaking temperatures to 2750°F, and up to 3000°F for transient exposure.

(U) A fused silica hard insulation was evaluated for application to the lower surface as a monolithic heat shield concept. This material was selected for its inherent strength and its low conductivity. The insulation is self-sustaining and does not require encapsulation but does present mechanical attachment difficulties. Bonding to the substrate structure is considered to be feasible.

2.1.4.6 (U) Final Parametric Thermostructural Evaluation. A comparison of the basic structure plus thermal protection system weight for each candidate thermostructural concept is presented for the complete vehicle in Table 6, and in Figure 36. Figures 37 through 42 present details of separate upper and lower surface thermostructural weights, as well as the weight variation of each thermostructural concept with material and panel configuration selection.

(U) Basic structure plus thermal protection system average unit weight ratios for the various candidate concepts are presented in Figure 43. These data provide a comparison of each candidate concept with the minimum weight concept (hot load carrying) for various backface temperatures. The results indicate that for a given backface temperature requirement near 70°F, the weights of the concepts considered vary by no more than ±12 percent from the average value of about 1.6 times the minimum weight hot load carrying value.

(U) The effect of thermostructural concept selection on vehicle gross weight is presented in Figure 44. A comparison is made of total vehicle weights utilizing the various thermostructural concepts for a range of backface temperatures. For a typical vehicle requiring a 70°F internal environment within the complete vehicle (Sta 36 to Sta 360) a 9 percent to 17 percent increase in total vehicle weight results when comparing it with a minimum weight concept (hot load carrying). Only a 2 percent weight difference results when comparing an insulated and cooled aluminum structural concept to a hot load carrying structural concept that uses internal insulation and active cooling.

(U) The structural analysis defines the parametric relationship of the candidate thermostructural concepts. A similar parametric analysis performed by AFFDL is reported in Ref. 2. Several important factors in the thermostructural design of vehicles of this class have emerged from these parametric investigations.

These include the following:

- (U) Critical design loading condition occurs during boost,
- (U) Loading intensities for the design loading conditions are relatively low,

- (U) Minimum gage criteria play an important role in panel selection and weight,
- (U) Structural stability rather than strength dictates the panel design,
- (U) Stiffened axial load carrying panel configurations are lower weight than honeycomb sandwich panels of minimum gage,
- (U) Post-supported honeycomb sandwich heat shields are lower weight than corrugation stiffened or integrally stiffened heat shield designs.

2.2 (U) STRUCTURAL DESIGN

2.2.1 (U) Selection of Structural Concept

(U) An evaluation of the results from the structural parametric analysis in the preceding paragraphs produces the following conclusions:

- (U) The radiation-cooled, hot-load-carrying structural concept provides the minimum vehicle weight providing that no limit is placed on inside wall temperature.
- (U) A vehicle with an insulated and actively cooled concept with an internal temperature of 70°F weighs 11 percent more than a vehicle which uses hot-load-carrying structure with no internal insulation or cooling.
- (U) For an internal temperature of 70°F, the vehicle weight for a hot-load-carrying structural concept, with internal insulation and active cooling, is 2 percent more than with the use of an insulated and cooled primary load carrying aluminum structure.
- (U) The insulated concept is competitive weight-wise with the other concepts, however, this concept has a lower volumetric efficiency and may require cooling provisions upon landing.
- (U) Considerations of such functional provisions as access panels and landing gear doors tend to alter the selection of a structural concept from that concept which is lightest on a weight-per-square-foot basis to that concept which provide the lightest total vehicle weight, highest internal volume, or practical design approach.
- (U) Manned operational vehicles will require thermal control to 70°F or less of most of the vehicle usable internal volume.

(U) Based on the foregoing conclusions and considerations, the insulated and actively cooled concept was selected for the flight test vehicle primary structure. Initially, a combination using the insulated and actively cooled concept, and the hot-load-carrying concept was considered in order to test both concepts. It was subsequently concluded, however, that the vehicle structural arrangement should be characteristic of an operational application, with the entire internal area being controlled to 70°F. It was further concluded that the test vehicle structure should be based upon a 35-foot length. This length was selected on the basis of performance and ultimate adaptability to manned flight.

2.2.2 (U) Detailed Structure Design Arrangement

(C) The vehicle structural design is based upon an insulated and actively cooled structural concept applied to a 35-foot-long, unmanned, FDL-5 flight test vehicle. The structural arrangement of the flight test vehicle is shown in Figure 45. The vehicle uses a 2219-T81 aluminum alloy internal primary load-carrying structure. The external surface consists of 15 x 15 inch edge-supported heat shield panels. The lower surface uses Cb-752/R512E coated columbium alloy, and the upper surface uses Inconel-625 nickel alloy predominantly.

(U) Inconel-625 nickel alloy was selected for the upper surface heat shields on the basis of minimum weight, and adequate oxidation-resistance for the small number of missions involved; however, Haynes 25 cobalt alloy could be used.

(U) The more commercially available and more oxidation-resistant cobalt alloy Haynes 25 offers increased mission life at the expense of a small weight increase. It was estimated that the total structural weight increase from the use of Haynes 25 is 60 pounds compared with the use of Inconel-625.

2.2.2.1 (U) Internal Shell Structure. (C) The internal shell structure uses 2219-T81 aluminum alloy throughout, except that 8Al-1Mo-1V, titanium alloy is used locally to accommodate thermal-structural requirements. The shell structure uses 0.016 inch minimum gage single corrugation skin. Corrugations are aligned in the axial (fore and aft) direction. The shell structure is supported by internal hat section frames at a spacing of approximately 15 inches. This spacing is compatible with wide-column buckling allowables for minimum-gage, single-corrugation skin over most of the vehicle length. The spacing also corresponds with heat shield support spacing. Three longerons are used, one on the lower surface and two at the top surface. The longerons provide pick-up points for attachment to the booster, and are designed to carry the full axial and bending loads. The longeron design permits convenient structural arrangement of access doors at the upper surface. Internal hat section frames are designed to provide shell stability and to transmit panel loads. At the nose landing gear bay, the main landing gear bay, the flaps and the elevons, the frames are reinforced to satisfy local loading conditions. The forward frame depth for the upper surface is 1.0 inch, increasing to 1.25 inch at Sta 200, and remaining constant to the aft end. For the lower surface the forward depth is 1.0 inch increasing to 3.0 inches at Sta 200, and remaining constant to the aft end. Frame gages vary from 0.040 to 0.065 inches.

(C) The longerons are located externally to the basic shell structure. They are designed to carry vehicle axial loads, bending loads, and panel loads. They also furnish lateral support to the shell frames. The two longerons at the upper surface use hat sections which are 1.50 inches deep at Sta 145 and 1.75 inches deep at Sta 420. Longeron gages vary from 0.045 inch to 0.090 inch. The lower longeron varies in depth from 1.75 inches at Sta 145 to 2.0 inches at Sta 420. Gages vary from 0.075 to 0.160 inch.

(C) The unit weight of 0.97 lb/ft^2 for the primary load-carrying shell structure (skin, frames, longerons), as established by the parametric analysis, is adequate for the present vehicle. Additional structural weight requirements have been obtained as a result of stress analyses in the following areas:

- (C) Nose attachment
- (C) Nose landing gear bay
- (C) Main landing gear bay

  Including bulkheads and longitudinal webs.

- (C) Access doors
- (C) Elevon support
- (C) Flap support
- (C) Fin attachment

Structural weight items not accounted for by stress analysis, such as attachments and local doublers, are included in the nonoptimum factors as used in the overall weight analysis. Conventional joining processes such as riveting, fusion and resistance welding are used.

2.2.2.2 (U) Thermal Protection System. (C) The thermal protection system consists of external heat shields, supports, high-temperature fibrous insulation, and an active cooling system. Figure 45 presents the thermal protection concept for the flight test vehicle. The lower surface uses Cb-752/R512E coated columbium alloy, unflanged, integrally stiffened, edge-supported heat shield panels. These panels are extended approximately 4 inches beyond the leading-edge tangency point on the upper surface. The remainder of the upper surface uses Inconel-625 nickel alloy skin-corrugation edge-supported heat shield panels. The Inconel-625 panels are also used at the close-off bulkhead on the rear of the vehicle.

(C) The panels have an integrally stiffened edge along one transverse side. The panels are designed to carry normal loads only. The 15 x 15 inch panels have overlapping edges and oversize attachment holes on one edge to allow for relative expansion and contraction. Each panel uses four stand-off's at 9.0-inch spacing. Flush-head countersunk attachment screws, with a shank diameter of 3/16 inch, of the same material as the heat shields are used. A hard insulation glass rock spacer, approximately 1 inch deep for the lower surface, is used with each stand-off. The 1-inch-diameter tubular stand-off is made of 0.010-inch-thick Haynes 25 cobalt alloy. A square nut is encased into a glass rock spacer internally within the stand-off. The stand-off attaches directly to the internal shell structure. Cooling tubes are bonded or clip-supported to the crest of the single corrugation skin. The 0.10-inch inner-diameter tubes are spaced a minimum of 2.50 inches. High temperature fibrous insulation is bonded directly to the aluminum alloy internal shell. The insulation consists of 6 lb/ft^3 dynaflex and 3.5 lb/ft^3 microquartz. The lower surface uses dynaflex adjacent to the heat shields and microquartz adjacent to the shell. The upper and rear surfaces use only microquartz over most of their area (1600°F limit for microquartz is assumed).

(U) The heat shield, support, insulation, cooling, and cooling system weights are obtained from the parametric analysis. Following is a summary of these weights for the insulated and cooled concept:

Element	Lower Surface (lb/ft^2)	Upper Surface (lb/ft^2)
Heat Shield	1.66	1.62
Support	0.50	0.40
Insulation	1.04	0.66
Coolant	0.60	0.21
Cooling System	0.62 (0.70)	0.32 (0.35)

The insulation weights for the parametric analysis are based on the use of 4.5 lb/ft^3 dynaquartz. Dynaflex and microquartz insulation are substituted for the flight test vehicle because of improved structural integrity and reliability of these materials. Study results per Ref. 3 show that the average weight requirements for dynaflex and microquartz, used on an unmanned high L/D vehicle, are comparable with parametric study results.

The following data for the lower surface are obtained from Ref 3:

VEHICLE STA.	X/L = .20		X/L = .60		X/L = .90	
Insulation Data	Thickness (in.)	Weight (lb/ft^2)	Thickness (in.)	Weight (lb/ft^2)	Thickness (in.)	Weight (lb/ft^2)
Dyna-Flex	1.46	.73	.74	.37	1.20	.60
Micro-Quartz	1.50	.44	1.90	.56	1.35	.40
Total	2.96	1.17	2.64	.93	2.55	1.00
Average Values	Thickness 2.71 in. Weight 1.04 lb/ft^2					

(U) The cooling system weight requirement includes the weight of the tubing, plumbing, pump, motor, tanks, supports, controls, heat exchanger, water residuals and carry-over, and power penalty (batteries). Bracketed values show finalized results as reflected in the weight statement.

2.2.2.3 (U) Nose Attachment. The nose attachment concept presented for the flight test vehicle offers a solution to local thermal-structural requirements. The approach is to restrict heatloads through the use of minimum heat leak attachment concepts and an insulated and actively cooled support structure.

(U) Two attachment points are used at the lower surface and one attachment point is used at the upper surface. The attachments at the lower surface carry axial loads only. The attachments at the upper surface carry both shear loads and axial loads.

(C) All attachments are surrounded by a layer of dynaflex and microquartz insulation. The lower attachments use a Cb-752/R512E coated columbium alloy fitting at the nose. A Haynes-25 cobalt alloy tubular linkage is used. An Inco-718 nickel alloy fitting at the shell attaches to a 8 Al-1Mo-1V titanium alloy longeron, using a layer of glass rock hard insulation. The linkage uses a 1.0 inch diameter 0.035 inch wall tube, 4.0 inches long.

(C) The upper attachment uses Inco-718 nickel alloy fittings at both the nose and the shell structure. The fittings incorporate a shearface at the vertical webs. The shell structure fitting attaches to two 8 Al-1Mo-1V titanium alloy longerons at the upper surface. A layer of glass rock hard insulation is used at the shearface and the longerons. A vertical stiffener at the aluminum alloy bulkhead reacts fitting kick loads.

2.2.2.4 (U) Nose Structure. The nose structure comprises a modified boundary layer silicide (e.g. TNV-13) coated 90Ta-10W tantalum alloy weldment, or forging, and a tungsten-2 percent thoria nose cap. The nose structure weight also serves to balance the vehicle, permitting the use of heavy material gages with low active stresses. The design incorporates integrally stiffened webs and attachment fittings. The tungsten-2 percent thoria nose cap attaches by means of a locked-in bolt and nut, and transmits shearloads through integral shoulders.

(U) An alternate approach for the nose structure of the flight test vehicle would be the use of hot structure forward of Sta 97.0. The structure forward of Sta 58.0 would be as described, with direct backface attachment. From Sta 58.0 to 97.0, a hot structural shell would be used, including:

- (U) Lower Surface - Unflanged, integrally stiffened panels
  - Internal hat section frames
  - Cb-752/R512E coated columbium alloy

- (U) Upper Surface - Skin-corrugation panels
  - Internal hat section frames
  - Haynes 25 cobalt alloy
- (U) Leading Edges - Rib-supported curved sheet
  - 90Ta-10W modified boundary layer silicide coated tantalum alloy

The hot structural shell attachment concept would be the same as described before.

2.2.2.5 (U) Landing Gear Bays. The nose landing gear bay is contained between Sta 97.0 and Sta 146.5. The primary structure in this area uses 2219-T81 aluminum alloy throughout. External surfaces use Cb-752/R512E coated columbium alloy and Inconel 625 superalloy. Stiffened shearwebs are located at Sta 97.0 and Sta 146.5 which attach directly to the local frames. Two longitudinal webs are located between the transverse shearwebs at 16-inch spacing. These stiffened webs use single corrugation skin and incorporate local reinforcements to transmit landing gear loads.

(C) Figure 45 shows the structural concept for a landing gear door installation. The door uses the insulated and actively cooled concept. The same single corrugation, as used for the skin structure, serves as primary support structure for the door. Hat section beams at 15-inch spacing support the corrugations and serve as hinge attachment members. The heat shield support design, insulation and cooling system is the same as used for the lower surface in general. Flexible cooling system connections are required. The door edges are closed off by means of a minimum gage Haynes 25 cobalt alloy strip. The lower edge of the strip attaches to a scalloped Cb-752/R512E columbium alloy stiffener. The stiffener attaches to the edge of the heat shield and has washers of glass rock hard insulation at the attachments to the strip (avoids direct contact between columbium and super-alloy). The door has an outside and an inside seating face.

(C) The main landing gear bay is contained between Sta 283.0 and Sta 348.5. The structural arrangement of this bay is generally the same as described for the nose landing gear bay, except that two gear cavities extend outboard of the longitudinal webs. The landing gear door installation concept is shown in Figure 45.

2.2.2.6 (U) Access Doors. (C) The external access doors are located on the upper surface between the longerons. The locations include:

- (C) Sta. 147-175
- (C) Sta 178-205
- (C) Sta 238-281
- (C) Sta 364-394
- (C) Sta 401-419

Although the external access doors differ in size, they are identical in structural concept. Figure 45 presents an access door installation concept. An internal and an external installation panel is used. The internal panel uses aluminum alloy single corrugation, the same as the cut-out skin structure in this area. The same insulated and cooled concept is used as for the upper surface in general. The external panel is of the same superalloy structural design as the heat shield panels at the upper surface, except that larger sizes are used. Both panels are screw mounted to attachment flanges with nut plates.

(C) The access door concept as described is characterized by available shear load capability and a minimum weight and heat leak approach. An alternate access door concept would be similar to the landing gear door concept. The door then attaches at the external surface only. An advantage of this concept is single face attachment and improved expansion capability of the external surface (narrower expansion gaps).

(U) Longerons in the area of access doors are reinforced for longer column length capability where frames are under cut.

2.2.2.7 (U) Elevon Support. (C) The elevon support comprises a stiffened shear web which also serves as the aft structural bulkhead for the vehicle. The 2219-T81 aluminum alloy shear web has vertical stiffeners located at the elevon hinge attach points. The stiffeners also transmit heat shield loads resulting from pressures at the aft surface.

2.2.2.8 (U) Flap Support. (C) The flap support comprises a truss-web, located at Sta 395. The 2219-T81 tubular truss-structure has flap hinge attach points at truss intersections. Shear webs are located in the outboard areas for efficient load distribution.

2.2.2.9 (U) Fin Attachment. (C) The fin attachment concept is depicted in Figure 45. A three-point fin attachment is used corresponding with vehicle main frame locations. The central attachment carries loads in all directions. The forward and aft attachment fittings have longitudinal slotted holes for thermal expansion adjustments. The fin beams attach to Inco-718 nickel alloy fittings. These fittings attach to 8Al-1Mo-1V titanium alloy longerons on each side of the fin which extend over the fin chord length. A layer of hard glass rock insulation is used between fitting and longeron to retard the heat flow. Cooling tubes attach directly to the longerons. The attachments are surrounded by microquartz fibrous insulation.

2.2.2.10 (U) Fin, Elevons, and Flaps. (C) The fin, elevons, and flaps are similar in construction. A hot structure multi-spar multi-rib design is used with vierendeel truss-webs. The fin and flaps use Haynes 25 cobalt alloy skin-corrugation panels predominantly. The elevons use Cb-752/R512E coated columbium alloy integrally stiffened panels at the lower surface and Haynes 25 cobalt alloy skin-corrugation panels at the upper surface.

2.2.2.11 (U) Leading Edges. The leading edges for the fin and lower body surface have a 1.5-inch radius and are made of 90Ta-10W modified boundary layer silicide coated tantalum alloy. They consist of a curved sheet, supported by ribs and leading edge beam sections. The aft body upper leading edges have a 3-inch radius and are made of Cb-752/R512E coated columbium alloy.

SECTION 3

(U) SYSTEMS

(U) This section of the report discusses the various systems and subsystems required for the high-L/D unmanned flight test vehicle. These include:

- (U) General Arrangement
- (U) Guidance and Navigation System
- (U) Flight Control System
- (U) Data Management System
- (U) Tracking and Command System
- (U) Environmental Control System
- (U) Electric Power System

3.1 (U) GENERAL ARRANGEMENT

(U) The general arrangement drawing, Figure 46, indicates the equipment locations established in the preliminary design effort. The systems equipment is categorized for arrangement purposes as follows:

A. (U) Those items whose locations are largely dictated by dynamic or radiation considerations (landing gear, antennas, reaction control motors and destruct devices).

B. (U) Internal pieces of equipment, the location of which is discretionary (guidance and control system, environmental control system, power supply and expendables).

(U) The landing gear arrangement is proportioned for minimum rollout then adjusted to provide internal stowage clearances and to meet the HIAD turnover requirement of 27°. The turnover angle is met with wholly noncompressed shock attenuators which is conservative and should provide a satisfactory margin to compensate for the non-steerable nose gear. The landing gear consists of dual metal nose wheels and main gear metal brush skids. Shock attenuation is accomplished by employment of crushable honeycomb cylinder inserts. All three spring-initiated free fall struts are installed so as to deploy aft, thus exploiting the free stream drag force. The down-lock braces are sized to carry a considerable longitudinal load component, but the side load must be carried to the strut trunnions, as sway braces are eliminated to minimize door sizes.

(U) The reaction control and landing motors are mounted on the exterior of the base end as there is no specific aerodynamic requirement for flushness there. The four 500-pound thrust landing motors are closely clustered on the longitudinal centroidal axis as they may be selectively fired to provide a landing equivalent L/D ratio range of 2.0 to 6.0. The motors are placed so as to minimize jet impingement upon the spacecraft.

(U) A destruct system is provided to assure ballistic rather than aerodynamic flight, should a control failure occur. Loss of an elevon will cause a substantial roll rate, thus negating aerodynamic lift. The left-hand elevon is supported by torque tube drive splines through the bearing/seal assemblies. The spline can be driven out of the elevon torque tube by pyrotechnic charges, thus severing the elevon.

(U) The remaining items of equipment (Category B) are all located within the cooled inner structure. Their locations have been selected using these criteria: Minimize nose ballast by placement of those items such as batteries far forward; minimize trim changes by placement of expendables at or near the center of gravity; minimize access doors or hatches and their attendant sealing problems, and establish an orderly grouping of like components for ease of installation and serviceability.

(U) In consideration of the first criterion, the four battery boxes have been placed adjacent to and immediately aft of the nose landing gear assembly. Also, a liquid system for the landing rockets was selected over solids in order to place the propellant with the other expendables. The expendables, hydrogen peroxide, environmental control system water and ammonia, have been placed about the center of gravity.

(U) Access doors along both sides of the vehicle were investigated but they provided less efficient accessibility and resulted in a greater overall door perimeter. Hatches on the vehicle upper surfaces were selected as they provide good access to critical compartments and much easier access when the vehicle is in the launch position. Landing gear doors are utilized in lieu of access hatches for their respective sections of the airframes and all routing is through the wheel wells. The nose wheel well provides transit for instrumentation wiring to the nose and contains two battery boxes mounted on webs on either side of the strut. The battery boxes can pivot about their lower aft corners to provide access to the batteries inside without complete removal of the container. Access to the mid fuselage is through a main landing gear well inboard web.

(U) An access hatch is identified by the equipment intrinsically associated with it, thus the remaining equipment placements are described on a hatch-to-hatch basis, proceeding aft along the vehicle.

3.1.1 (U) Antenna Hatch

(U) The antenna hatch on the upper right hand side of the fuselage at stations 63 to 81 covers the landing X-band antenna and provides access to instrumentation between the hot nose structure and the nose wheel well.

3.1.2 (U) Battery Hatch

(U) The battery compartment hatch is located on the upper fuselage and between the longerons and immediately aft of the nose wheel bay at stations 147 to 175. Removal of this door exposes the remainder of the battery complement. Each of the two battery boxes may be pulled out on extension rails. This is particularly convenient in the launch position where the batteries are installed. For removal after landing, when the vehicle is horizontal, it is necessary to employ a small hoist. Also in this compartment on a pull-out rack are a number of the avionics packages particularly selected as those most likely to be serviced or inspected. A cable take-up reel is supplied on the left-hand side of the rack to accommodate wire bundles, so that it is not mandatory to disconnect the system for maintenance. Major items on this rack are the main junction box, the voltage regulator and inverter, the PCM deck and tape recorder, and the destruct system interlock, safe arm and squib battery.

(U) An alternate arrangement with fuselage side doors was investigated and found to be unsatisfactory. Such an arrangement would require a structural centerline web and result in a less efficient packaging of the battery complement.

3.1.3 (U) Avionics Hatch

(U) The avionics compartment occupies the next two ring spaces aft, with sills at stations 178 and 205. This compartment contains the larger units of avionic equipment such as the inertial measurement unit, computer, and communications. A horizontal mounting surface is provided 20 inches below the side sills, well within reach. To further facilitate access, the outboard portions of the panel are canted upwards. Each unit is spaced so that it may be extracted without requiring the displacement of its neighbors.

3.1.4 (U) Expendables Hatch

(U) The expendables compartment hatch, station 238 to 281, is located just forward of the main landing gear wells. This compartment occurs at the center of gravity of the vehicle by selection, and where, fortuitously, the inner structure is deep enough to accept a man. To exploit this, the area has been provided with a lightweight floor and expendable tanks are arranged about the hatch opening but not directly beneath it. Should a technician kneel in this compartment he has, to his right, two of five hydrogen peroxide tanks, a water tank, and the pressurant tank. To his left are similar tanks and the hydraulic pump and reservoir. A large 5-cubic-foot water tank is at the forward end of this compartment. All ancillary fluid systems equipment are peripherally mounted about the compartment so as not to interfere with access to the large tanks. To achieve these features, it is necessary to utilize multiple tankage. All fluid and electrical lines are routed from here to the main landing gear wells for transport past the mid-fuselage.

3.1.5 (U) Jpper Actuator Hatch

(U) The upper actuator hatch at stations 364 to 394, just outboard of the fin on the right hand upper surface is employed to gain access to the major aft fuselage. Items in this area are the upper flap actuator, and the extension aft of fluid and electrical routing from the landing gear bays. The fuselage is deep enough for entrance at this point and occasional foot and knee support structure will protect the inner spacecraft skin.

3.1.6 (U) Aft Actuator Hatches

(U) The aft actuator hatches are in the upper structure beneath the upper flaps and provide access to the fuselage aft of the main truss frame supporting the flaps. To use these doors, it is necessary to disconnect the linkage from the actuator (via the upper actuator hatch), or to drive the flaps to the maximum open position, thus exposing a disconnect point. The bellows seal under the piano hinge must be capable of sustaining the subsequent extreme positioning of the flaps. In this area all reaction motor connections are available. The elevon actuators are directly below the hatches.

(U) Hatches in the base surface of the vehicle were seriously considered, but not employed. Reasons for the rejection included the possibility of damage to the elevons which might result when using the hatches, and the possibility of aerodynamic fairings being added in that region at a later date.

3.2 (U) GUIDANCE AND NAVIGATION SYSTEM

(U) The Guidance and Navigation System consists of an inertial platform and a general-purpose digital computer. The inertial navigator has been selected on the basis that an autonomous continuous means of determining the research vehicle position, velocity vector, and attitude is necessary. For a three-quarter orbit, continuous ground tracking is generally not available, thus making self-contained on-board guidance and navigation mandatory. Precise attitude control is required because the research vehicle enters the atmosphere in a preprogrammed glide maneuver which reduces the energy of the vehicle without causing an excessive rise in skin temperature in the lower atmosphere.

(U) The selected guidance and navigation system consists of:

- (U) Inertial platform (4-gimbel) or a strap-down platform (no gimbals)
- (U) General-purpose digital computer.

3.2.1 (U) Altitude Divergence

(U) When inertial platforms are employed in orbital or near orbital altitudes and altitude measurements are not available external to the inertial system, a difficulty arises because of altitude divergence. Although a radar altimeter could provide this measurement, a situation may arise where no radiation emanating from the vehicle can be reliably accomplished or permitted.

(U) The vertical channel accelerometer measures the sum of gravitational and kinematic accelerations. To separate the kinematic accelerations from gravity, a computation is made in the computer in which the exact value of gravity for a given altitude is subtracted from the total measured vertical acceleration. Also, involved in the computation are the earth's mass and the distance from the center of the earth to the present location of the platform. If altitude information is unavailable from an external source, the computer must compute altitude from the vertical acceleration measured in the vertical channel. Any error introduced into this computation leads to a divergence in computed altitude. For example, assume the computed altitude is higher than the actual altitude. Since gravity is less at a higher altitude a smaller value of gravity than that required for the actual altitude is subtracted from the measured vertical acceleration. Even though the actual altitude has not varied, the system senses a net acceleration in the vertical which indicates an apparent increase in altitude. This apparent increase in altitude requires a smaller value of gravity to be used in the next computation, thus indicating a further increase in apparent altitude. Thus, the altitude computation diverges with time.

(U) The altitude divergence induces position error in the horizontal channels (latitude and longitude) because the gain of these channels contains a term which is a function of altitude. Specifically, this term is "distance from the center of the earth to the platform" which is the sum of earth radius plus altitude above the earth.

(U) The position error of a one-nautical-mile-per-hour CEP class system can be as large as 100 nautical miles toward the end of a three-quarter orbital mission due to altitude divergence. Various methods are available for removing or minimizing this effect. Some methods are:

- (U) Radar altimeter measurements
- (U) Open loop programming of altitude flight profile
- (U) Aerothermodynamic sensing of vehicle environment
- (U) Discrete altitude measurements via satellite-tracking networks with updates transmitted to research vehicle.

(U) With respect to the method using a radar altimeter it may be noted that suitable narrow pulse radar altimeters are available. These altimeters have a pulse width of 100 nanoseconds, operate on C-Band, and yield an accuracy of less than 100 ft, using digital outputs. For a maximum altitude of 400,000 feet, a dish antenna of approximately 20 inch diameter would be used. The details of the antenna and installation require further investigation. The total system weight is in the order of 20 pounds.

3.2.2 (U) System Requirements

(C) During launch and boost only the navigation function (position monitoring) of the guidance and navigation system is operating. The guidance functions remain passive until after the research vehicle has been separated from the booster. After separation, the guidance system is activated so that the reaction control system (RCS) stabilizes the vehicle in the attitude necessary for entry and removes angular rates imparted by staging.

(C) During the glide portion of the flight, the vehicle performs maneuvers according to a predetermined flight profile. The flight profile program is stored in the computer. This program calls for a continuous reduction of vehicle energy as constrained by the aerodynamic and structural capability of the vehicle. When the vehicle approaches to within 150 n. mi. of the landing area, the controller decouples the vehicle guidance system and vectors the vehicle to the runway. The automatic landing system (Sperry RCRS-X-20 remote control system) utilizes a control radar at the landing site. It is a "ground control approach" system with "fly-to-nominal flight path" mode. The navigation system is required to navigate the research vehicle from launch through glide to interception of the approach window which is 500 n. mi. off the coast of California. The 500 n. mi. acquisition is accomplished by the tracking radar located at Pt. Mugu. The dimensions of the approach window are constrained by vehicle design parameters to be:

Down Range	±50 n. mi.
Cross Range	±20 n. mi.
Velocity Vector	50 ft/sec

To meet these constraints an inertial navigator in the 1 to 5 nautical mile per hour CEP class has been selected.

(C) Inertial navigators in the 1 to 5 nautical-mile-per-hour class are available off-the-shelf. A listing of some representative systems is given in Table 7. The accuracies of the systems listed below vary from 0.5 nautical mile per hour to 5 nautical mile per hour CEP. All the systems listed are gimbal platforms with the exception of the Honeywell SIGN III which is a strap-down inertial navigator. While a strap-down system appears more desirable at this time because of the inherent advantage* over gimbal systems, a selection would require a computer simulation to determine the performance of the various platforms with several forms of altitude divergence suppression.

*No gimbals, in most cases lower weight, smaller size, high immunity to shock, high reliability.

TABLE 7

(C) PARTIAL LIST OF INERTIAL GUIDANCE SYSTEMS

Company	Weight (lb)	Volume (ft^3)	Comments
AC Electronics	44.0	1.10	Carousel 5 - military version of Carousel 4 slated for Boeing 747
Autonetics	100.0	2.58	N-16 Selected for MK II (F-111)
Kearfott	57.0	1.50	In production for Lockheed P-3C
Litton	60.0	1.50	LN-15
Minneapolis-Honeywell	41.7	0.61	SIGN III STRAP-DOWN Inertial
Nortronics	123.0	2.82	In production for Lockheed C5A
Teledyne	12.3	535 in^3	CONFIDENTIAL

3.2.3 (U) Adequacy of Inertial Guidance

(C) Prior experience with computer simulations of inertial navigators in the one-nautical mile class for different flight profiles, indicate that this class of navigator is adequate. It is noted, however, that platforms such as Bell Aerosystems Hypernas III are available in the "one tenth nautical mile per hour CEP class." Thus, it can be assured with a high degree of confidence that an inertial navigator without a need for position updating is adequate for this mission.

3.3 (U) FLIGHT CONTROL SYSTEM

(U) The Flight Control System is composed of an autopilot, a reaction jet control system, an aerodynamic control system, and a landing propulsion system. Signals from the vehicle guidance and navigation system and body motion sensors are blended and conditioned by the autopilot. Output signals from the autopilot actuate the reaction and aerodynamic control systems. The reaction control system produces moments about the vehicle roll, pitch and yaw axes for stabilization and control at altitudes where aerodynamic control surfaces are ineffective. Reaction control about the yaw axis is also used during flight within the atmosphere. The aerodynamic control system actuates control surfaces for atmospheric maneuvering. The landing propulsion system produces thrust to increase the equivalent lift/drag ratio during landings, and is controlled by the guidance and navigation system.

3.3.1 (U) Autopilot

(U) The system incorporates the same autopilot selected as a result of previous studies of the F-5 configuration (Ref.1) stability and control requirements. It is a fixed-gain general-purpose unit being developed by the Sperry Company for maneuverable reentry vehicle applications, and provides the following functions:

- (U) Three-axis attitude stabilization
- (U) Operational mode logic
- (U) Gain and bandwidth control
- (U) Stored programs for trajectory control and insertion of disturbance transients
- (U) Provisions for accepting and storing remote control commands
- (U) Signal conditioning for telemetered flight control system data

(U) A signal conditioning unit is included for use in missions which require changes in autopilot gain as a function of air data or inertial navigation signal inputs.

3.3.2 (U) Reaction Control System

(U) A schematic of the Reaction Control System is shown in Figure 47. A monopropellant fuel (hydrogen-peroxide) system is used, in accordance with results from a previous analysis (Ref. 1) which show that this type of system is best suited to FDL-5 mission requirements. The minimum number of reaction thrust motors required for control about all three vehicle axes are shown. Thrust motor propellant inlet valves are controlled by signals from the autopilot. Volumetric considerations dictate the use of three propellant storage tanks. The tanks also store fuel for the landing propulsion system. A single pressurization gas tank is used to pressurize the propellant tanks and the environmental control cooling system.

(U) System design criteria are:

- (U) Off-the-shelf equipment will be used whenever possible to minimize costs, employ proven components, and avoid procurement problems.
- (U) A rate of change of vehicle attitude on the order of 10 deg/sec about all three axes may occur upon separation from the launch vehicle.

- (U) The reaction control system must provide an angular acceleration capability on the order of 5 deg/sec^2 about all three axes for satisfactory vehicle control.
- (U) Total missior time from launch vehicle separation to recovery is 2400 seconds.
- (U) Reaction control of pitch and roll attitude is required only during the first 300 seconds of the flight regime, after which aerodynamic control surfaces are used.
- (U) Reaction control in yaw is employed during the entire flight regime.
- (U) During reaction control periods, the system duty cycle is on the order of 10 percent.
- (U) An extra propellant margin of 10 to 15 percent is provided.
- (U) Hydrogen peroxide specific impulse is 150 seconds.
- (U) Environmental control cooling is available as required to protect system components from the reentry environment.

3.3.3 (U) Aerodynamic Control System

(U) The vehicle is aerodynamically stabilized and controlled by two elevons and two flaps. The elevons and flaps are located on the aft lower and upper surfaces, respectively. The elevons are deflected symmetrically for pitch control and deflected differentially for roll control. The flaps are deflected symmetrically and in conjunction with symmetrical elevon deflections to provide more effective aerodynamic control in pitch at low supersonic and transonic speeds. Stability augmentation commands are added to control and guidance commands to stabilize the vehicle. Previous studies (Ref. 1) of aerodynamic characteristics show the dutch roll mode is lightly damped and that if artificial damping is introduced by means of aileron action, the oscillations are transferred into the yaw plane in the form of a "flat dutch roll." It is impossible to completely damp this motion without rudder action or similar lateral forces. However, for the unmanned vehicle this motion is not considered so serious as to require the addition of rudder action. In addition, the yaw reaction control system, which is operative during the entire flight regime, will provide forces acting to damp out lateral oscillations.

(U) Control surface actuation systems capable of operation at high temperatures were investigated to determine their present state of development and future growth potential. Candidate systems are:

- (U) Mechanical
- (U) Electrical
- (U) Pneumatic
- (U) Hydraulic
- (U) Liquid Metal

3.3.3.1 (U) Mechanical. Mechanical systems employ a high speed, low torque power takeoff from a prime mover which is converted to a low speed, high torque output in a gear box. Motion is applied to the control surface hinge point through mechanical clutches. A system developed by Curtiss-Wright Corporation has been tested at 600°F. Failures caused by clutch material swelling and changes in the characteristics of clutch springs limit sustained operation of the system to periods of about ten hours. Development activity in this field appears to be very limited.

3.3.3.2 (U) Electrical. There is no evidence that development of high temperature electrical flight control systems is currently sponsored. Atomics-International has developed motors, actuators, position transducers and other components for nuclear reactor control which can be operated continuously at temperatures of 1000°F to 1300°F; however, these components are not directly applicable to flight control systems.

3.3.3.3 (U) Pneumatic. Pneumatic systems include hot gas systems, in which the working fluid is obtained by burning a solid propellant. These are basically one-shot, short-duration systems and are not applicable when sustained control is required. Dynamic sealing and lubrication problems occur in systems designed for prolonged high-temperature, high pressure operation. Gas compressibility may result in degraded system performance. There are apparently no high pressure pneumatic servo systems capable of satisfactory operation at 1000°F.

3.3.3.4 (U) Hydraulic. There is currently considerable activity in the development of high temperature hydraulic systems. Hydraulic fluids, seals, and bearings capable of operation at high temperatures are being investigated. The XB-70 hydraulic system is designed to operate at 4000 psi pressure with fluid temperatures from -65° to +450°F. It is expected that the upper temperature limit of hydraulic systems will be approximately 800°F.

3.3.3.5 (U) Liquid Metal. General Electric is currently engaged in sponsored development of liquid metal systems. A eutectic alloy of sodium-potassium-cesium which remains liquid from -102°F to +1332°F is used. Pumps, servo-valves, actuators and accumulators have been tested at 1000 psi and 1000°F with promising results. The feasibility of a highly reliable servo valve with no moving parts is being investigated. A unified system package is scheduled to be flight tested in 1969. It is planned to produce an operational system by 1972.

(U) On the basis of current industrial activity, availability of standard, proven components, anticipated system operating temperatures and future growth potential, a cooled hydraulic system was selected for the vehicle control surface actuation system. This will permit an initial design using available components, which can be upgraded as the state of the art of high temperature hydraulic systems advances. Replacement of the hydraulic system with a liquid metal system capable of operation at higher temperatures will not require extensive changes in the design of the control system.

(U) A schematic of the surface actuation system is shown in Figure 48. The system is powered by a constant pressure, variable delivery 3000 psi pump which is driven by a dc motor. An accumulator is provided to protect against pressure surges and ensure adequate surface response rates under high load conditions. Pressure and return line filters are used to protect system components from particle contamination. Electro-hydraulic servo-valves are used to position control surface actuators according to command signals from the autopilot. All hydraulic system components which will be cooled or used under low temperature conditions can be selected from off-the-shelf components for existing aircraft.

3.3.4 (U) Landing Propulsion System

(U) The Landing Propulsion System employs hydrogen peroxide thrust motors, catalyst beds, and solenoid operated propellant inlet valves similar to those used in the reaction control system. Propulsion fuel and pressurization gas are stored in tanks common to both systems. Four landing propulsion motors are mounted at the aft end of the vehicle. The motors are controlled by the guidance and navigation system.

(U) Design criteria are:

- (U) Off-the-shelf equipment will be used whenever possible to minimize costs, employ proven components and avoid procurement problems.
- (U) Each motor is controlled by a separate propellant inlet valve to permit changes in thrust level in increments of 25 percent of maximum thrust output.
- (U) Thrust output of each motor is 500 pounds.
- (U) Sufficient propellant is stored to permit a burn time of 20 seconds with all four thrust motors operating.
- (U) An extra propellant margin of 10 to 15 percent is provided.
- (U) Hydrogen peroxide specific impulse is 150 seconds.
- (U) Environmental control cooling is available as required to protect system components from the reentry environment.

3.4 (U) DATA MANAGEMENT SYSTEM

(U) The Data Management System includes all aspects of flight data acquisition, transmission, postflight data handling, and postflight data processing. The basic functions of the system are to obtain basic research objectives of the vehicle, make accurate vehicle flight performance measurements, make vehicle evaluation measurements, obtain diagnostic data, and produce accurate records from which meaningful vehicle flight performance analyses and evaluations can be made.

(U) Existing USAF equipment without modification may be used, whenever possible, thereby minimizing development and new equipment procurement. The basic requirements for the data management system are determined from the data mission profile and the objectives established for the program.

(U) For any given vehicle test, the time cycle for data acquisition and postflight data processing including the issuance of reports is estimated to be 30 to 60 days after vehicle launch. A typical data time sequence and reporting plan is given as follows:

1. (U) Prelaunch, calibration and checkout	30 days
2. (U) Launch countdown	6 hours
3. (U) Vehicle flight and landing	3 hours
4. (U) Issue quick-look TWX	*TD + 2 hours
5. (U) Issue quick-look report	TD + 24 hours
6. (U) Deliver data tapes to processing facility	TD + 4-11 days
7. (U) Preliminary diagnostic report	TD + 5-12 days
8. (U) Issue diagnostic report	TD + 6-13 days
9. (U) Issue time history report	TD + 8-15 days
10. (U) Issue flight performance report	TD + 20
11. (U) Issue design analysis report	TD + 30
12. (U) Issue test evaluation report	TD + 60

*TD - Denotes vehicle touch-down on landing field.

(U) Data will be acquired by six media:

- (U) Ground radar plus optical trackers
- (U) Preflight wired calibrations and countdown
- (U) Real-time data transmitted from the vehicles to the telemetry ground receiving sites.
- (U) Delayed-time data recorded on board during flight and played back through a radio link to the telemetry ground sites.
- (U) Delayed-time data recorded on board during flight and recovered from the vehicle after landing.
- (U) Visual monitoring of ground tracking instruments, vehicle flight, and real-time telemetered functions by observer personnel in the launch blockhouse and downrange sites.

(U) It is required that data be handled in several forms, including analog time plots, analog data tapes, teletype, verbal communications, digital tabular readouts and digital tapes. All data acquired will be gathered from the recording sites, transported to the data processing sites, edited and refined, correlated with tracking data, adjusted for sensor installation factors, converted to computer format. analyzed by computer programs, and converted to analysis, performance, and evaluation reports. A prime factor in meeting the data schedule is transporting the data to the processing site within six days. This can be accomplished by effective planning and utilization of range transportation facilities.

3.4.1 (U) Functional Description

(U) The Data Management System operation is simple and straightforward. The vehicle sensors are calibrated by applying standard physical functions from the sensor calibration ground support equipment during the preflight checkout and calibration period. The output of the sensors is fed into the PCM multiplexer encoder and RF transmitters. The transmitted signals are received by the launch-site receivers and the calibration data are recorded. The PCM monitor is used to check the quality of the PCM waveforms. After launch, the sensors respond to the vehicle physical input parameters and feed varying sensor signals to the multiplexer and PCM encoder. The PCM serial wavetrain is fed simultaneously to the VHF transmitter, microwave transmitter, and on-board tape recorder. The transmitted radio frequency signal is received by one or more of the receiving sites. During VHF blackout due to thermal plasma, the microwave frequency is expected to penetrate and provide data. After blackout the VHF frequency provides a link for data again. The data received by one or more of the ground sites are combined with the standard IRIG 17-bit time code and recorded on the ground tape recorder. The time-code correlation between receiving sites is maintained at 1 msec per hour using standard range equipment referenced to National Bureau of Standards radio transmissions.

(U) During flight, certain selected measurements are demodulated after reception and recorded on a 15-channel analog recorder for supplying quick-look data. After the vehicle has landed, the on-board tapes recorded during flight are copied, then forwarded to the data processing site. The tapes from the intermediate sites are also picked up and forwarded to the data processing site. At the data processing center the large number of tapes are edited and condensed to a single master data tape. After editing, the serially transmitted PCM and analog data are recorded and formated, then recorded on a tape for use with an IBM 7090, 7094, or 360 computer. The master computer tape is then used with the computer to add data calibrations, sensor installation corrections, and tracking data. The correlated and adjusted data are then used to print out a time history plot for each measurement.

(U) Further computation is accomplished for thermal analysis, structural analysis, flight control analysis, vehicle performance analysis, and design evaluation analysis. The results of each analysis are printed out by digital plotters wherever applicable.

3.4.2 (U) Data Recording and Format

(U) The parameters to be measured have been defined as shown in Table 8. A total of 315 measurements are suggested, of which only 285 can be transmitted via the telemetry system. The remaining 30 measurements are high-temperature strain measurements made with scratch gages which have to be retrieved from the vehicle after landing for manual readout.

3.4.3 (U) Telemetry System

(U) Analysis of the sensor requirements indicates that the telemetry system signal stimuli voltage levels and types of functions to be telemetered can be grouped into low-level voltages, high level voltages, binary-coded pulse trains, and analog vibration waveforms as shown in Table 9. Since a data accuracy of ±0.5 percent is required, it is recommended that the PCM system be employed.

(U) Missile and space programs thus far have had little reason to develop communication systems to penetrate plasma blackout. Payload shapes are ballistic in nature and the length of time that the vehicle is undergoing plasma conditions is short. Thus, the decision has been to live with the blackout and reestablish communications when it is over. Two programs (Asset and Dyna-Soar) have had occasion to need communication equipment that minimizes the plasma attenuation.

(U) The Dyna-Soar telemetry system operated on K_u band and thus would not be affected by the plasma. Development of the equipment, primarily the ground equipment, was not completed as it is highly complicated and would be expensive. In addition, the airborne equipment is large and heavy for a vehicle such as the flight research vehicle.

A comparison of ASSET and PRIME telemetry system with this vehicle's telemetry system is presented in Table 10.

TABLE 8

(U) SENSOR REQUIREMENTS

Measurement	Sensor Range	Sensor Type	Output Range	CPS Response	Number Req.
Temperatures A & B	100-1700°F	T/C PT/Pt 10% RH	0.24-8.7 MV	0.05	64
Temperature C	100-1500°F	T/C Chromel/Alumel	1.61-34 MV	0.2	25
Temperature D	100-600°F	T/C Iron/Const	5.3-17 MV	0.2	7
Temperature E	100-2880°F	T/C PT/PT/10% RH	0.24-14.7 MV	0.015	16
Temperature F	100-4000°F	T/C W 5% R_e/ W26% R_e	0.13-7.6 MV	0.008	3
Skin Pressure (A-B-C)	0-0.2 psia	Potentiometer	0-5V	0.01	50
Vibration	±10G	Piezo Electric	0-12 MV P-P	4000	9
Structural Strain (A-B-C)	---	120 ohm Bridge	0-10 MV	10	40
Structural Strain (D-E)	---	Scratch	No electrical output	--	30
Pressure (ECS)	0-50 psia	Potentiometer	0-5V	0.1	2
Pressure (ECS)	0-500 psi	Potentiometer	0-5V	0.1	1
Pressure (ECS)	0-0.2 psia	Potentiometer	0-5V	0.1	1
Differential Press. (ECS)	0-50 psia	Potentiometer	0-5V	0.1	1
Flow Rate (ECS)	0-750 lb/hr	Magnetic	0-5V	0.1	2
Temperature (ECS)	30-100°F	T/C Iron/Const	0-3 MV	0.2	5
Cold Junction	30 ± 5°F	T/C Iron/Const	±0.1 MV	0.05	1
X, Y, Z, Angle Rate	±20° sec	Rate Gyro	±2.5 V	5	3
X, Y, Z Acceleration	±200 ft/sec^2	Diff Xfmr	0-5V	5	3
Airspeed, Subsonic	0-500 knots	Diff Xfmr	0-5V	1	1
Altitude, Barometric	0-20,000 ft.	Diff Xfmr	0-5V	1	1
Battery Voltage	0-40V	Voltage Divider	0-5V	0.05	3
Converter Voltage	0-150 VAC	Rectifier Divider	0-5V	0.05	1
Current	0-10 MA	Magamp	0-5V	0.1	15
R.F. Power	0-20 watts	Diode & Divider	0-5V	0.05	4
Vehicle Attitude	±80°	Binary Coded	0-5V		3
Altitude (Computer)	0-250,000 ft.	Binary Coded	0-5V		1
Velocity Forward	0-20,000 ft/sec	Binary Coded	0-5V		1
Control Surface Position	±40°	Shaft Encoder	0-5V		4
Velocity Up/Down	0-10,000 ft/sec	Binary Coded	0-5V		1
Velocity Right/Left	0-10,000 ft/sec	Binary Coded	0-5V		1
Up Range Distance Position	0-5000 N.M.	Binary Coded	0-5V		1
Up Range Position	0-500 N.M.	Binary Coded	0-5V		1
Miscellaneous			0-5V		14
TOTAL					315

UNCLASSIFIED

TABLE 9

(U) TELEMETRY SYSTEM INPUT REQUIREMENTS

Type	Input Voltage	No. of Channels
Low Level (PCM)	0-20 mv	136
Low Level (PCM)	0-40 mv	25
High Level (PCM)	0-5 v	102
Binary Code (PCM)	0-5 v	13
Total PCM Channels		276
Vibration	0-12 mv peak-to-peak	9
Total Channel Transmitter		285
Non Transmitted Data (Scratch Gages)		30
Total Measured Functions		315

UNCLASSIFIED

TABLE 10

(U) COMPARISON OF TELEMETRY SYSTEMS

	Selected for High L/D	Asset	Prime
Type of Telemetry	PCM-FM-FM, PCM	PDM-FM-FM PPM	PDM-FM-FM
Frequency	VHF and MICROWAVE (X or KuBand)	VHF and X-Band	VHF
Channels	285	150	270
Accuracy	0.1%	2.0%	2.0%

UNCLASSIFIED

(U) The Asset telemetry system operates at X-band and has been used operationally. It is smaller, lighter and requires less complicated ground equipment. However, this system uses an analog method of transmission, and for a high altitude injection, may undergo some blackout.

(U) Plasma attenuation and communication ranges are important factors in the choice of the radio frequency used and power output requirements. The severity of the ionized sheath in terms of transmission frequency and degrees of attenuation and reflection, is a function of altitude, velocity, angle of attack, and vehicle shape.

(U) The telemetry system characteristics are:

1. (U) The communication frequency for the hypersonic portions of the flight is between 3.0 and 13.5 GH_z.

2. (U) The modulation is PCM with eight-bit quantization with one-bit parity. Precise outputs such as stable platform outputs will be encoded in two digital words. Vibration and acoustic data are transmitted as FM/FM using either translated subcarriers or non-IRIG subcarriers.

3. (U) Means for transmission of data, accumulated during periods of no ground communication, are provided by play back at an accelerated rate as soon as ground contact is possible. Because vehicle-generated RF power is limited, too great an increase of bandwidth would deteriorate the data, so that the delayed-time data may need to be selected.

4. (U) Both microwave and VHF transmitters are provided for both real and delayed time links.

5. (U) A "hard copy" recorder is installed to provide a complete record of the flight. This tape is recovered at the completion of the flight and will be the best data source for the long-range postflight data reduction and analyses.

(U) The VHF transmitters are currently available as off-the-shelf items from a number of vendors.

(U) The microwave transmitters are of one design. The transmitter power output requirement is tailored to the conditions of the delayed time link. The real-time transmitter has an excess of power. Even though microwave transmitters have a low overall efficiency, the few minutes of operation is short enought not to place too great a burden on the power system. A programmed shutoff of the microwave system can be employed and the VHF transmitter alone can be used where possible.

(U) The microwave transmitter output power requirements at candidate frequencies for the real-time link, high-altitude injection mode are 12.5 watts at X-band and 20 watts at K_u band. If a low-altitude injection is possible, a frequency of 4 or 5 GH_z can be used. This would reduce the transmitter output requirements to 8 watts. The delayed-time links will require 2.7 times the output power of the real-time links because of the wider bandwidths necessitated by the accelerated tape playback.

(U) To keep the number of antennas to a minimum, RF multiplexers are used to enable the transmitters and the receivers of the command link to use the same antenna simultaneously. One RF multiplexer is required for microwave and one for VHF. All of the antennas, whether microwave or VHF are flush with the outer surfaces of the vehicle. The data link and telemetry antenna primary radiation is oriented toward the ground station.

(U) The antenna dielectrics are of high-temperature quartz or equivalent material and offer a smooth surface in order to void any adverse affects on flight characteristics or aerodynamic heating. The antenna elements are fabricated of materials capable of withstanding the high temperatures experienced in hypersonic flight. Typical construction techniques for the individual antennas are documented in Ref. 4. Figure 49 illustrates the tentative installation of all antennas required for telemetry, tracking, terminal guidance and the destruct system of the research vehicle.

3.4.4 (U) Ground Equipment Required

(U) A summary of the Data Management System major equipment requirements is given in Table 11. The greatest portion of the equipment complement is Government-furnished (GFE) and presently exists at E.T.R. Use of the equipment will be requested at the time the vehicle test is scheduled. Special equipment for the specific vehicle such as the sensor calibrator, launch-site telemetry control, vehicle sensor installation and the vehicle telemetry installation must be designed and constructed.

3.5 (U) TRACKING AND COMMAND SYSTEM

3.5.1 (U) C-Band Transponder/Skin Track

(U) Tracking is accomplished by a C-Band transponder and by skin track. Existing radars such as the AN/FPS-16 and the AN/TPQ-18 can be used to perform tracking.

3.5.2 (U) Approach and Landing System

(U) The Approach and Landing System proposed is that developed by Sperry-Phoenix Company under USAF Contract AFF33(657)-9614, "Remote Control Recovery System." In general, a mobile remote control radar at the landing site is used as a "ground control approach" utilizing a "fly-to-nominal-flightpath" mode. Early energy-management control is accomplished by the onboard guidance. The vehicle is automatically acquired when within range of the ground control radar. The letdown is controlled to an equilibrium glide as described previously. At an altitude of 1000 feet, a flare maneuver is initiated automatically by altitude. This pitches the vehicle from the steep glide angle to a deceleration glide path which intersects the ground at the intended landing point. Although many of the more critical modes of flight are controlled by ground computers, a human ground controller can override the automatic control by observing the radar profile plots and flight data telemetered from the vehicle.

(U) The ground station is van-mounted and is both mobile and air-transportable. The van contains the tracking radar, the command control equipment, the flight data telemetering equipment and, in addition, is provided with UHF communications, recording equipment, air conditioning equipment and a water cooler, making it completely self-sustaining. Three operating positions are provided:

TABLE 11

(U) MAJOR EQUIPMENT SUMMARY

Equipment Item	Quantity Required				Total	Avail-ability
	Vehicle	Data	Launch Site	Enroute Sites (Estimated)		
Sensor Calibrator			1		1	Make
PCM Monitor			1		1	Buy
Telemetry Control			1		1	Make
VHF Receiver			1	4	5	GFE
Microwave Receiver			1	4	5	Buy
PCM Demodulator			1	4	5	GFE
Quick Look Recorder			1	4	5	GFE
PCM Tape Recorder		1	1	4	6	GFE
Vehicle TM Installation	1				1	Make
Vehicle Sensor Installation	1				1	Make
Time Code Generator			1	4	5	GFE
PCM Tape Playback		1	1		2	GFE
Analog Tape Playback		1	1		2	GFE
Visual Tape Editor		1	1		2	GFE
Analog Tape Recorder		1	1	4	6	GFE
PCM to 7094 Converter		1			1	GFE
A/D 7094 Digital Computer		1			1	GFE
7094 Printer		1			1	GFE
7094 Card Punch		1			1	GFE
Card Sorter		1			1	GFE
7094 Digital Plotter		1			1	GFE
Stromberg Carlson Microfilm		1			1	GFE

UNCLASSIFIED

(1) radar operator, (2) preliminary controller, and (3) auxiliary controller. Although the radar is provided with automatic acquisition, controls are provided to accomplish manual acquisition.

(U) The flight control console contains the command and control panel, the real-time flight data, a two-pen plotting board, and communication panels.

(U) The command X-Band data link is provided by pulse-code modulating the radar; up to 31 commands can be provided. The telemeter link is provided by pulse code modulating the transponder reply. Eight channels of data are provided. This data link is used to transmit on-board sensed data such as attitude, angle of attack, etc.; and is in addition to the instrumentation telemetry link. The airborne transponder weighs 18 pounds, has a volume of 475 cu in. and requires 60 watts from a 28 vdc source. The antenna for the landing transponder is a small flush mounted antenna.

3.5.3 (U) Destruct System

(U) The Destruct System, which is designed to meet range safety requirements, uses a combination of command and autonomous controls to activate a liquid explosive charge that removes the left elevon causing the vehicle to assume a rolling ballistic flight condition.

(U) The standard 400-MH_z destruct system now installed on both test ranges is effective during ascent, and transponders can be installed on both the launch vehicle and the payload. Primary transmissions are to the launch vehicle prior to separation. Ground facilities downrange of the high plasma region of the flight will have 400-MH_z destruct capability.

(U) During the high plasma portion of the flight, an autonomous system which monitors roll maneuvers will terminate the flight if range safety boundaries are endangered. The autonomous system also functions during periods of tracking gaps. The destruct time constant in the autonomous mode has to be carefully evaluated in order to reduce false alarms.

(U) The above system is supplemented by two additional features: a microwave command link using the telemetry system to augment the autonomous destruct system during the severe plasma period, and a system which monitors inertial guidance information which back-up the autonomous system during tracking gaps.

(U) A liquid explosive is used to sever the left elevon of the vehicle making the vehicle aerodynamically unstable. ASSET components and operational procedures can be used throughout, modified as appropriate to reflect the differences in configuration.

3.6 (U) ENVIRONMENTAL CONTROL SYSTEM

3.6.1 (U) System Description

(U) The Environmental Control System (ECS) of the research vehicle is designed to maintain an average vehicle internal temperature of 70°F during the flight. Thermal protection system weight optimization studies show that approximately 400,000 Btu will penetrate the insulation of a high L/D vehicle using a minimum weight protection system. An additional cooling load of 28,000 Btu is imposed by the vehicle electrical and electronic systems. Since the heat capacity of the airframe is negligible compared to heat loads of this magnitude, an active environmental control system is used to absorb the heat and reject it from the vehicle. The results from related studies reported in Ref. 3 shows that active cooling of the vehicle internal wall with an expendable coolant is a light-weight method for positive vehicle internal temperature control.

(U) Figure 50 shows a schematic of the active cooling system using water and ammonia as expendable coolants. The internal wall and electrical equipment are cooled by means of a water-glycol transport loop. The heat load is transported to the heat exchangers where it is absorbed by the vaporization of water or ammonia and water and exhausted from the vehicle. Temperature control during entry is achieved by controlling the pressure of the vapor space in the water and ammonia evaporators. To maintain an average water-glycol transport fluid temperature of 70°F assuming an allowable 50°F temperature increase in the transport fluid temperature around the loop, the water-glycol temperature as it emerges from the heat exchangers must be about 45°F at the time of maximum heat load. This means that the heat exchanger fluid must be maintained at approximately 40°F. As shown in Figure 51, the vapor pressure of water at 40°F is 0.122 psia which corresponds to a 1962 U.S. Standard Atmosphere altitude of 106,000 ft. To provide a water-glycol temperature of 45°F at altitudes below 106,000 ft, the ammonia boiler is used to augment the cooling capacity of the water boiler. The vapor pressure of ammonia at 40°F, also shown in Figure 51, is 73 psia. Temperature control during groundhold is achieved by the use of an externally supplied coolant which cools the water-glycol transport fluid by means of a wrap-around heat exchanger as shown in the lower right of the schematic. An estimated one-ton cooling system is required for ground checkouts.

(U) A pump is required to maintain coolant circulation in the transport loop. An expansion tank is necessary to accommodate volume changes in the water-glycol loop. The temperature sensors and flow meter are necessary for system control. Water-glycol (60 percent ethylene glycol, 40 percent water) is selected for the transport loop because of its relatively low viscosity over a wide range of temperature and pressure. Water is used as the primary expendable coolant because of its large latent heat of vaporization. Ammonia is used as the auxiliary expendable coolant because of its higher vapor pressure at the temperatures of interest and its latent heat of vaporization is the highest at the required conditions.

3.6.2 (U) Typical System Design Parameters

(U) Figure 52 shows a history of the heat flux into the active coolant at two lower surface locations for a typical entry trajectory. The cooling system design is based on the maximum conditions encountered during entry. For the lower surface and much of the side area of the vehicle, this amounts to 0.255 Btu/sec-ft^2. Since the maximum upper vehicle surface temperatures attained during entry are roughly two-thirds that of the lower surfaces, the corresponding maximum upper surface cooling requirement is approximately two-thirds that for lower surface, or 0.168 Btu/sec-ft^2.

(U) For the reference trajectory, the vehicle is below 106,000 feet altitude for the final 800-1000 seconds. During this period the heat load dissipation function is gradually taken over by the ammonia boiler. The average heat flux from aerodynamic heating and electrical equipment cooling during this period is estimated to be 10 Btu/sec, or a total load of 10,000 Btu, requiring 20 pounds of ammonia coolant.

(U) Table 12 is a tabulation of the data and parameters used in the design of the ECS system.

TABLE 12

(U) ENVIRONMENTAL CONTROL SYSTEM PARAMETERS

Wetted area, A (ft^2) (does not include fins and nose)	551
Transport loop fluid (60% ethylene glycol - 40% H_2O)	
Density (lb/ft^3)	67
Heat Capacity (Btu/lb-°F)	0.74
Expendable Water Coolant	
Density (lb/ft^3)	62.4
Latent heat of vaporization (Btu/lb)	1000
Expendable Ammonia Coolant	
Density (lb/ft^3)	40
Latent Heat of Vaporization (Btu/lb)	500
Aerodynamic Heating	
Operating period, θ (sec)	8000
Peak Heat Flux, Q/A (Btu/sec-ft^2)	0.226
Peak Heat Load, Q (Btu/sec)	125
Average Heat Flux, $\bar{Q}/A$ (Btu/sec-ft^2)	0.09
Average Heat Load, $\bar{Q}$ (Btu/sec)	50
Total Heat Load, $\bar{Q}\,\theta$ (Btu)	400,000
Electrical Energy	
Average Heat Load, $\bar{Q}_e$ (Btu/sec)	3.5
Total Heat Load, $Q_e\,\theta$ (Btu)	28,000

UNCLASSIFIED

(U) For estimating the system power requirements, the following operation conditions were assumed:

- (U) The maximum allowable coolant loop temperature rise from the outlet to the inlet of the water boiler is 50°F. (The wall temperature differential along the tube length will be minimized by multi-pass circuits.)
- (U) The maximum loop pressure drop is 50 psi.
- (U) The maximum wall temperature rise between adjacent cooling tubes 20°F.
- (U) The combined efficiency of the pump and motor is 60 percent.

3.6.3 (U) System Design Requirements

(U) Table 13 presents a tabulation of the estimated weight and power requirement of the ECS subsystem for vehicle internal temperature control during entry. These estimates are based on the parameters and assumptions summarized in the preceding section. Weight estimation factors for actively cooled wall systems reported in Ref. 5 were used in determing the system weights.

3.7 (U) ELECTRICAL POWER SYSTEM

(U) The Electrical Power System provides all the DC and AC energy required by the research vehicle for operation during 8000 seconds, i.e., approximately the time from three minutes before start until landing.

(U) Basically, silver-zinc batteries with a cell capacity of 200 amp-hr and an energy/weight ratio of approximately 50 watt-hours per pound were selected as opposed to rechargeable batteries (from a weight standpoint).

(U) The Electrical Power System consists of the batteries (two strings of 19 cells each, with a combined cell capacity of 400 ampere-hours, and suppling a total energy content of approximately 11,000 watt hours), a DC/AC inverter, and a DC/AC regulator. The selected system supplies 28.4V DC ± 2V at approximately 100 amperes continuous load with 250 amperes peak loading. Regulated 5V DC and 115V AC 400 Hz power is supplied for instrumentation. A more detailed weight breakdown of the on-board electrical power system is given in Table 14.

TABLE 13

(U) COOLING SYSTEM SUMMARY

System Requirements	
Max loop Δ T, °F	50
Max Δ T between tubes, °F	20
Tube length between manifolds, ft	10
Minimum tube spacing, inches	2.5
Tube diameter, inches	0.1
Max loop pressure drop, psi	50
Horsepower (η = 0.6), hp	1.10
Weight Estimates	
H_2O evaporated, lb	418
H_2O residual and carryover, lb (10% of H_2O evap)	42
NH_3 evaporated, lb	20
NH_3 residual and carryover, lb (10% of NH_3 evap)	2
60% Ethylene - 40% Glycol, lb	67
Loop distribution system, lb (0.15 lb/ft^2)	83
Pump and motor, lb (12,100 lb/hr cap.)	10
Battery*, lb (18.0 lb/hr-hr)	45
Tanks, supports, controls, plumbing, lb (12% of lbs evap)	53
Heat exchanger weight, lb (1.15 x 10^{-4} lb/(Btu/hr))	53
TOTAL WEIGHT, lb	793

*This weight is included in the 208 lbs of total battery weight.

UNCLASSIFIED

TABLE 14

(U) ELECTRICAL POWER SYSTEM

	Pounds
Batteries	208
DC/AC Inverter	20
DC/AC Regulator	10
J-Box	12
Umbilical	15
Squib Battery	6
Wiring	25
Miscellaneous	4
TOTAL	300

UNCLASSIFIED

(U) The expected average load distributions are shown below in Table 15. The average power figures were obtained from Volume 3 of Reference 1. The increased vehicle size is reflected in an increase in average power during glide of 250 watts. The average power during Approach and Landing also reflects this increase. A high capacity environmental control system has been added (1.1 horsepower). The total flight duration of 8000 seconds was rounded off to facilitate the calculations of energy and weight estimates. The actual mission time will be something less and dependent on the particular flight profile selected on a given mission.

TABLE 15

(U) ELECTRICAL POWER REQUIREMENTS

Mission Phase	Average Power (KW)	Duration (sec)	Energy (KWh)
Prelaunch	2.9	180	0.145
Boost and Injection	1.5	180	0.075
Glide (Total)	3.0	7160	5.980
Aerodynamic Control* (Flight below 260,000 ft Altitude)	0.825	5600	1.290
Environmental Control System (1.1 horsepower)	0.825	8000	1.840
Approach and Landing	2.65	480	0.353
Total Flight Duration		8000	
Total Electrical Energy Required			9.683
Total Electrical Energy Required (Including 10% Safety)			10.650

*The controls require 1250 watts average and a peak of 5000 watts with a 30% duty cycle. The 1250 watts average has been included in the 3 KW average during the glide phase.

UNCLASSIFIED

SECTION 4

(U) WEIGHTS

(U) The results of the structural, thermal and systems analyses were evaluated to develop a complete weight breakdown for the test vehicle. A summary of the weights is presented in Tables 16 and 17. To allow for unforeseen design changes a 10 percent contingency on the structure and equipment weights has been added as a separate item to derive nominal vehicle weights.

4.1 (U) STRUCTURE

(U) Primary loads are carried by insulated and cooled structural skins, longerons and frames, while the surface heat shield panels carry airloads only. The upper surface heat shield panels are Inconel 625 with an installed unit weight of 2.02 psf. Due to the higher temperatures, lower surface heat shield panels are Cb 752 with an installed unit weight of 2.16 psf. A leading edge heat shield (Ta-10W) is used where the peak temperatures exceed 2500°F. The nose section to Station 60 is also Ta-10W.

(U) The insulation is composed of dynaflex and microquartz. The thickness varies with the body location. Typical unit weights run from 0.50 psf on the upper surface to 1.11 psf on the forward lower surface. The total of thermal protection and internal structure weights average 5.39 psf. For comparison, an F-104G has a fuselage unit weight of 4.28 psf.

(U) The weight of doors, bulkheads, frames and longerons are based on loads and stress analyses. A non-optimum factor of 20 percent is included in all cases except for the nose and main landing gear frames and doors where a 40 percent factor is used. These installed weight allowances are considered to be minimum.

(U) Equipment support structure is estimated at approximately two percent of the total equipment weight. Separation fittings and reinforcement are equal to one percent of the vehicle landing weight.

(U) Fins, rudders and elevons consist of hot load-carrying structure with tantalum alloy leading edges. Surface panels and substructure of the centerline mounted fin are constructed of Haynes 25. The corrugation stiffened surface panels have a unit weight of 1.94 psf, including a 20 percent non-optimum factor. This non-optimum factor accounts for gage tolerance, edge members and attachments. The total fin unit weight is 8 psf.

(U) Body flaps on the upper part of the aftbody consist of Haynes 25, while the elevons on the lower aftbody are Cb 752. The unit weights of the upper and lower surfaces are 8 psf and 12 psf, respectively. For comparison, the F-4C has a horizontal and vertical tail unit weight of 6.70 psf and 3.84 psf, respectively.

(U) The landing gear weight, including support fittings in the body, has been estimated at 4 percent of the total vehicle landing weight. This allowance is adequate for a skid-type main gear and a wheel-type nose gear. The X-15 and F-4C landing gear weights, by comparison, are approximately equal to 3.0 percent and 6.0 percent, respectively, of the total landing weight.

4.2 (U) EQUIPMENT

(U) The Environmental Control System provides sufficient cooling to keep the aluminum inner structure temperature at 70°F. Water-glycol is pumped through tubes on the single corrugated skin, the fluid is circulated through a water/ammonia boiler heat exchanger.

(U) The Electric Power System uses silver-zinc batteries sized for 200 ampere-hours. The Data Management, Guidance and Navigation, Tracking, and Command and Autopilot System weights are based on the analyses in the systems section.

(U) The destruct system ignites the pyrotechnic device severing one elevon. This causes the vehicle to roll, following a ballistic trajectory.

(U) The monopropellant Reaction Control System provides the pitch, roll and yaw control of the vehicle when out of the atmosphere. The hydrogen peroxide is pressure fed by a 3,000 psi nitrogen tank. Included in the propellant weight is approximately 250 lbs for the four 500 lbs thrust motors of the Landing Propulsion System.

(U) The Aerodynamic Control System provides the hydraulic power to actuate all the aerodynamic control surfaces.

4.3 (U) BALLAST

(U) The vehicle has sufficient ballast to move the landing center of gravity forward to 62 percent of the body reference length. The effect of changing the ballast weight is plotted in Figure 53.

TABLE 16

(U) WEIGHT STATEMENT

	Weight (Lb)		C.G. (Sta)*	
STRUCTURE:				
Thermal Protection		1973		258.05
Heat Shield (212 sq ft, Cb 752)	458		281	
Heat Shield (343 sq ft, Inco 625)	693		287	
Leading Edge (64 ft, Ta)	256		240	
Nose Section (Sta 60, Ta)	160		36	
Lwr Surf. Insulation (212 sq ft)	217		277	
Upr Surf. Insulation (343 sq ft)	189		287	
Internal Structure (2219-T81)		1018		284.90
Skins, Single Corrugation	204		285	
Longerons	239		285	
Frames	203		285	
Nose Gear Bay and Doors	48		122	
Main Gear Bay and Doors	109		316	
Penalties for Access Doors (7)	40		281	
Fittings - Elevons	37		419	
Fittings - Body Flaps	11		394	
Fittings - Vertical Fins	11		389	
Gussets and Attachments	60		285	
Equipment Supports	56		235	
Elevons (17.2 sq ft)		241		441.00
Elevon Seals		25		419.00
Body Flaps (11.6 sq ft)		93		412.00
Vertical Fin (34.3 sq ft)		274		397.00
Separation Ring (Incl.Hard Points)		65		430.00
Nose and Main Landing Gears		270		275.00
Contingency		395		294.00
EQUIPMENT: (See Table 17)		2787		247.14
BALLAST: (To 62 percent)		525		50.00
Gross Weight		7666		260.42
Less:				
Water		-418		249.00
Ammonia		- 20		226.00
Hydrogen Peroxide		-438		276.00
Minimum Landing Weight		6790		260.22

*Nose of vehicle = 0
Body reference length = 420 in.

UNCLASSIFIED

TABLE 17

EQUIPMENT BREAKDOWN

	Weight (Lb)		C.G. (Sta)	
GUIDANCE AND NAVIGATION SYSTEM:		69		184.00
IMU and Electronics	40		184	
Computer and Rate Gyros (3)	19		184	
Cabling and Connectors	10		184	
FLIGHT CONTROL SYSTEM				
AERODYNAMIC CONTROL SYSTEM:		270		306.07
Hydraulic Pump, Accumulator, Reservoir	140		250	
Hydraulic Actuators	70		402	
Plumbing and Valves	60		325	
LANDING PROPULSION SYSTEM:		50		411.80
Thrust Motors (4)	40		427	
Plumbing and Valves	10		351	
Propellant included in RCS	--			
AUTOPILOT SYSTEM:		50		198.00
REACTION CONTROL SYSTEM:		686		287.42
Thrust Motors	54		425	
Tanks (H_2O_2)	113		276	
Tank (N_2)	28		246	
Hydrogen Peroxide	460		276	
Nitrogen	16		246	
Plumbing and Valves	15		350	
DATA MANAGEMENT SYSTEM:		190		181.13
Real-Time Microwave Transmitter	14		184	
Microwave Power Divider	3		184	
Real Time VHF Transmitter	3		184	
Delayed Time VHF Transmitter	4		184	
VHF Multiplexer	2		184	
PCM Deck	20		172	
Subcarrier Oscillator, Sampler, and Translator	2		184	
Telemetry Tape Recorder	13		172	
Microwave Antenna	1		184	

UNCLASSIFIED

(Continued)

TABLE 17

EQUIPMENT BREAKDOWN (Continued)

	Weight (Lb)		C.G. (Sta)	
VHF Antenna	3		184	
Instrumentation	100		184	
Cabling and Connectors	25		178	
TRACKING AND COMMAND SYSTEM:		86		164.90
C-Band Transponder	7		396	
C-Band Diplexer	4		396	
C-Band Power Divider	3		396	
C-Band Antenna	1		[illegible]	
Transponder - Landing	18		78	
Diplexer - Landing	4		78	
Power Divider - Landing	3		78	
Antenna and Corner	8		78	
Receiver Decoder	20		78	
Antenna	1		78	
Cabling, Connector, etc	17		237	
ENVIRONMENTAL CONTROL SYSTEM:		748		253.86
Water	460		249	
Ammonia	22		226	
Ethylene - Glycol	67		285	
Loop Distribution	83		285	
Pump and Motor	10		226	
Tanks and Supports	53		252	
Heat Exchanger	53		226	
ELECTRIC POWER SYSTEM:		300		146.44
Batteries	208		138	
Inverter and Regulator	30		172	
Umbilical and J Box	27		172	
Wiring	35		155	
DESTRUCT SYSTEM:		24		172.00
LANDING GEAR SYSTEM:		60		275.00
Landing Gear (included in structure)				
Controls	60			
CONTINGENCY		254		247.00
TOTAL		2787		247.14

UNCLASSIFIED

REFERENCES

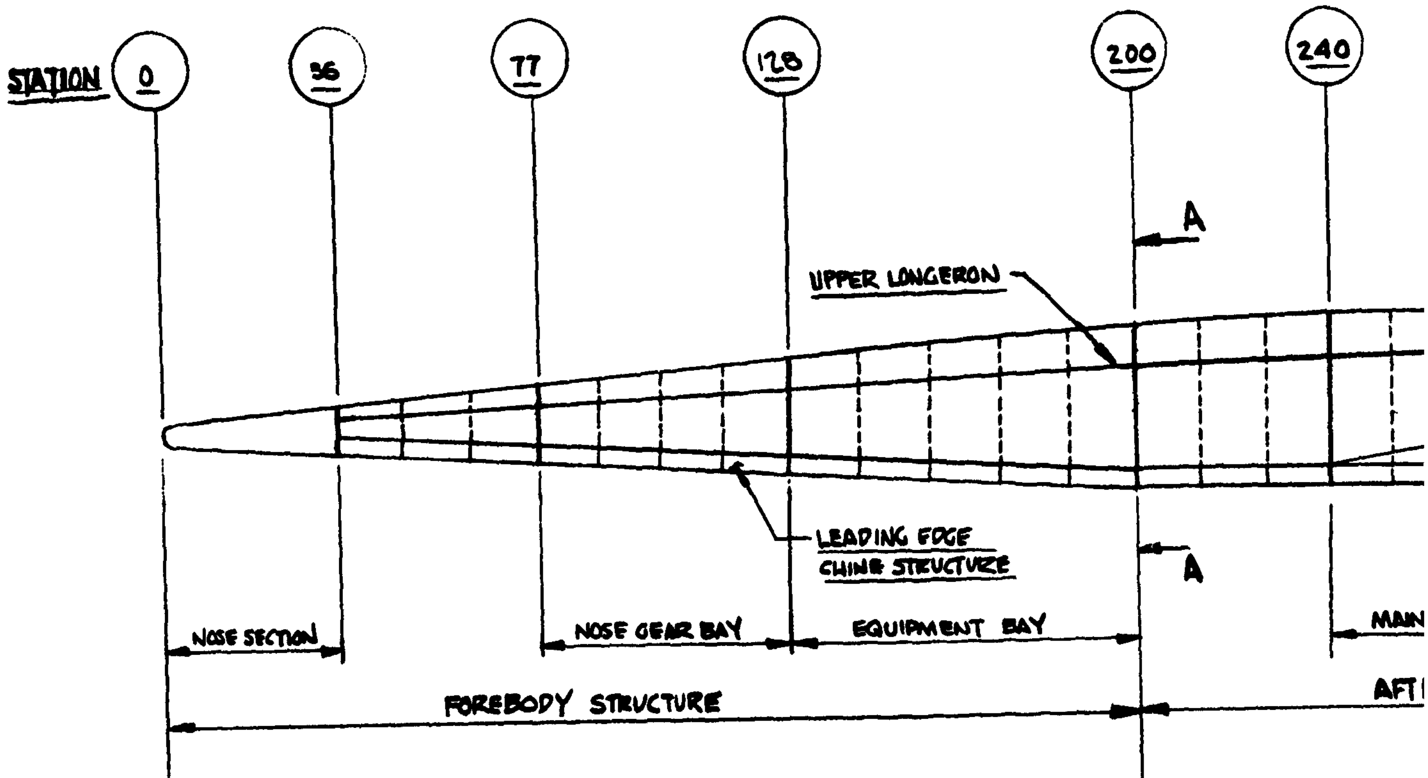
1. Lockheed-California Company, Preliminary Design of Hypersonic High L/D Test Vehicles (U), J. T. Lloyd, et al, AFFDL TR-66-12, May 1966 (C)

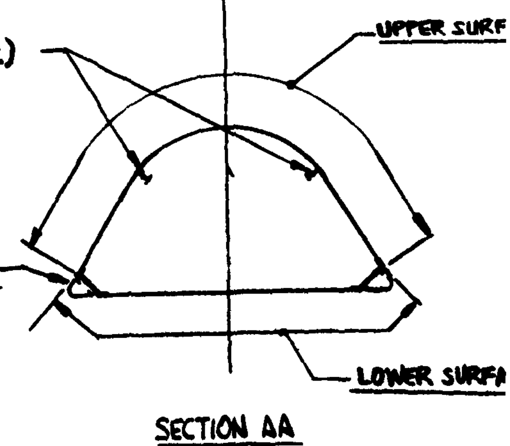
2. W. H. Goesch, "Critical Evaluation of Thermal Protection Systems for Entry Vehicles," presented at the AIAA Annual Meeting, October 23-27, 1967

3. Lockheed-California Company, Preliminary Advanced Structural Evaluation of Six Reusable Spacecraft (U), I. F. Sakata and P. P. Plank, July 10, 1967 (C)

4. Cornell Aeronautical Laboratory, Inc., Design Handbook for High Temperature Aerospace Antennas, Godfrey F. Buranich, AFAL TR-65-267, October 1965

5. Bell Aerosystems, Analytical Evaluation of Actively Cooled Modified Monocoque Structural Sandwich Concepts, F. M. Anthony and R. D. Huff.

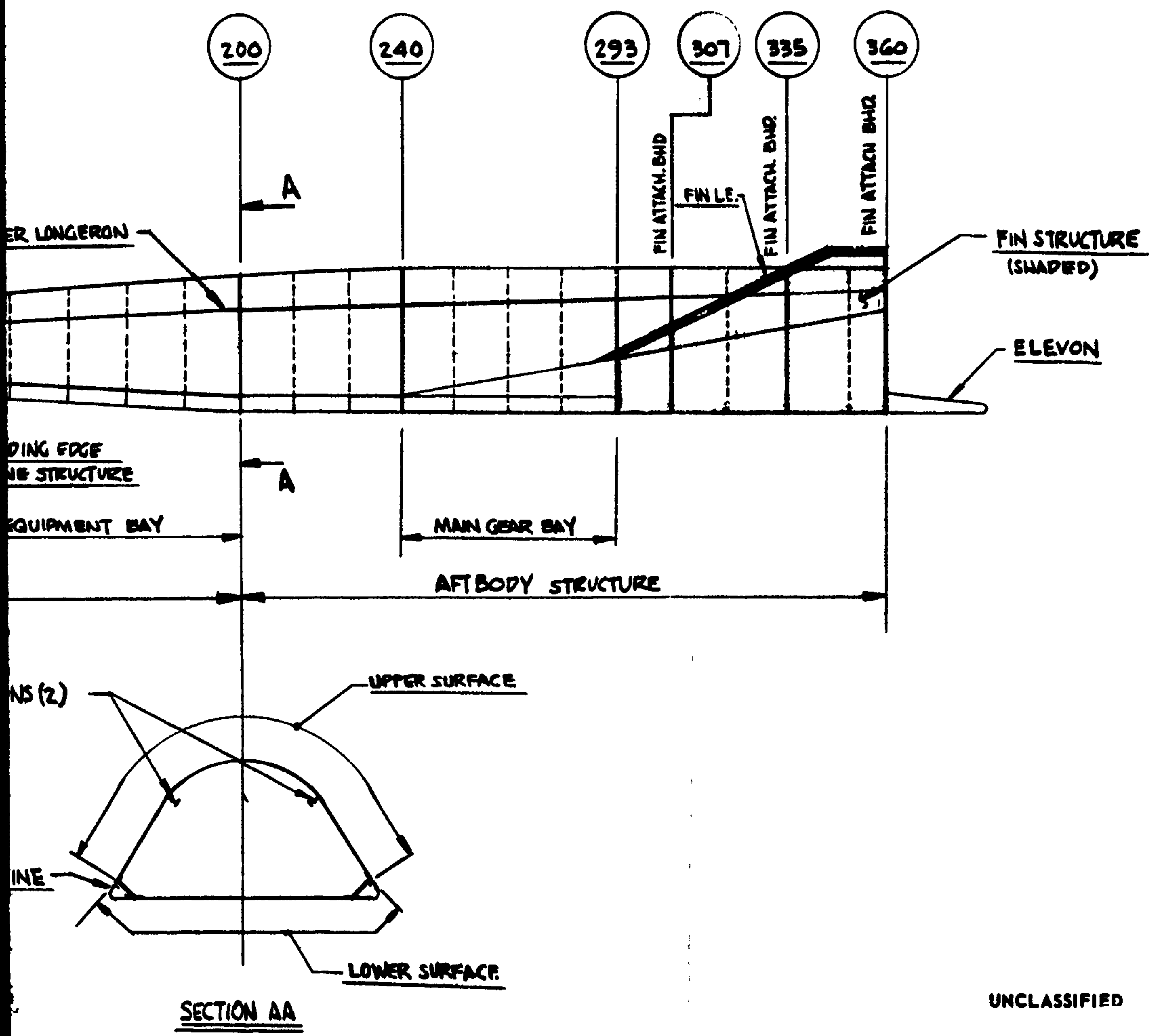




SECTION AA

NOTES: 1. SOLID VERTICAL LINES DENOTE MAIN FRAMES.
2. DASHED VERTICAL LINES DENOTE INTERMEDIATE FRAME.
3. HOT LOAD CARRYING CONCEPT SHOWN AS REFERENCE.

UNCLASSIFIED

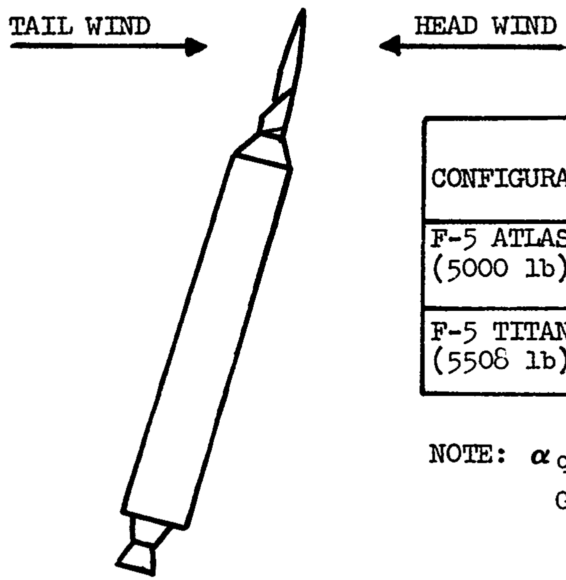


UNCLASSIFIED

FIGURE 1 (U) STUDY VEHICLE (F-5) STRUCTURAL ARRANGEMENT

(REVERSE SIDE IS BLANK)

UNCLASSIFIED



CONFIGURATION	ALTITUDE (FT)	MACH NO.	HEAD WIND			TAIL WIND		
			αq	N_z	GIMBAL ANGLE	α_q	N_z	GIMBAL ANGLE
F-5 ATLAS (5000 lb)	41,000	2.2	3,880	0.3	2.23	3260	0.252	1.87
F-5 TITAN IIIB (5508 lb)	37,000	1.526	4,500	0.3	4.60	3780	0.252	3.86

NOTE: α_q (deg-PSF)
GIMBAL ANGLE (deg)

UNCLASSIFIED

FIGURE 2 (U) COMPARISON OF DESIGN PARAMETERS - EXIT

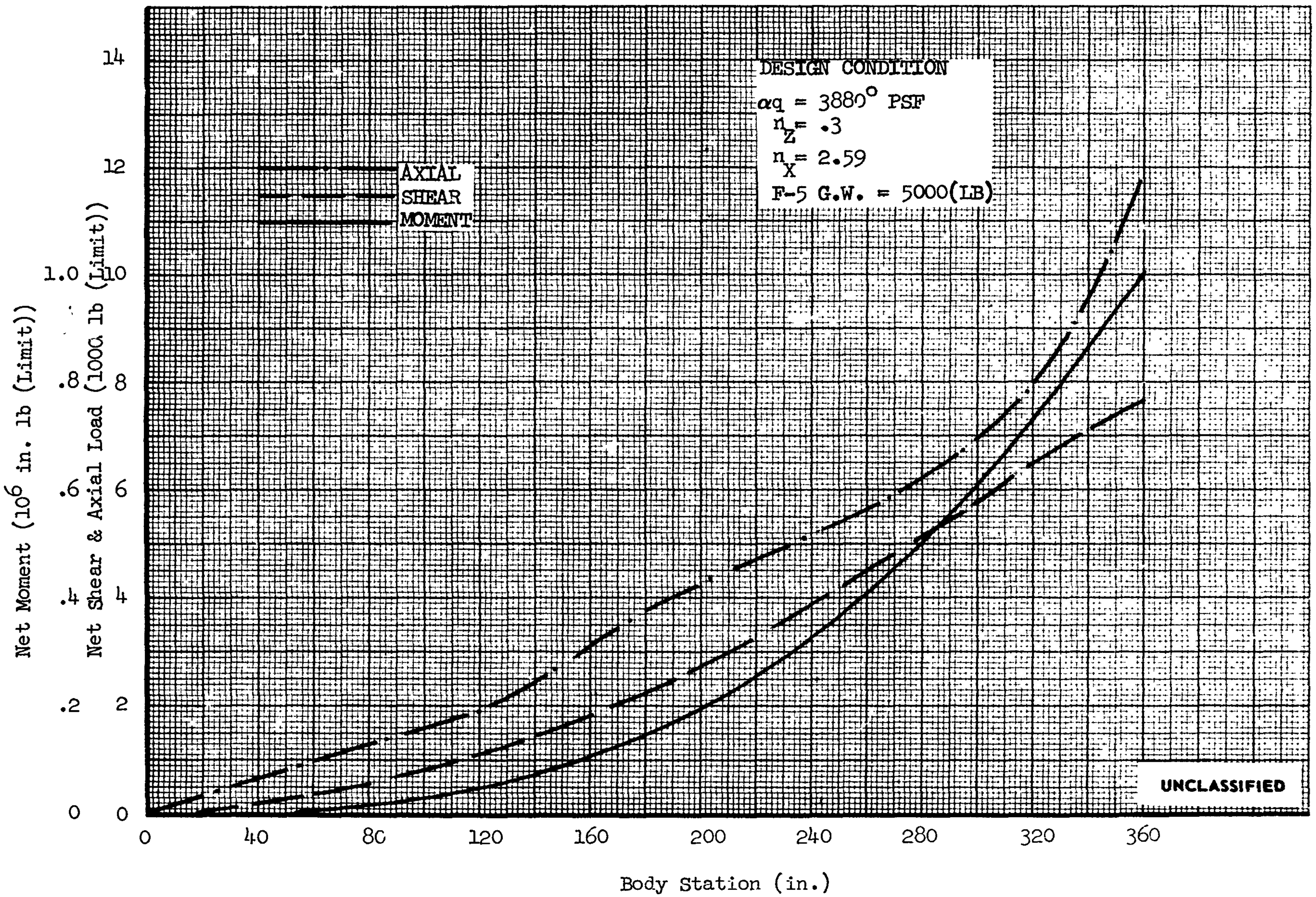


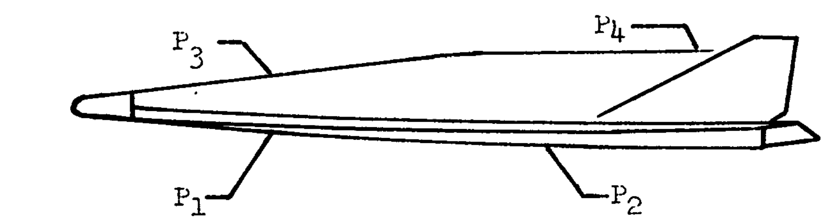
FIGURE 3 (U) LIMIT LOADS FOR F-5 CONFIGURATION ON ATLAS BOOSTER

F-5/ATLAS DESIGN PRESSURES - EXIT PHASE

LOCATION	LIMIT PRESSURE (PSI)	
CASE	HEADWIND	TAIL WIND
αq	3880° PSF	-3260° PSF
P_1	1.20	0.06
P_2	0.50	-0.40
P_3	0.68	1.16
P_4	-0.50	0.40

NOTE: + COMPRESSION;- SUCTION

F-5/TITAN IIIB DESIGN PRESSURES - EXIT PHASE



LOCATION	LIMIT PRESSURE (PSI)
NOSE	12.7
LEADING EDGE	2.6
P_1	1.73;- 0.31
P_2	1.21;- 0.81
P_3	1.64
P_4	0.81; - 1.21

NOTE: + COMPRESSION;- SUCTION

UNCLASSIFIED

FIGURE 4 (U) DESIGN PRESSURES - FLIGHT TEST VEHICLE

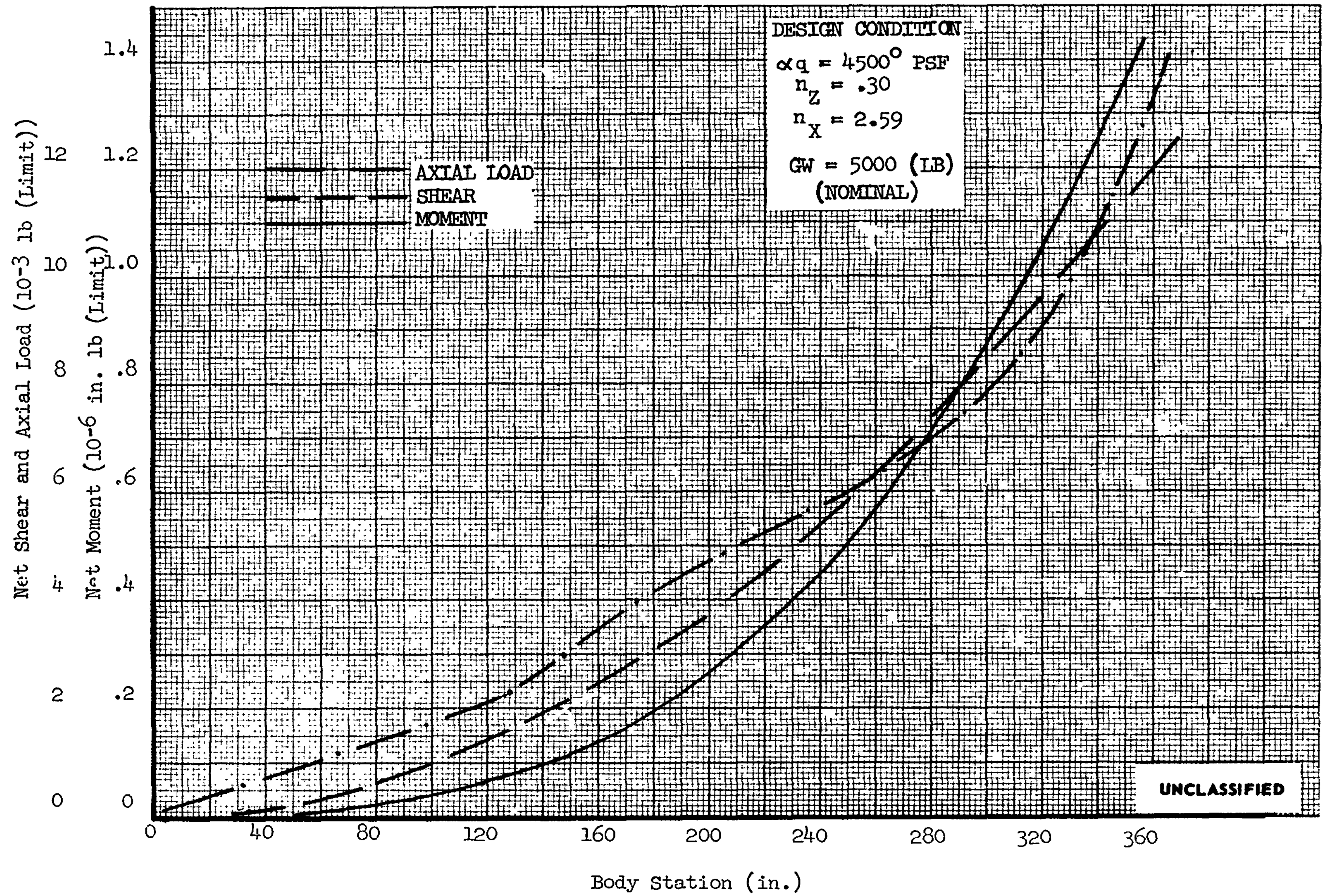


FIGURE 5 (U) LIMIT LOADS FOR F-5 CONFIGURATION ON TITAN III BOOSTER

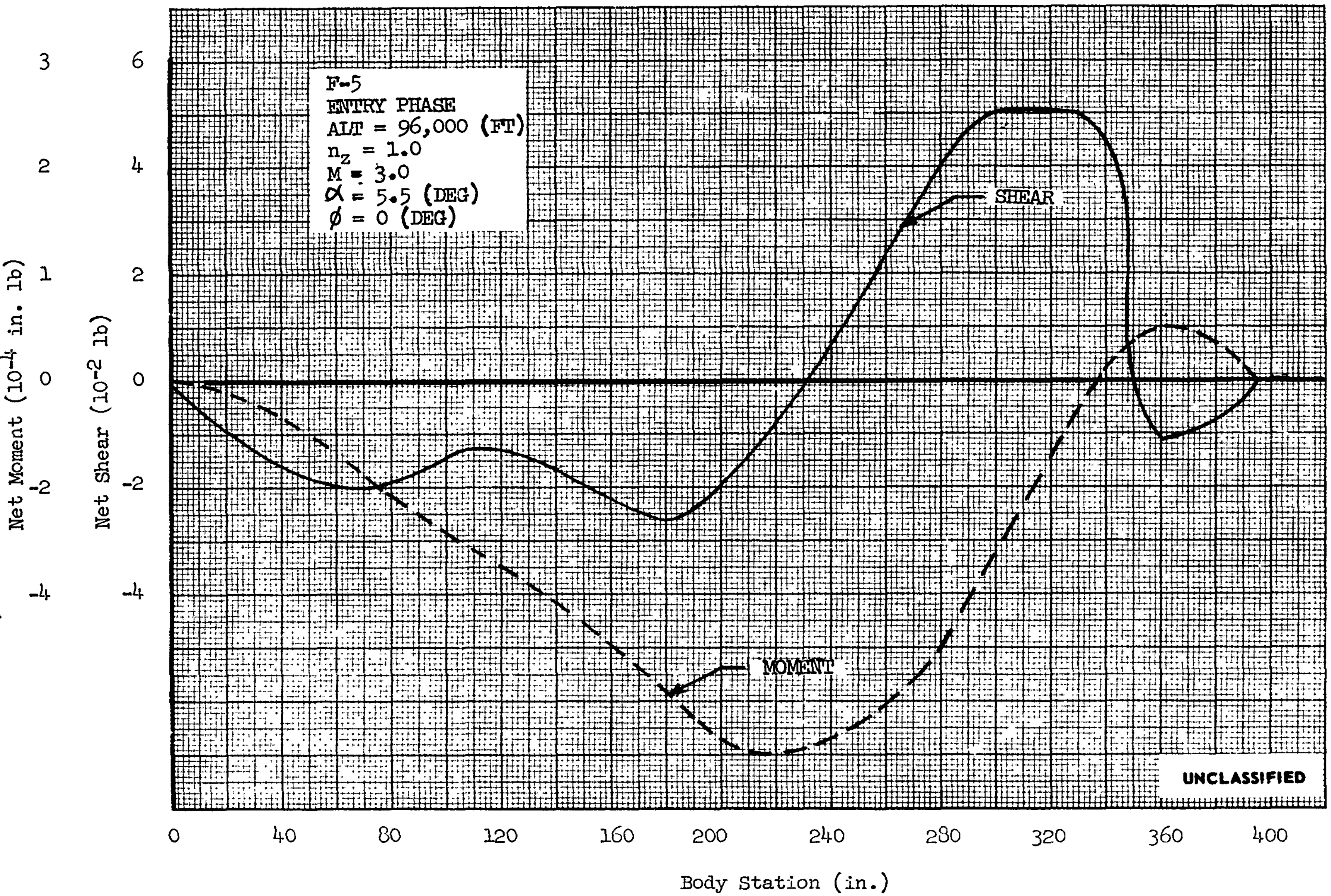


FIGURE 6 (U) F-5 ENTRY LOADS AT END OF GLIDE PHASE

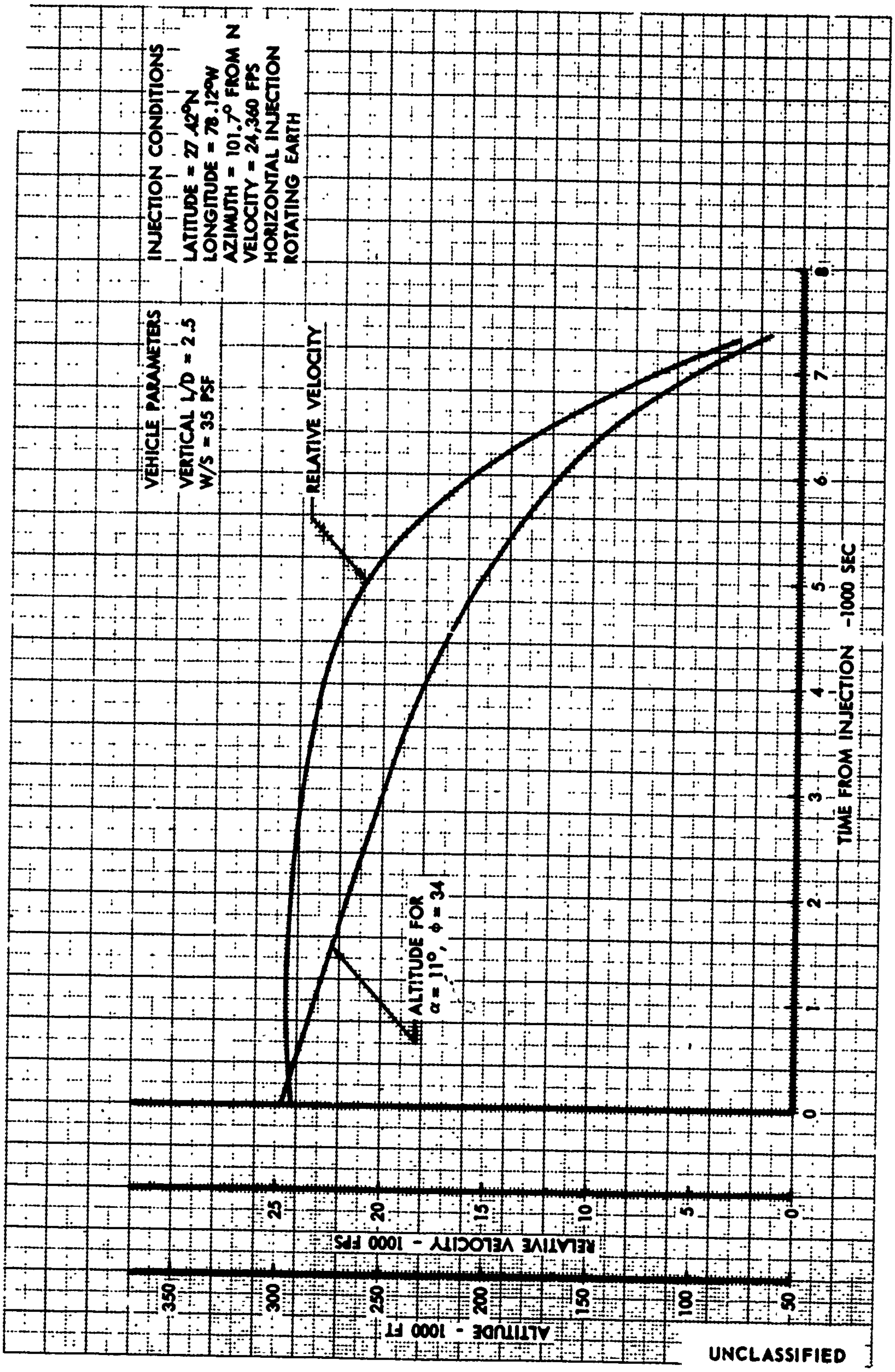


FIGURE 7 (U) REFERENCE TRAJECTORY

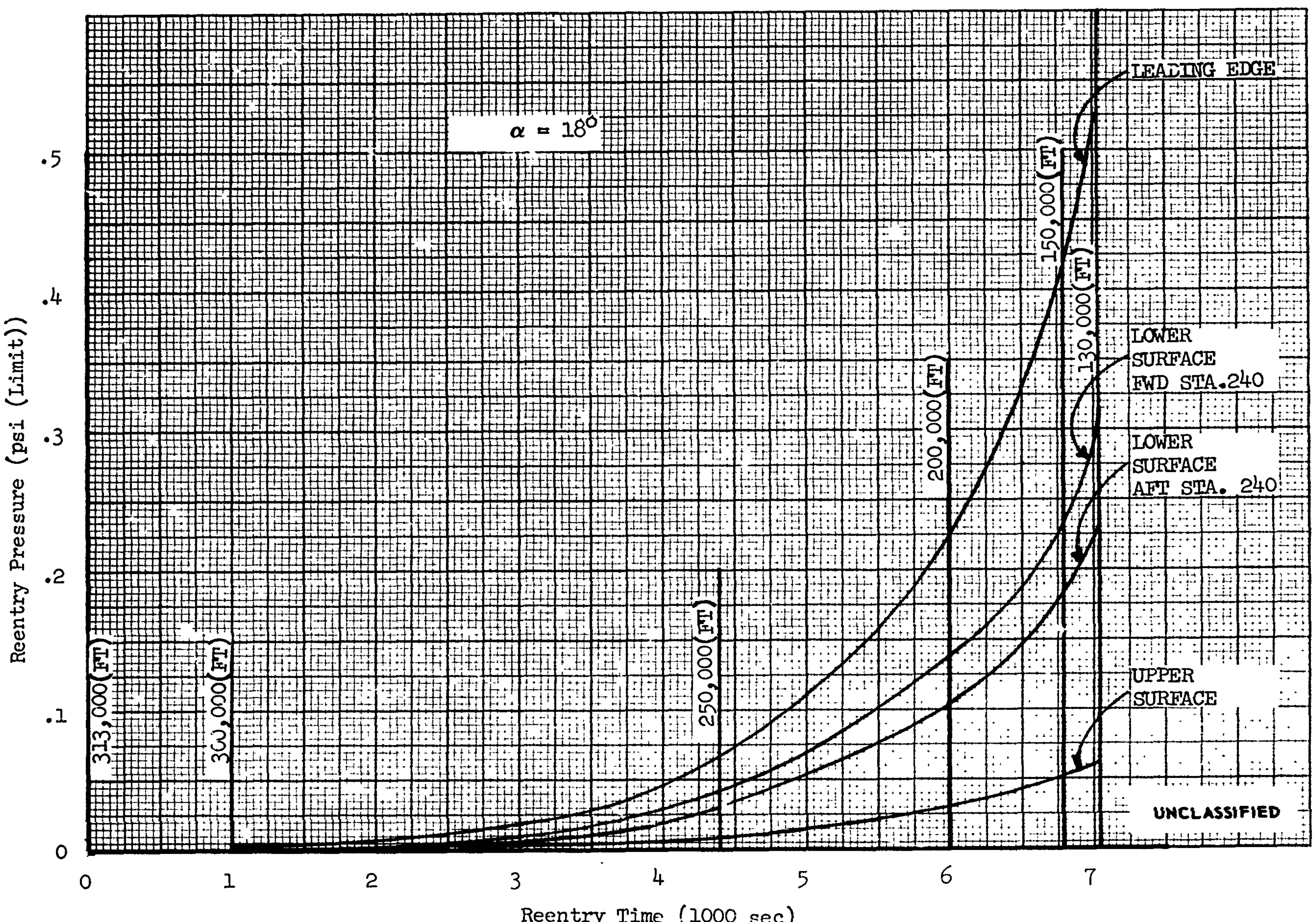


FIGURE 8 (U) F-5 LIMIT PRESSURE ENVELOPES FOR ENTRY TRAJECTORIES

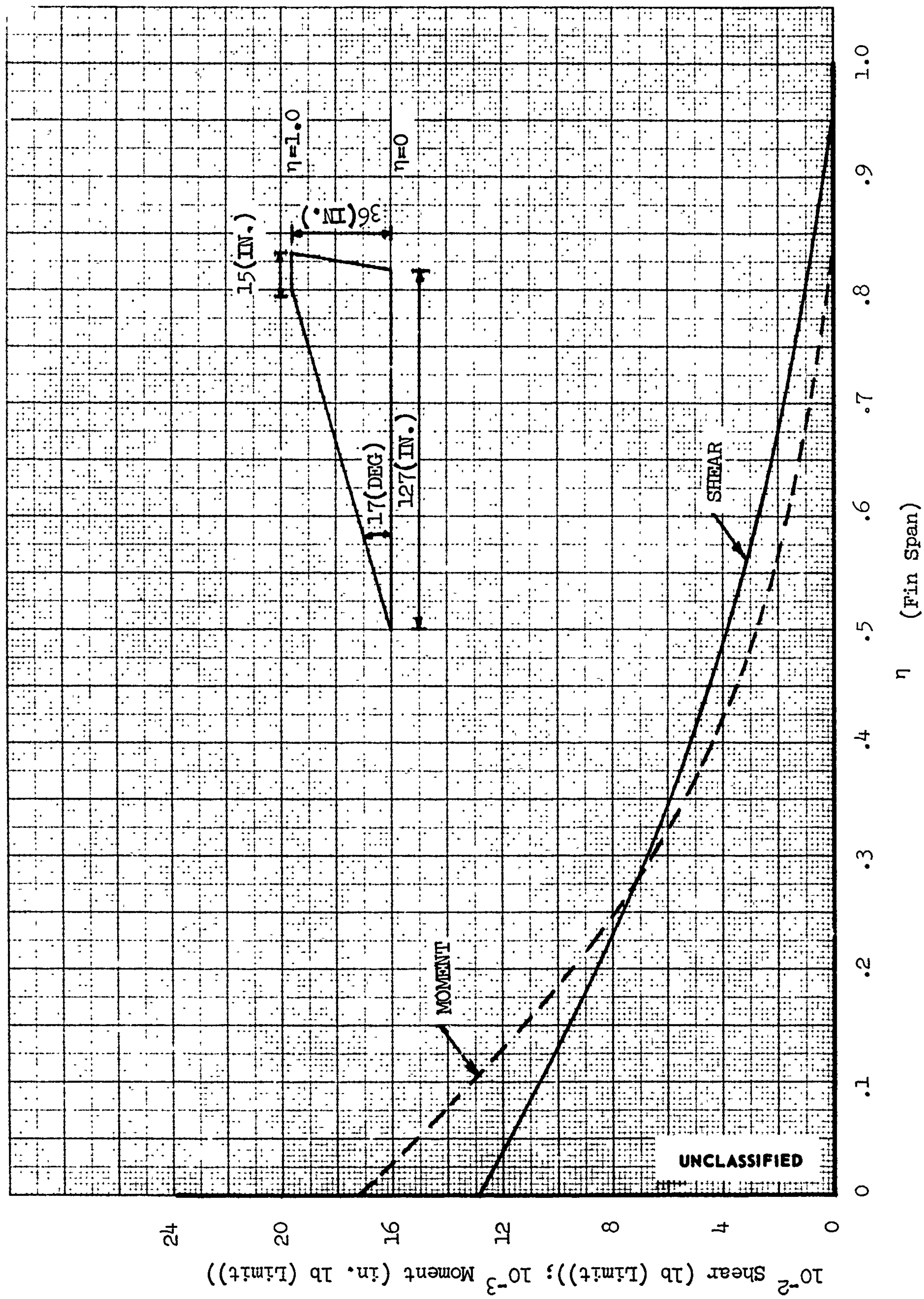


FIGURE 9 (U) LIMIT LOADS FOR F-5 FIN - DURING ENTRY

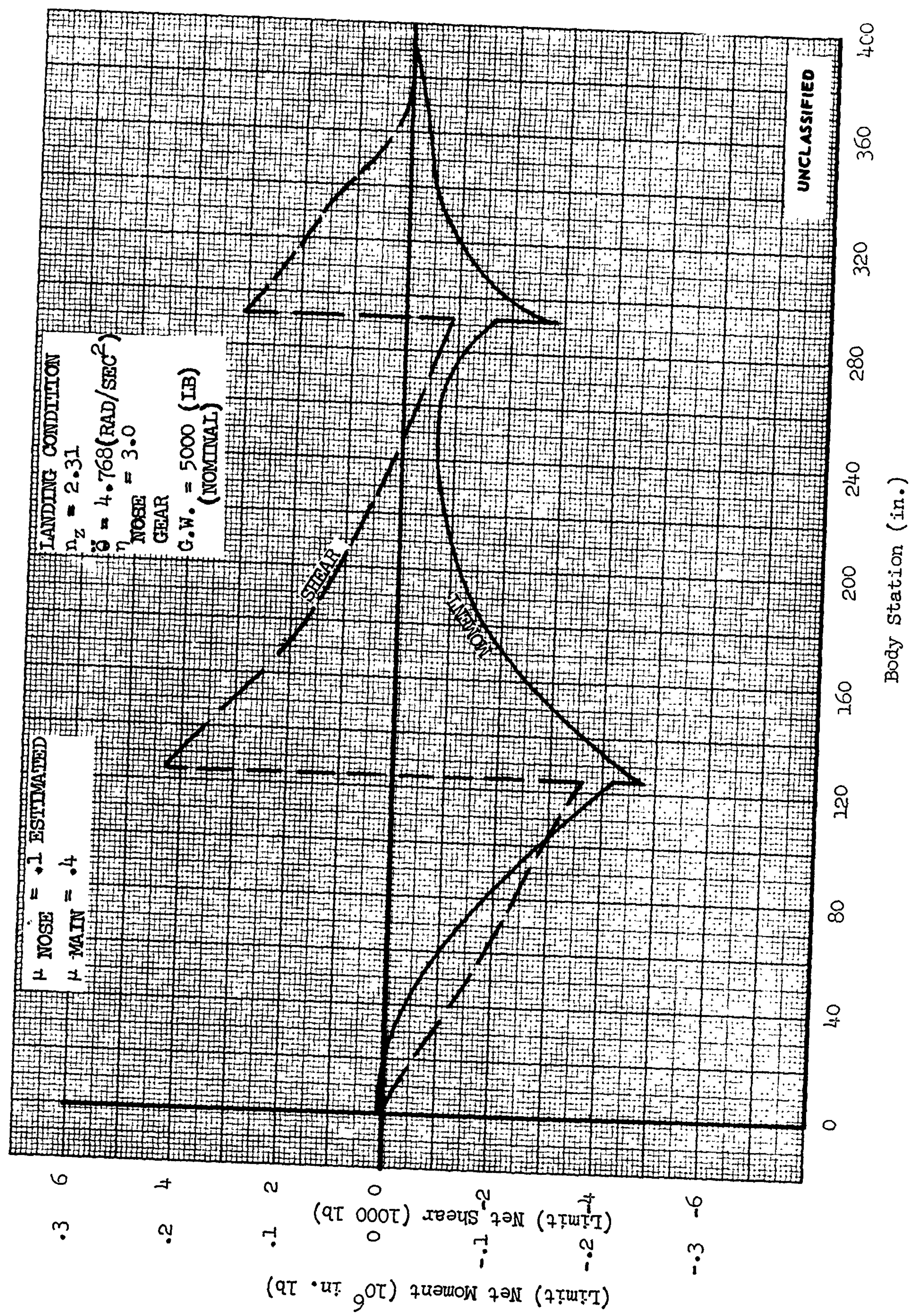


FIGURE 10 (U) F-5 LIMIT LOADS AT LANDING CONDITION

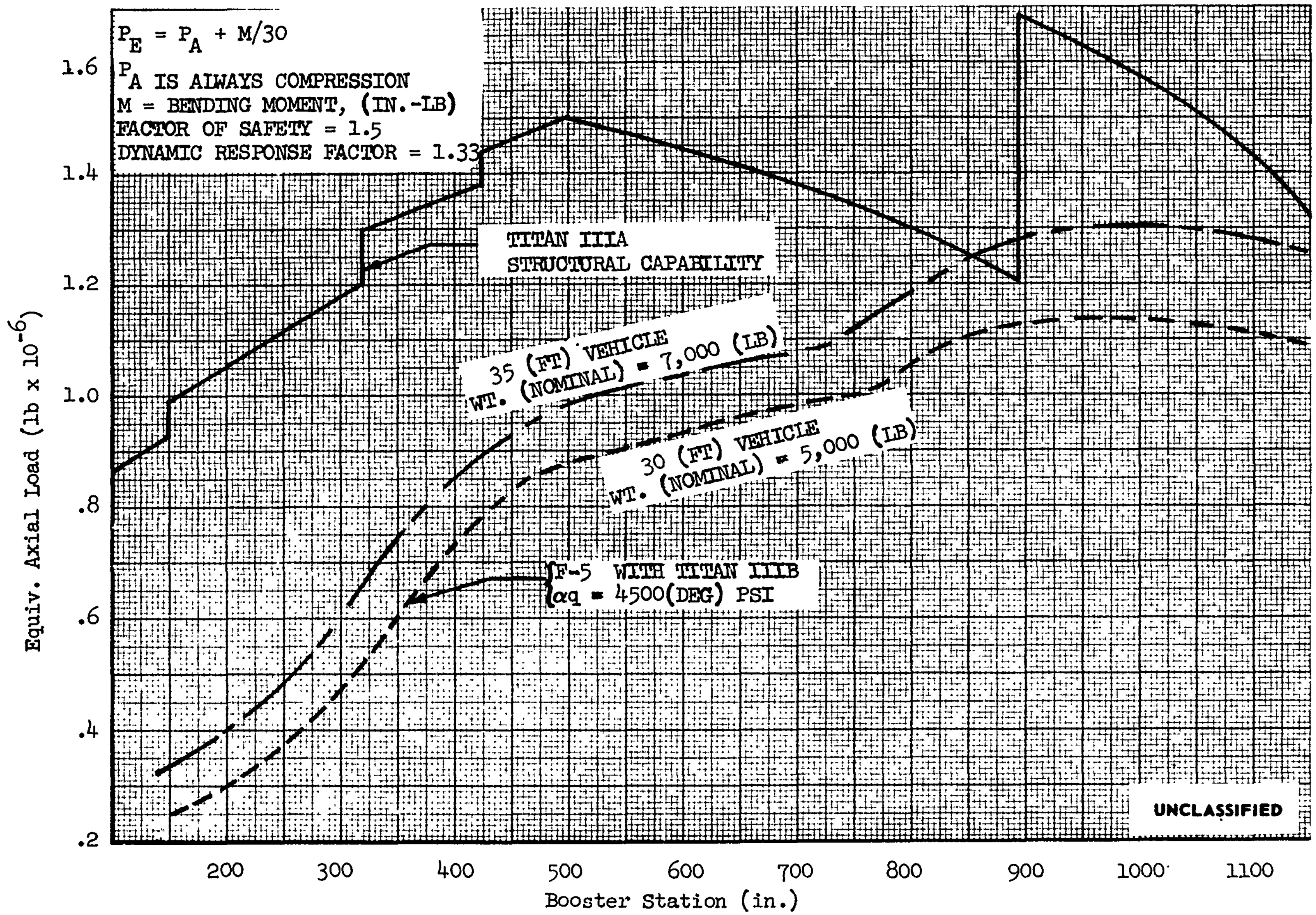
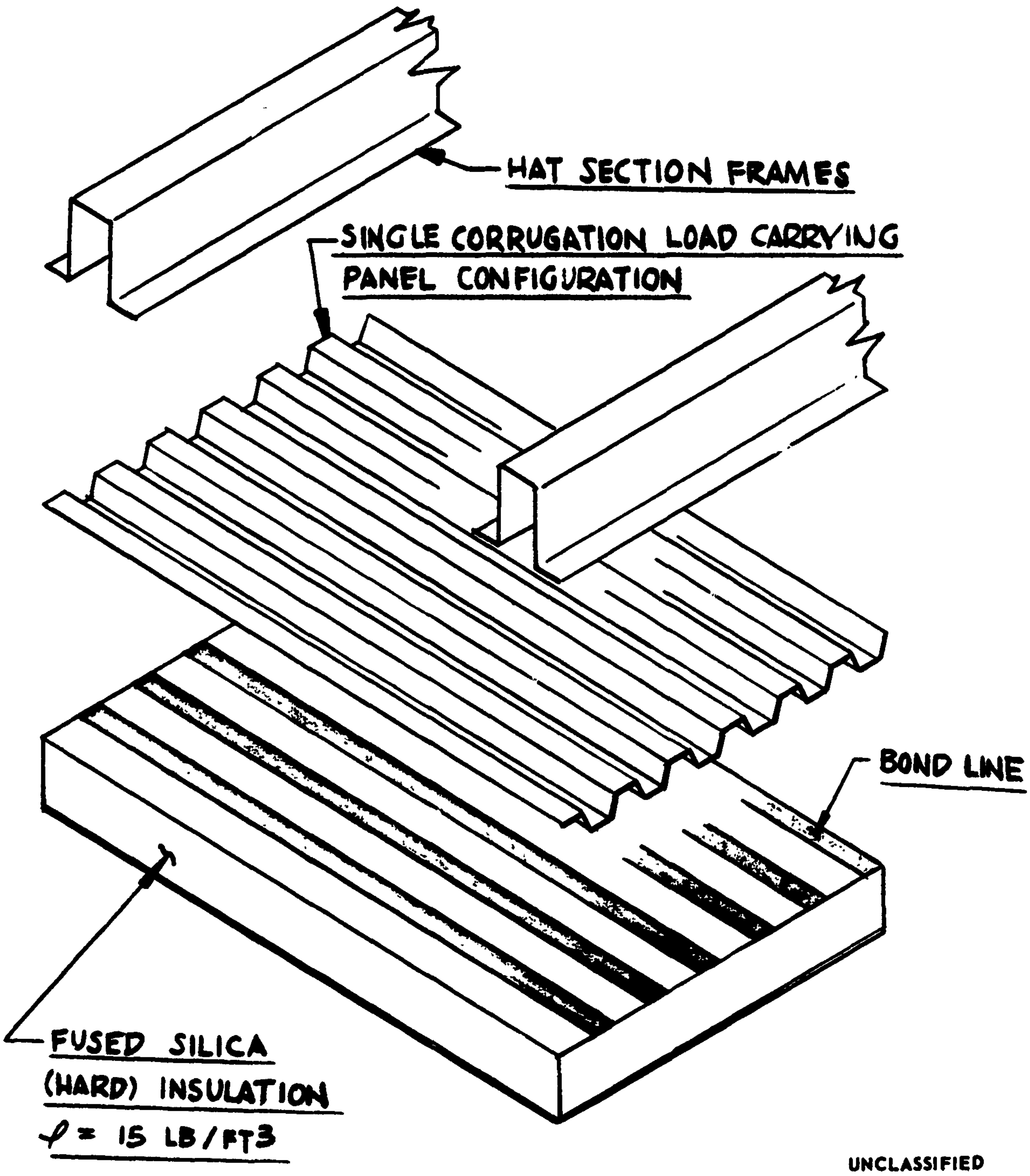


FIGURE 11 (U) COMPARISON OF MAXIMUM FLIGHT STRUCTURAL LOAD CAPABILITIES



UNCLASSIFIED

FIGURE 12 (U) INSULATED STRUCTURAL CONCEPT - MONOLITHIC HEAT SHIELD

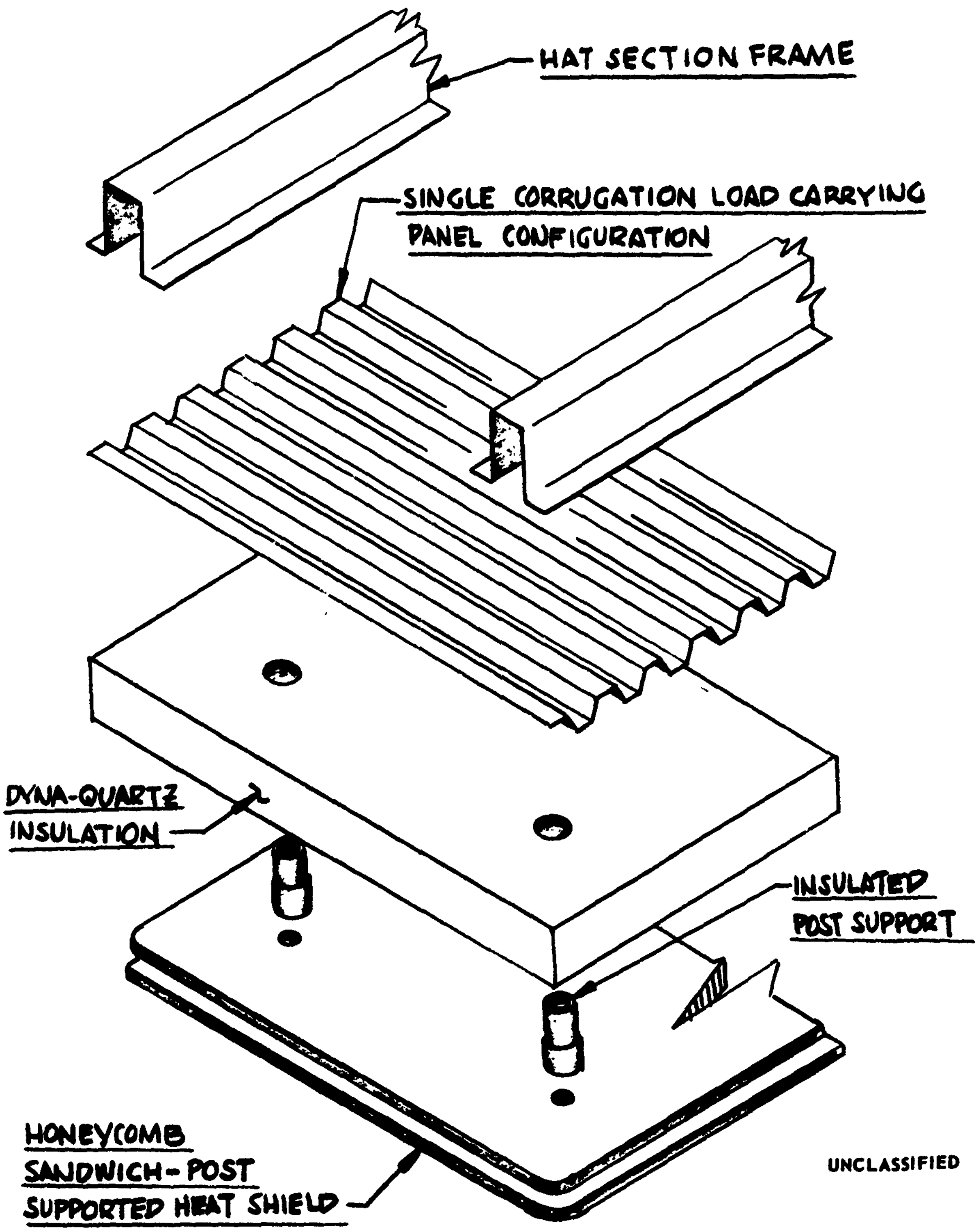


FIGURE 13 (U) INSULATED STRUCTURAL CONCEPT - METALLIC HEAT SHIELD - INSULATION

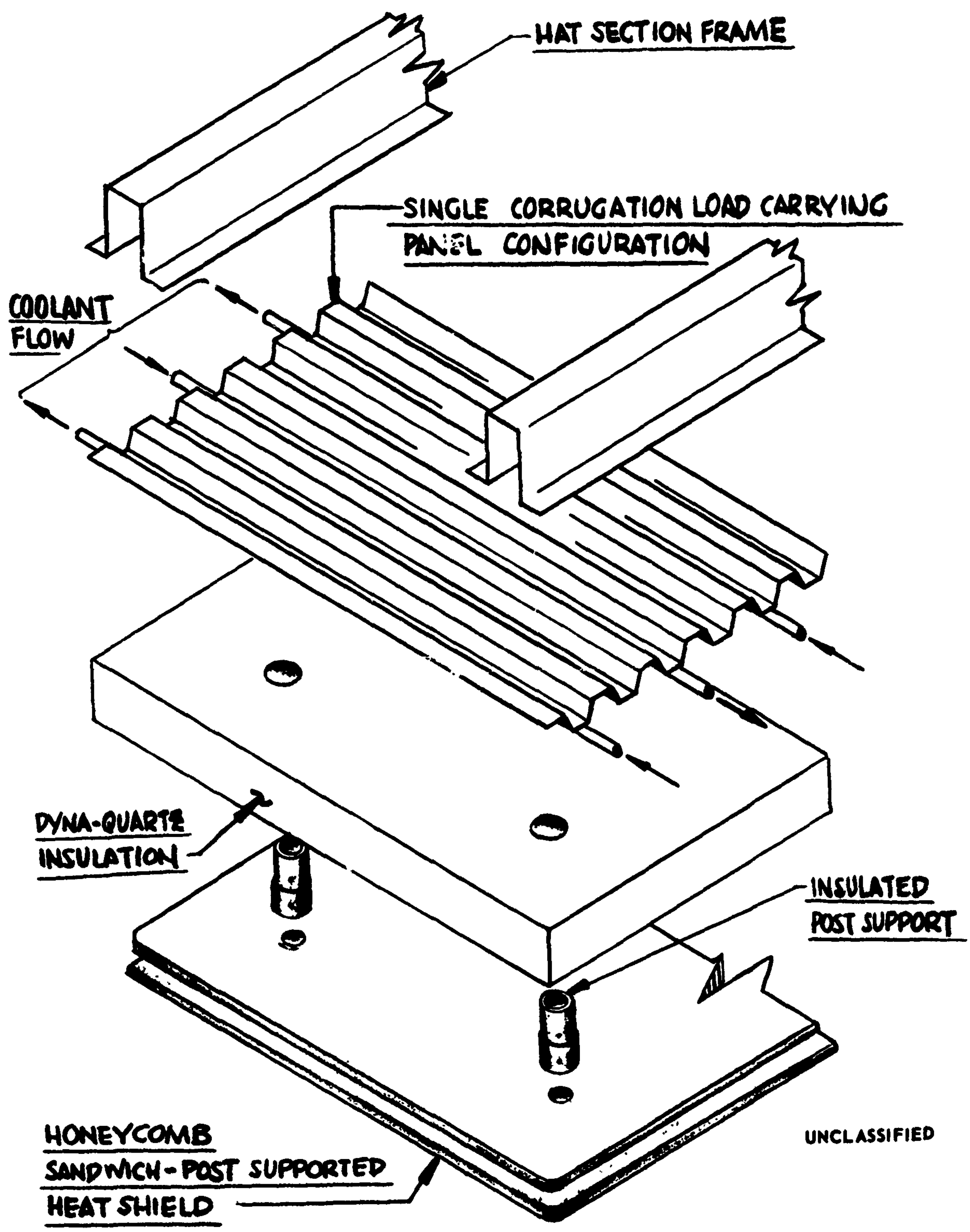


FIGURE 14 (U) INSULATED AND COOLED STRUCTURAL CONCEPT - EXTERNAL OR EXTERNAL-INTERNAL INSULATION

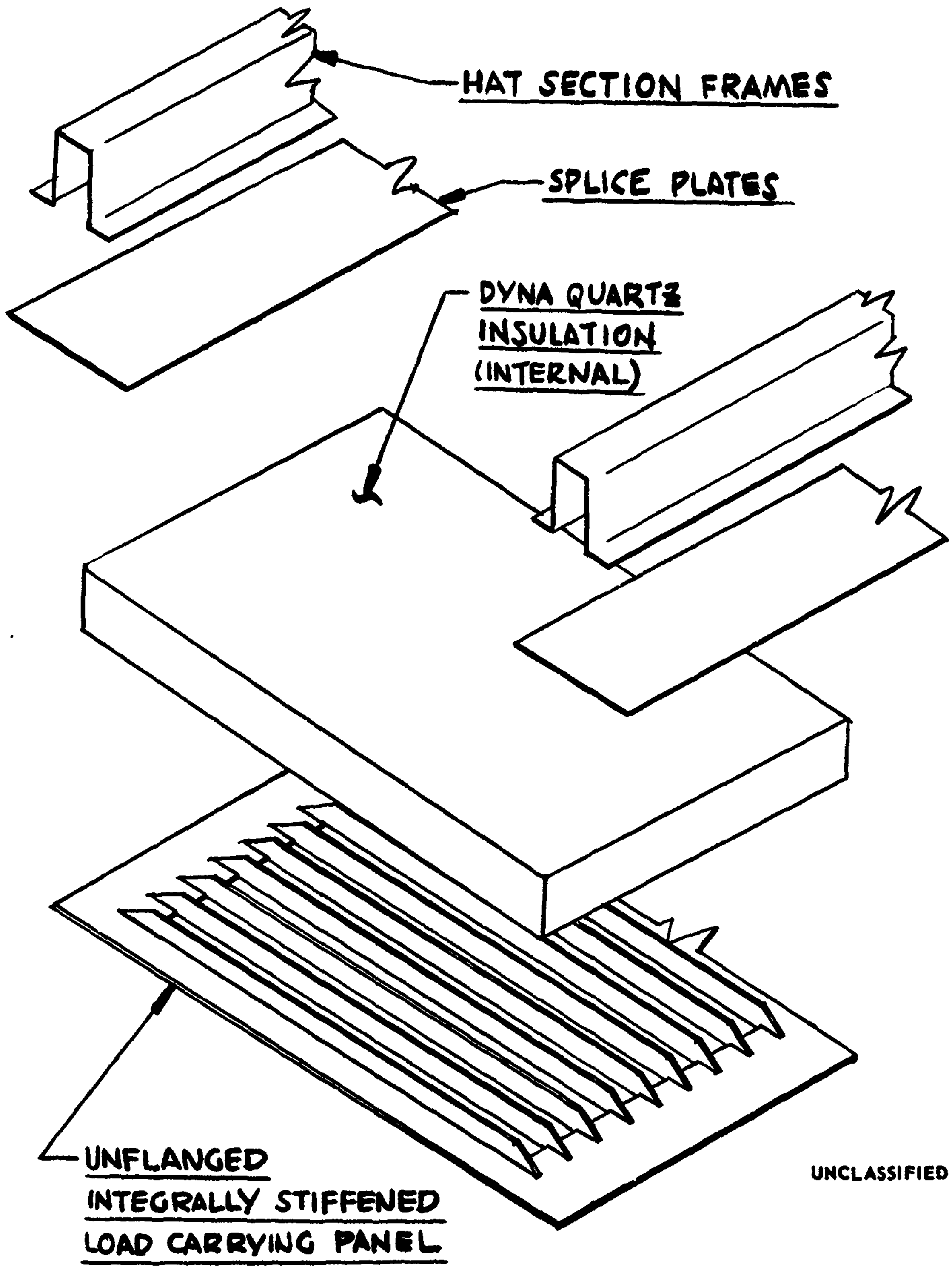
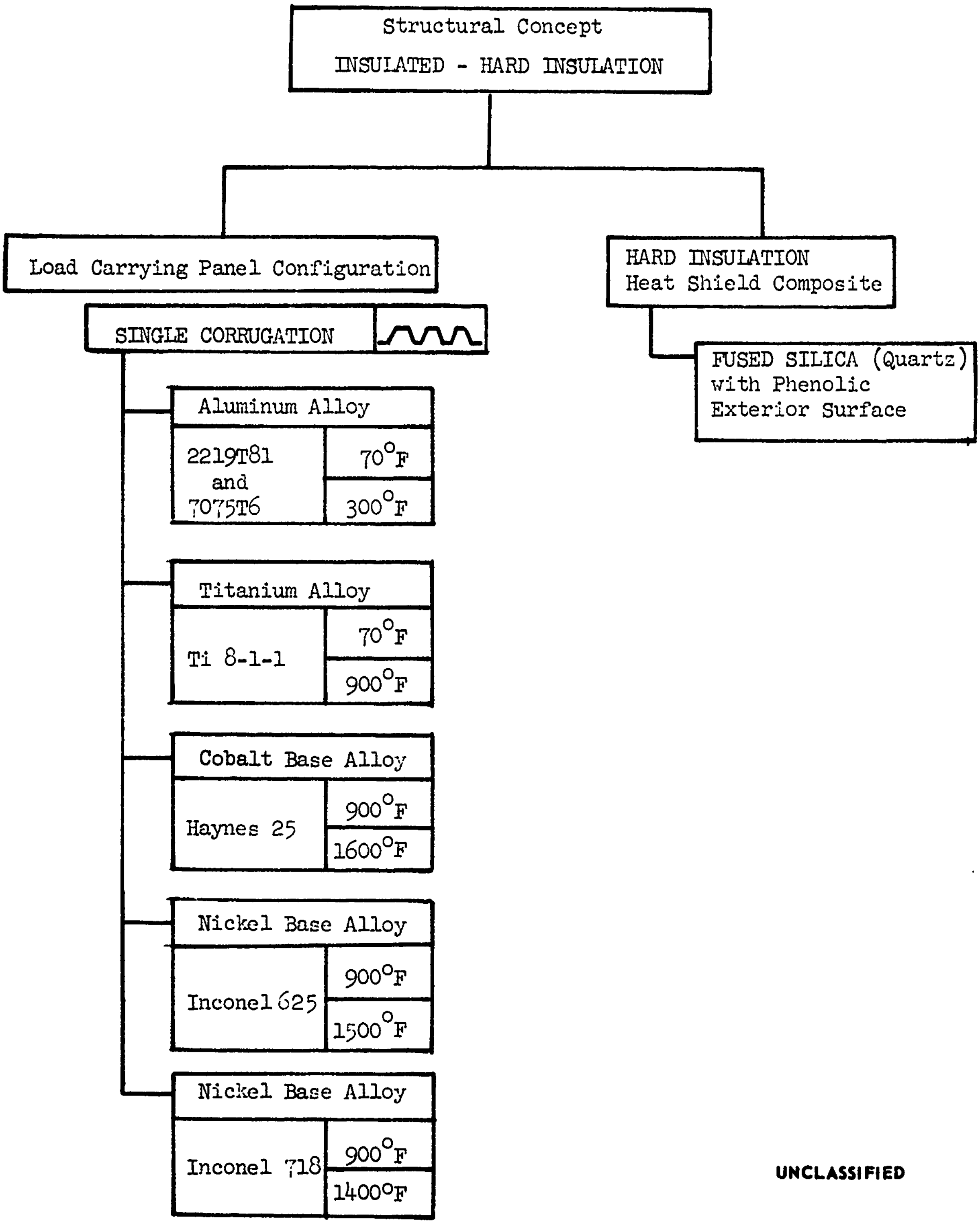
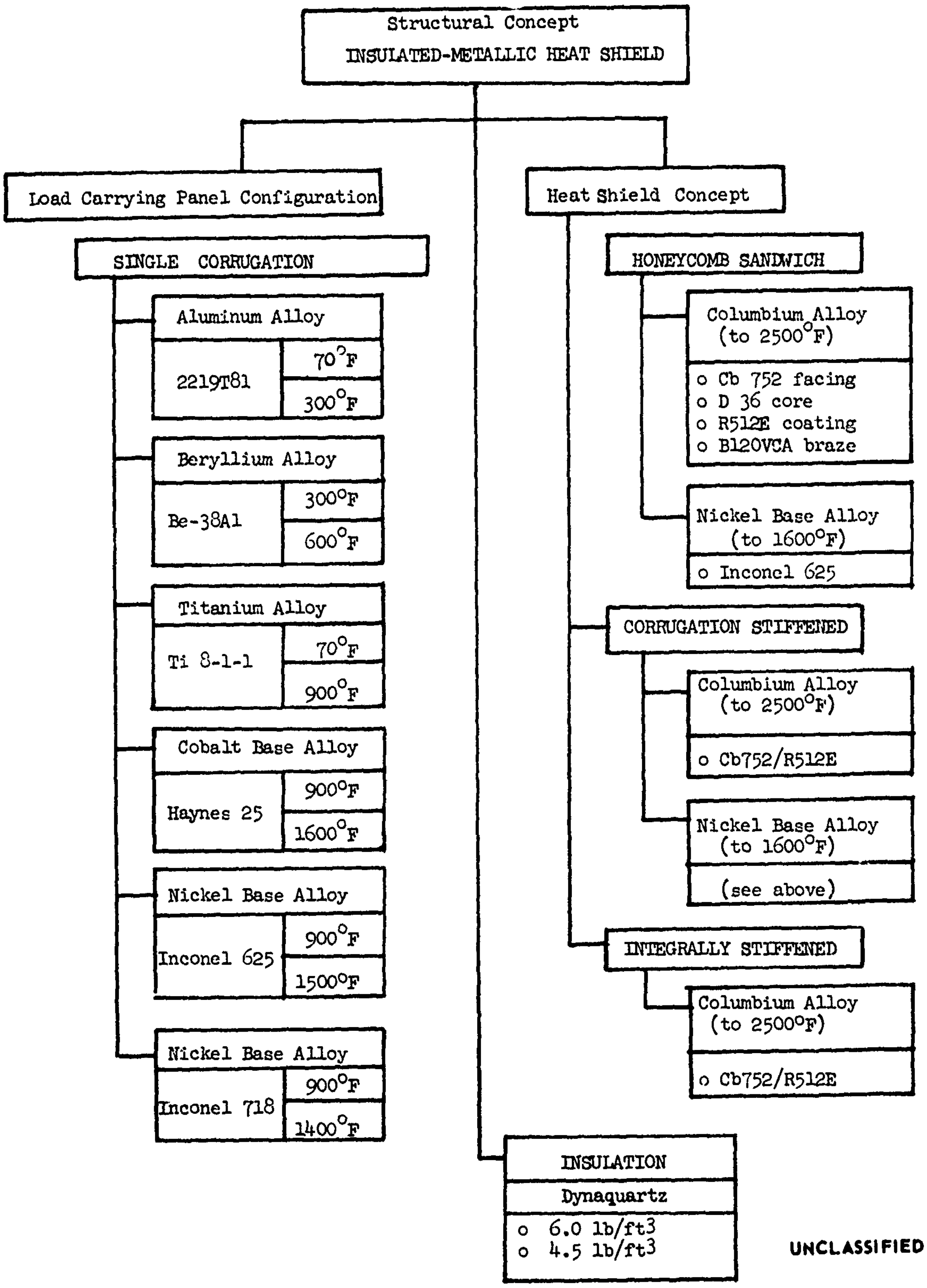


FIGURE 15 (U) HOT LOAD CARRYING STRUCTURAL CONCEPT - INTERNAL INSULATION OR INTERNAL INSULATION WITH ACTIVE COOLING



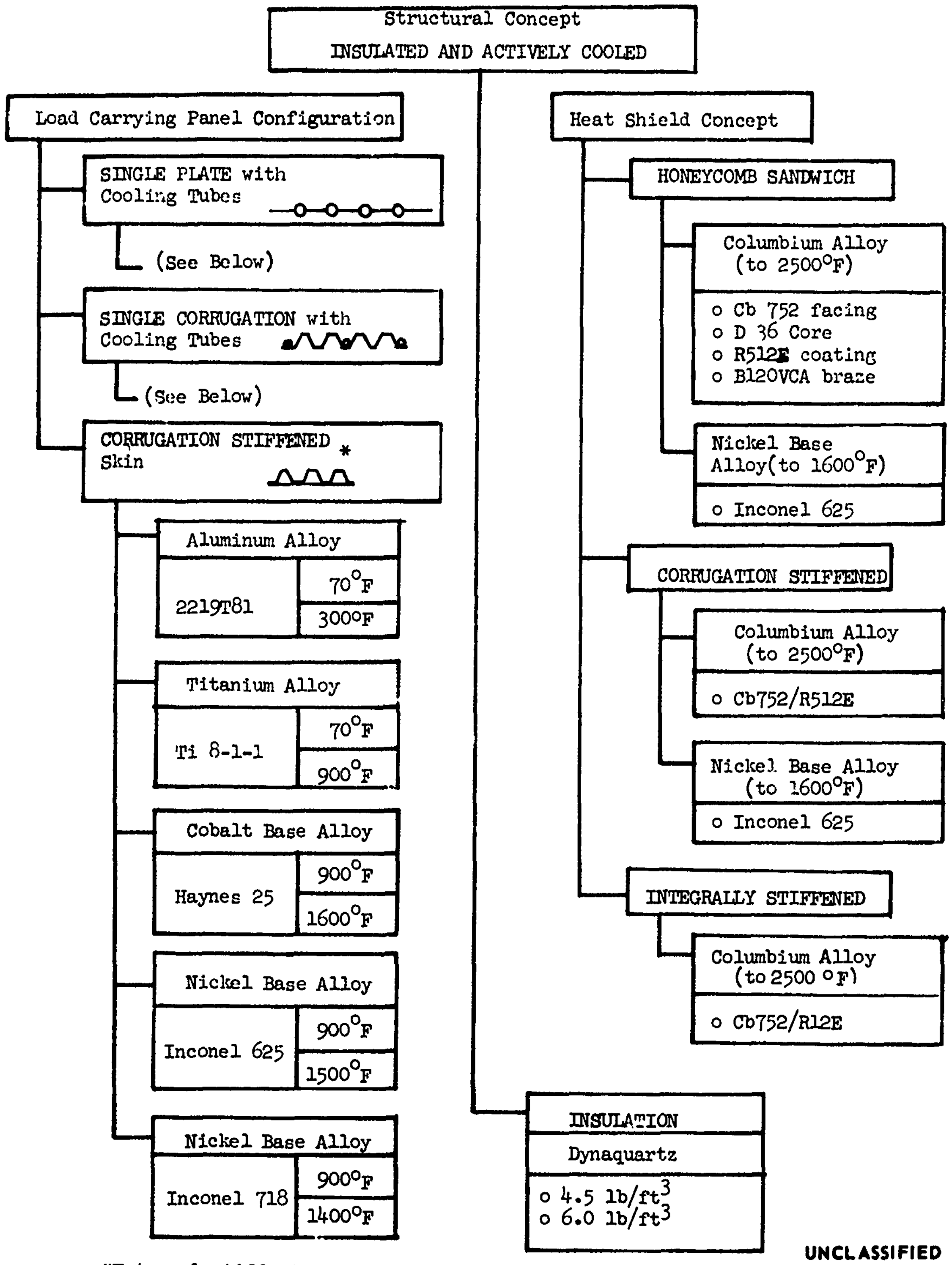
UNCLASSIFIED

FIGURE 16 (U) STRUCTURAL CONCEPT - INSULATED - HARD INSULATION



UNCLASSIFIED

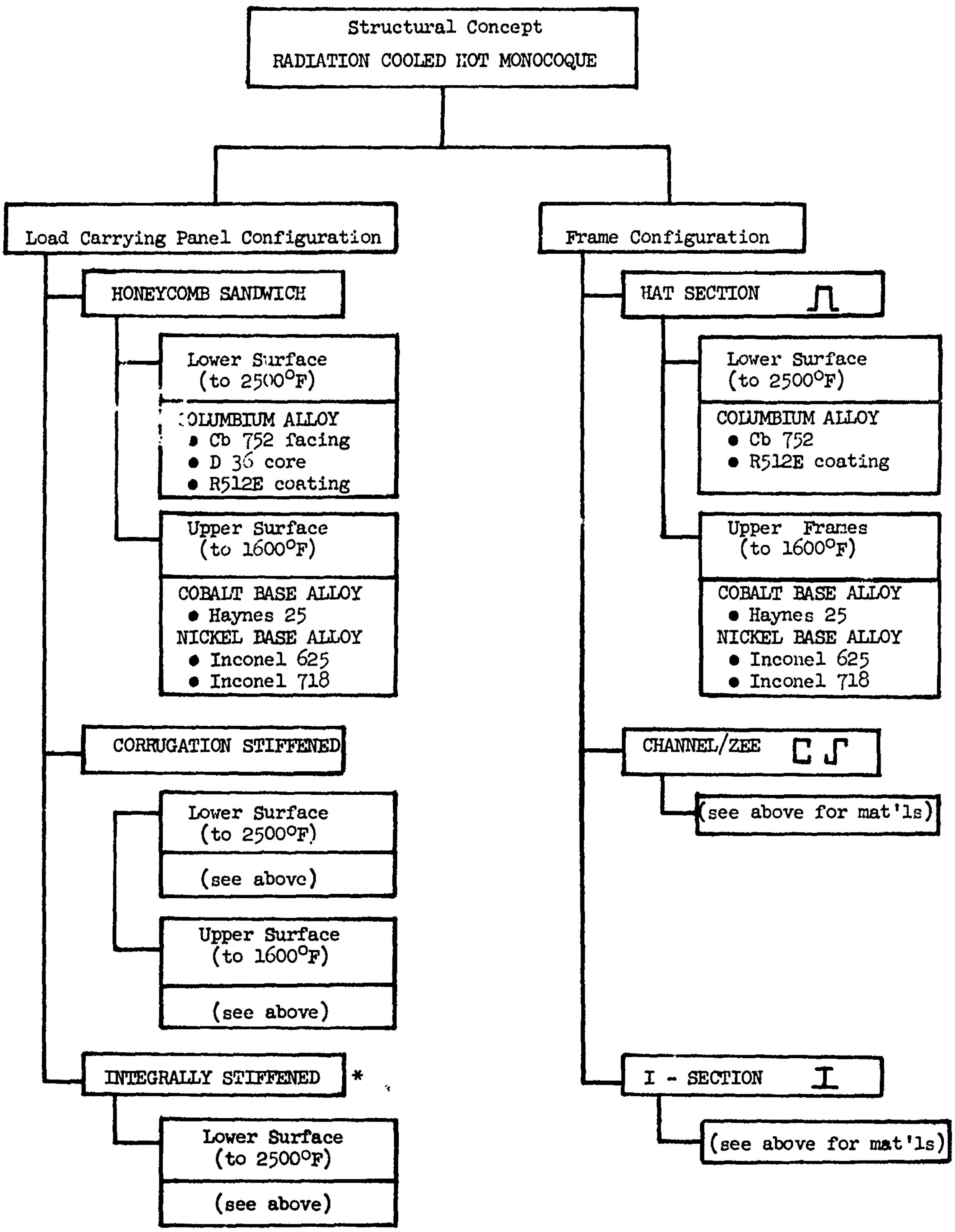
FIGURE 17 (U) STRUCTURAL CONCEPT - INSULATED-METALLIC HEAT SHIELD



UNCLASSIFIED

*External stiffening, integral cooling tubes

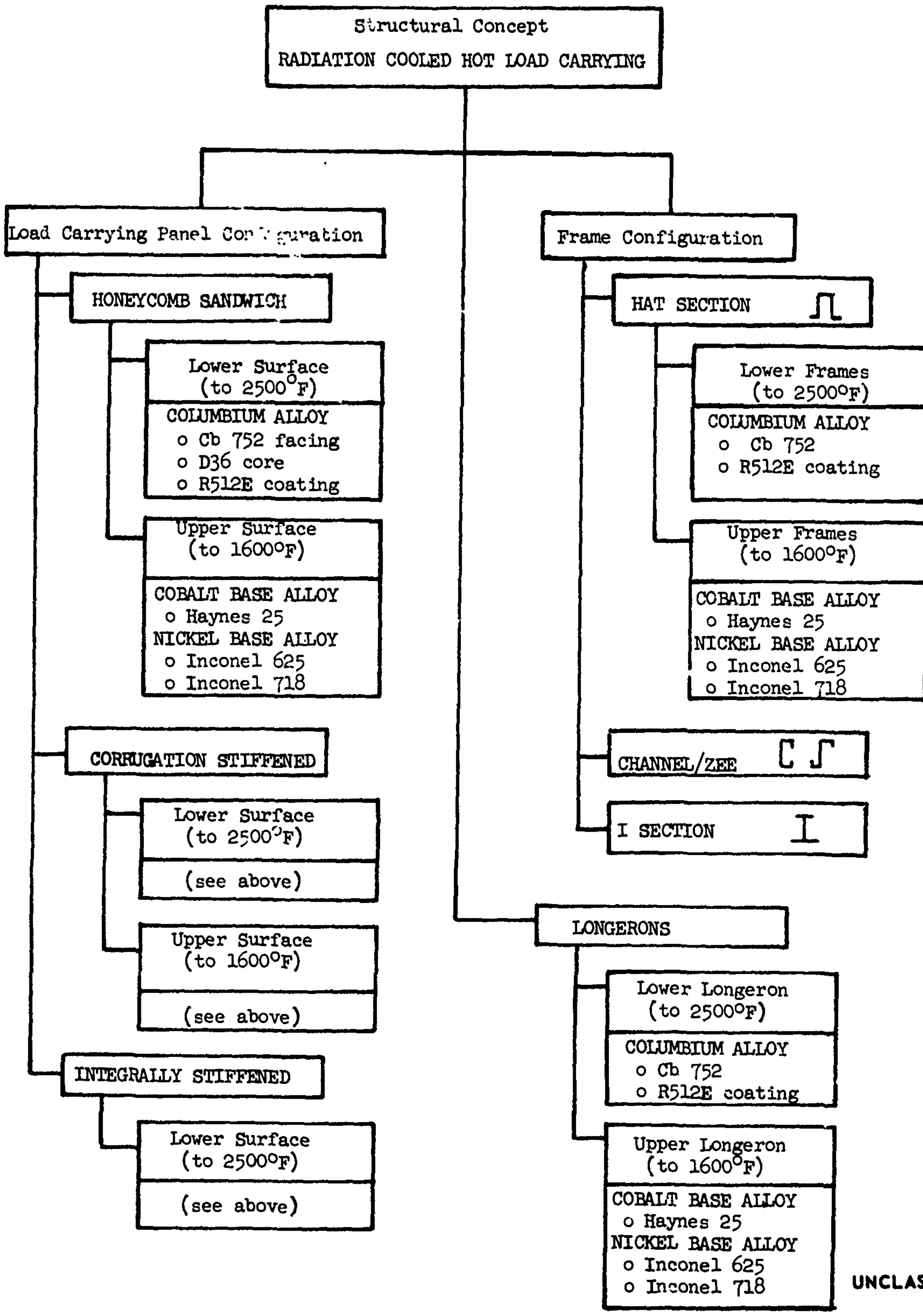
FIGURE 18 (U) STRUCTURAL CONCEPT - INSULATED AND ACTIVELY COOLED



UNCLASSIFIED

* Not applicable to upper surface - cannot fabricate to gage requirements

FIGURE 19 (U) STRUCTURAL CONCEPT, RADIATION COOLED HOT MONOCOQUE



UNCLASSIFIED

FIGURE 20 (U) STRUCTURAL CONCEPT - RADIATION COOLED HOT LOAD CARRYING

PANEL CONFIGURATION	GEOMETRY	WIDE COLUMN EFFICIENCY FACTOR
ZEE STIFFENED		0.911
SKIN-CORRUGATION	 *	0.685
BEADED SKIN-CORRUGATION	 *	0.590
HAT STIFFENED		0.928
UNFLANGED INTEGRALLY STIFFENED	 *	0.656
INTEGRAL TEE		1.000
INTEGRAL ZEE	 *	1.030
TRUSS CORE SANDWICH		0.605
HONEYCOMB SANDWICH	 *	1.60 - 3.20
SINGLE CORRUGATION	 *	1.600

UNCLASSIFIED

*Evaluated for flight test vehicle application

FIGURE 21 (U) CANDIDATE STRUCTURAL PANEL CONFIGURATIONS

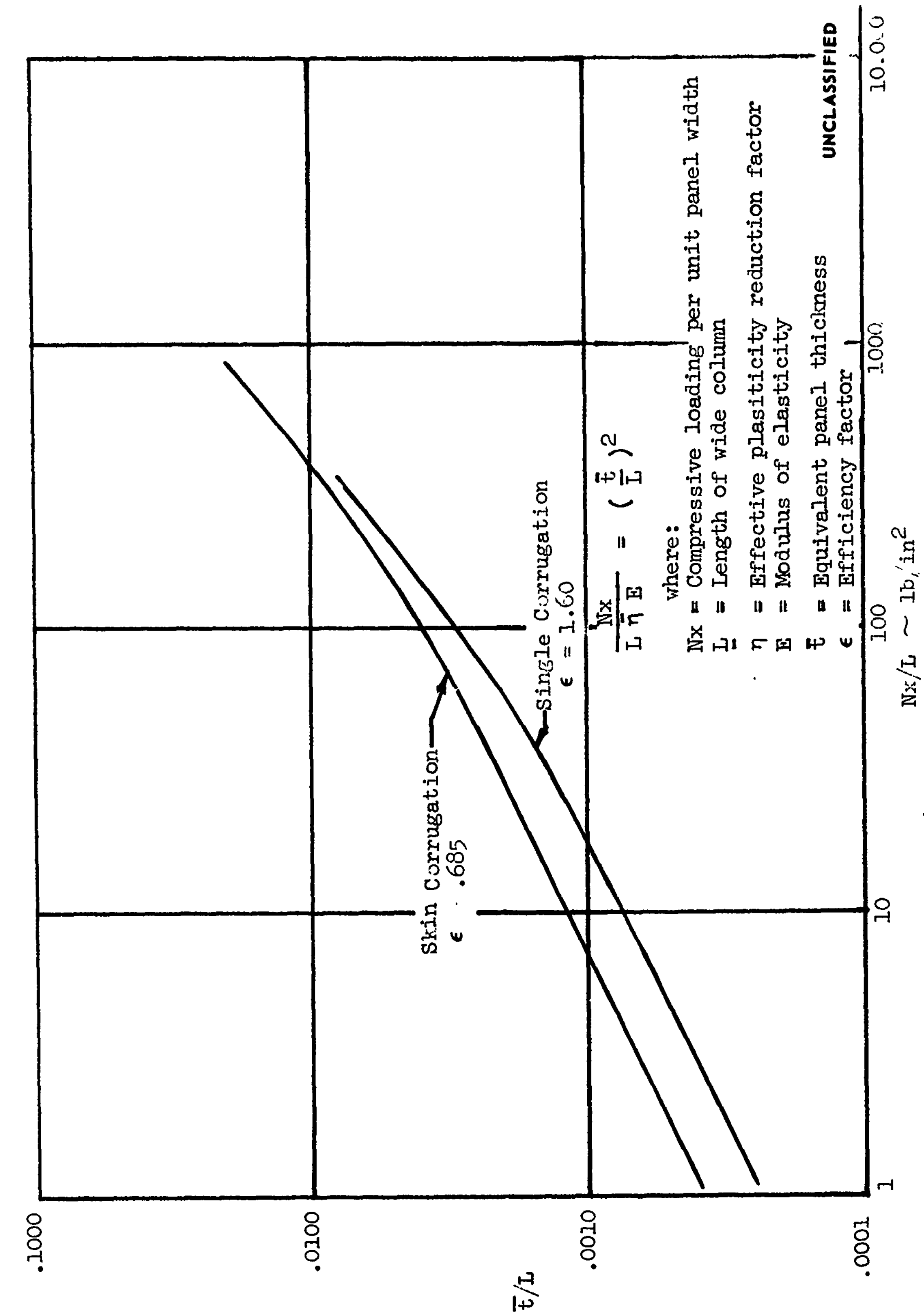


FIGURE 22 (U) WIDE COLUMN DESIGN - ALUMINUM 2219-T81 AT ROOM TEMPERATURE CONDITIONS

UNCLASSIFIED

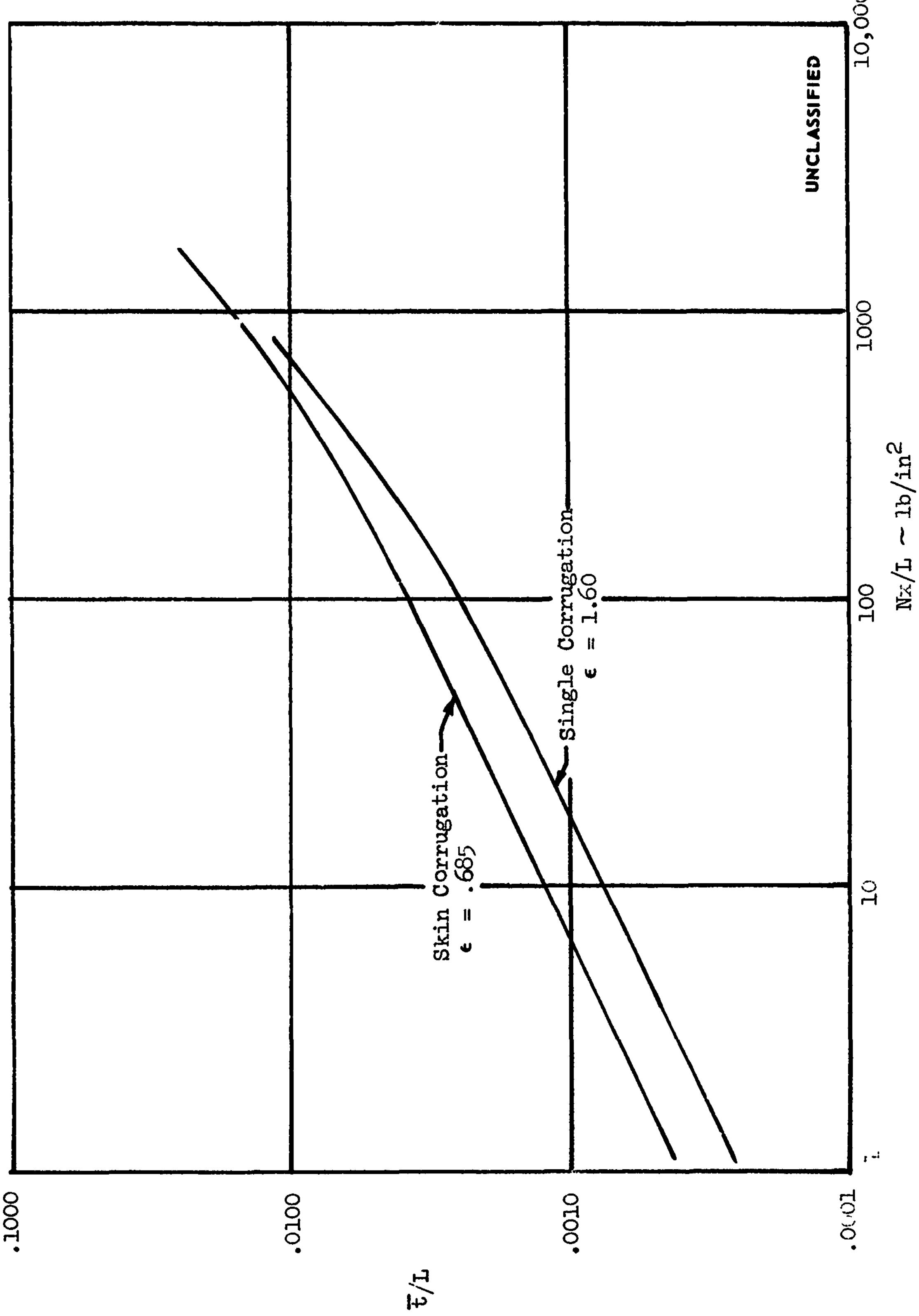


FIGURE 23 (U) WIDE COLUMN DESIGN - ALUMINUM 7075-T6 AT ROOM TEMPERATURE CONDITIONS

UNCLASSIFIED

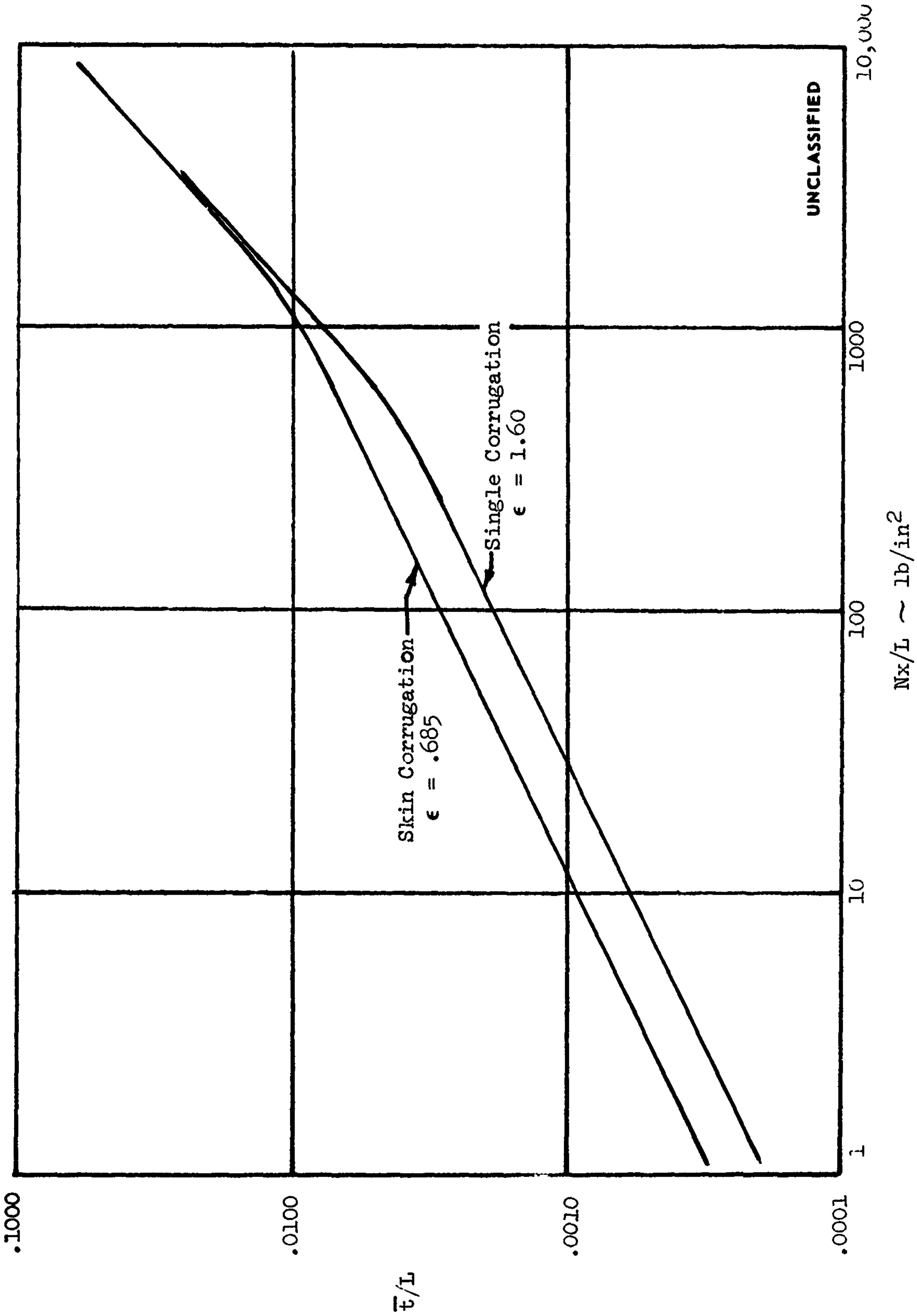


FIGURE 24 (U) WIDE COLUMN DESIGN - TITANIUM 8Al-1Mo-1V AT ROOM TEMPERATURE CONDITIONS

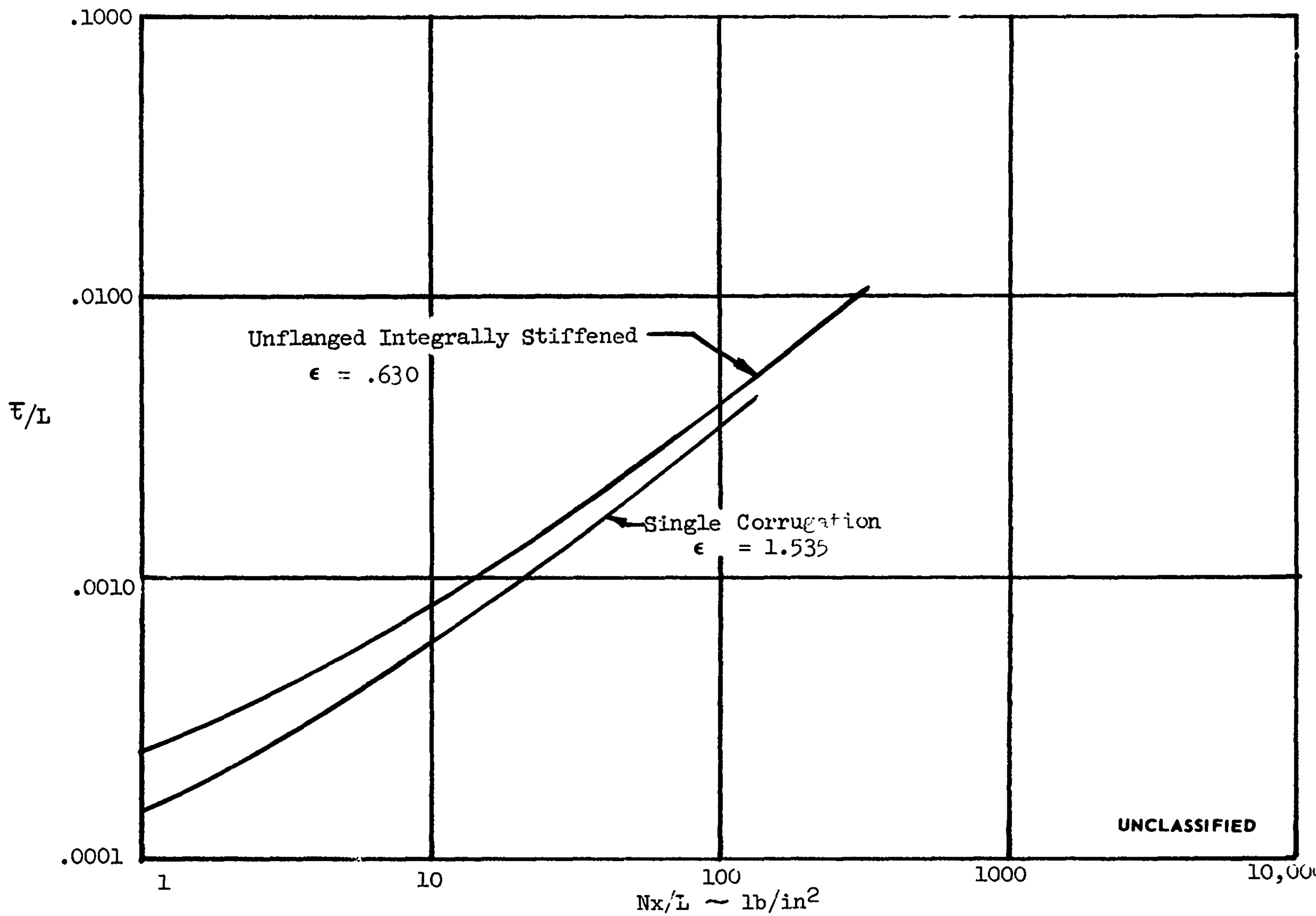


FIGURE 25 (U) WIDE COLUMN DESIGN - BERYLLIUM-ALUMINUM Be-38Al AT ROOM TEMPERATURE CONDITIONS

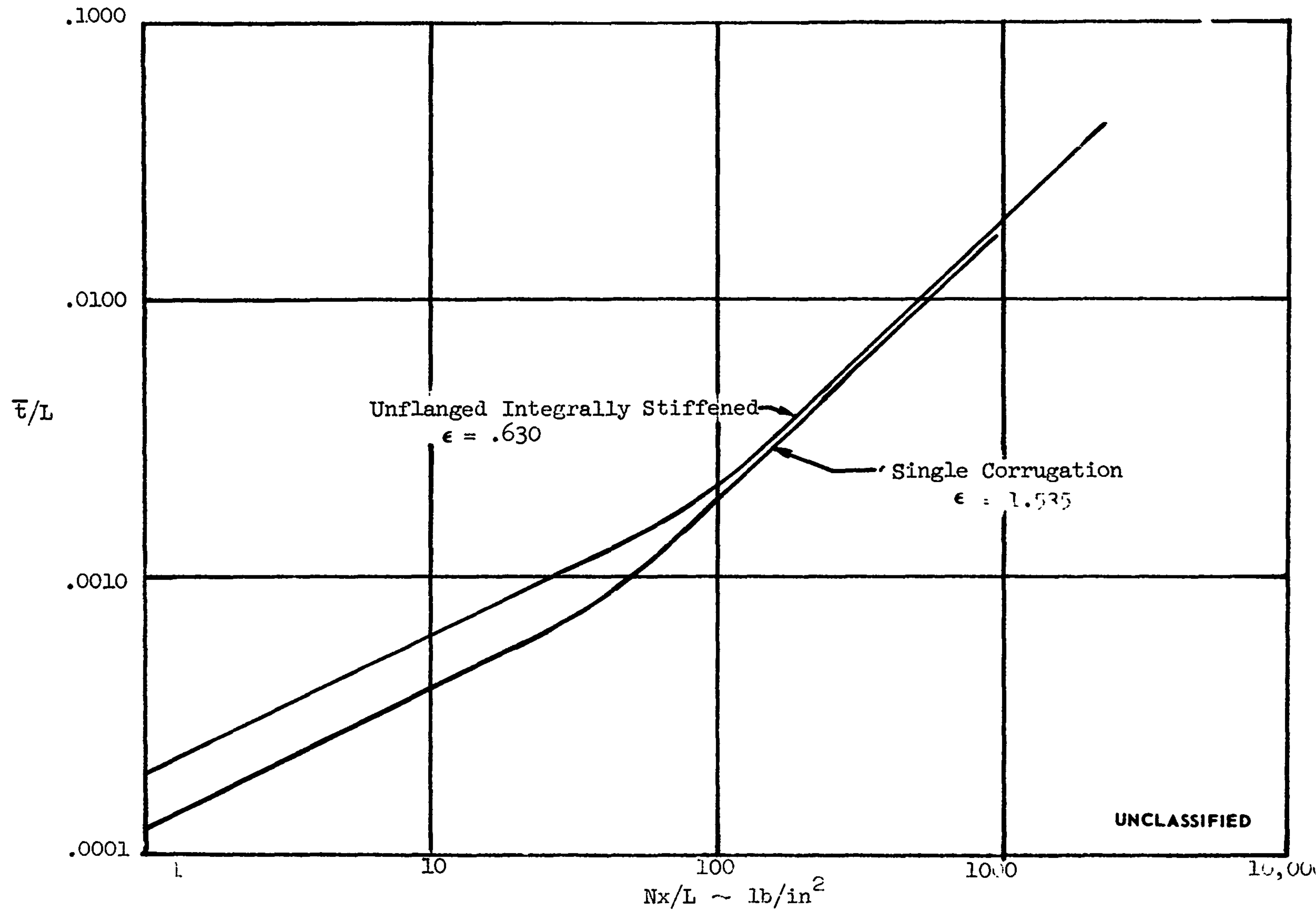


FIGURE 26 (U) WIDE COLUMN DESIGN - BERYLLIUM AMS-7902 AT ROOM TEMPERATURE CONDITIONS

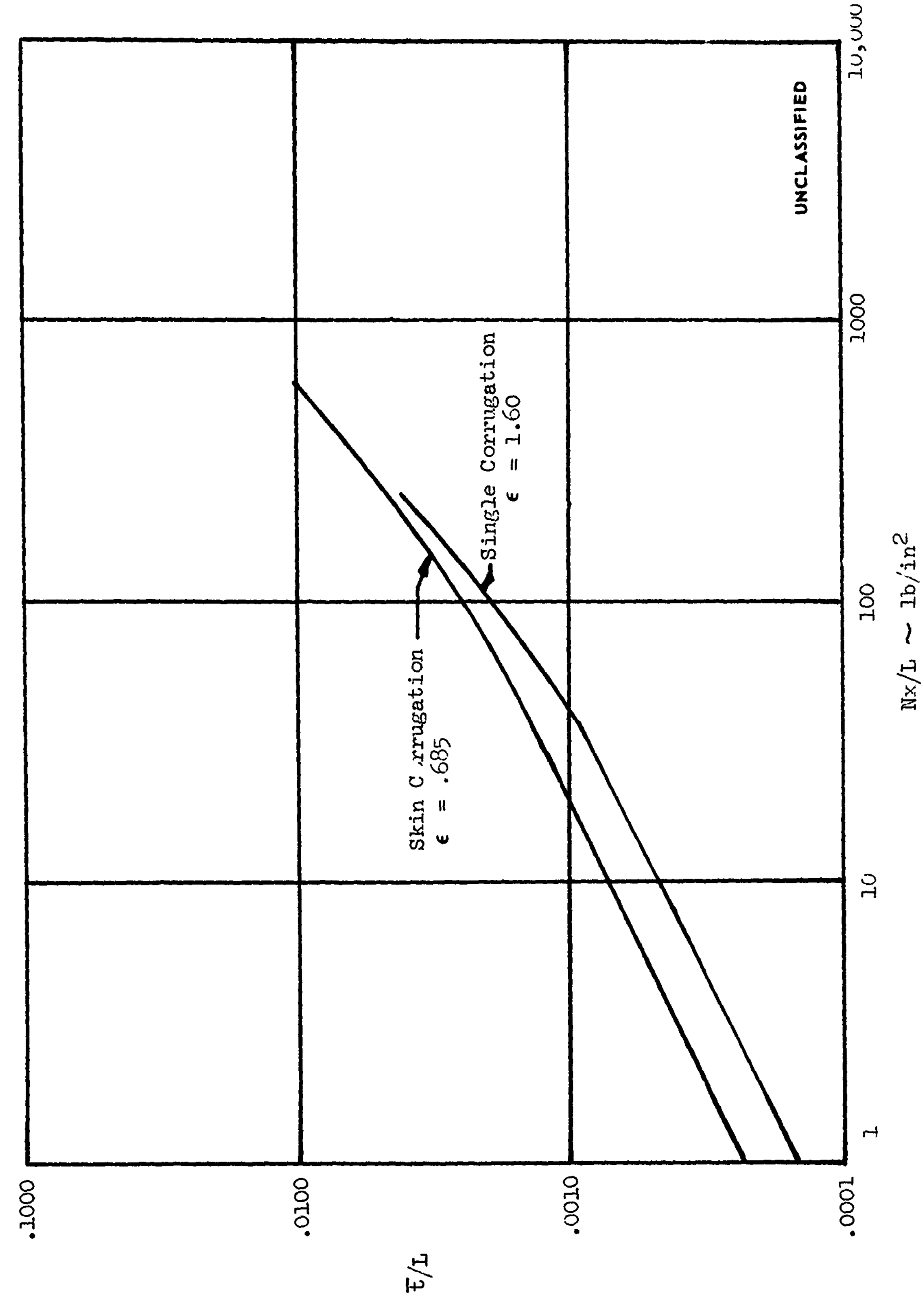


FIGURE 27 (U) WIDE COLUMN DESIGN - INCONEL 625 AT ROOM TEMPERATURE CONDITIONS

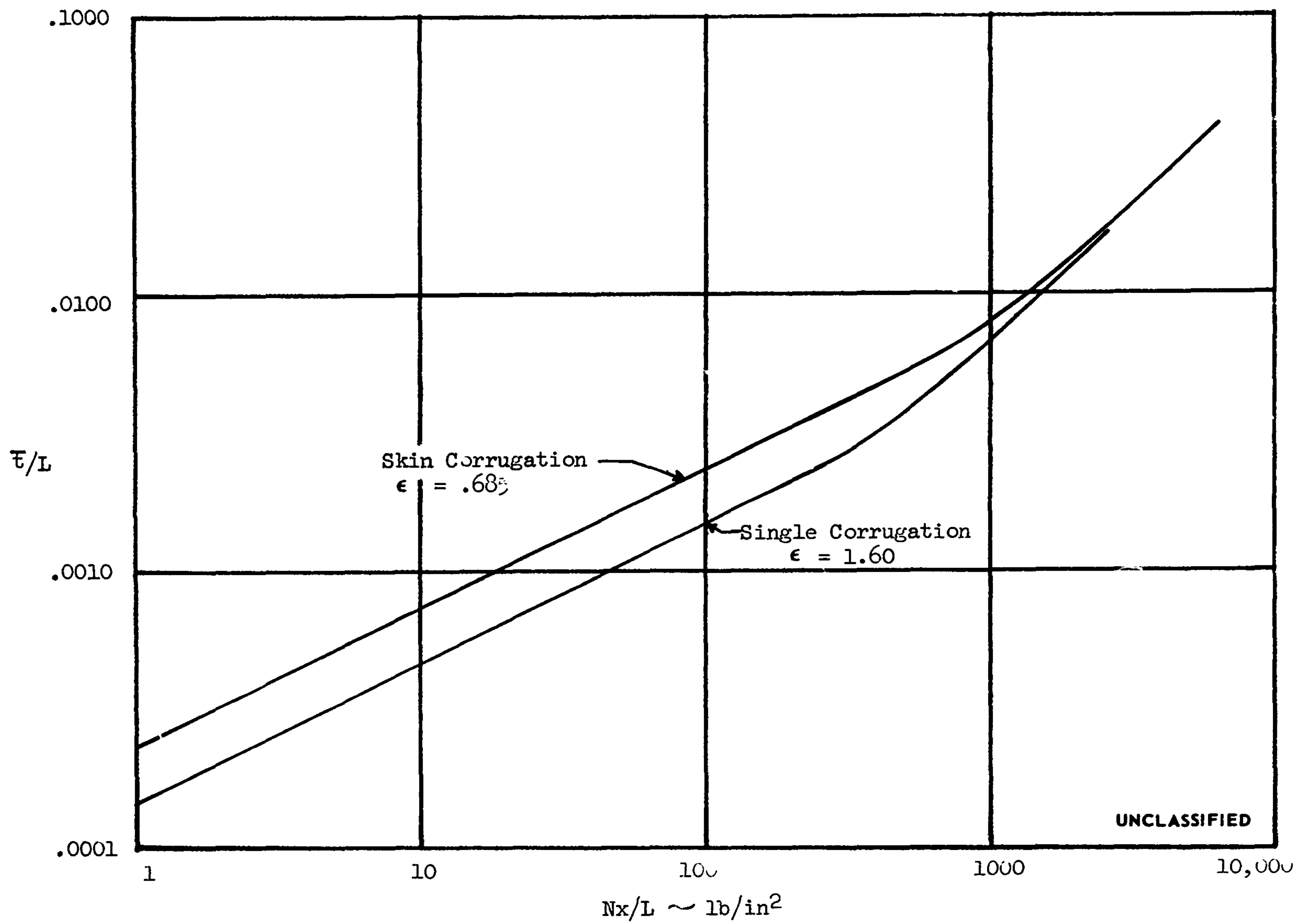


FIGURE 28 (U) WIDE COLUMN DESIGN - INCONEL 718 AT ROOM TEMPERATURE CONDITIONS

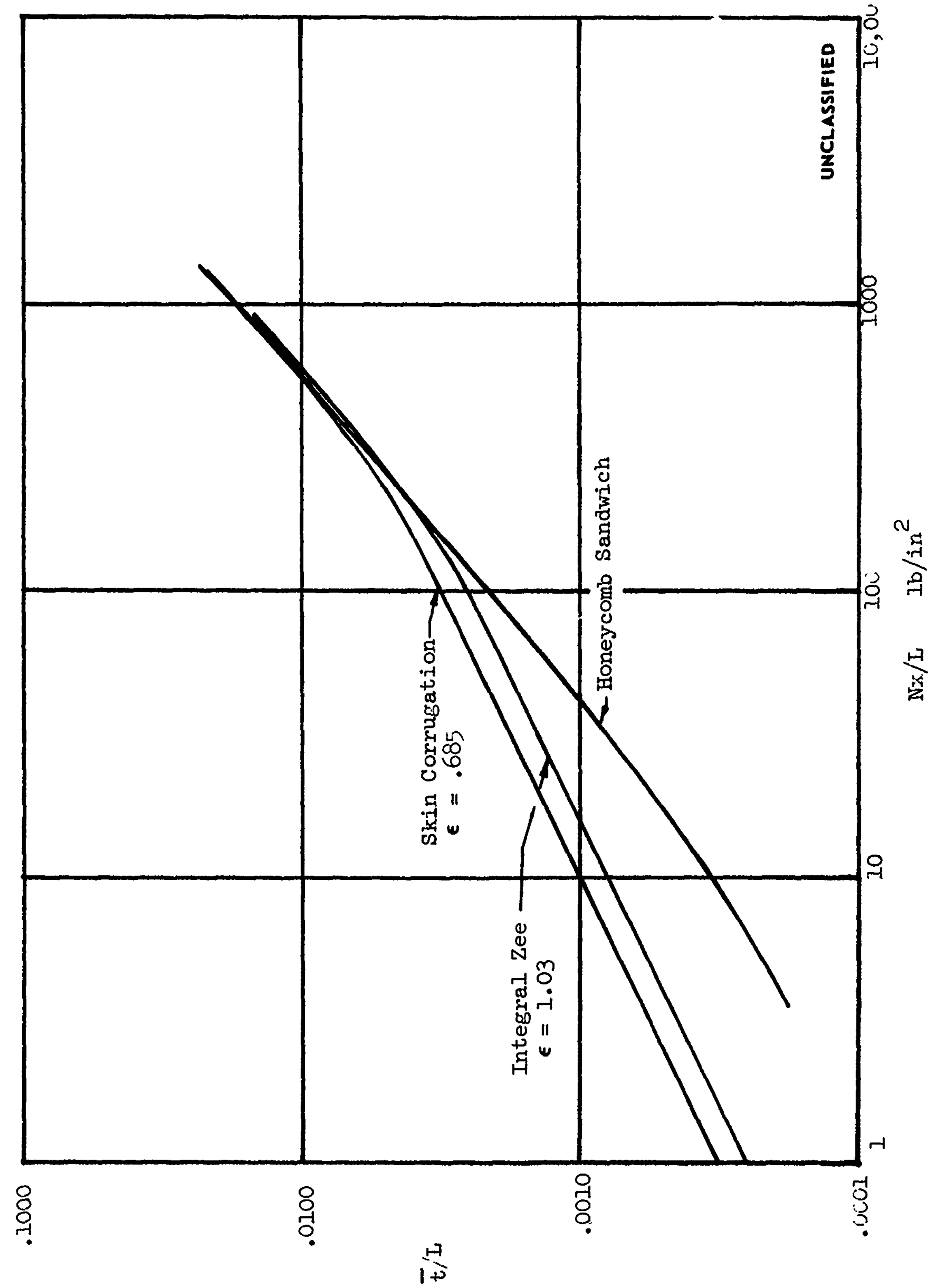


FIGURE 29 (U) WIDE COLUMN DESIGN - COLUMBIUM Cb752 AT ROOM TEMPERATURE CONDITIONS

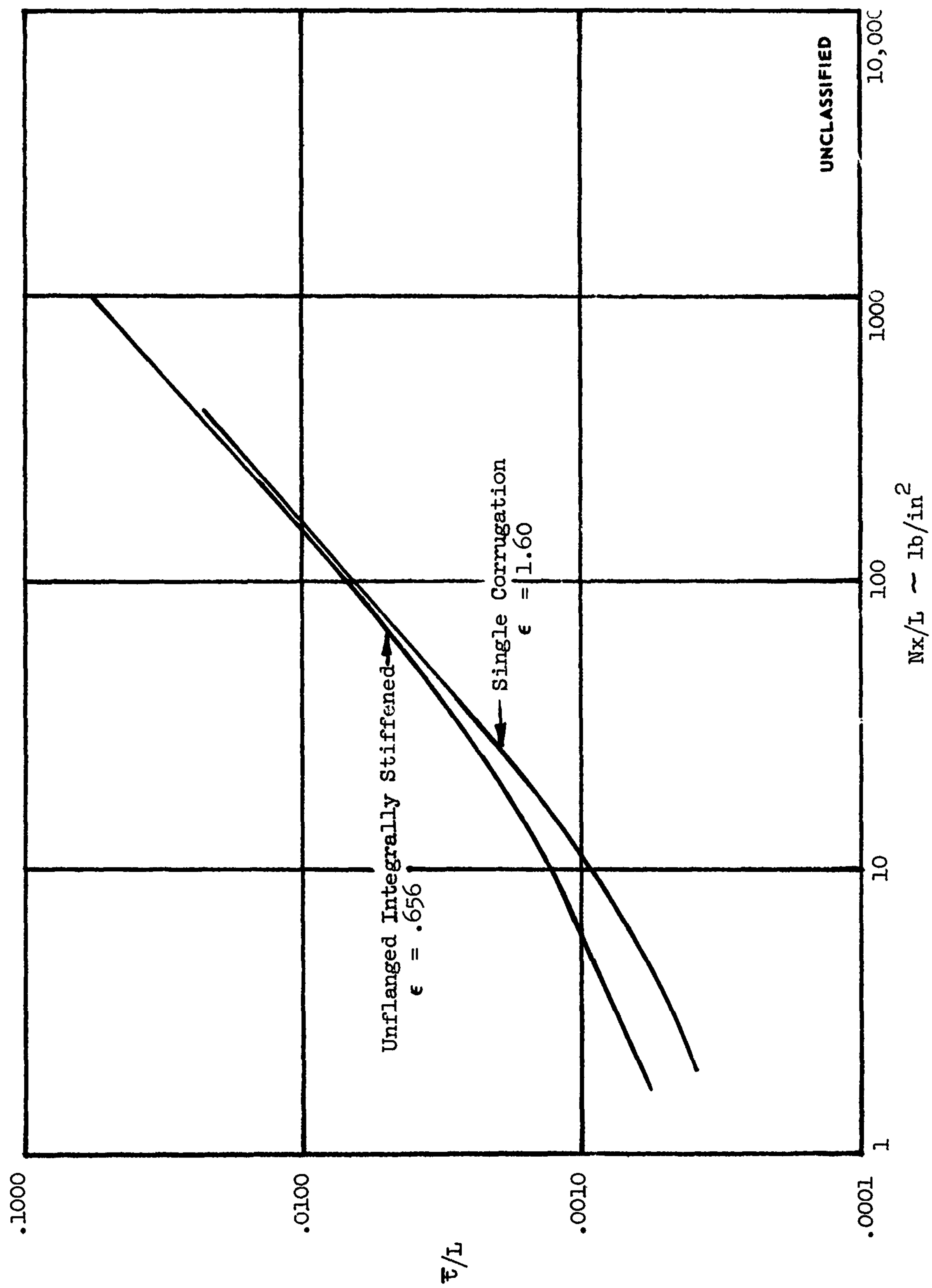
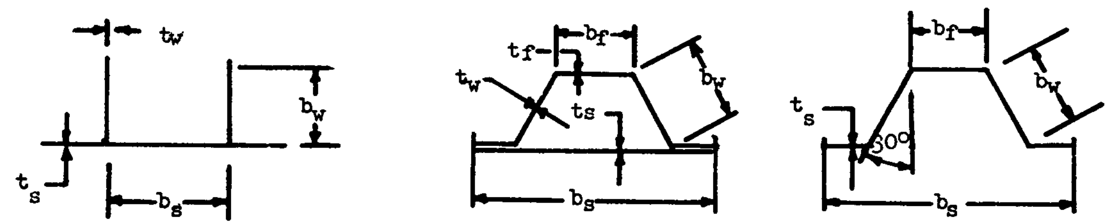


FIGURE 30 (U) WIDE COLUMN DESIGN - COLUMBIUM Cb752 AT 2500°F

Panel Type: Axial Load Carrying

Geometry:



MATERIAL:	ALUMINUM ALLOY; TITANIUM ALLOY; BERYLLIUM ALLOY												
ITEM / PANEL CONFIGURATION	MATERIAL ALLOY DESIGNATION	ρ DENSITY	t_s (SKIN)	t_w (WEB)	t_f (FLANGE)	b_s (SKIN)	b_w (WEB)	b_f (FLANGE)	$\bar{t}_s$ (STRUC)	$\bar{t}_t$ (PANEL)	$\bar{t}_{nof}$ NON OPTIMUM	$\bar{t}_T$ TOTAL PANEL	W_T PANEL WT. (LB/FT^2)
UNFLANGED INTEGRALLY STIFFENED	Be-38Al	.0756	.016	.036	--	.927	.604	--	.0394	.0394	.0099	.0493	0.537
	Be(AMS7902)	.066	.016	.036	--	.927	.604	--	.0394	.0394	.0099	.0493	0.470
SKIN CORRUGATION	Al.Al.2219T81	.102	.016	.012	.012	1.569	.580	.492	.0329	.0329	.0066	.0395	0.580
	Ti 8Al-1Mo-1V	.158	.016	.012	.012	1.569	.580	.492	.0329	.0329	.0066	.0395	0.900
SINGLE CORRUGATION	Al.Al.2219T81	.102	.016	--	--	1.811	.670	.569	.0219	.0219	.0022	.0241	0.354
	Ti 8Al-1Mo-1V	.158	.016	--	--	1.811	.670	.569	.0219	.0219	.0022	.0241	0.548
	Be-38Al	.0756	.016	--	--	1.811	.670	.569	.0219	.0219	.0022	.0241	0.262
	Be(AMS 7902)	.066	.016	--	--	1.811	.670	.569	.0219	.0219	.0022	.0241	0.229

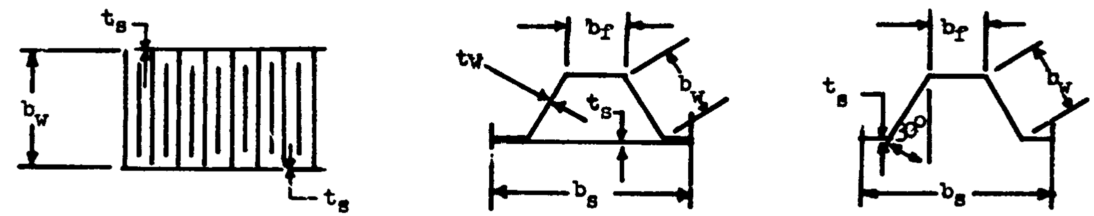
UNCLASSIFIED

FIGURE 31 (U) WEIGHT COMPARISON OF MINIMUM GAGE STRUCTURAL PANELS - ALUMINUM, TITANIUM, BERYLLIUM

UNCLASSIFIED

Panel Type: Axial Load Carrying

GEOMETRY:



MATERIAL:	NICKEL BASE AND COBALT BASE ALLOYS												
ITEM / PANEL CONFIGURATION	MATERIAL ALLOY DESIGNATION	t_s (SKIN)	t_w (WEB)	b_s (SKIN)	b_w (WEB)	b_f (FLANGE)	$\bar{t}_s$ (STRUC)	$\bar{t}_h$ (CORE)	$\bar{t}_b$ (BRAZE)	$\bar{t}_s$ (PANEL)	$\bar{t}_{nof}$ NON OPTIMUM	$\bar{t}_T$ TOTAL PANEL	W_T PANEL WT (LB/FT^2)
HONEYCOMB SANDWICH	Inconel 718	.008	--	--	.788	--	.0160	.0076	.0040	.0276	.0082	.0358	1.530
	Inconel 625	.008	--	--	.937	--	.0160	.0093	.0040	.0293	.0087	.0380	1.670
	Haynes 25	.008	--	--	1.245	--	.0160	.0120	.0040	.0320	.0095	.0415	1.970
SKIN-CORRUGATION	Inconel 718	.016	.012	1.569	.580	.492	.0329	--	--	.0329	.0082	.0411	1.750
	Inconel 625	.010	.010	1.432	.530	.450	.0246	--	--	.0246	.0062	.0308	1.350
	Haynes 25	.010	.010	1.432	.530	.450	.0246	--	--	.0246	.0062	.0308	1.464
SINGLE CORRUGATION	Inconel 718	.016	--	1.811	.670	.569	.0219	--	--	.0219	.0022	.0241	1.025
	Inconel 625	.010	--	1.432	.530	.450	.0137	--	--	.0137	.0014	.0151	0.664
	Haynes 25	.010	--	1.432	.530	.450	.0137	--	--	.0137	.0014	.0151	0.717

MATERIAL DENSITY:

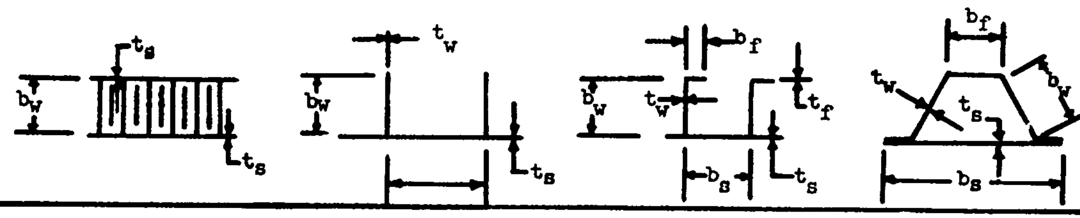
Inconel 718 $\rho = 0.296$ lb/in.^{3}
Inconel 625 $\rho = 0.305$ lb/in.^{3}
Haynes 25 $\rho = 0.330$ lb/in.^{3}

UNCLASSIFIED

FIGURE 32 (U) WEIGHT COMPARISON OF MINIMUM GAGE STRUCTURAL PANELS - SUPERALLOYS

Panel Type: Axial Load Carrying

GEOMETRY:



MATERIAL:	COLUMBIUM ALLOY Cb752 R512 GT&E COATING										ρ, DENSITY = 0.326 lb/in.^{3}			
ITEM / PANEL CONFIGURATION	t_s (SKIN)	t_w (WEB)	t_f (FLANGE)	b_s (SKIN)	b_w (WEB)	b_f (FLANGE)	$\bar{t}_s$ (STRUC)	$\bar{t}_h$ (CORE)	$\bar{t}_c$ (COATING)	$\bar{t}_b$ (BRAZE)	$\bar{t}_t$ (PANEL)	$\bar{t}_{nof}$ NON OPTIMUM	$\bar{t}_T$ TOTAL PANEL	W_T PANEL WT (LB/FT^2)
HONEYCOMB SANDWICH	.010	--	--	--	.603	--	.0200	.0088	.0015	.0022	.0325	.0099	.0422	1.980
UNFLANGED INTEGRALLY STIFFENED	.012	.027	--	.803	.521	--	.0296	--	.0025	--	.0321	.0080	.0401	1.885
INTEGRAL ZEE	.012	.012	.012	.530	.530	.159	.0276	--	.0034	--	.0310	.0077	.0387	1.820
SKIN-CORRUGATION	.012	.012	.012	1.569	.580	.492	.0282	--	.0031	--	.0313	.0079	.0392	1.840

NOTES: t_n, actual thickness (in.)
$\bar{t}_n$, effective thickness (in.)

UNCLASSIFIED

FIGURE 33 (U) WEIGHT COMPARISON OF MINIMUM GAGE STRUCTURAL PANELS - COLUMBIUM ALLOY

PANEL CONFIGURATION	MATERIAL	$\bar{t}_s$ (2)	t_T (3)	N_x/L	N_x	W_T (4)	F_c (5)
	(Table 4-1)			--	($15N_x/L$)		($\frac{N_x}{\bar{t}_s}$)
HONEYCOMB (6) SANDWICH	Inconel 625	.016	.0380	47.5	713	1.670	44,600
	Inconel 718	.016	.0358	105.0	1578	1.530	98,600
	Haynes 25	.016	.0415	49.0	735	1.970	46,000
	Cb 752	.020	.0422	66.0	990	1.980	49,500
UNFLANGED INTEGRALLY STIFFENED	Be-38Al	.0394	.0493	55.0	825	0.537	20,900
	Be(AMS7902)	.0394	.0493	125.0	1875	0.470	47,600
	Cb752	.0296	.0401	38.0	570	1.885	19,250
INTEGRAL ZEE	Cb752	.0276	.0387	50.0	1750	1.820	27,200
SKIN- (1) CORRUGATION	AL.2219T81	.0329	.0395	38.9	58.4	0.580	17,750
	Ti8Al-1Mo-IV	.0329	.0395	74.3	1114	0.900	33,900
	Inconel 625	.0246	.0308	47.5	713	1.350	29,000
	Inconel 718	.0329	.0411	105.0	1578	1.750	47,900
	Haynes 25	.0246	.0308	49.0	735	1.464	28,300
	Cb 752	.0282	.0392	49.5	743	1.840	26,400
SINGLE CORRUGATION	AL 2219T81	.0219	.0241	35.0	525	0.354	24,000
	Ti8Al-Mo-IV	.2019	.0241	61.0	915	0.548	41,800
	Be-38Al	.0219	.0241	35.0	525	0.262	24,000
	Be(AMS7902)	.0219	.0241	75.0	1125	0.229	51,300
	Inconel 625	.0137	.0151	47.0	630	0.664	46,000
	Inconel 718	.0219	.0241	100.0	1500	1.025	68,500
	Haynes 25	.0137	.0151	41.3	620	0.717	45,300

NOTES:

(1) MINIMUM GAGE RESULTS IN OFF-OPTIMUM CONDITION.
(2) EQUIVALENT THICKNESS OF LOAD CARRYING MATERIAL.
(3) EQUIVALENT THICKNESS OF TOTAL PANEL INCLUDING NON-OPTIMUM FACTOR.
(4) UNIT WEIGHT OF TOTAL PANEL INCLUDING APPROPRIATE NON OPTIMUM FACTOR.
(5) ALLOWABLE WIDE COLUMN STRESS.
(6) SUPERALLOY HONEYCOMB PANELS - NOT OPTIMUM. BASED ON EQUAL LOAD CARRYING CAPABILITY AS CORRESPONDING SKIN-CORRUGATION.

UNCLASSIFIED

FIGURE 34 (U) STABILITY OF MINIMUM GAGE STRUCTURAL PANELS (L = 15.0 INCHES)

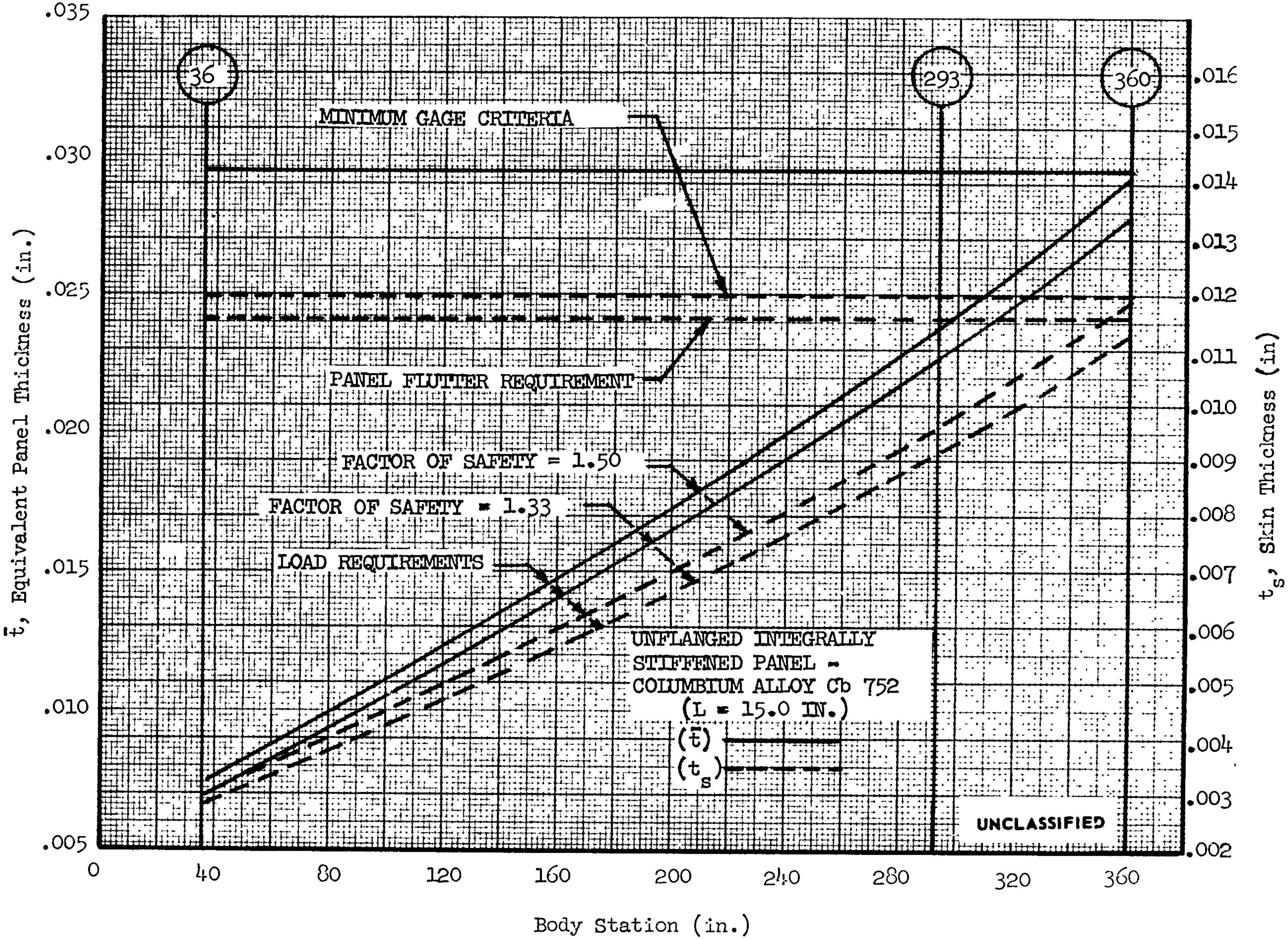


FIGURE 35 (U) THICKNESS REQUIREMENTS FOR LOWER SURFACE PANELS (L = 15.0 INCHES)

UNCLASSIFIED

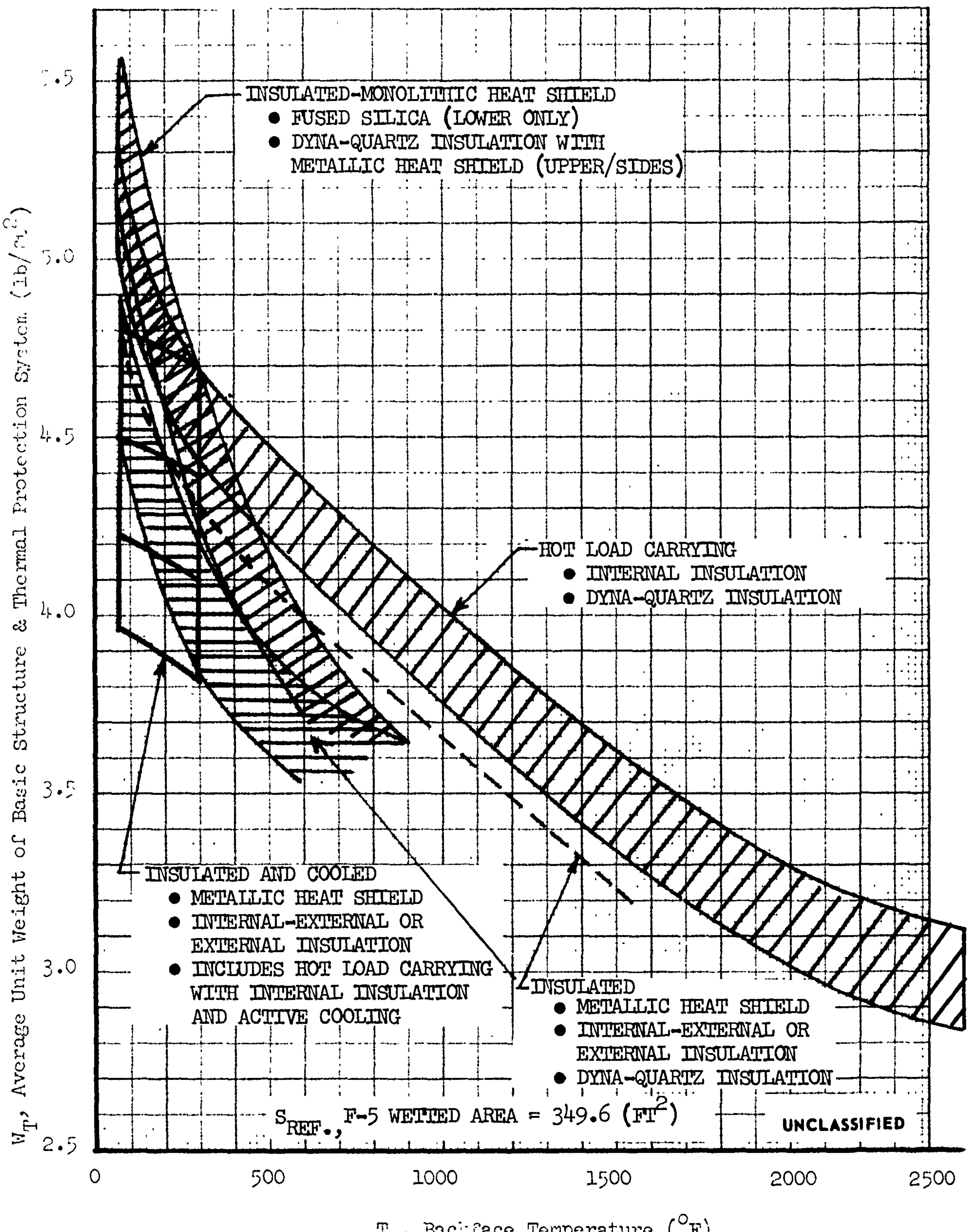


FIGURE 36 (U) WEIGHT COMPARISON OF CANDIDATE THERMOSTRUCTURAL CONCEPTS - F-5 STUDY VEHICLE

UNCLASSIFIED

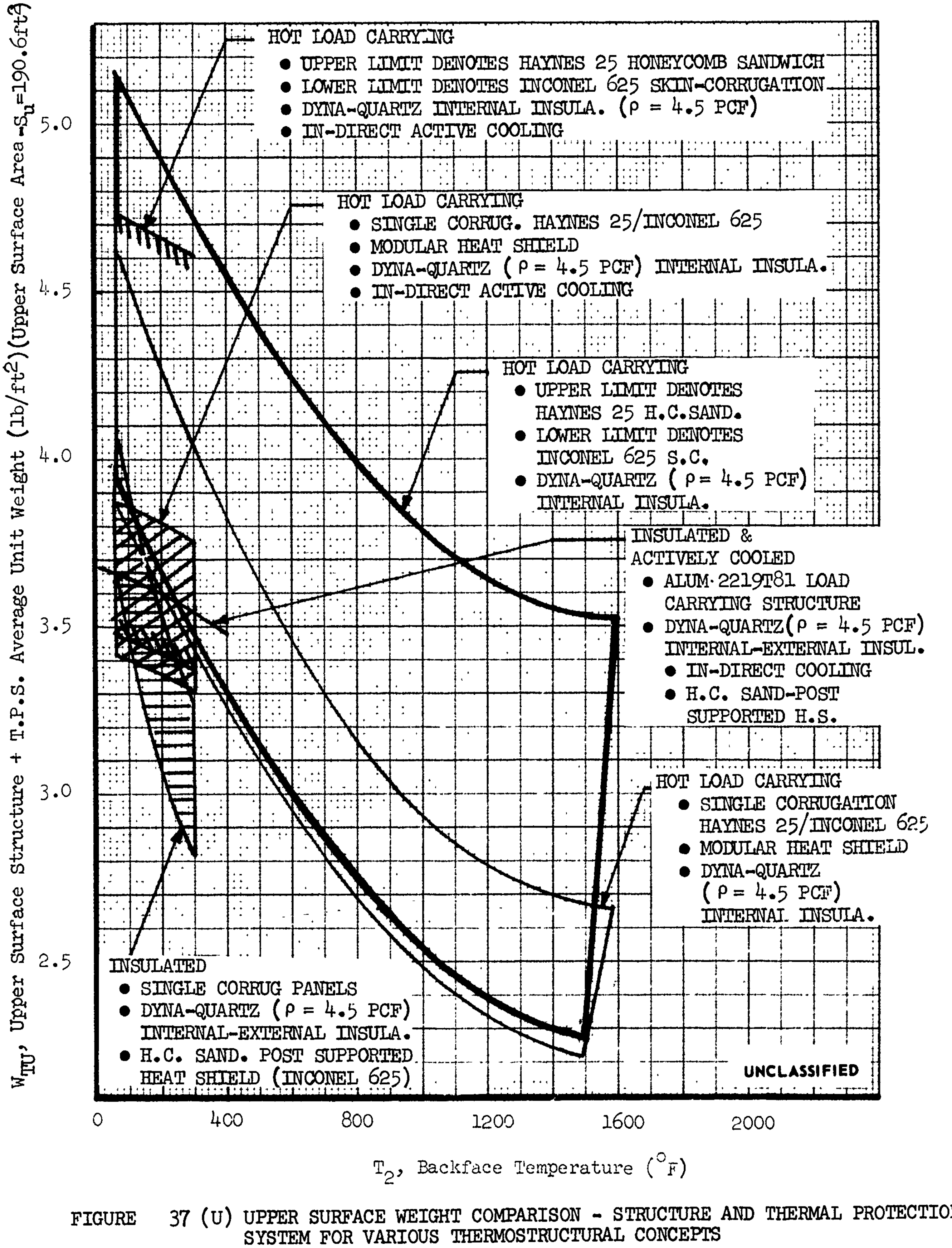


FIGURE 37 (U) UPPER SURFACE WEIGHT COMPARISON - STRUCTURE AND THERMAL PROTECTION SYSTEM FOR VARIOUS THERMOSTRUCTURAL CONCEPTS

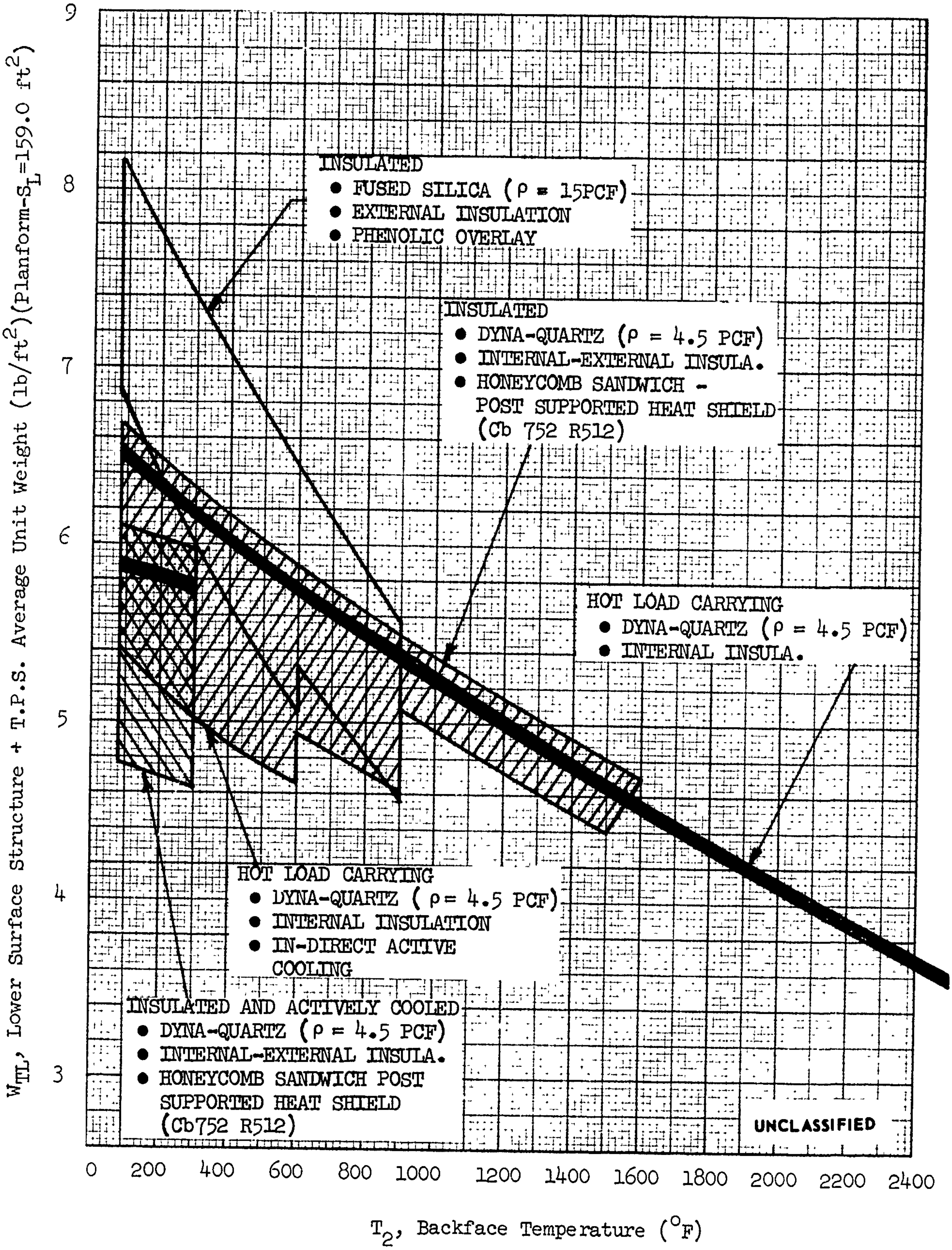


FIGURE 38 (U) LOWER SURFACE WEIGHT COMPARISON - STRUCTURE AND THERMAL PROTECTION SYSTEM FOR VARIOUS THERMOSTRUCTURAL CONCEPTS

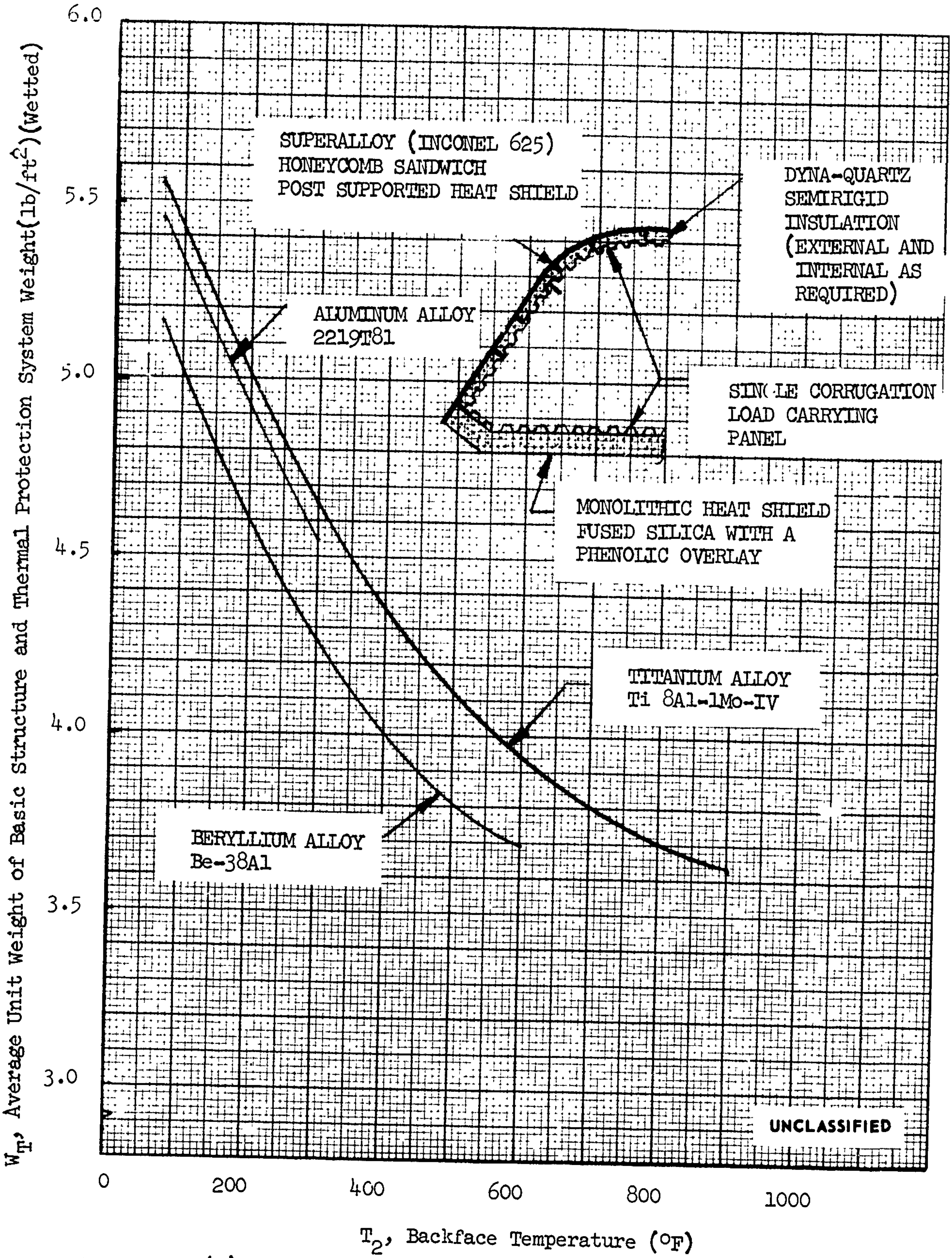


FIGURE 39 (U) WEIGHT COMPARISON - INSULATED (FUSED SILICA & DYNA-QUARTZ) THERMOSTRUCTURAL CONCEPT

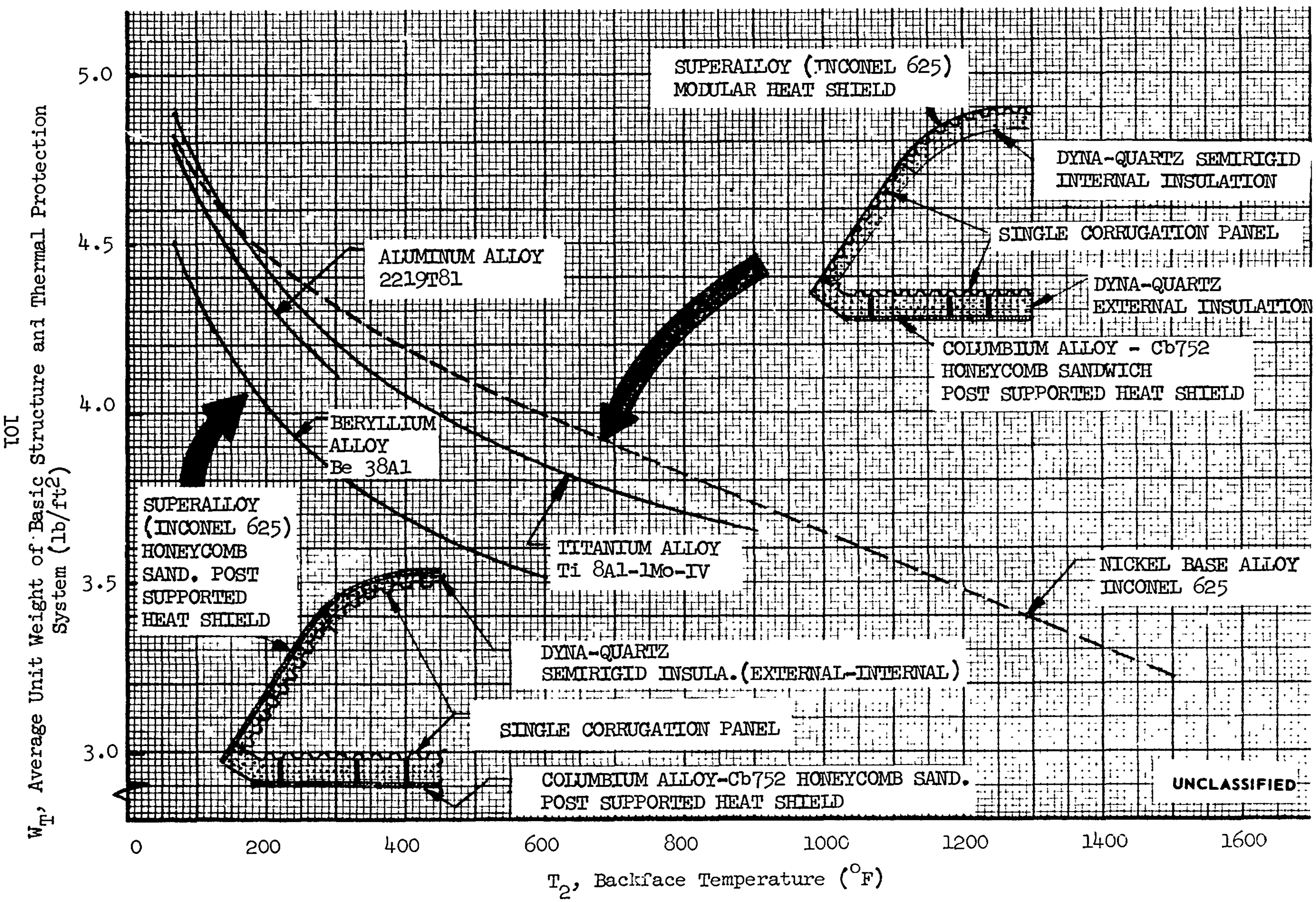


FIGURE 40 (U) WEIGHT COMPARISON - INSULATED THERMOSTRUCTURAL CONCEPT (DYNA-QUARTZ-INTERNAL EXTERNAL INSULATION)

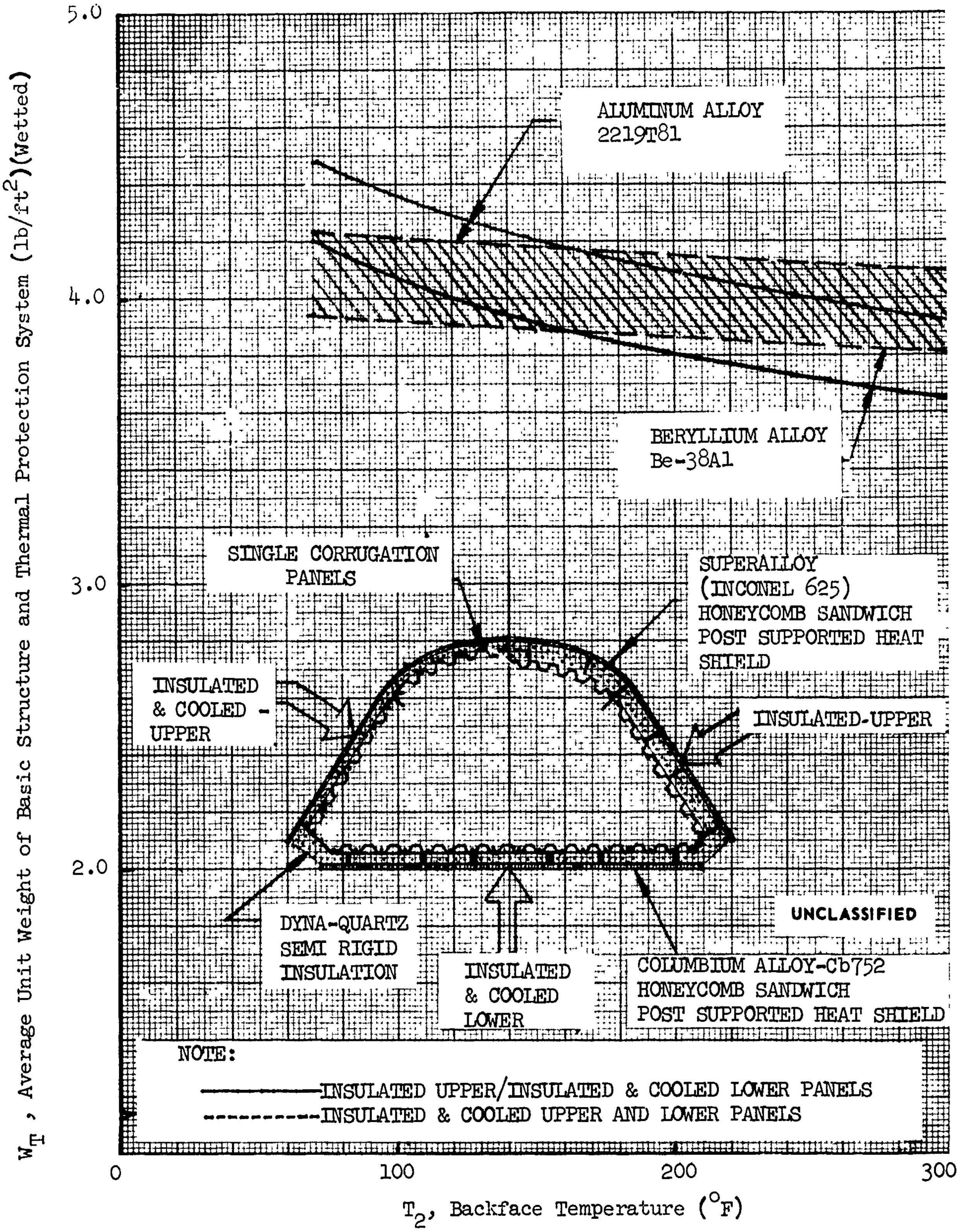


FIGURE 41 (U) WEIGHT COMPARISON - INSULATED AND COOLED THERMOSTRUCTURAL CONCEPT (INDIRECT ACTIVE COOLING USING EXPENDABLE WATER)

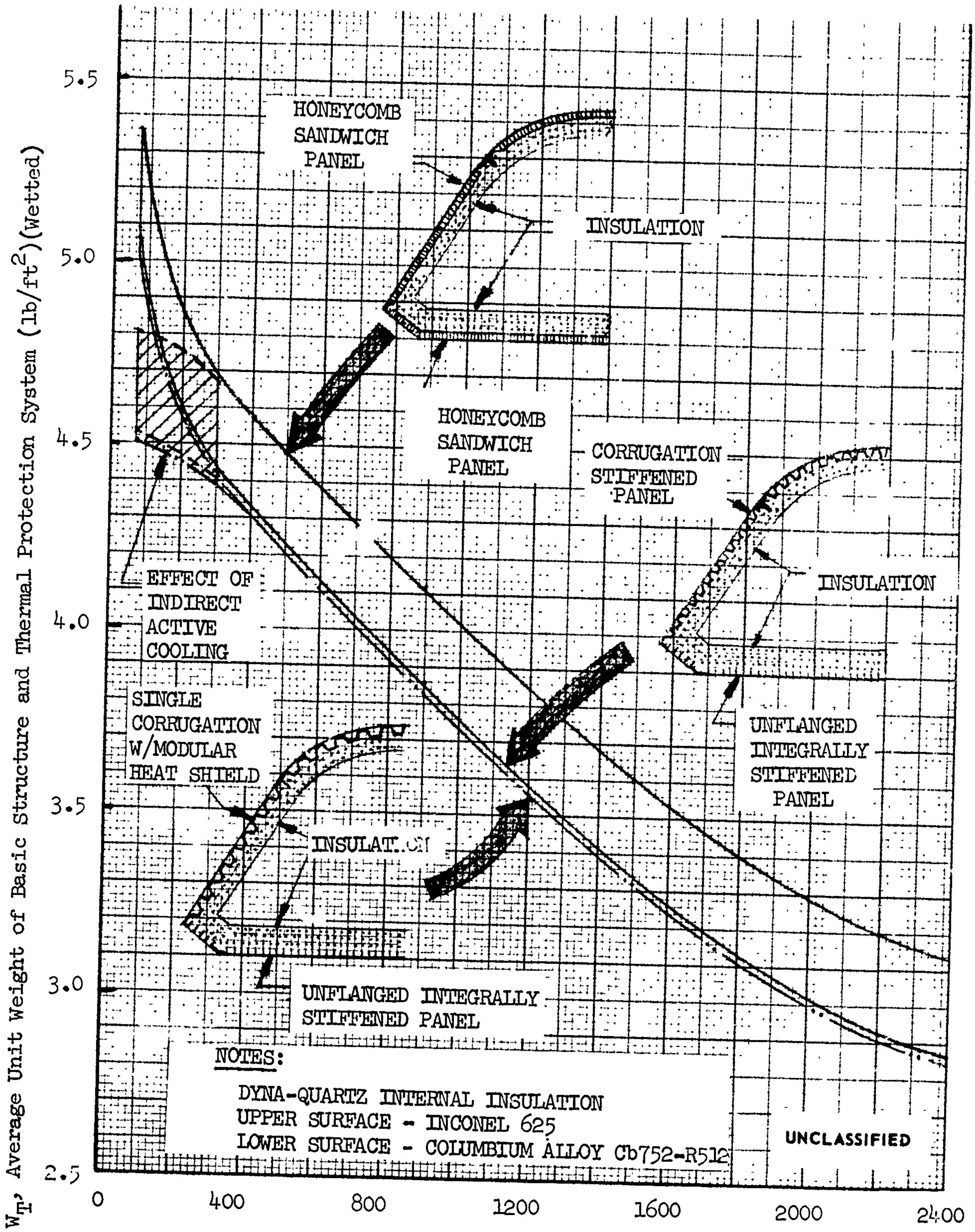


FIGURE 42 (U) WEIGHT COMPARISON - HOT LOAD CARRYING THERMOSTRUCTURAL CONCEPT

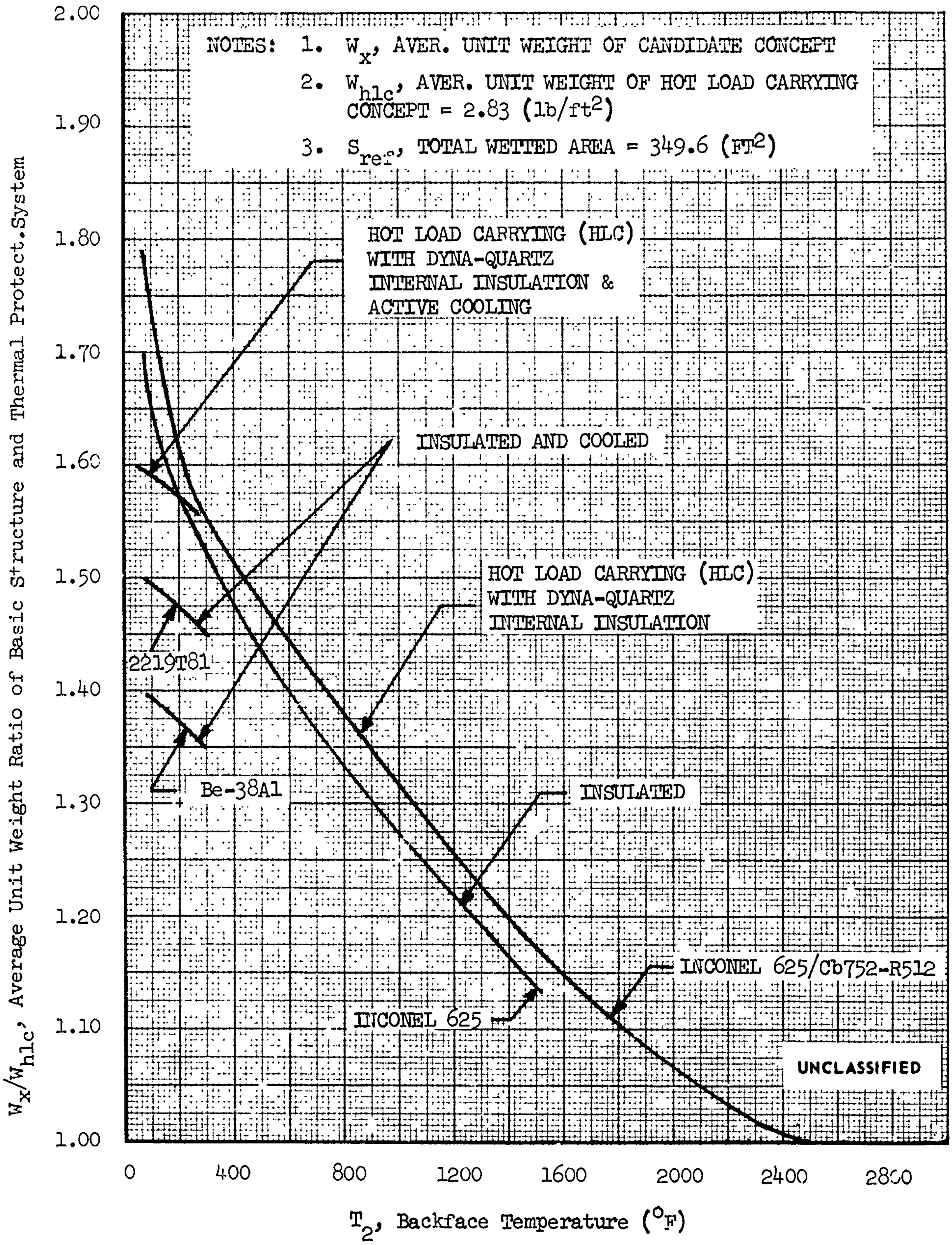


FIGURE 43 (U) THERMOSTRUCTURAL CONCEPT SELECTION EFFECT ON AVERAGE UNIT WEIGHT OF BASIC STRUCTURE AND THERMAL PROTECTION SYSTEM

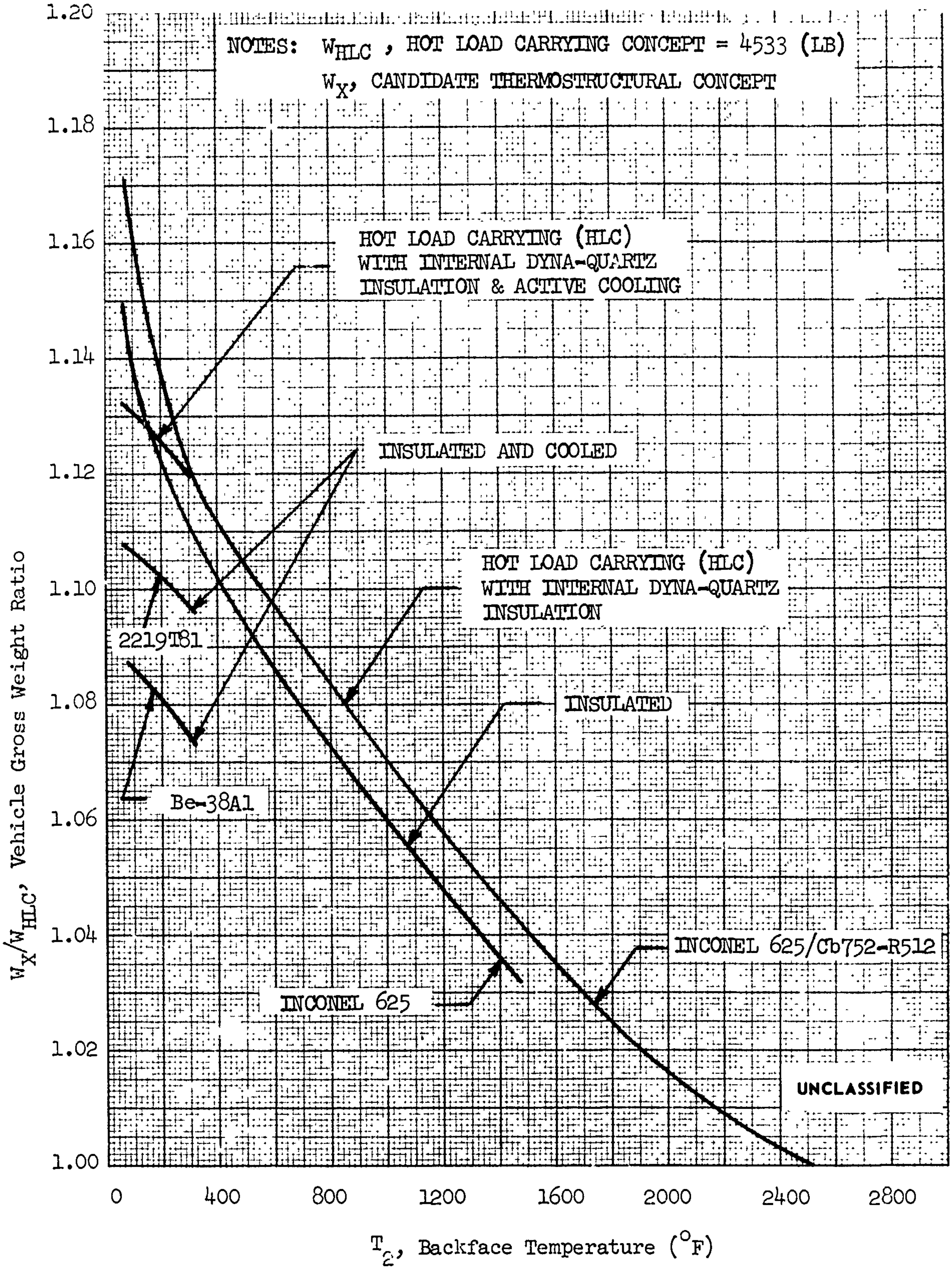
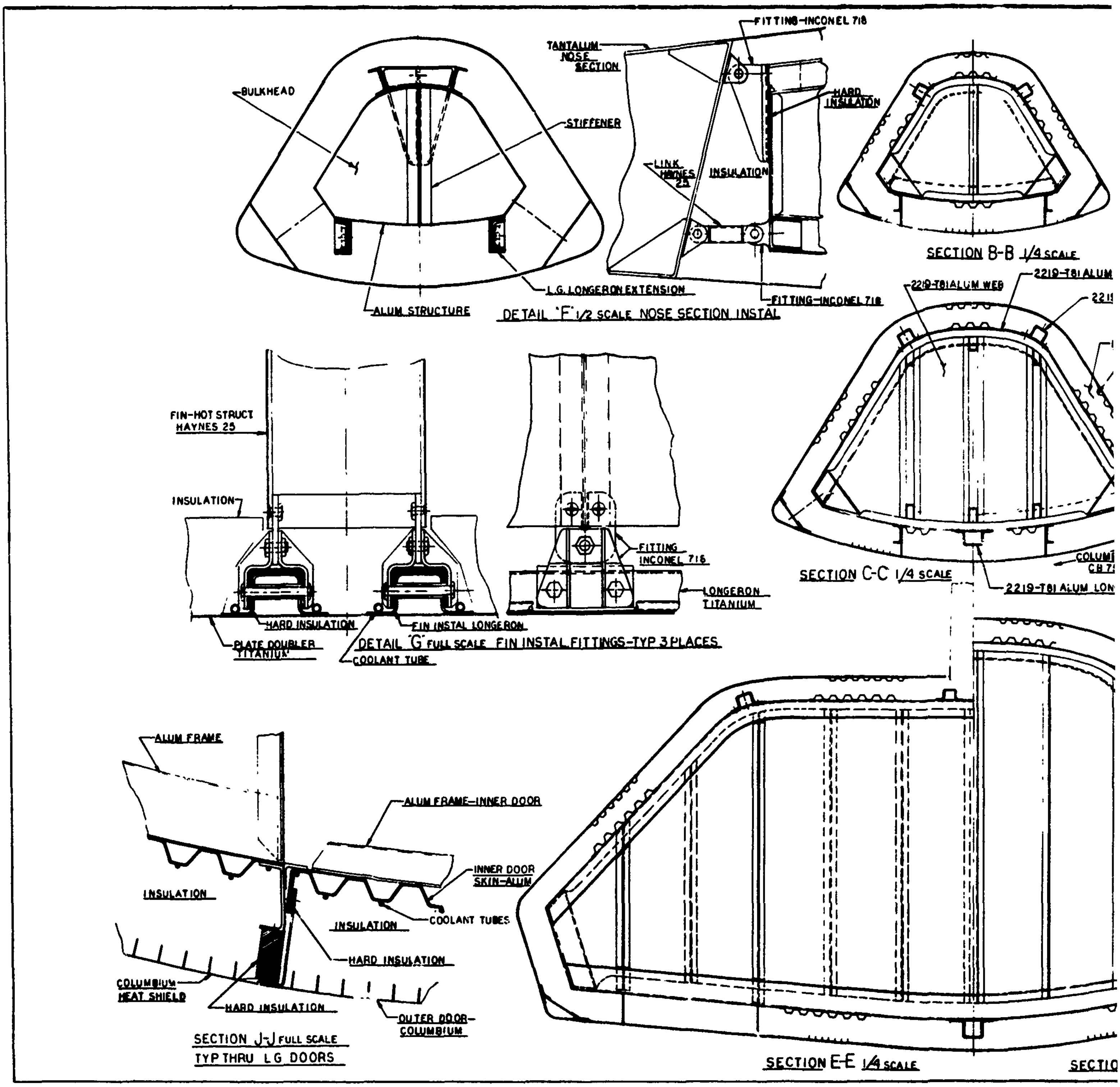
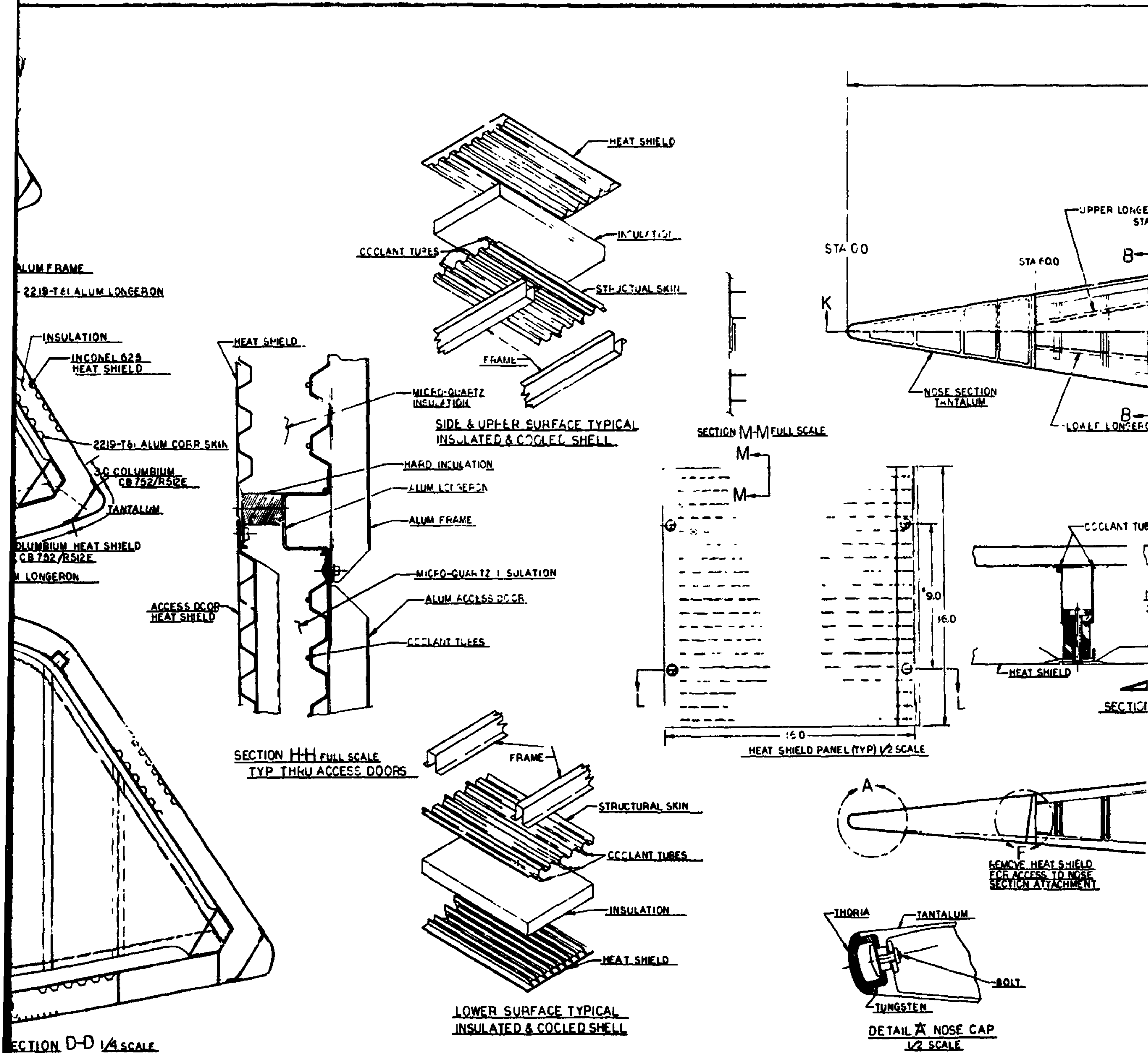


FIGURE 44 (U) THERMOSTRUCTURAL CONCEPT SELECTION EFFECT ON VEHICLE GROSS WEIGHT

(REVERSE SIDE IS BLANK)

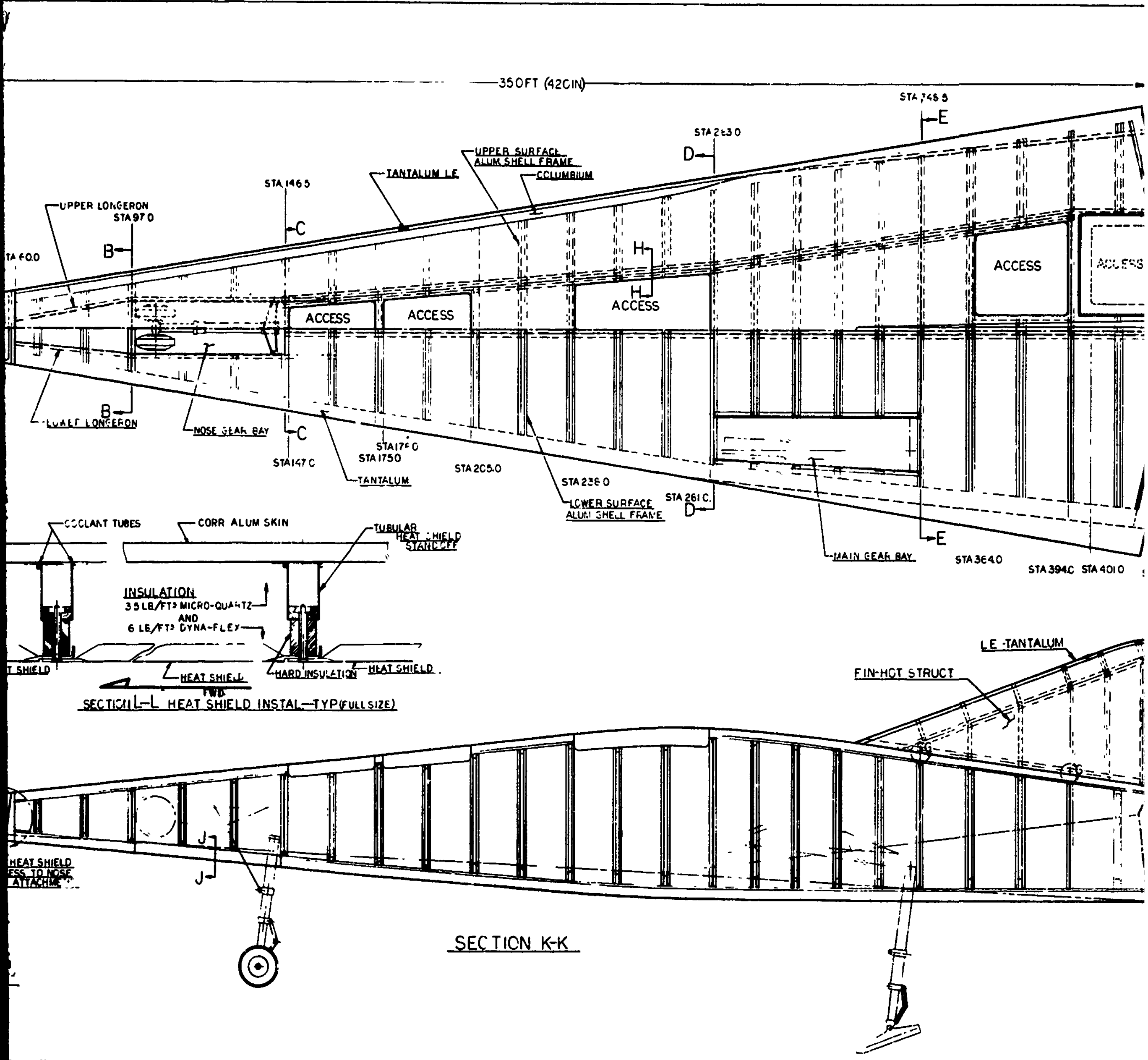

FITTING-INCONEL 718
TANTALUM NOSE SECTION
BULKHEAD
STIFFENER
HARD INSULATION
LINK HAYNES 25
INSULATION
L.G. LONGERON EXTENSION
FITTING-INCONEL 718
ALUM STRUCTURE
DETAIL 'F' 1/2 SCALE NOSE SECTION INSTAL
SECTION B-B 1/4 SCALE
2219-T81 ALUM WEB
2219-T81 ALUM
FIN-HOT STRUCT HAYNES 25
INSULATION
FITTING INCONEL 718
LONGERON TITANIUM
HARD INSULATION
FIN INSTAL LONGERON
PLATE DOUBLER TITANIUM
DETAIL 'G' FULL SCALE FIN INSTAL FITTINGS-TYP 3 PLACES
COOLANT TUBE
SECTION C-C 1/4 SCALE
2219-T81 ALUM LON
ALUM FRAME
ALUM FRAME-INNER DOOR
INNER DOOR SKIN-ALUM
INSULATION
INSULATION
COOLANT TUBES
HARD INSULATION
COLUMBIUM HEAT SHIELD
HARD INSULATION
OUTER DOOR-COLUMBIUM
SECTION J-J FULL SCALE
TYP THRU L G DOORS
SECTION E-E 1/4 SCALE
SECTIO



ALUM FRAME
2219-T81 ALUM LONGERON
INSULATION
INCONEL 625
HEAT SHIELD
2219-T81 ALUM CORR SKIN
CB 752/R512E
TANTALUM
COLUMBIUM HEAT SHIELD
CB 752/R512E
LONGERON
ACCESS DOOR
HEAT SHIELD
SECTION D-D 1/4 SCALE
HEAT SHIELD
MICRO-QUARTZ
INSULATION
HARD INSULATION
ALUM FRAME
MICRO-QUARTZ INSULATION
COOLANT TUBES
SECTION H-H FULL SCALE
TYP THRU ACCESS DOORS
HEAT SHIELD
COOLANT TUBES
FRAME
SIDE & UPPER SURFACE TYPICAL
INSULATED & COOLED SHELL
FRAME
STRUCTURAL SKIN
COOLANT TUBES
INSULATION
HEAT SHIELD
LOWER SURFACE TYPICAL
INSULATED & COOLED SHELL
SECTION M-M FULL SCALE
M
M
L
L
9.0
16.0
16.0
HEAT SHIELD PANEL (TYP) 1/2 SCALE
STA 0.0
STA 0.0
K
B
B
UPPER LONGER
NOSE SECTION
TANTALUM
COOLANT TUB
HEAT SHIELD
SECTIO
A
F
REMOVE HEAT SHIELD
FOR ACCESS TO NOSE
SECTION ATTACHMENT
THORIA
TANTALUM
BOLT
TUNGSTEN
DETAIL A NOSE CAP
1/2 SCALE

2

PRECEDING PAGE BLANK--NOT FILMED



3

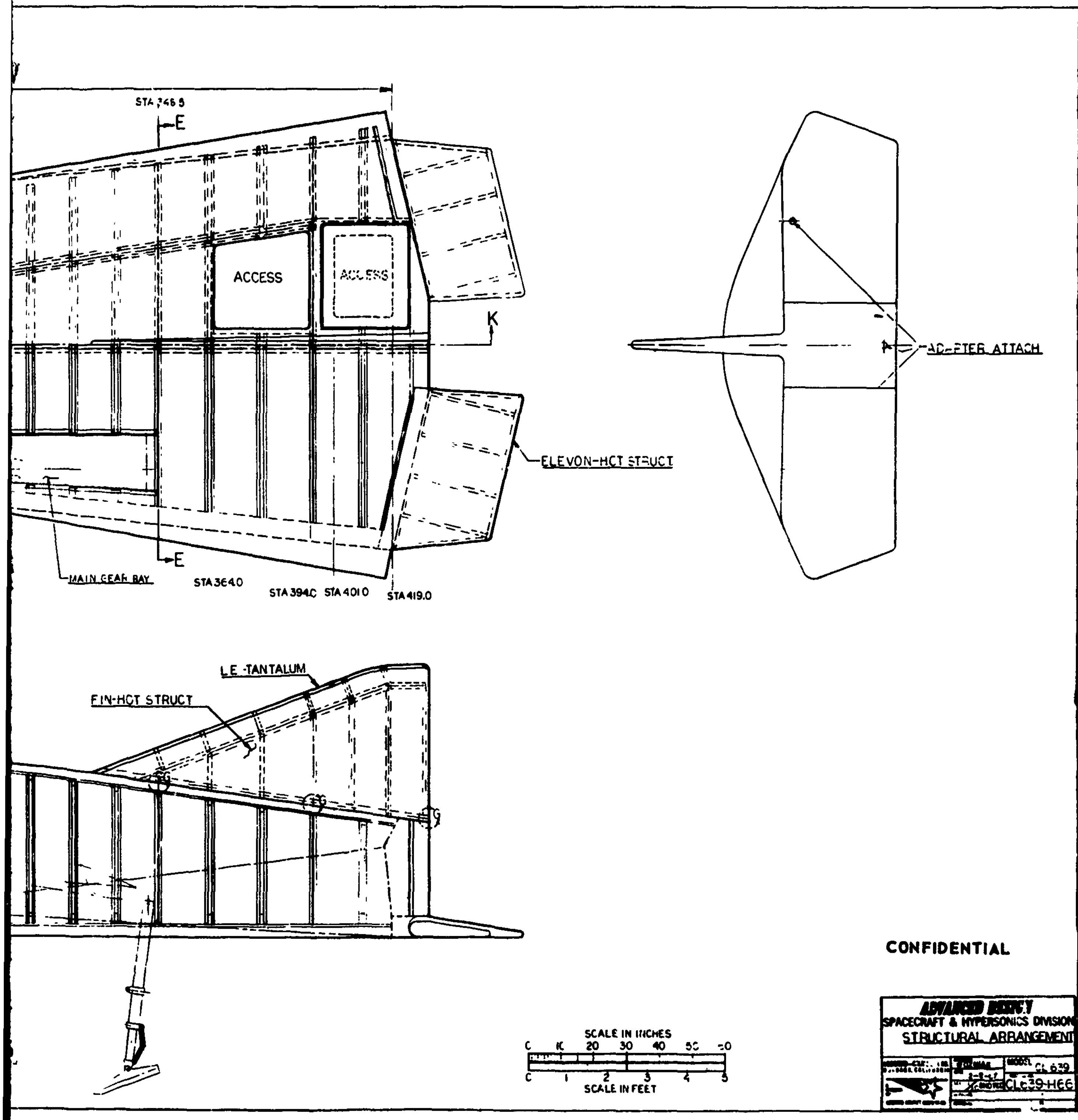
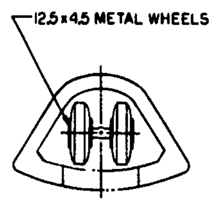
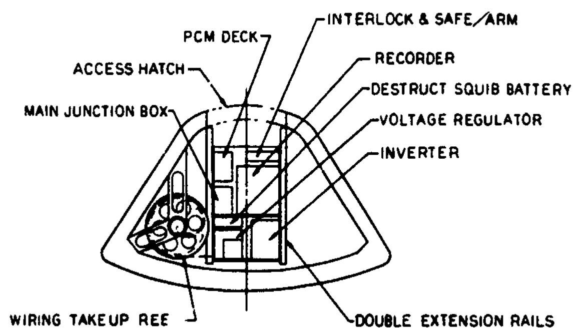


FIGURE 45 (C) FLIGHT TEST VEHICLE STRUCTURAL ARRANGEMENT

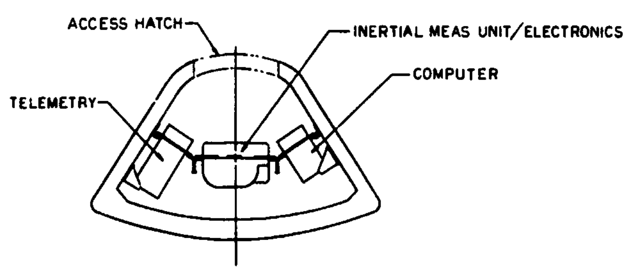
(REVERSE SIDE IS BLANK)



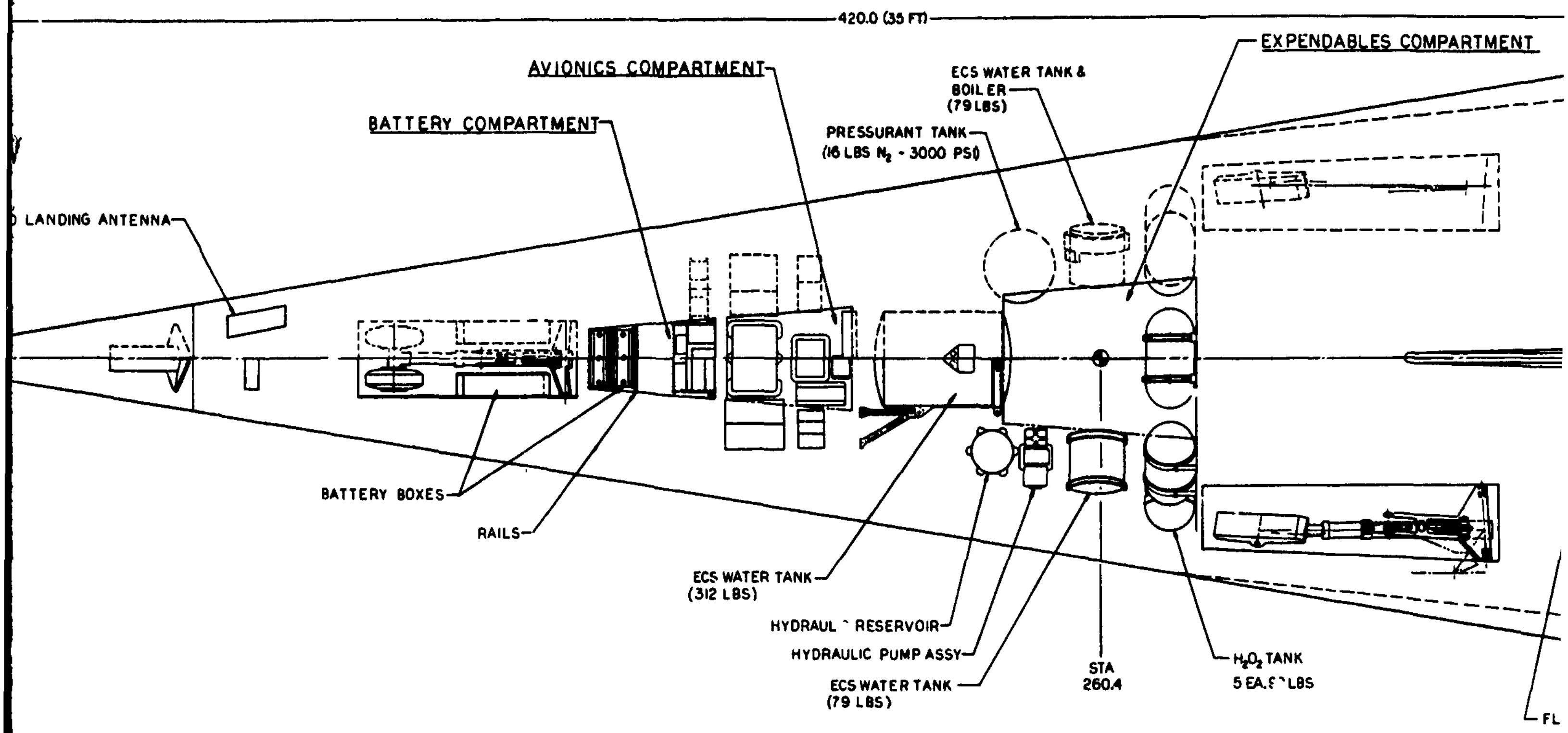
SECTION A-A
STA 105



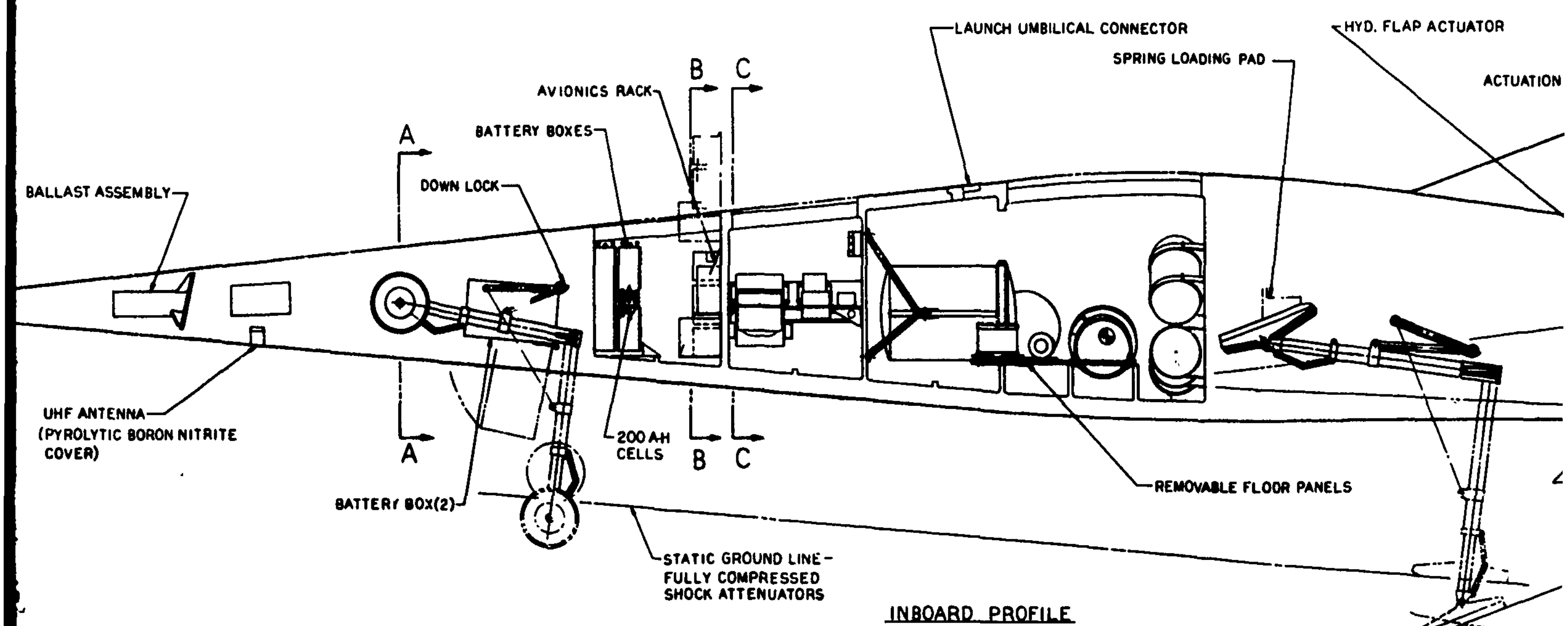
SECTION B-B
STA 176
AVIONICS RACK-SLIDE
OUT FOR ACCESS



SECTION C-C
STA 180
PANEL MOUNTED AVIONICS
EQUIPMENT



PLAN VIEW



INBOARD PROFILE

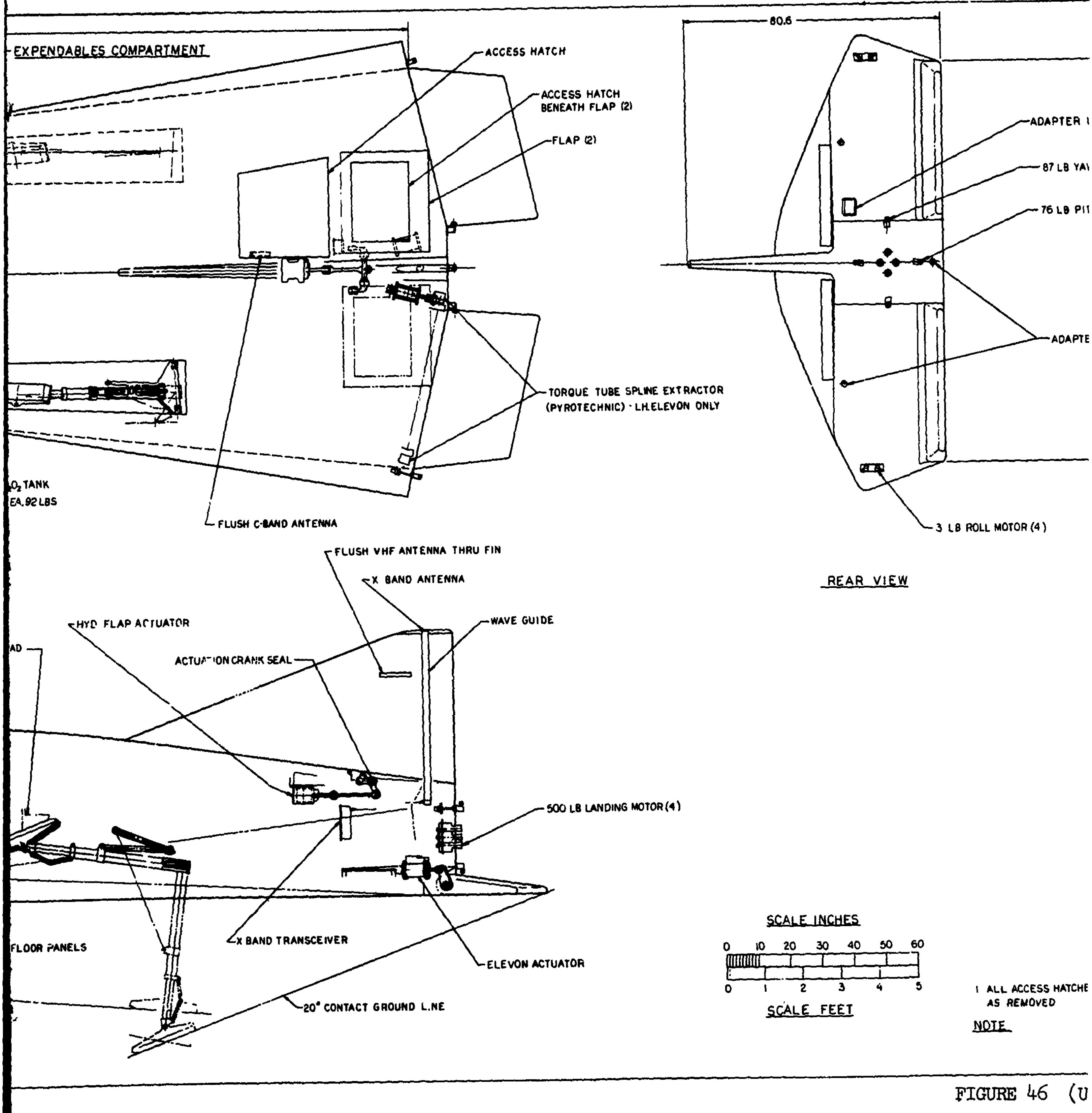


FIGURE 46 (U

(REVERSE SIDE

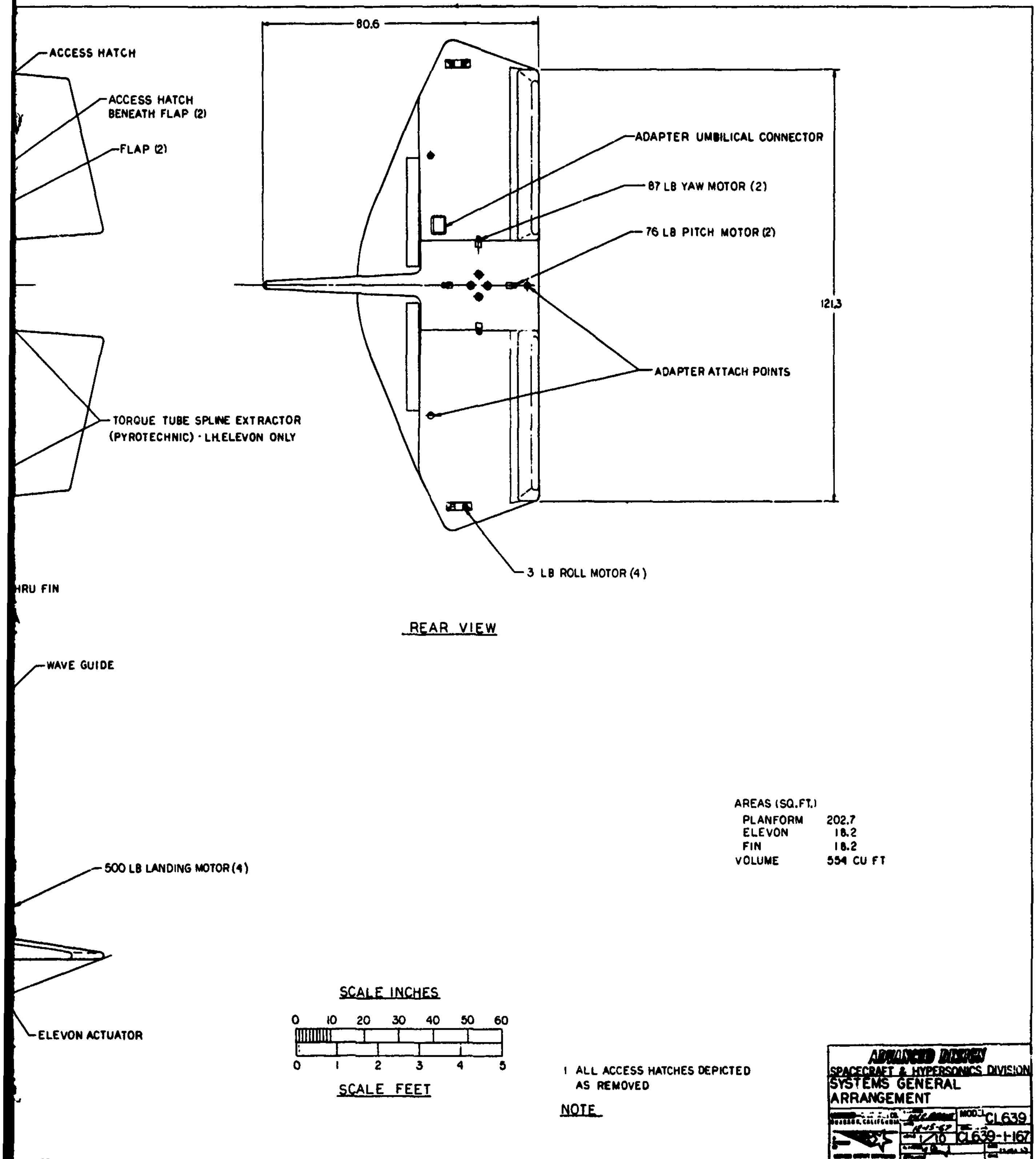


FIGURE 46 (U) SYSTEMS GENERAL ARRANGEMENT

(REVERSE SIDE IS BLANK)

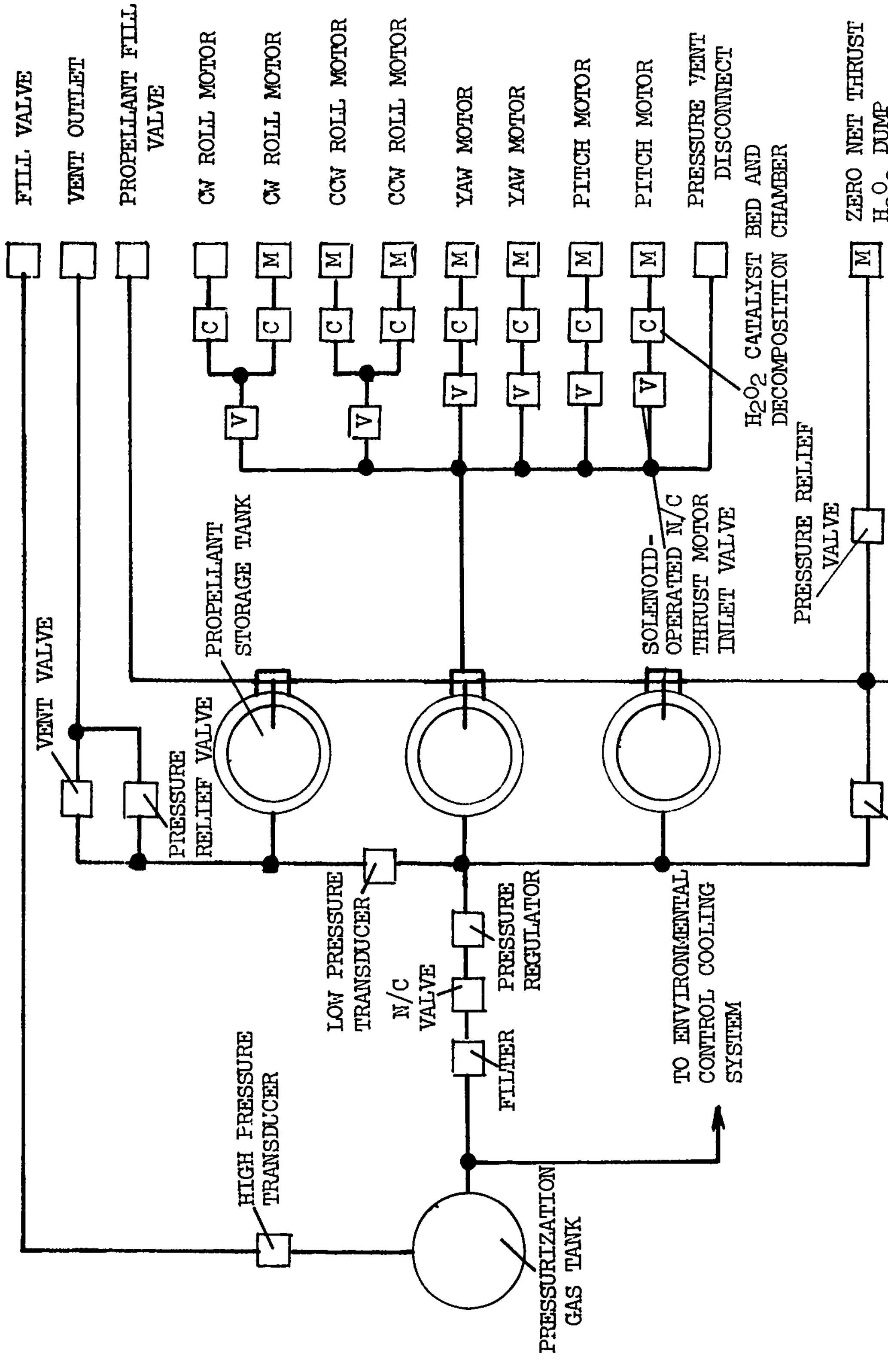


FIGURE 47 (U) REACTION CONTROL SYSTEM

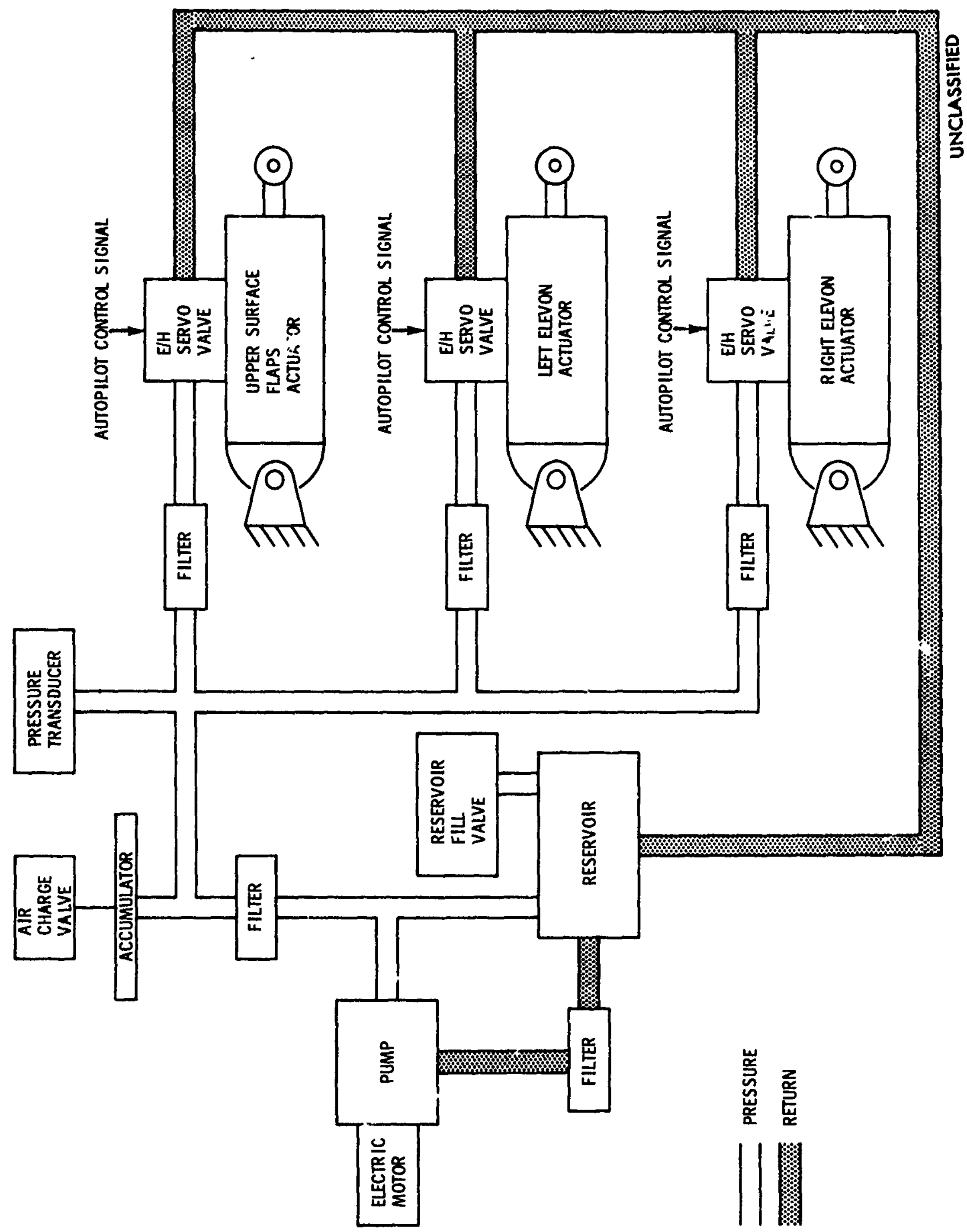
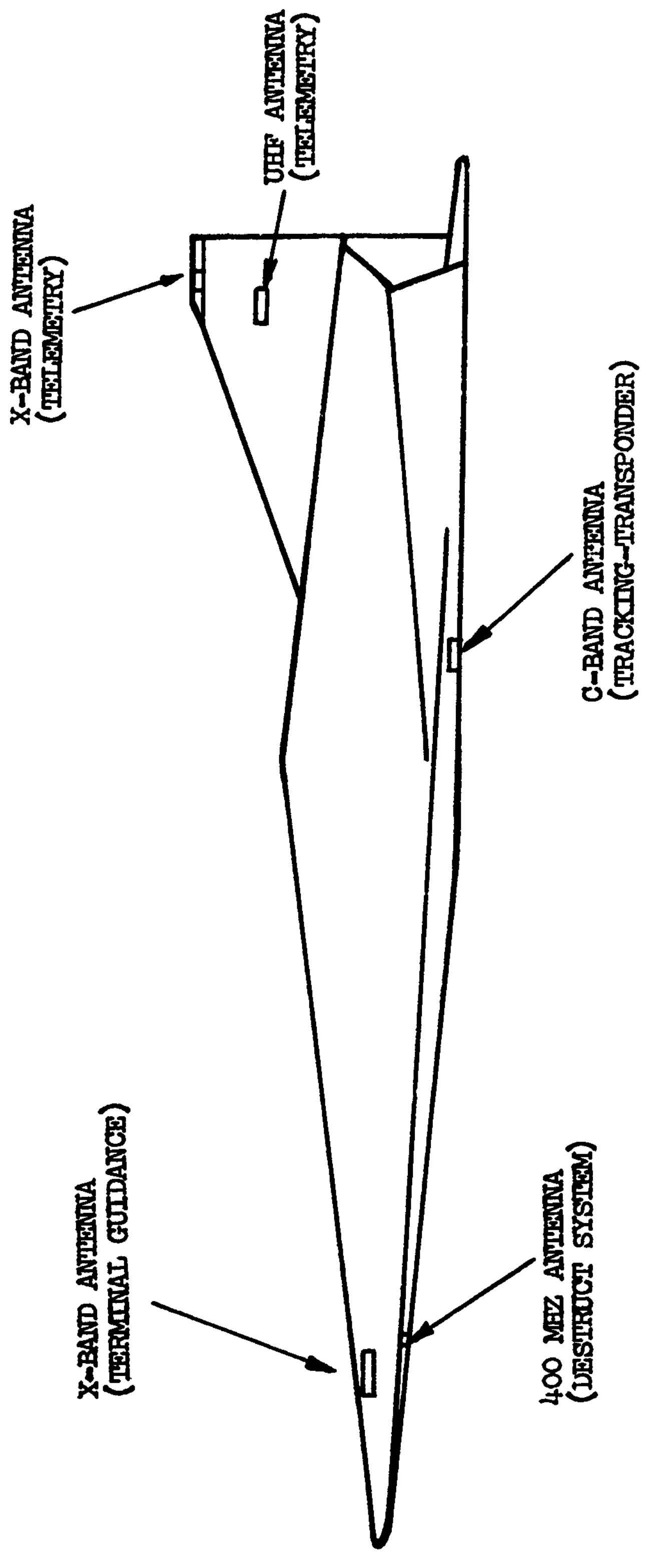
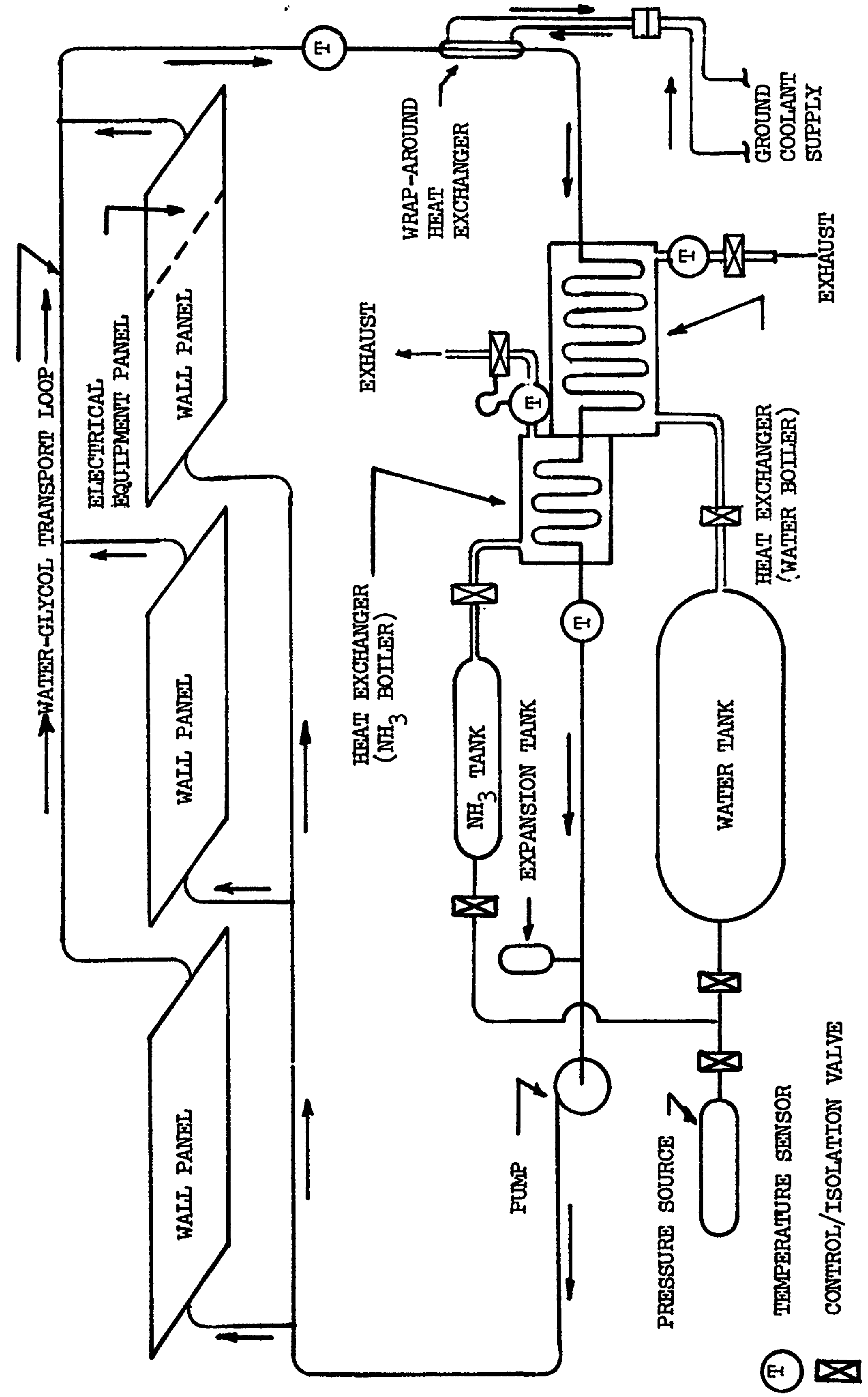


FIGURE 48 (U) HYDRAULIC CONTROL SYSTEM



UNCLASSIFIED

FIGURE 49 (U) INSTALLATION OF ANTENNAS



UNCLASSIFIED

FIGURE 50 (U) ENVIRONMENTAL CONTROL SYSTEM

UNCLASSIFIED

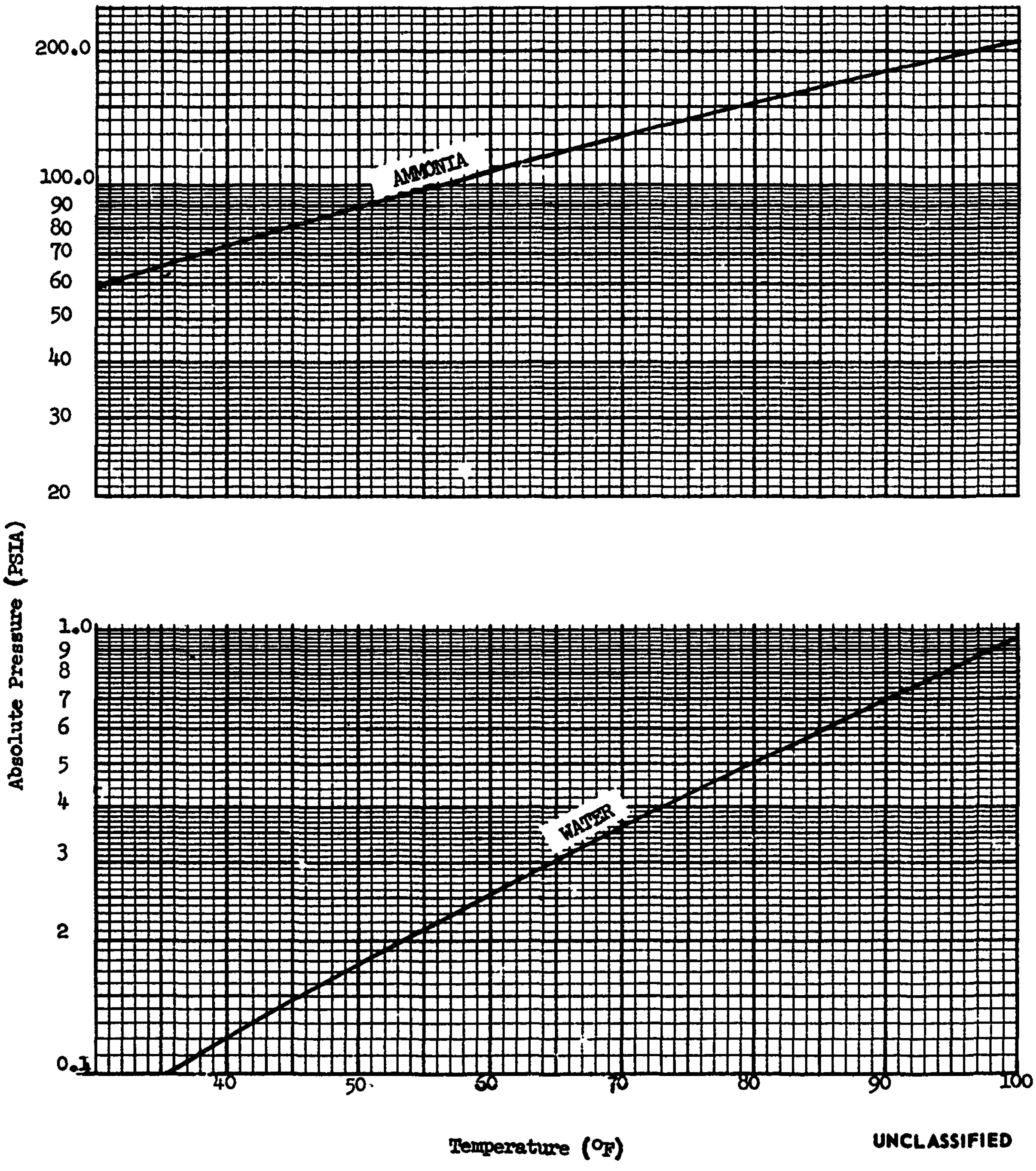


FIGURE 51 (U) VAPOR PRESSURE OF SATURATED LIQUIDS

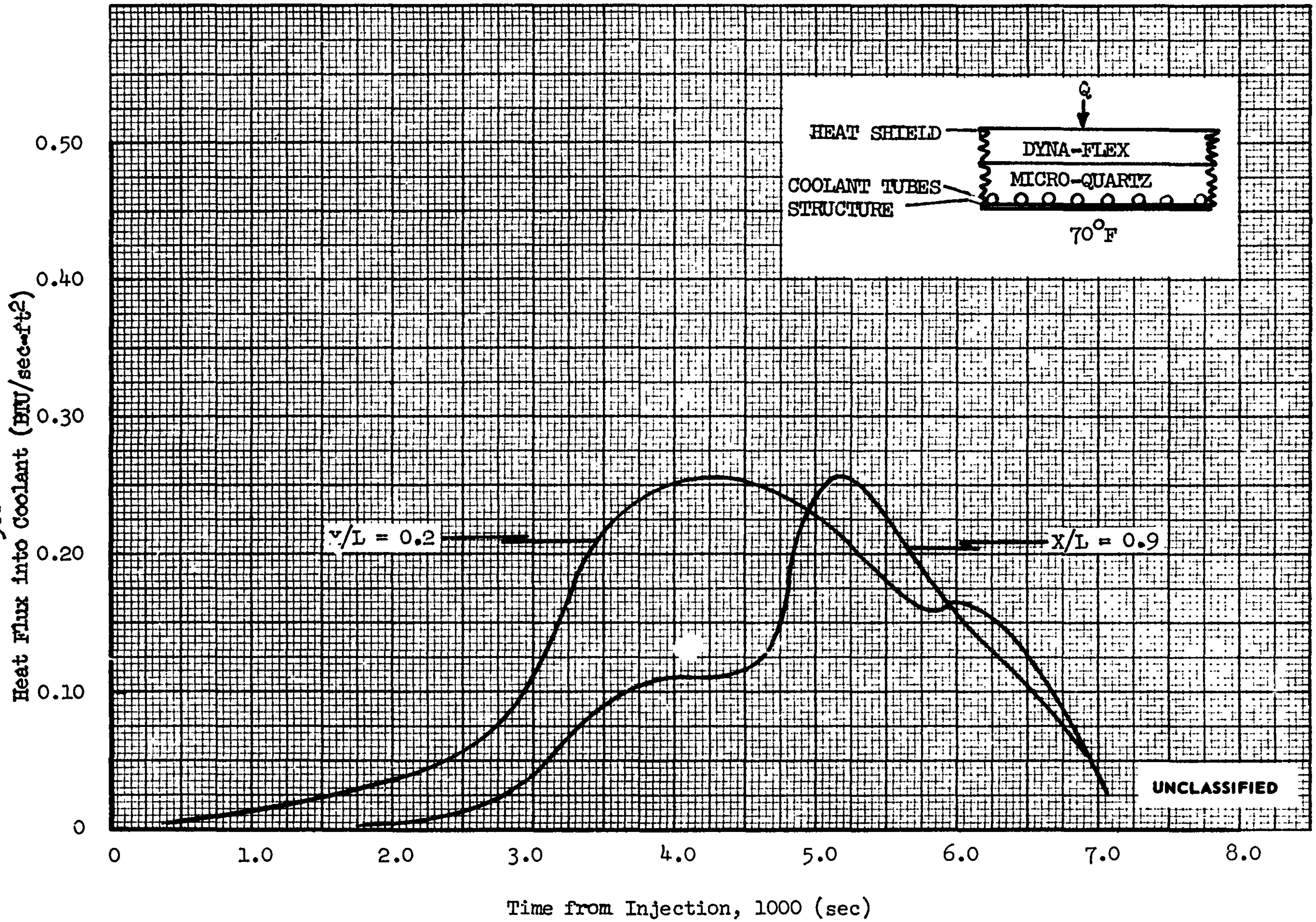


FIGURE 52 (U) ACTIVE COOLANT HEAT FLUX PROFILE, TYPICAL FOR LOWER SURFACE

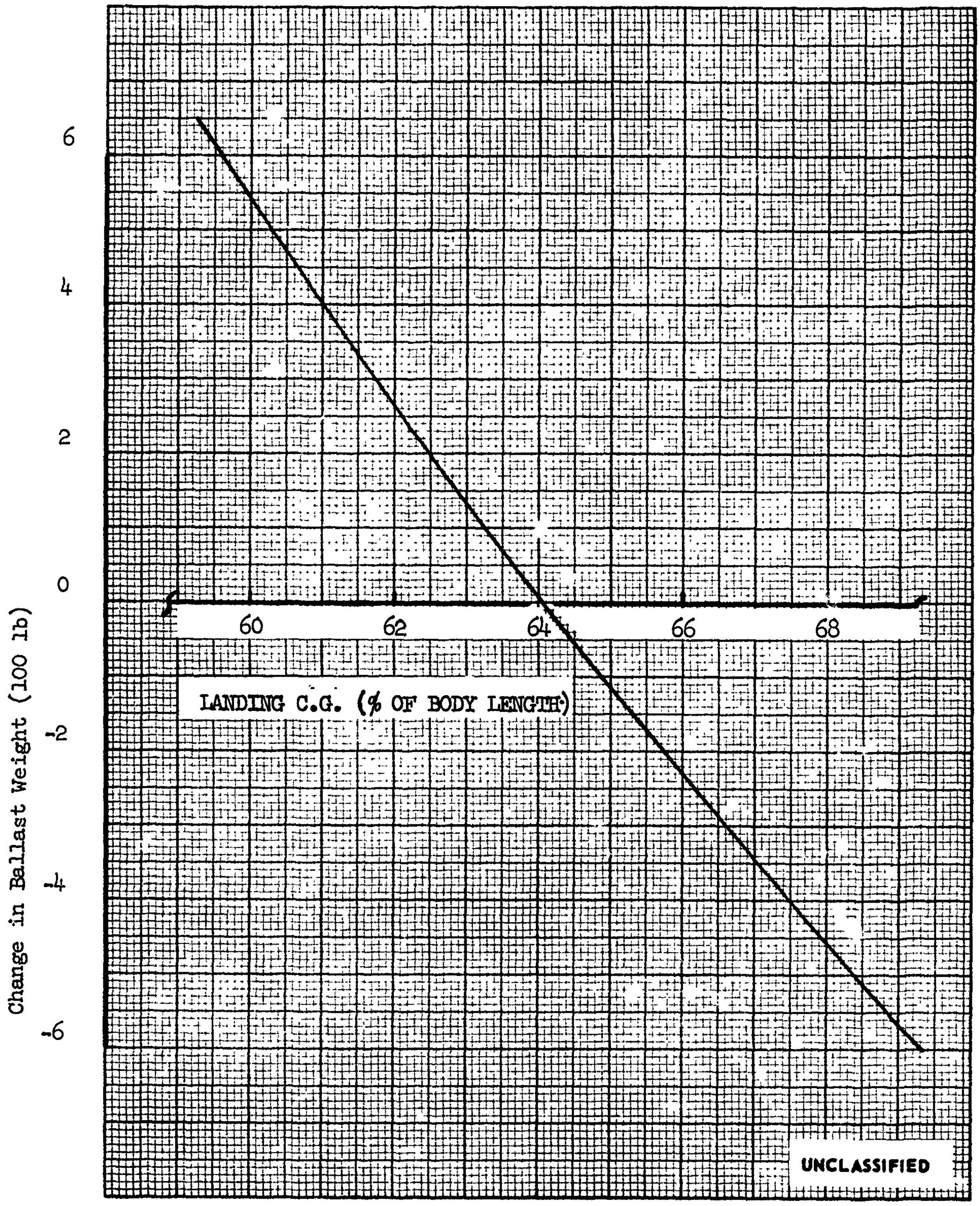


FIGURE 53 (U) BALLAST EFFECT

(REVERSE SIDE IS BLANK)

UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY *(Corporate author)* Lockheed Aircraft Corporation Burbank, California	2a. REPORT SECURITY CLASSIFICATION CONFIDENTIAL 2b GROUP 4

3 REPORT TITLE

PRELIMINARY DESIGN AND EXPERIMENTAL INVESTIGATION OF THE FDL-5A UNMANNED HIGH L/D SPACECRAFT - PART V - VEHICLE DESIGN (TITLE UNCLASSIFIED)

4. DESCRIPTIVE NOTES *(Type of report and inclusive dates)*

Final Report (1 July 1966 through 31 March 1968)

5. AUTHOR(S) *(Last name, first name, initial)*

Plank, P. P., Sakata, I. F., and Verhaegh, M.

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
March 1968	112	5

8a. CONTRACT OR GRANT NO. Contract No. AF 33(615)-5241 b. PROJECT NO. 1366 c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) LAC/619519 LR 21204 (PART V) 9b. OTHER REPORT NO(S) *(Any other numbers that may be assigned this report)* AFFDR-TR-68-24 Part V

10. AVAILABILITY/LIMITATION NOTICES

This report is subject to special export controls and each transmittal to foreign governments or foreign nationals may be made only with prior approval of the Air Force Flight Dynamics Laboratory (FDMS), Wright-Patterson Air Force Base, Ohio 45433.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Air Force Flight Dynamics Laboratory Air Force Systems Command Wright-Patterson Air Force Base, Ohio 45433

13. ABSTRACT

(U) A parametric analysis of a broad spectrum of thermostructural concepts is presented. A complete structural concept for the 35-foot entry test vehicle using the FDL-5 configuration is defined. Subsystems for the entry test vehicle are selected. A weight breakdown and a weight summary for the test vehicle are presented.

DD FORM 1 JAN 64 1473

UNCLASSIFIED

Security Classification

UNCLASSIFIED

Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
* High L/D Entry Vehicles						
* Thermostructural Concepts						
* FDL-5 Configurations						

INSTRUCTIONS

1. ORIGINATING ACTIVITY: Enter the name and address of the contractor, subcontractor, grantee, Department of Defense activity or other organization *(corporate author)* issuing the report.

2a. REPORT SECURITY CLASSIFICATION: Enter the overall security classification of the report. Indicate whether "Restricted Data" is included. Marking is to be in accordance with appropriate security regulations.

2b. GROUP: Automatic downgrading is specified in DoD Directive 5200.10 and Armed Forces Industrial Manual. Enter the group number. Also, when applicable, show that optional markings have been used for Group 3 and Group 4 as authorized.

3. REPORT TITLE: Enter the complete report title in all capital letters. Titles in all cases should be unclassified. If a meaningful title cannot be selected without classification, show title classification in all capitals in parenthesis immediately following the title.

4. DESCRIPTIVE NOTES: If appropriate, enter the type of report, e.g., interim, progress, summary, annual, or final. Give the inclusive dates when a specific reporting period is covered.

5. AUTHOR(S): Enter the name(s) of author(s) as shown on or in the report. Enter last name, first name, middle initial. If military, show rank and branch of service. The name of the principal author is an absolute minimum requirement.

6. REPORT DATE: Enter the date of the report as day, month, year; or month, year. If more than one date appears on the report, use date of publication.

7a. TOTAL NUMBER OF PAGES: The total page count should follow normal pagination procedures, i.e., enter the number of pages containing information.

7b. NUMBER OF REFERENCES: Enter the total number of references cited in the report.

8a. CONTRACT OR GRANT NUMBER: If appropriate, enter the applicable number of the contract or grant under which the report was written.

8b, 8c, & 8d. PROJECT NUMBER: Enter the appropriate military department identification, such as project number, subproject number, system numbers, task number, etc.

9a. ORIGINATOR'S REPORT NUMBER(S): Enter the official report number by which the document will be identified and controlled by the originating activity. This number must be unique to this report.

9b. OTHER REPORT NUMBER(S): If the report has been assigned any other report numbers *(either by the originator or by the sponsor)*, also enter this number(s).

10. AVAILABILITY/LIMITATION NOTICES: Enter any limitations on further dissemination of the report, other than those imposed by security classification, using standard statements such as:

(1) "Qualified requesters may obtain copies of this report from DDC."

(2) "Foreign announcement and dissemination of this report by DDC is not authorized."

(3) "U. S. Government agencies may obtain copies of this report directly from DDC. Other qualified DDC users shall request through ________________."

(4) "U. S. military agencies may obtain copies of this report directly from DDC. Other qualified users shall request through ________________."

(5) "All distribution of this report is controlled. Qualified DDC users shall request through ________________."

If the report has been furnished to the Office of Technical Services, Department of Commerce, for sale to the public, indicate this fact and enter the price, if known.

11. SUPPLEMENTARY NOTES: Use for additional explanatory notes.

12. SPONSORING MILITARY ACTIVITY: Enter the name of the departmental project office or laboratory sponsoring *(paying for)* the research and development. Include address.

13. ABSTRACT: Enter an abstract giving a brief and factual summary of the document indicative of the report, even though it may also appear elsewhere in the body of the technical report. If additional space is required, a continuation sheet shall be attached.

It is highly desirable that the abstract of classified reports be unclassified. Each paragraph of the abstract shall end with an indication of the military security classification of the information in the paragraph, represented as *(TS), (S), (C), or (U).*

There is no limitation on the length of the abstract. However, the suggested length is from 150 to 225 words.

14. KEY WORDS: Key words are technically meaningful terms or short phrases that characterize a report and may be used as index entries for cataloging the report. Key words must be selected so that no security classification is required. Identifiers, such as equipment model designation, trade name, military project code name, geographic location, may be used as key words but will be followed by an indication of technical context. The assignment of links, roles, and weights is optional.

UNCLASSIFIED

Security Classification