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MCR-78-582

STUDY OF ADVANCED ATMOSPHERIC ENTRY SYSTEMS FOR MARS

JULY 1978

Approved

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Prepared Under Contract No. 955075
by
Martin Marietta Corporation
for the
Jet Propulsion Laboratory

FOREWORD

This final report has been prepared in accordance with requirements of JPL Contract No. 955075 to present the results of a brief study for the Jet Propulsion Laboratory by the Martin Marietta Corporation, Denver Division.

This report contains information prepared by Martin Marietta Denver Division under JPL subcontract. Its content is not necessarily endorsed by the Jet Propulsion Laboratory, California Institute of Technology, or the National Aeronautics and Space Administration.

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SUMMARY

The purpose of this study was to estimate the mass and performance characteristics of entry system designs required in support of 1) a Mars sample return mission, 2) a simple hard lander, and 3) a Mars airplane mission. Viking technology or logical extensions of that technology were used as a basis for these entry system designs thus providing a low risk technology concept and realistic estimates of mass relationships and performance capability.

For the Mars sample return mission, various mission combinations were evaluated including different Mars Ascent Vehicle (MAV) masses and the use of 2 through 4 stages of the IUS launch vehicle. The primary constraint in this task was the shuttle payload bay geometry limitation (4.27 m diameter limit) in conjunction with the very large and massive MAV requirements. These constraints tend to result in entry vehicles with excessively high ballistic coefficients and correspondingly high descent velocities near the surface. However, within these constraints the following conclusions resulted:

- An entry/landing system can be developed to deliver a landed-science/earth return system of the order of 6,000 - 7,000 kg.
- Such a system is based on Viking technology but will require further engineering development in areas of:
  - Extendible aeroshell segments
  - Larger, higher dynamic pressure parachute systems
- Target elevation capability appears limited to 3 km or less -- site alteration may be a problem due to the massive terminal propulsion system effects in the landing area.

For the simple hard lander mission, the vehicle was required to enter at 6.0 km/s from direct approach and decelerate down to 20 m/s at surface impact. A main parachute, deployed at Mach 2 and an altitude of 10 km, rapidly slows the vehicle to about 60 m/s at 7.5 km and at that point a solid rocket, ignited by a signal from a proximity radar, decelerates the lander to the required 20 m/s at surface impact. The following conclusions resulted for the hard lander entry system:

- Direct entry system mass of about 65 kg is required to support a 50 kg lander.
- Either solid rocket or parachute can be used to limit the terminal descent and impact below 20 m/s.
- The solid rocket is the recommended terminal descent decelerator approach.
- The solid rocket decelerator approach requires a proximity radar altimeter for ignition timing.

For the Mars airplane mission, a design goal was to minimize the total entry capusle/airplane mass by combining entry functions and hardware into the airplane system wherever practical. Also various approaches were examined for supporting the folded airplane within the aeroshell in an efficient manner. The following conclusions were developed for the Mars airplane entry system:

- A 200 kg from orbit entry system will support a 300 kg airplane entry and provide proper airplane deployment conditions.
- A base cover truss supported airplane concept is preferred.
- The shuttle payload envelope is compatible with launch of four to seven airplane entry systems depending on support spacecraft size requirements.
- Entry systems command and control functions can be incorporated into the airplane.
- The entry system must contain as a minimum the radar altimiter, RCS subsystem, valve drive circuitry, and supplemental battery power functions, other entry system functions can be performed by airplane systems.

I. INTRODUCTION

This report describes entry system designs for various advanced Mars missions including -- 1) sample return, 2) hard lander, and 3) Mars airplane. The Mars exploration systems for sample return and the hard lander defined in this study require deceleration from direct approach entry velocities of about 6 km/s to terminal velocities consistent with surface landing requirements. The Mars airplane entry system is decelerated from orbit at 4.6 km/s to deployment near the surface.

This study was performed by Martin Marietta Corporation in support of the Jet Propulsion Laboratory activities associated with the definition of future Mars missions. The data generated herein will be used by JPL in their selection of the more promising approaches for the future exploration of Mars.

The purpose of the study was to estimate the mass and performance characteristics of major elements of the required entry systems using Viking technology or logical extensions of that technology in order to provide a common basis of comparison for the three mission mode approaches. The entry systems, although not optimized, are based on Viking designs and reflect current hardware performance capability and realistic mass relationships.

Each of the three different Mars exploration concepts is defined and evaluated in a separate chapter as an independent task in the following order:

Chapter II - Sample Return Vehicle Entry System;
Chapter III - Hard Lander Entry System;
Chapter IV - Mars Airplane Entry System.

II. TASK 1 - SAMPLE RETURN VEHICLE ENTRY SYSTEM

A. TECHNICAL CONSIDERATIONS

The Mars exploration systems specified in this study require deceleration from entry velocities of about 6 km/s to terminal velocities consistent with surface landing requirements. The sample return vehicle concept uses a direct Mars entry and direct return to earth mission mode and, therefore, the entry systems must be designed for direct entry velocities of 6 km/s as compared to the Viking entry from orbital velocities of about 4.6 km/s. The entry systems have been designed using Viking technology modified as necessary to enhance landed mass performance.

Various entry system concepts were designed to accommodate a range of possible payloads and these cases are defined under Section II.C.2. The IUS vehicle requirements range from two to four stages. The Shuttle payload bay geometry constrains the maximum stowed diameter to 4.27 m (14 ft) for all configurations. The diameter constraint turned out to be a severe limitation on packaging and affected ballistic coefficient design capabilities. With increasing entry mass and ballistic coefficient it becomes more difficult to slow the entry vehicle down sufficiently for safe main parachute deployment. Alternate aerodynamic shapes, extendible aeroshell flaps and ballutes were considered during the study as potential means of reducing ballistic coefficient and, therefore, velocity reduction prior to main parachute deployment, aeroshell staging, and terminal propulsion ignition.

B. TASK 1 OBJECTIVE

The objective of this task was to establish concepts for entering and landing a direct-return sample return vehicle using the Shuttle/IUS launch system. The delivered mass was varied as a function of the number of the number of IUS stages considered which ranged from two to four. Concepts evaluated included constant lift/drag entries (similar to Viking) as well as constant lift/drag entries with the lift vector modulated by roll maneuvers in order to minimize entry loads and parachute deployment velocities.

Aeroshell designs included conventional (fixed-diameter) as well as deployable designs that extend to diameters larger than the 14 ft allowed by the Shuttle bay.

C. ANALYSIS

1. Guidelines and Ground Rules

The guidelines and ground rules for the design of the Mars Sample Return entry system are as follows.

1) Direct entry from approach.
2) Both constant L/D and roll modulated lift at constant L/D entry modes to be considered.
3) Initial analysis (Cases 1 through 4) constrained to entry mass compatible with payload of twin-stage IUS (5,500 kg).
4) Subsequent analyses based on determining entry mass required by JPL direct-return Mars Ascent Vehicle (MAV). Mass of MAV 6,000 kg including 400 Kg margin.
5) Determine available landed mass using three- and four-stage IUS payload capability.
6) Apply Viking technology or extensions as applicable.
7) Launch payload capability based on JPL handout (Nagorski data of Reference 1).
8) Entry and descent performance based on the Viking reconstructed atmosphere model.
9) Assume terminal descent propulsion similar to the Viking liquid descent propulsion system.
10) Assume wind profiles equal to 1/2 of Viking design winds (which proved to be too conservative). Direction of wind assumed to be in adverse direction:
    a) Head wind at entry (increases loads and heating);

b) Tail wind at descent (increases total velocity requirement on terminal propulsion system).

2. Case Definitions

The case definitions for the Mars Sample Return entry system task are shown in Table II-1. The cases were evolved during the study as new data were developed both here and at JPL.

Cases 1, 2, and 3 result from a preliminary analysis of the use of Viking technology for both the 11.5 ft "as built" aeroshell and for increased entry masses and scaled up aeroshell diameters. Cases 1 and 2 were designed to the nominal entry environment whereas Case 3, at a greater mass and ballistic coefficient than Case 2, has a beefed-up aeroshell to withstand the greater load and heating.

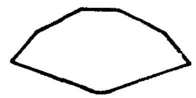
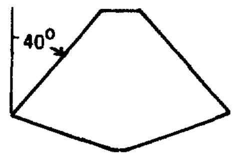
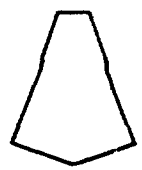
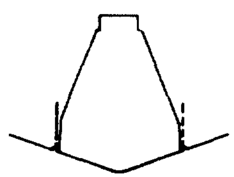
Case 4 has the 4.72 m (14 ft) diameter that is the maximum allowable within the Shuttle payload bay dynamic envelope. This case assumed a twin stage IUS launch vehicle which resulted in a 5,343 kg mass available at Mars entry. The mass and volume available for a MAV were calculated assuming a scaled-up Viking configuration for both the aeroshell and the 40° return angle afterbody. Both constant L/D and roll modulated lift, also with constant L/D, were considered in this case. The roll orients the lift vector as required by the guidance laws to control altitude, however the vehicle is fixed in trim at a constant L/D ratio.

In Case 5 the entry mass required to support 6,000 kg MAV mass was estimated assuming the maximum allowable diameter of 4.27 m (14 ft) and a three-stage IUS launch vehicle. These large masses and volumes resulted in high ballistic coefficients and required the addition of a ballute to slow the vehicle down sufficiently to deploy the main chute.

Case 6 was a proposed alternate solution to the problems encountered in Case 5. Extendible flaps were incorporated around the circumference of the aeroshell. When stowed, the flaps were within the 4.27 m (14 ft) diameter Shuttle payload limits yet, when extended, the flaps provided sufficient drag area to reduce the entry ballistic coefficient, minimize entry weight, and provide an acceptable altitude above the surface for chute deployment at M = 2.2.

Cases 7 and 8 are scaled up versions of Case 6 designed to evaluate landed weight available using a four-stage IUS launch vehicle for a 1990 and a 1988 launch opportunity.

TABLE II-1 CASE DEFINITION

CASE NO.	NO. OF IUS STAGES	AEROSHELL DIAMETER m (ft)	CONFIGURATION	MASS AVAILABLE FOR ENTRY (Kg)	MASS AVAILABLE FOR MAV (Kg)	CASE DEFINITION
1	2	3.5 (11.5)	Viking "As Built" 	1,148	N/A	1. Viking "as built" system 2. 14 ft diameter version of Viking with same entry environment 3. 14 ft diameter - higher ballistic coefficient, beefed-up aeroshell Preliminary study of Viking technology--out of orbit entry
2	2	4.27 (14)	Increased Diameter	1,700	N A	
3	2	4.27 (14)	Increased Diameter	3,100	! A	
4	2	4.27 (14)	 Main Chute	5,343	3,568 (See Section IIc6b for derivation)	Estimate mass available for MAV starting with 5,343 Kg mass available for entry. Assume MAV may be packaged in a 40° return angle afterbody. a) Constant L/D entry mode; b) Roll modulated, constant L/D mode.
5	3	4.27 (14)	 Ballute/Chute	9,690 (See Section IIc6c for derivation)	6,000	Estimate entry mass required and entry vehicle configuration based on 4.27 m (14 ft) diameter aeroshell (starting with 6,000 Kg MAV).
6	3	Fixed A/S, 4.27 (14) With Flaps, 7.64 (25)		9,400 (See Section IIc6d for derivation)	6,000	Estimate entry mass required and entry vehicle configuration based on extendible flap aeroshell (starting with 6,000 Kg MAV).
7	4	Fixed A/S 4.27 (14) With Flaps, 8.4 (27.7)	Same as 6	10,750	---	Determine landeo mass available if full four-stage IUS is used. 1990 Launch Year.
8	4	Fixed A/S 4.27 (14) With Flaps, 9.2 (30.1)	Same as 6	12,000	---	Determine landed mass available if full four-stage IUS is used. 1988 Launch Year.

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3. Launch Vehicle Capability

Shuttle/IUS payload performance is shown on Figure II-1, and is from Ref.1. Payload mass in kg is shown as a function of C_3 for twin-stage, three-stage, and four-stage IUS configurations. Payload is defined as total mass injected into trans-Mars trajectory. Cases 7 and 8 are based on maximum payload capability for the four-stage IUS for the 1990 and 1988 launch opportunities. Other cases are not based on specific performance capability points. Mars arrival velocity and entry velocity for the decades of the 1980's and 1990's are shown on Figure II-2. For the pertinent opportunities during those decades, an entry velocity of 6.0 km/sec represents a critical entry velocity condition and was used for the MSR mission.

4. Entry and Descent

Entry analysis was made from direct approach (except for Cases 1, 2, and 3). Entry conditions are defined at 243 km above the planet surface with an entry velocity of 6.0 km/sec. Trajectory simulation is made up of the aeroshell phase, aerodynamic decelerator (chute and ballute) phase, and a terminal phase propulsion system.

A flight path corridor uncertainty of 1.5° was assumed which resulted in an entry corridor from -16.5° to -18.0°. The steep end of the corridor (-18°) produced the critical conditions for both maximum dynamic pressure and critical parachute deployment altitude at 2.2 Mach number when combined with roll angle modulation.

The atmosphere model used was a Viking reconstructed density profile with a surface pressure representing a near minimum of the Mars year pressure cycle. The model is therefore conservative from a landing site elevation standpoint for most of the Mars year.

A maximum lift to drag (L/D) ratio of 0.30 was used. For the configuration considered (a 70° blunted cone) the required angle of attack of $2[illegible]^{\circ}$ put the stagnation point near the cone edge which resulted in high local heating and stability problems under vehicle dynamic conditions. Entry performance and heating rate calculations are therefore optimistic for a 0.30 L/D ratio.

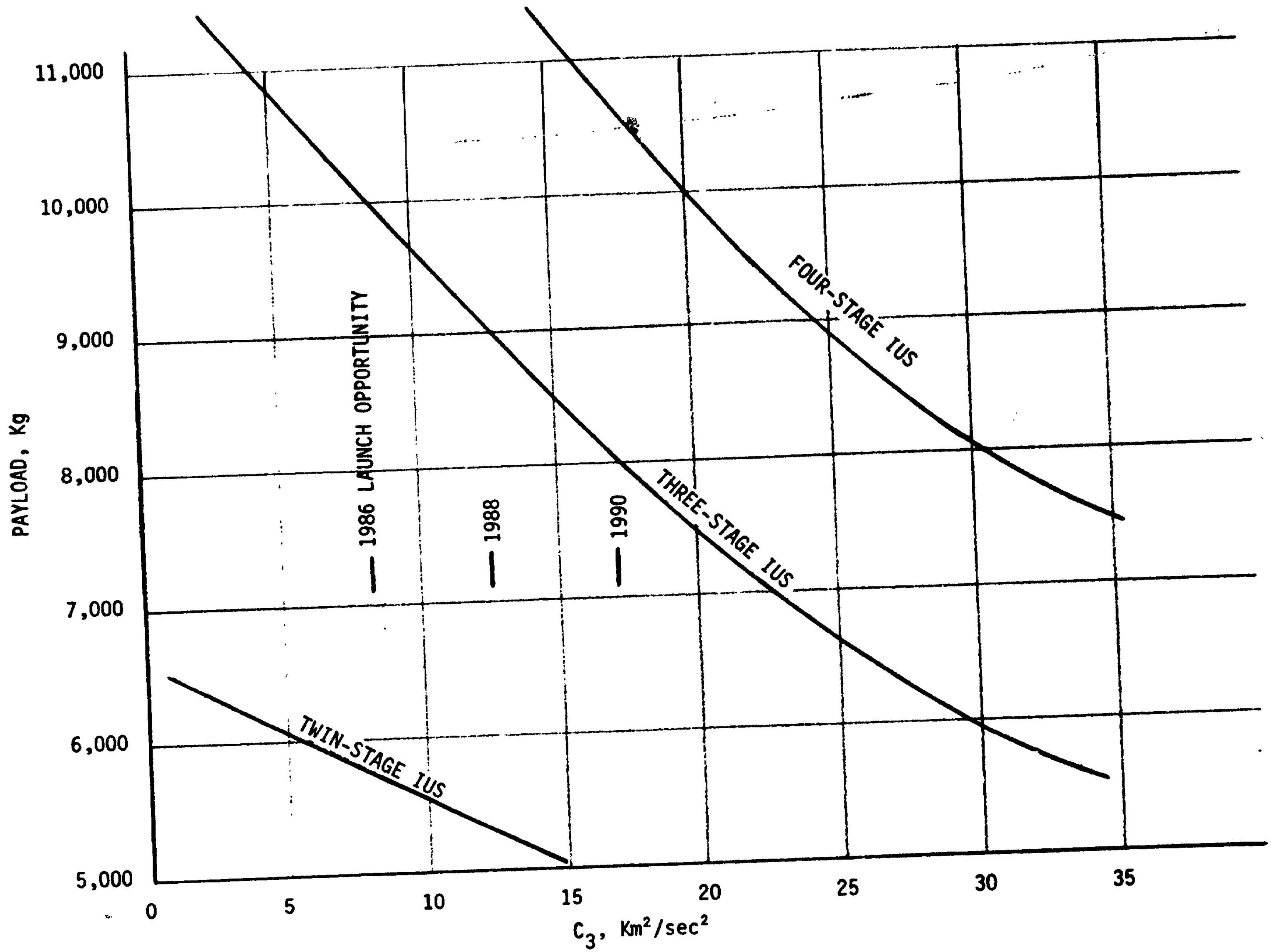


Figure II-1 Shuttle/IUS Performance

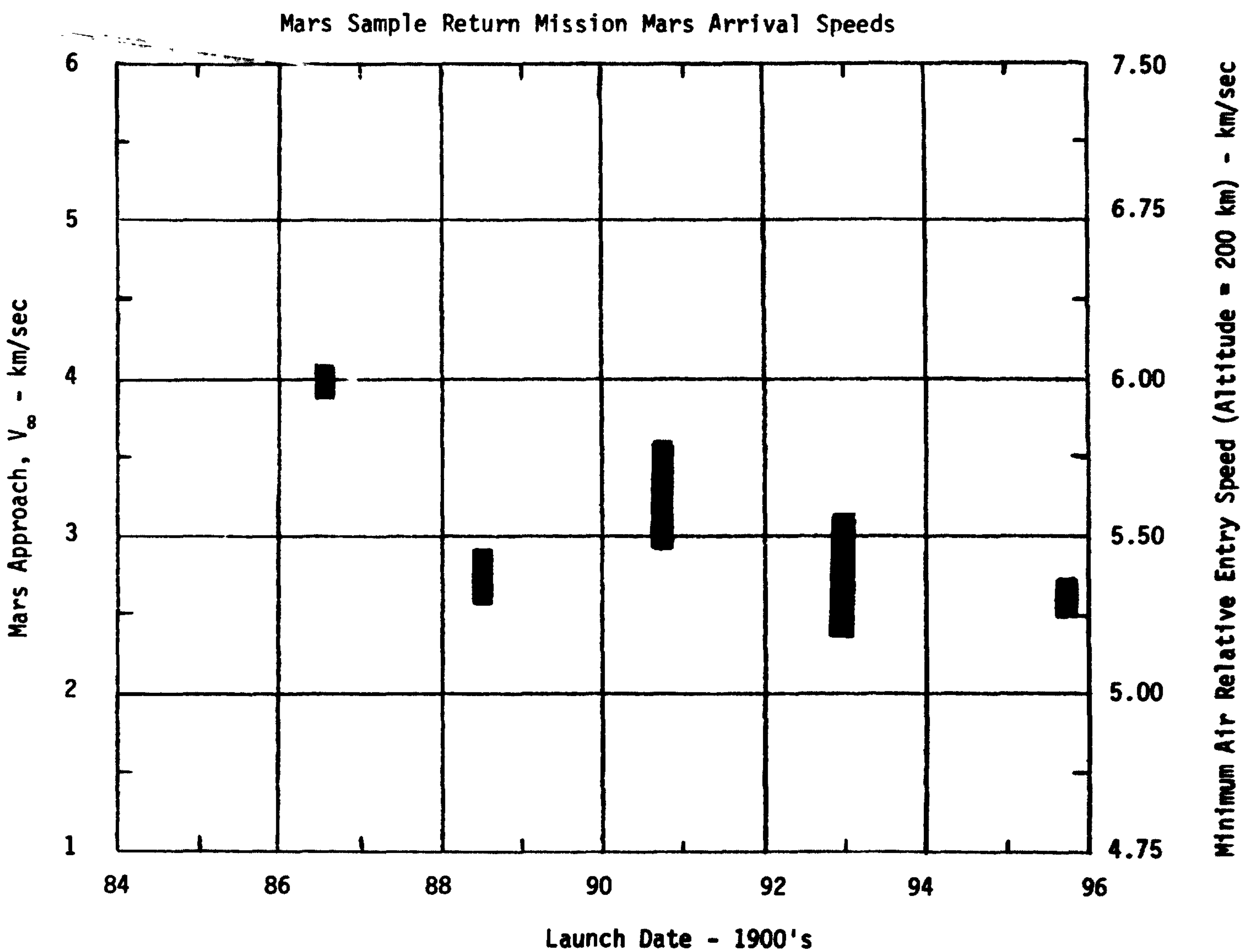

Mars Sample Return Mission Mars Arrival Speeds
Mars Approach, V_∞ - km/sec
Minimum Air Relative Entry Speed (Altitude = 200 km) - km/sec
Launch Date - 1900's
1
2
3
4
5
6
4.75
5.00
5.50
6.00
6.75
7.50
84
86
88
90
92
94
96

Figure II-2

Parachute performance and weight estimates were based on Viking technology. Due to the much larger size of the main chute and the higher dynamic pressure at deployment, a mortared-out drogue chute was used to deploy the main chute. The main chute maximum deployment condition was established as Mach 2.2, based on Viking chute deployment test results. The drogue chute diameter was sized to be two times aeroshell diameter to overcome wake effects. Chute weight was estimated by ratioing the canopy area and deployment dynamic pressure of the Viking chute. Se Appendix B for details.

The high ballistic coefficient of Case 5 required the addition of a Mach 5.0 deployed ballute to the decelerator systems. A ballute of 28 ft diameter was selected to provide the required main chute deployment conditions at sufficient altitude for terminal phase operation. The ballute weight estimate was made by using the following equation developed on early Viking studies.

$$W_B = 3.375\ R^3\ q_D\ (2.02E - 4 + 1.413E - 5\ M_D + 1.256E - 6\ M_D^{\ 2}),\ \text{lb}$$

R = ballute radius

q_D = dynamic pressure at deployment, psf

M_D = Mach number at deployment

W_B = ballute weight-LBS

The ballute is also used to deploy the main chute when the vehicle has decelerated to chute deployment conditions.

The terminal phase propulsion system used in Cases 1, 2, 3, and 4 was the basic Viking system modified to a constant pressure feed system which replaced the blowdown system. As landed weight was increased, additional engines were added to maintain the Viking thrust-to-weight ratio. For Cases 5, 6, and 7 a hybrid solid propellant-liquid system was used. The solid propellant system was designed to remove the major protion of the remaining velocity after chute separation and the liquid propellant system then removed the wind (surface relative) velocity and any uncertainties to affect a soft landing.

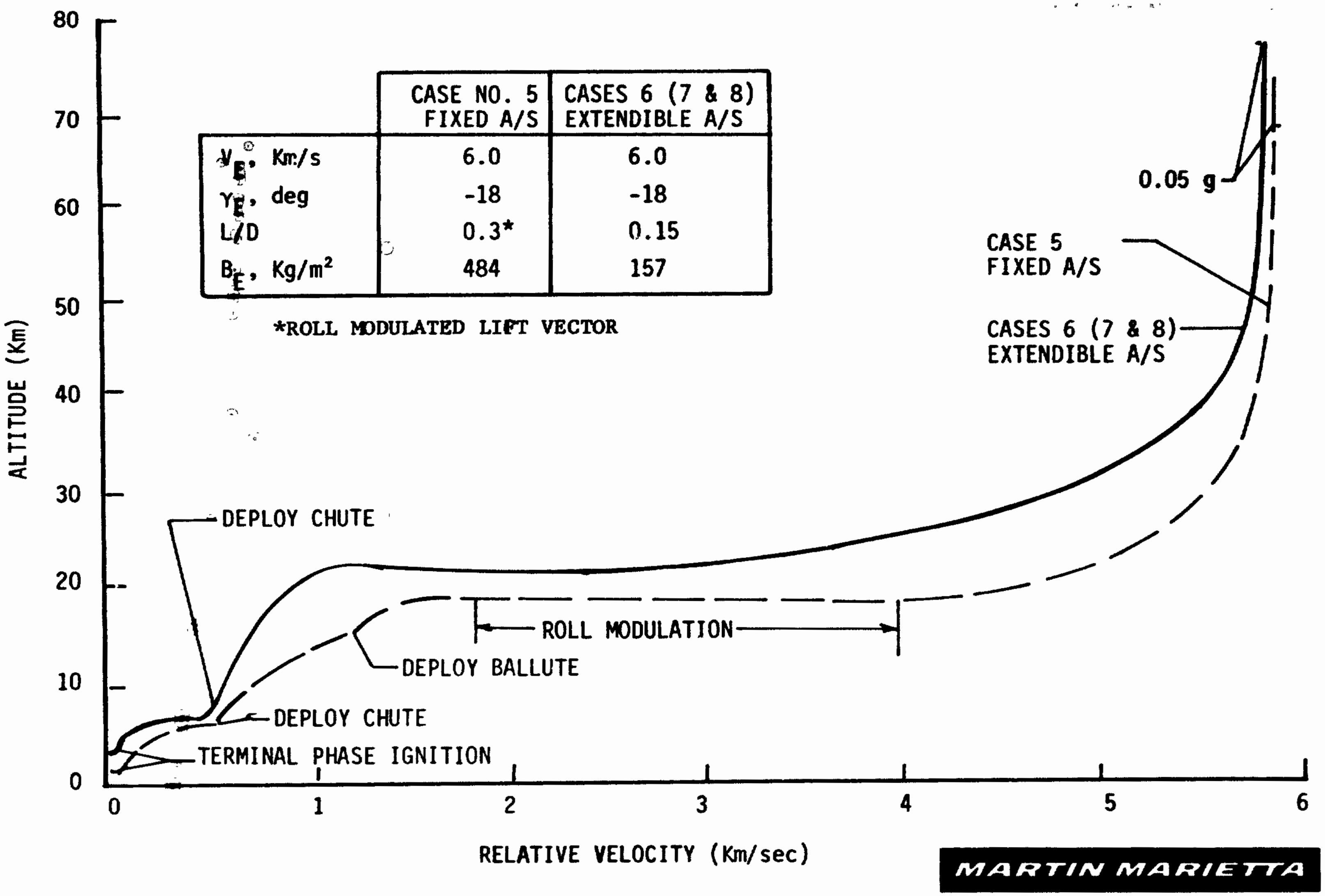
Entry trajectories for Cases 1, 2, and 3 were taken from references 2 and 3. For Case 4 entry trajectory environment and conditions at chute deployment were taken from JPL work, Reference 4. The later-data included the effect of lift vector roll modulation. Entry trajectories for Cases 5 and 6 (7 and 8) are shown on Figures II-3 and II-4. Figure II-4 shows the terminal phase ofthe entry trajectory. Cases 6 (7 and 8), which incorporate extendible aeroshell flaps, have flap sizes designed for constant entry ballistic coefficient and therefore have common entry trajectory characteristics. Parachute size and terminal phase propulsion thrust were increased in proportion to increased mass for Cases 7 and 8. Key entry/descent parameters are tabulated on Table II-2 for Cases 4 and 5 and on Table II-3 for Cases 6 (7 and 8).

An alternate analysis for Case 4 (entry mass = 5,343 kg) was made for entries with constant L/D ratio (non-roll modulated.) It became evident early in the analysis that entry system weights for this non-roll modulated concept were higher and landed dry weight lower than those of the roll modulated concept. Analysis for the non-roll modulated concept was therefore discontinued. Parametric entry data generated during the study are presented on Figure A-1 through A-4 in Appendix C to this report.

Roll modulation as used in the present analysis consisted of rolling the entry vehicle--which was trimmed to a specific angle of attack by c.g. offset--around the air relative velocity vector. This roll angle gives an effective modulation of the lift vector in the vertical plane of the trajectory. The effect of modulating the roll angle is:--1) to assure entry at the shallow end of the flight path corridor by rolling the lift vector down; 2) to minimize dynamic pressure and heating rate during the maximum deceleration portion of the trajectory, and 3) to modulate the vector to maintain a constant altitude after maximum dynamic pressure and thereby reach chute deployment at the highest possible altitude. Figure II-5 illustrates the effect of roll modulation on the altitude at which chute deployment can be achieved.

FIGURE II-3

ENTRY/DESCENT TRAJECTORIES FOR FIXED AND EXTENDIBLE AEROSHELLS

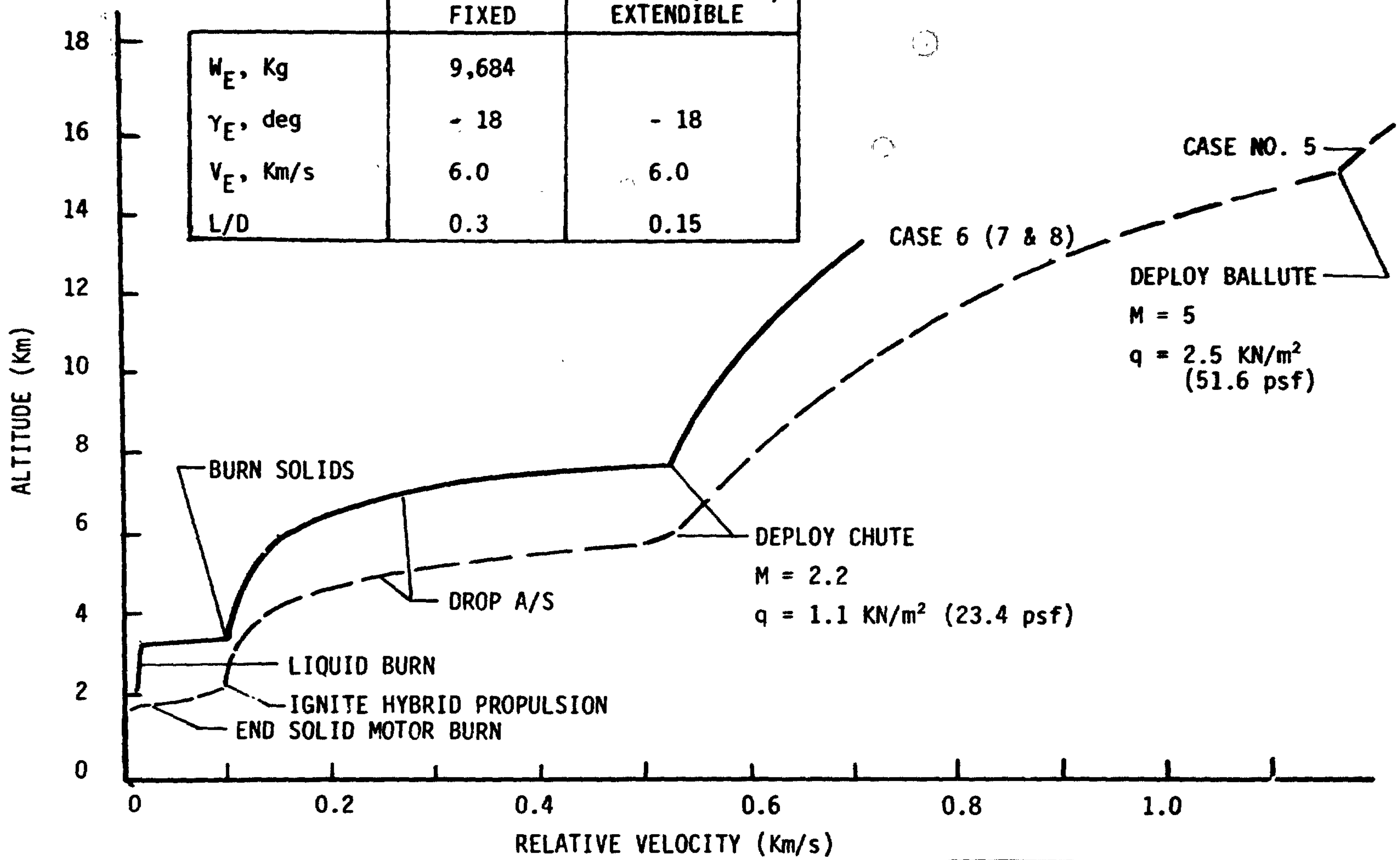


	CASE NO. 5 FIXED A/S	CASES 6 (7 & 8) EXTENDIBLE A/S
V_E, Km/s	6.0	6.0
γ_E, deg	-18	-18
L/D	0.3*	0.15
B_E, Kg/m²	484	157

*ROLL MODULATED LIFT VECTOR

FIGURE II-4

TERMINAL DESCENT TRAJECTORIES - FIXED AND EXTENDIBLE AEROSHELLS



	CASE NO. 5 FIXED	CASES 6 (7 & 8) EXTENDIBLE
W_E, Kg	9,684	
γ_E, deg	- 18	- 18
V_E, Km/s	6.0	6.0
L/D	0.3	0.15

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Table II- 2 Key Entry/Descent Parameters for Cases 4 and 5

V_E = 6.0 Km/sec

ENTRY CORRIDOR - 1-1/2 deg

ENTRY MODE - GUIDED (ROLL MODULATED)

L/D MAXIMUM - 0.3

LANDING SITE ELEVATION - 1.5 Km

AEROSHELL DIAMETER - 4.27 m

CASE	ENTRY MASS (Kg)	MAV MASS (Kg)	BALLIS. COEF. (Kg/m^2)	MAX. γ (Deg)	MAX. DYNAM. PRESS. (KN/m^2)	MAX. HEATING RATE (w/cm^2)	TOTAL HEAT LOAD (J/cm^2)	AFTERBODY SHAPE	BALLUTE DIA/MACH/q_{DYN} ($m/No./KN/m^2$)	MAIN CHUTE DIA/MACH/q_{DYN} ($m/No./KN/m^2$)
4	5,343	3,568	260	-18.0	14	43	5,414	40^o Return Angle (Rel. Vert.)	---	31.7/2.2/1.1
5	9,689	6,000 (including 400 Kg margin)	484	-18.0	24	72	5,115	20^o Return Angle (Rel. Vert.)	8.5/5.0/2.5	41.8/2.2/1.1

Table II- 3 Key Entry Parameters for Cases 6, 7, and 8

V_E = 6.0 Km/sec

ENTRY CORRIDOR	1-1/2 deg
ENTRY MODE	CONSTANT L/D, NOT ROLL MODULATED
L/D MAXIMUM	0.15
AEROSHELL DIAMETER (FIXED PORTION)	4.27 m
FLAP SIZE (8 REQUIRED):	

	CASE 6	CASE 7	CASE 8
CHORD/SPAN, m	1.67/1.80	1.67/2.16	1.67/2.54

CASE	ENTRY MASS (Kg)	MAV MASS (Kg)	BALLISTIC COEFF. (Kg/m²)	MAX. γ (deg)	MAXIMUM DYNAMIC PRESSURE (KN/m²)	MAX. HEATING RATE (W/cm²)	TOTAL HEAT LOAD (J/m²)	CHUTE DIAMETER/MACH/q_{DYN} (m/No./KN/m²)
6	9,402	6,000	157	-18	13	48	2,558	41.8/2.2/0.94
7	10,350	6,690	157	-18	13	48	2,558	42.0/2.2/0.94
8	11,600	7,549	157	-18	13	48	2,558	44.5/2.2/0.94

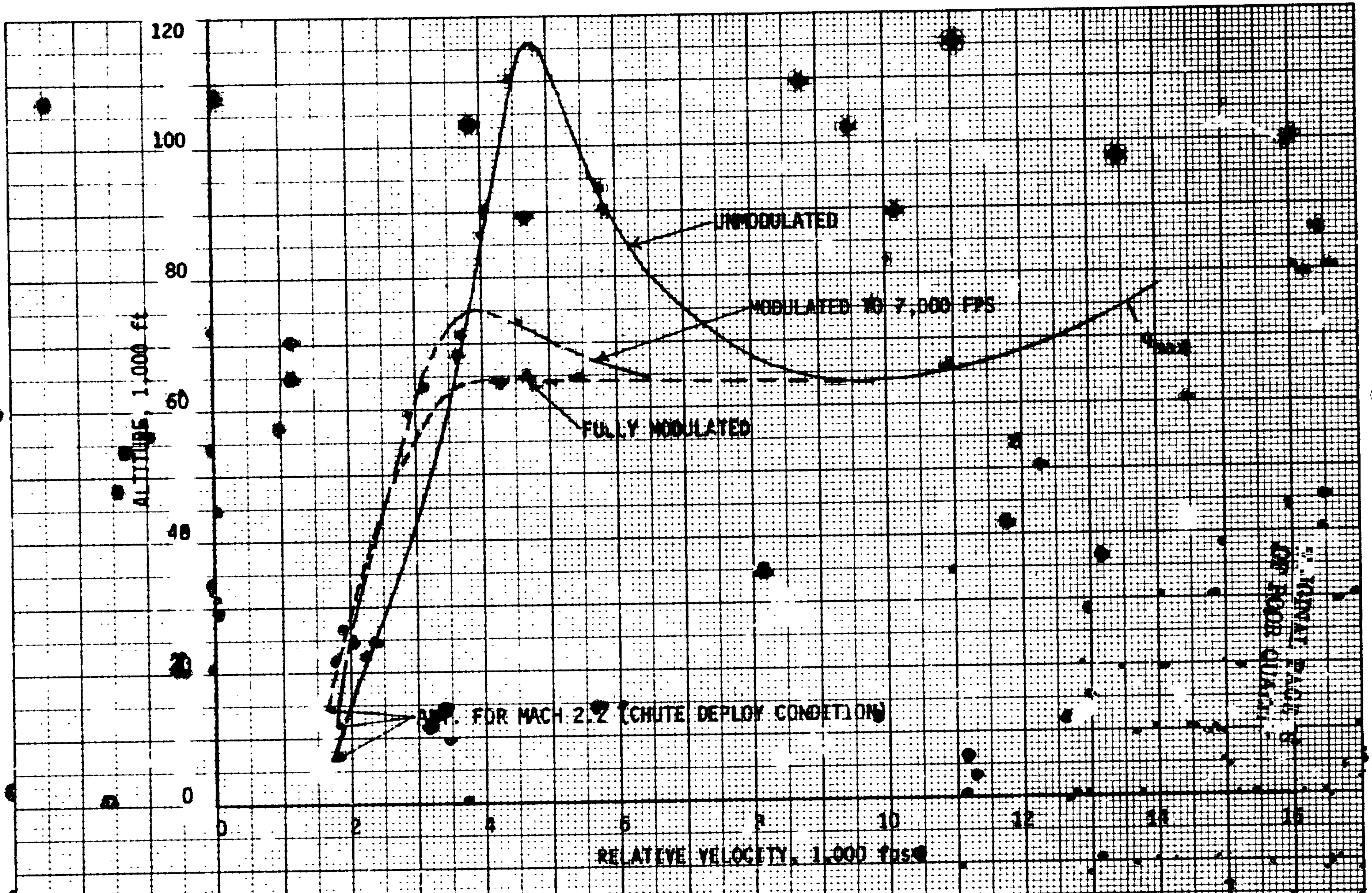

Figure II- Effect of Roll Modulation on Chute Deploy Conditions

The stagnation point heating rate pulse for Case 6 (7 and 8) is shown on Figure II-6. The integrated heat load, noted on the curve was used in estimating heat shield weights for cases 6, 7 and 8 based on the analysis for Case 5, discussed in Paragraph II.C.5, Heat Shield Design.

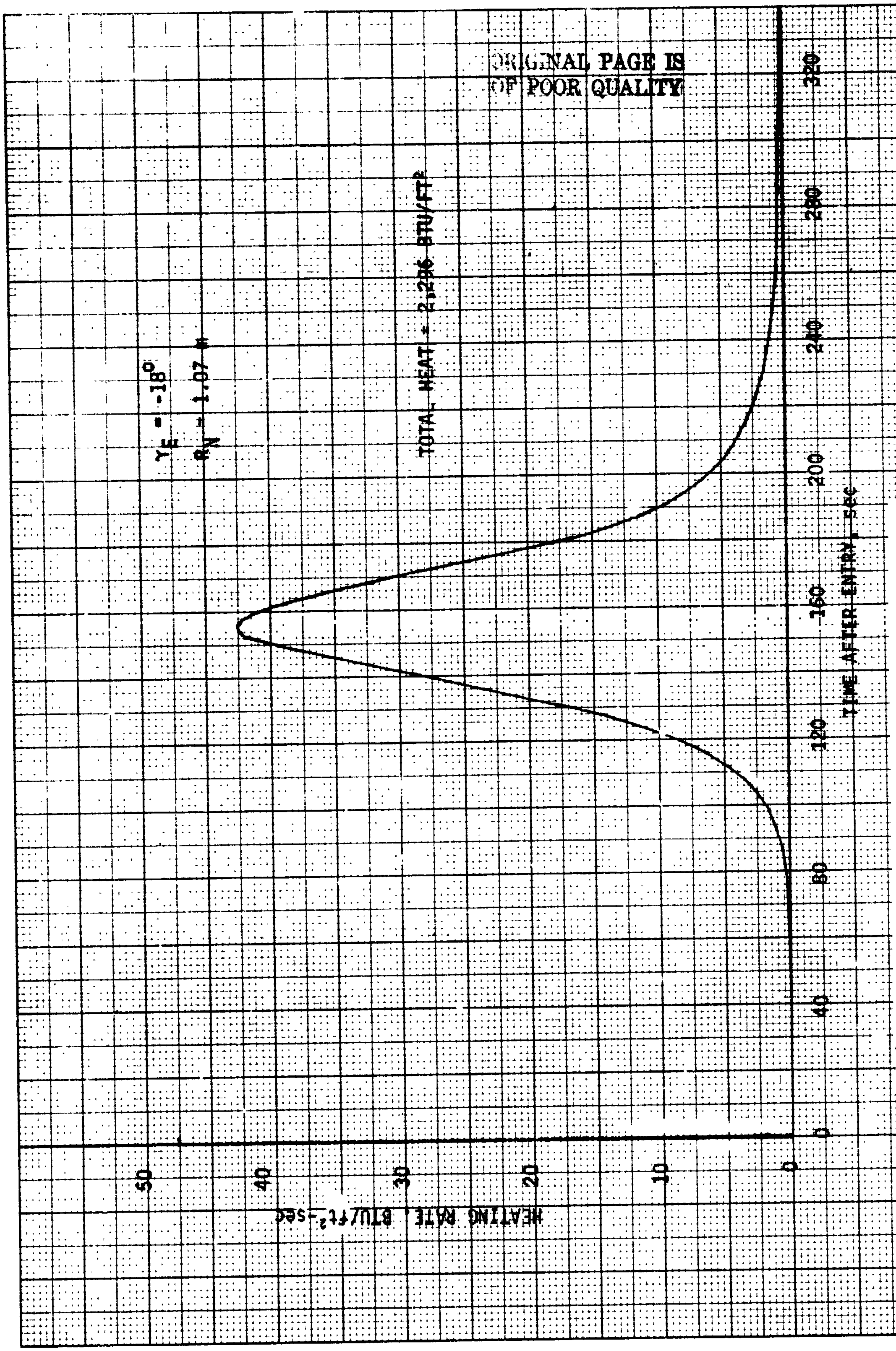

Figure II- 6 Stagnation Point Heating Rate, Case 6 (7 and 8)

5. Heat Shield Design

Protection of the entry vehicle from the aerodynamic heating environment was accomplished by using an ablative heat shield similar in concept to that used on Viking. Heat shield analysis was accomplished by means of the T-CAP computer simulation which calculated the ablation material thickness required such that the backface temperature would not exceed a specified temperature during entry. In the present analysis a maximum backface temperature of 600°F was specified. Heating rate input used was stagnation point rate based on a nose radius of 1.07 m. This approach is conservative because actual heating distribution is less than stagnation rate over a significant portion of the forebody diameter. This conservatism is countered by the much higher than Viking angle of attack (-20° vs -11°) which leads to high local heating at the tailing edge radius--windward side, and by nonevaluated dispersions in aerodynamic coefficient and atmospheric scale height characteristics. In addition, Viking heat shield calculated thickness required was increased by a safety factor of 1.5 while the present design has a factor of only 1.16. The present study heat shield designs were therefore considered to be only moderately conservative.

Heat shield thickness analysis and weight estimates were made for Cases 4 and 5. Heat shield weights for Cases 6, 7, and 8 were estimated from Case 5 analysis as discussed in Appendix B.

Heating rate time history for Case 4 is shown on Figure II-7. Based on an aluminum substructure equivalent to 0.04 in. thickness and ESA-3560 ablation material, the variation in backface temperature with heat shield thickness is as shown on Figure II-8. For the specified backface temperature of 600°F, a thickness of 0.433 in. was required. The area to be protected was 162 sq ft and the unit weight of the ESA 3560 material was 30 lb/ft^3. The resulting heat shield mass was 80 kg x 1.16 = 93 kg. ESA 3560 material was used because of its ability to withstand high heating rates. The Viking material, SLA 561, was tested to a maximum heating rate of 20 BUT/ft^2 sec, considerably below the 40 BTU/ft^2sec estimated for Case 4, and was therefore considered not proven for this application.

The afterbody for Case 4 was estimated to have a heating rate equal to 5% of the stagnation point heating rate. Maximum backface temperature as a function of heat shield material thickness is shown on Figure II-9. Practical ablation material installation considerations indicated a minimum thickness of 0.1 in. should be used. SLA 561 material provided superior insulation characteristics and was therefore used for this application. The afterbody area was 238 sq ft which at the material density of 15 lb/ft^3 resulted in a mass of 13.5 x 1.16 = 15.7 kg.

Heating rate time history for Case 5 is shown on Figure II-10. Based on an aluminum substructure equivalent to 0.05 in. thickness and ESA 3560 ablation material, the variation of backface temperature vs heat shield thickness shown on Figure II-11 was derived. For the specified 600°F backface temperature a heat shield thickness of 0.390 in. was required. The area to be protected was 162 sq ft and the unit weight of the ESA 3560 material was 30 lb/ft^3. The resulting heat shield mass was 72 kg x 1.16 = 84 kg.

The afterbody for Case 5 is estimated to have a heating rate equal to 5% of the stagnation point heating rate. Maximum backface temperature as a function of heat shield thickness is shown on Figure II-12. The minimum heat shield thickness of 0.1 in. SLA 561 material was used. The afterbody area was 468 sq ft which resulted in a heat shield mass of 26 kg x 1.16 = 20 kg.

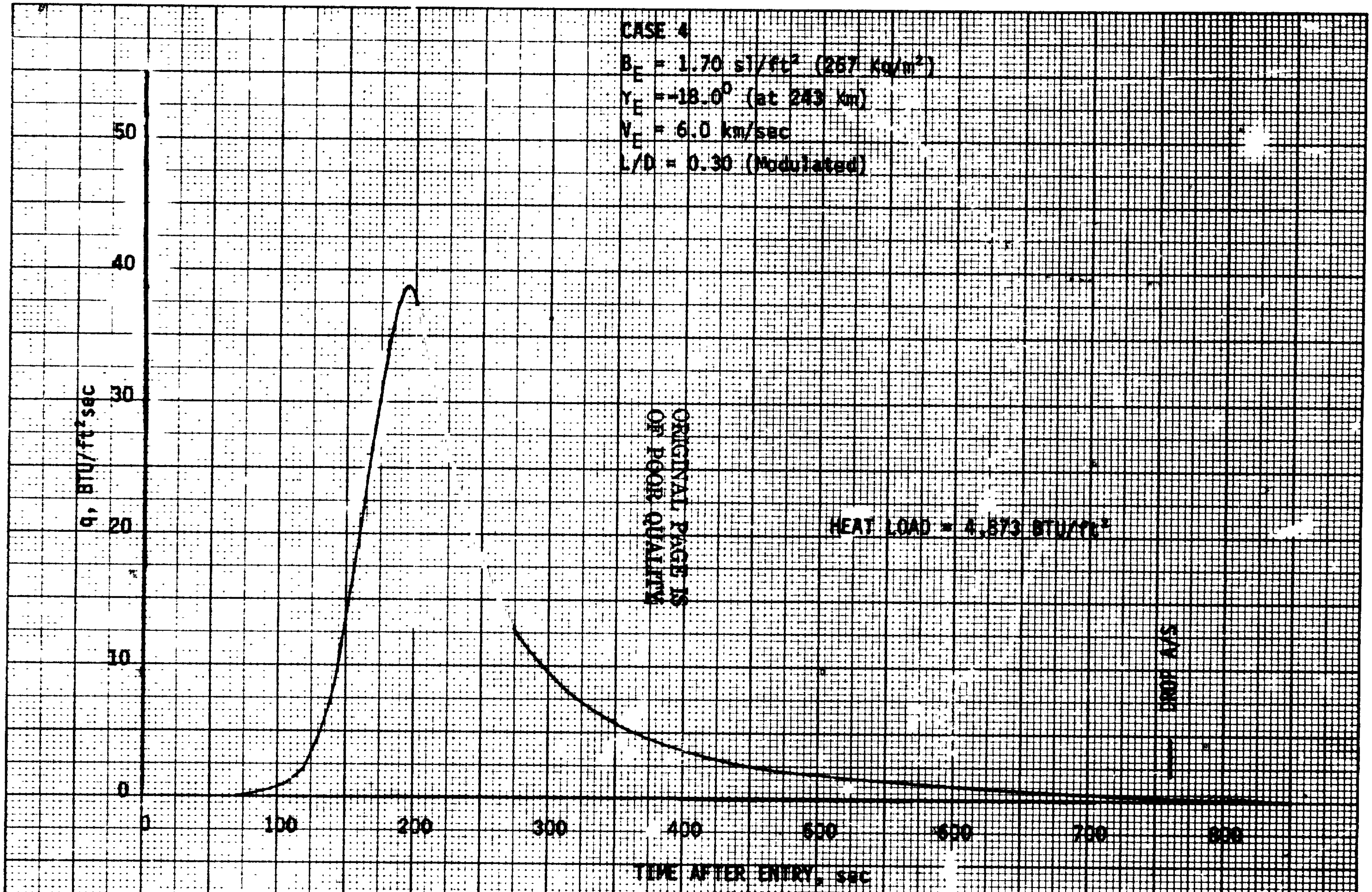

Figure II- 7 Entry Heating Rate vs Time After Entry

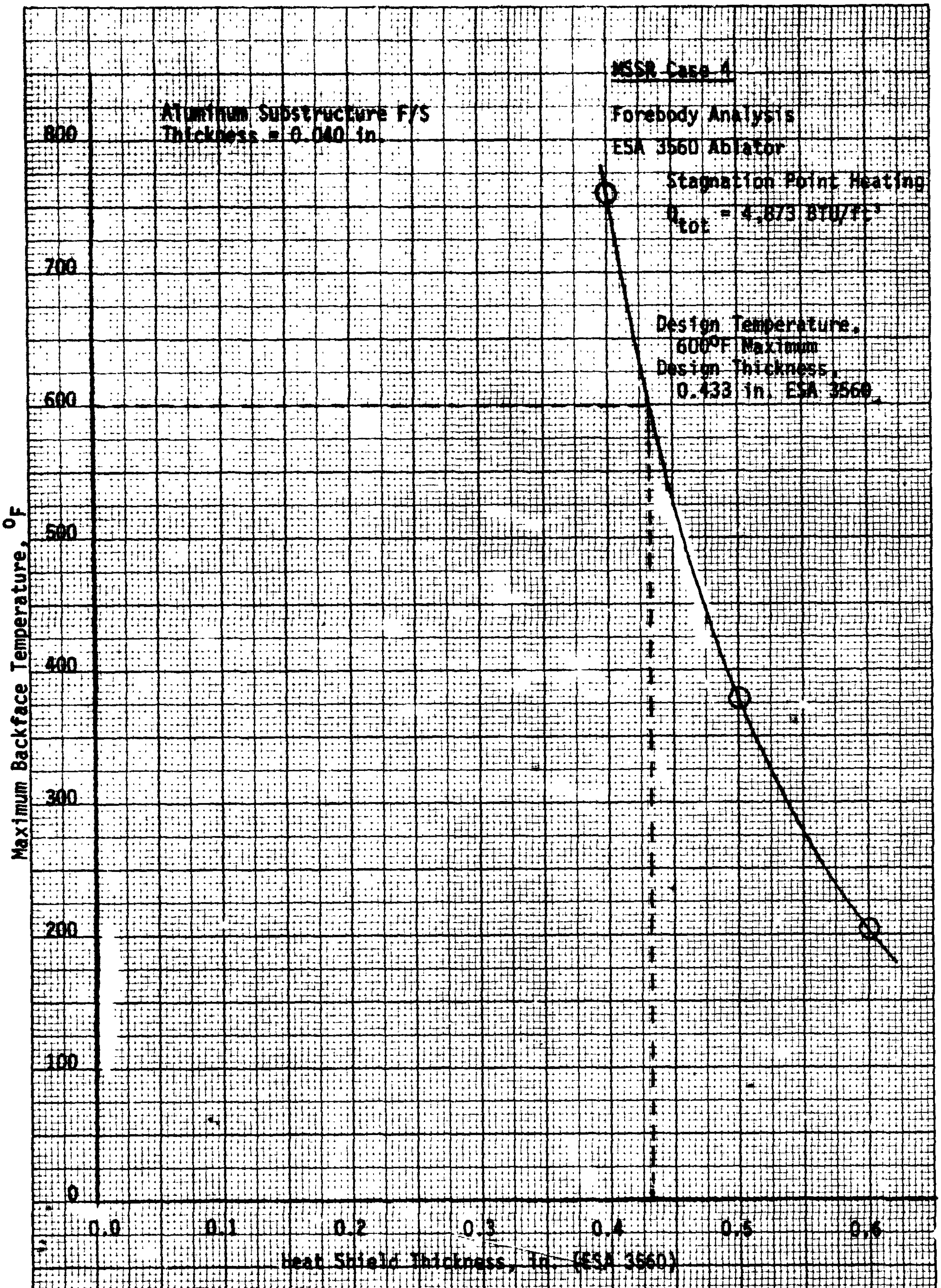

Figure II-8 Maximum Forebody Backface Temperature vs Heat Shield Thickness

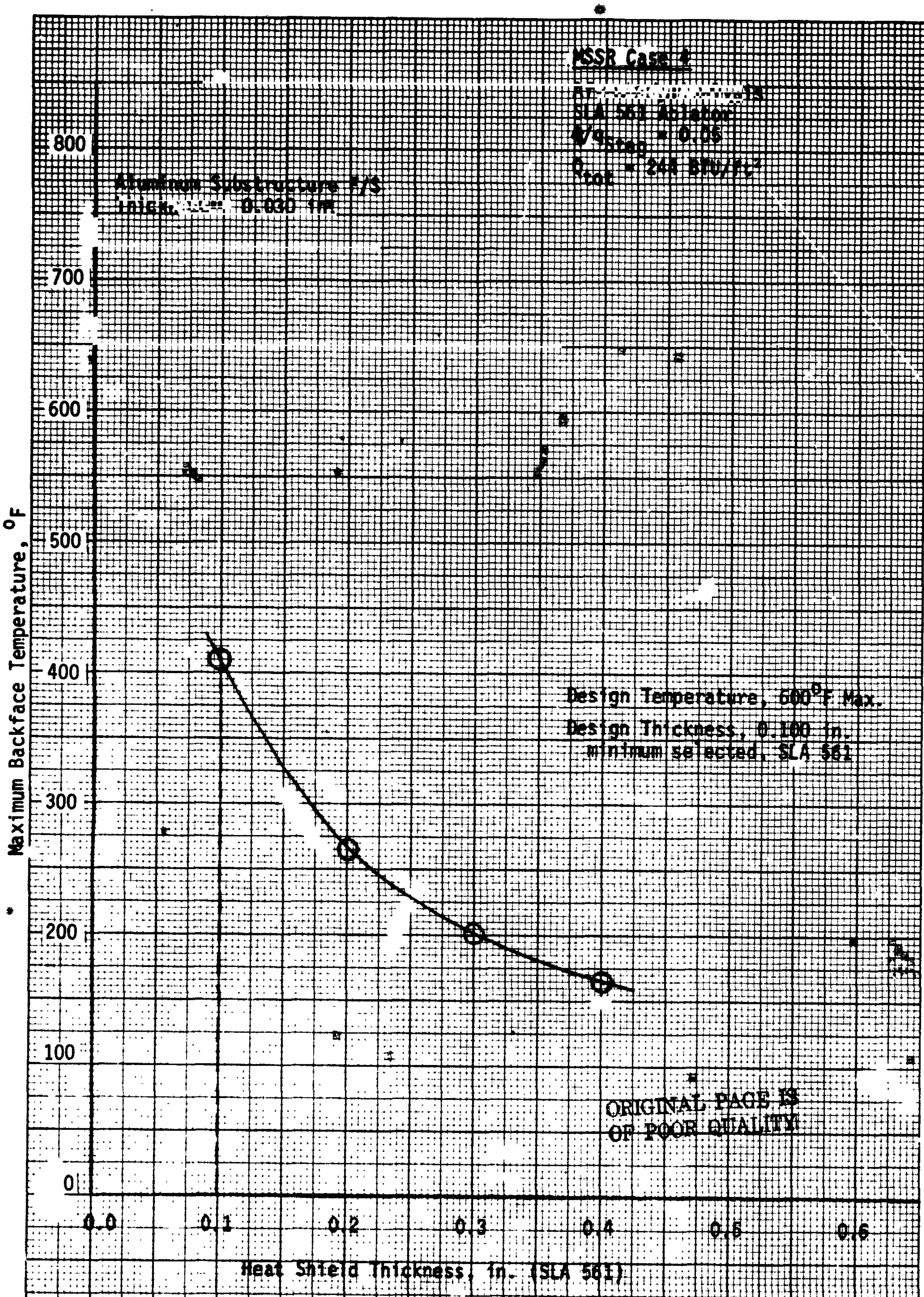

Figure II- 9 Maximum Afterbody Backface Temperature vs Heat Shield Thickness

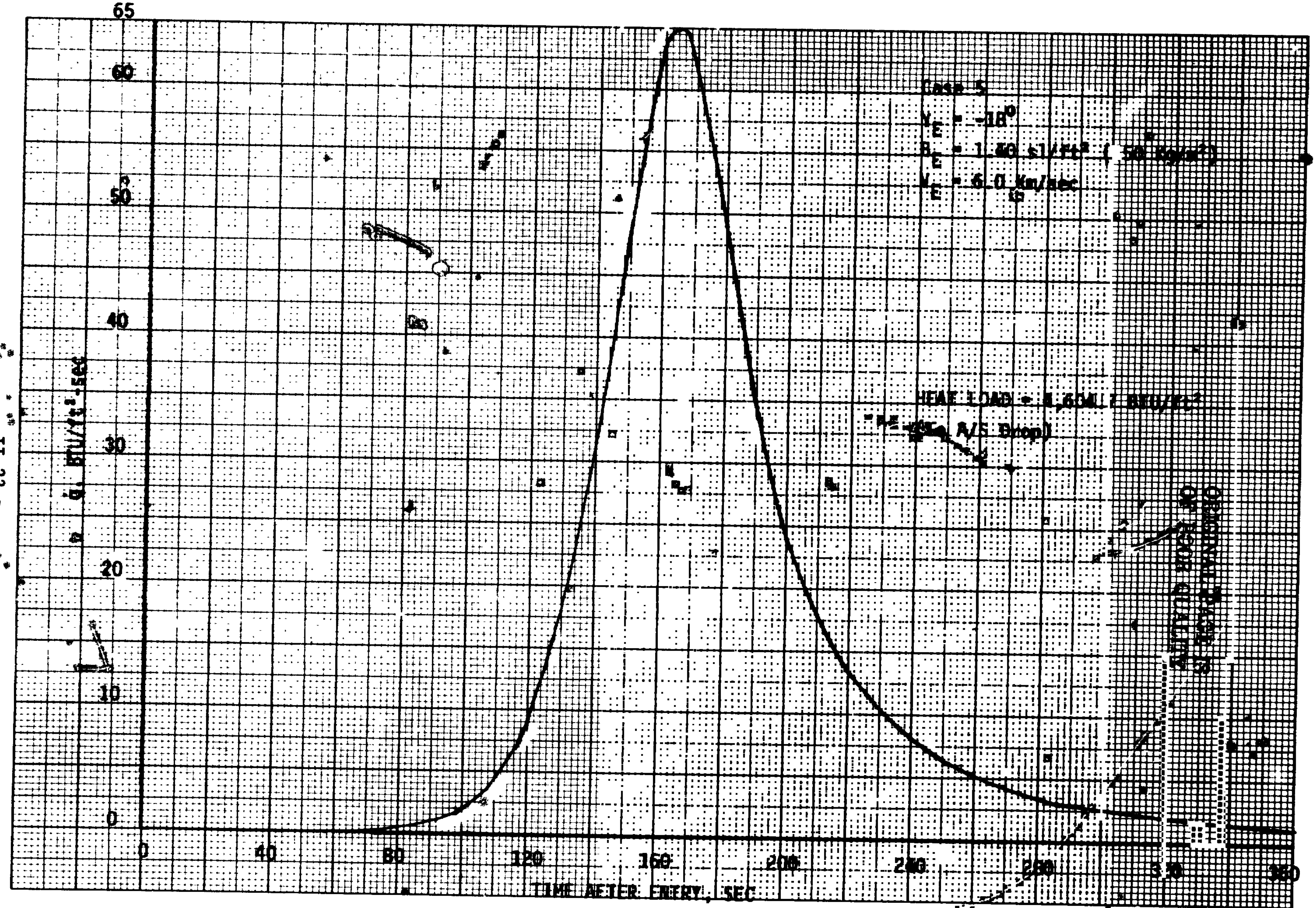

Figure II- 10 Entry Heating Rate vs Time After Entry

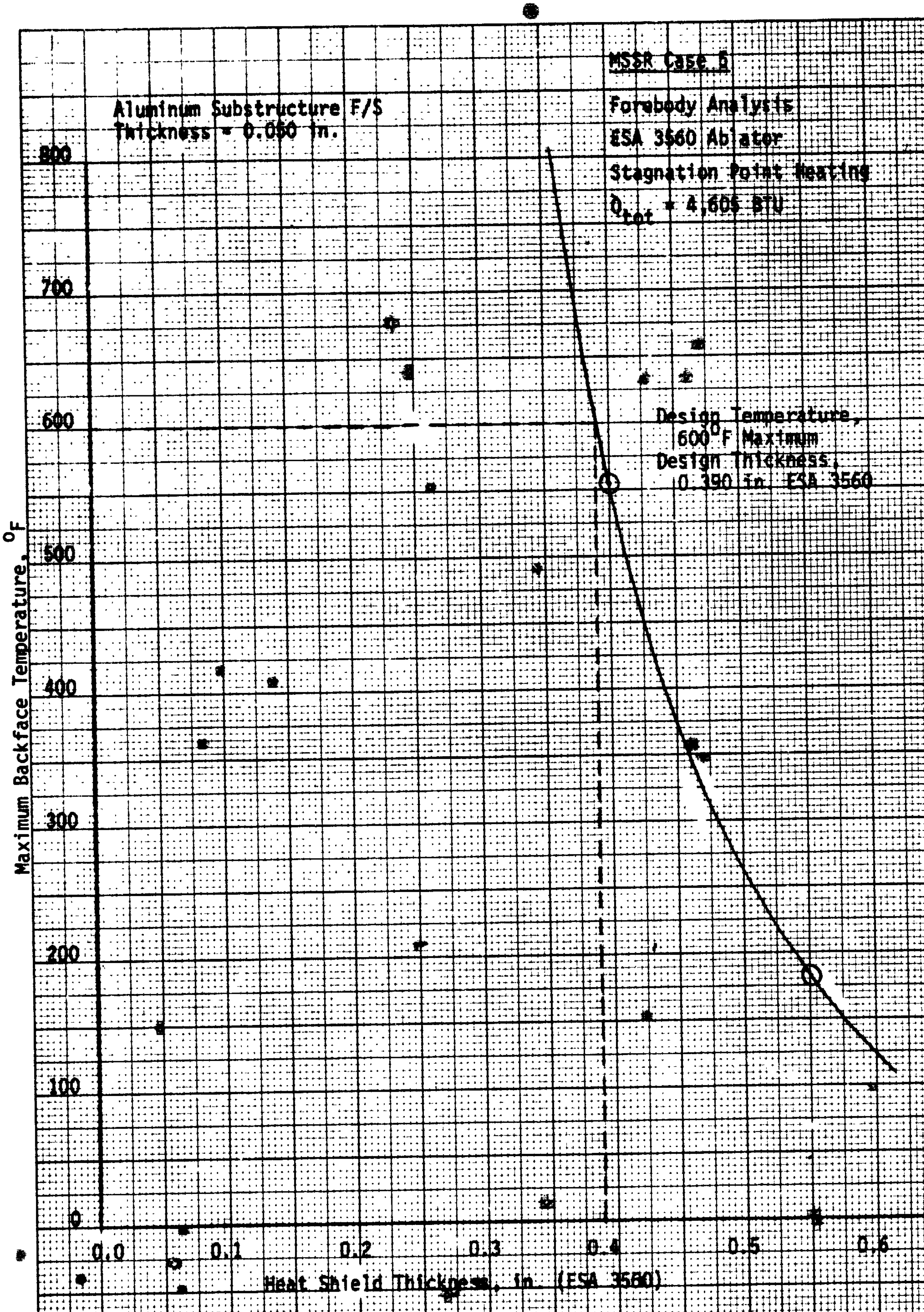

Figure II-11 Maximum Forebody Backface Temperature vs Heat Shield Thickness

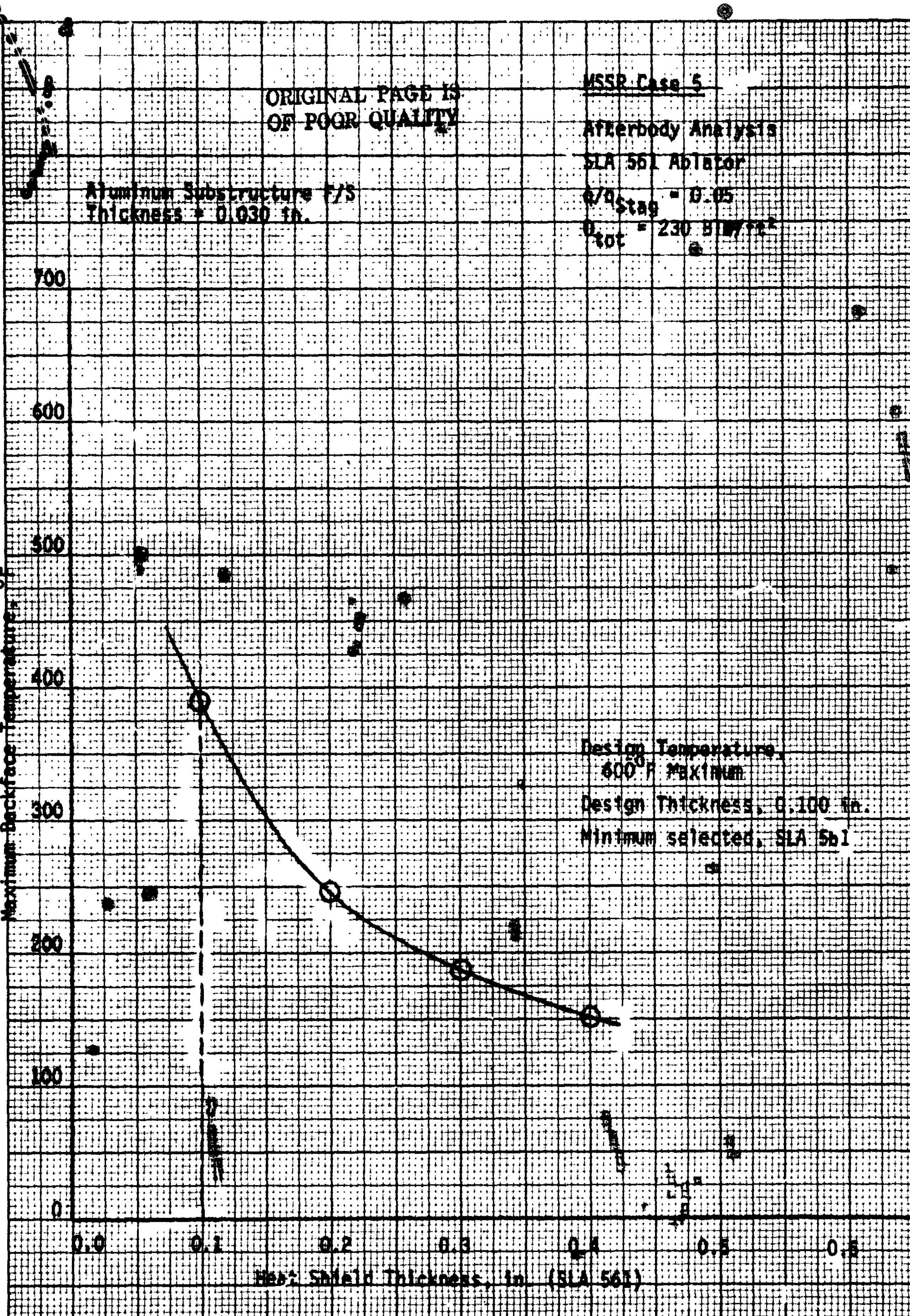

Figure II- 12 Maximum Afterbody Backface Temperature vs Heat Shield Thickness

6. Configuration and Mass Properties

a. Viking Technology (Cases 1, 2, and 3) - A brief preliminary evaluation was performed to determine the maximum mass which can be landed on the surface of Mars using Viking system concepts (see Appendix A for additional parametric data). As shown in Table II-4, the Case 1 design was consistent with the Viking '75 design entry environment limits of dynamic pressure, heating rate, and heat load obtained from out-of-orbit entry. This illustrates that the Viking '75 design, with its 3.5 m (11.5 ft) diameter aeroshell, can be extended from 981 kg of entry mass to 1,148 kg by taking advantage of the conservative design margins.

In Case 2 the diameter was increased to the full allowable limit of the Shuttle payload dynamic envelope, 4.27 m (14 ft), and the same entry environment and ballist coefficient were maintained as in Case 1. The allowable entry mass increased to 1,700 kg with a dry landed mass of 1,220 kg as compared to 800 kg for Case 1.

In Case 3 the entry environment(dynamic pressure and heating) were allowed to become more severe than those of Viking '75. An entry ballistic coefficient of 145 kg/m^2 was assumed based on the previous work done by NASA Langley (Ref. 3). Because of the increased severity of the entry environment, the aeroshell structure and heat shield masses were appropriately scaled up from that of the Viking '75 structure, resulting in entry and dry landed masses of 3,100 kg and 2,140 kg, respectively. The mass increases due to a larger parachute and added propulsion for the terminal descent phase are also included. The parachute was deployed at Viking '75 design conditions of Mach 2.2 and dynamic pressure 8.6 psf, however the chute is larger than that of Viking '75 and the target elevation margin is minimal.

b. Fixed Aeroshell with 5,343 kg Entry Mass (Case 4) - The objective in studying Case 4 was to estimate the mass and volume available for a MAV starting with a 5,343 kg entry mass (twin-stage IUS launch vehicle) and a 4.27 m (14 ft) diameter entry of configuration with a 40° return angle afterbody.

Both constant L/D and roll modulated lift entry modes were considered as discussed in Section II.C.4.

Table II-4 Landed Mass Capability Summary for Viking Class Systems

	CASE 1	CASE 2	CASE 3
AEROSHELL DIAMETER, m (ft)	3.5 (11.5)	4.27 (14)	4.27 (14)
ENTRY MASS, Kg	1,148	1,700	3,100
DRY LANDED MASS, Kg	800	1,220	2,140
ENTRY LOADS	SAME AS VIKING '75	SAME AS VIKING '75	GREATER THAN VIKING '75
ENTRY CONDITION	OUT OF ORBIT	OUT OF ORBIT	OUT OF ORBIT
ENTRY BALLISTIC COEFF., kg/m^2	78.5	78.5	145

Table II-5 presents the estimated mass breakdown for an entry weight of 5,343 kg and shows that the mass available for the MAV was 3,568 kg and for the lander was 960 kg. This total mass of MAV and lander, 4,528 kg, required a volume of 18.9 m^3 at 240 kg/m^3 (15 lbs/ft^3) packaging density. The available volume shown in Figure II-13 was only 12.2 m^3, therefore Case 4 was not a viable configuration.

Subsequent analysis at JPL indicated that a reasonable minimum MAV mass was about 5,600 kg which is considerably greater than the 3,568 kg available for this case.

c. Fixed Aeroshell with 6,000 kg MAV (Case 5) - the objective of this case was to estimate the entry mass required and the entry vehicle configuration based on a 4.27 m (14 ft) diameter aeroshell and a 6,000 kg MAV mass.

Table II-6 presents the entry vehicle mass breakdown with a 5,600 kg plus a 400 kg MAV margin. The entry subsystems (i.e., aeroshell, base cover, deployable decelerators, etc.) and lander (not including the MAV) had a mass of 3,690 kg. The all-up entry mass including the MAV was 9,690 kg. This configuration required a three-stage IUS launch vehicle.

Because of the aeroshell diameter constraint of 14 ft the large entry mass, and high ballistic coefficient, this configuration required the addition of a ballute to slow the vehicle down to a safe velocity and Mach number for main chute deployment. The ballute was deployed at Mach 5 and the chute was deployed at Mach 2.2, just providing minimum terrain clearance margin prior to ignition of the terminal descent propulsion. See Section II.C.4 for discussion of the descent trajectory.

In order to house the large MAV ascent propulsion stages and the lander hardware, the aft cover section was extended as shown in Figure II-14. With a shallow return angle of 21° the atmospheric flow during entry will tend to reattach to the aft cover and increase the local heating by a significant factor and, therefore, additional heat shield material was required on the base cover. About 60 kg of the 215 kg base cover mass is required to provide the necessary thermal protection.

This configuration also has a relatively aft center of gravity location compared to that of Viking; however it is still statically stable because of the blunt 70° half-angle aeroshell. The vehicle does need damping augmentation with this aft c.g. location and the estimated mass for the entry attitude control and damping system was 180 kg.

Although this configuration appears feasible, it must rely on the ballute to slow it down prior to chute deployment and the terrain clearance margin is essentially zero. Consideration of modifications to the configuration to improve the situation by lowering the ballistic coefficient led to the evaluation of Case 6 which is described in the next section (Section II.C.6.d).

TABLE II-5

ESTIMATE FOR MAV MASS AVAILABLE USING 5,343 Kg MASS AT ENTRY (CASE 4)

SUBSYSTEM		MASS, Kg
AEROSHELL		307
STRUCTURE	174	
HEAT SHIELD	93	
ACS	15	
MISCELLANEOUS (Insul., Pyro., etc)	25	
BASE COVER		113
STRUCTURE	82	
HEAT SHIELD	16	
MISCELLANEOUS	15	
DECELERATOR		234
DROGUE	21	
MAIN	213	
LANDER		960
STRUCTURE	293	
PROPULSION INERTS	181	
OTHER SS	100	
PROPELLANT	386	
SUBTOTAL		1,614
MARGIN		161
		1,775
MAV + MARGIN		3,568
TOTAL		5,343

SUMMARY

- MASS AT ENTRY = 5,343 Kg
- DRY LANDED MASS = 4,303 Kg
- MASS AVAILABLE FOR MAV = 3,568 Kg

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FIGURE II-13

VOLUME AVAILABLE FOR MAV AND LANDER - BASED ON TWIN-STAGE IUS PAYLOAD

5,343 Kg ENTRY MASS

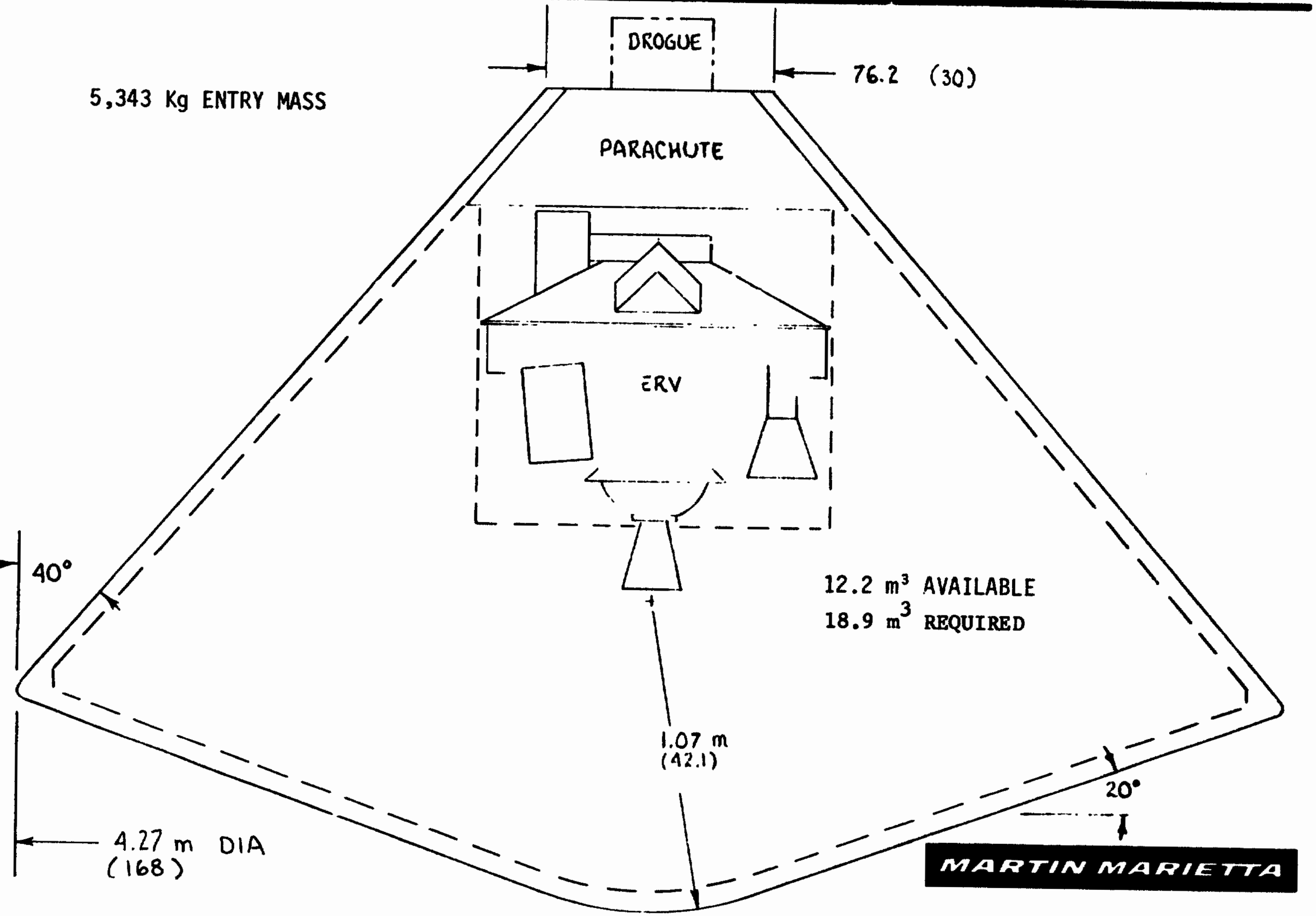

TABLE II-6

ENTRY MASS REQUIRED FOR FIXED AEROSHELL CONCEPT (CASE NO. 5)

6,000 Kg MAV - DIRECT RETURN MSR

ENTRY CHARACTERISTICS

TYPE ENTRY	GUIDED (L/D MODULATED FROM -0.3 TO +0.3)
BALLISTIC COEFFICIENT (Kg/m^2)	484 (3.0 SLUGS/FT^2)

MASS SUMMARY (Kg)

BASIC AEROSHELL	327
BASE COVER	215
ACS SYSTEM (ESTIMATED)	180
BALLUTE (MACH 5.0)	692
PARACHUTE	399
TERMINAL PROPULSION SYSTEM (SOLID + LIQUID)	892
LANDER	650
SUBTOTAL	3,355
ENTRY SYSTEM MARGIN	335
ENTRY/LANDING TOTAL	3,690
MAV (INCL. 400 Kg MARGIN)	6,000 (1)
TOTAL (ENTRY MASS)	9,690

(1) 6,000 Kg MAV per JPL DESIGN FOR WORST YEAR IN 1980's (i.e., 1988).

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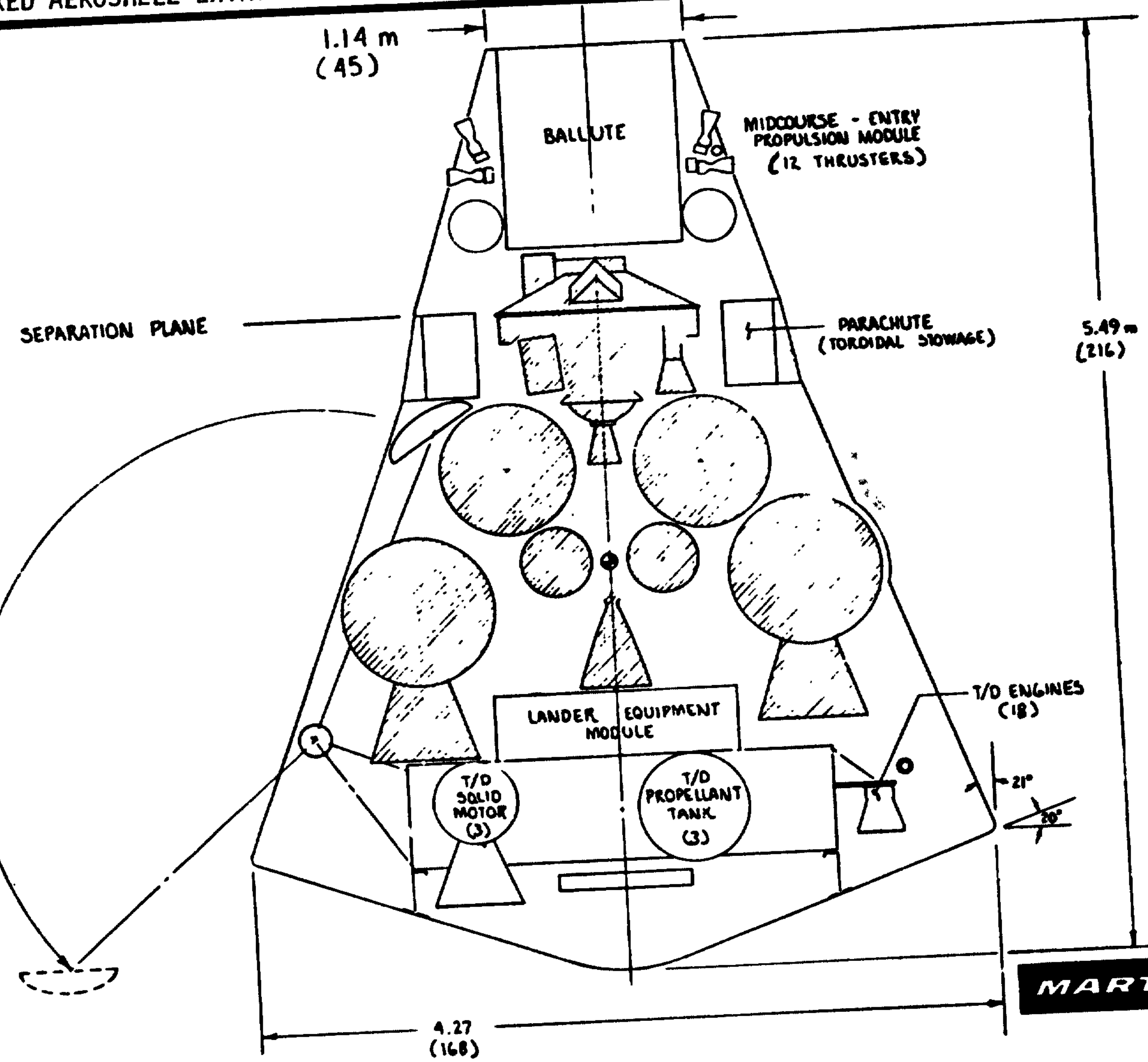
FIGURE II-14

FIXED AEROSHELL ENTRY SYSTEM (CASE NO. 5)



d. **Extendible Aeroshell (Case 6).** The objective of this task was to examine methods of lowering the ballistic coefficient of Case 5 by the addition of flaps which could be deployed at the edge of the aeroshell after leaving the Shuttle Bay. An additional objective was to estimate the entry mass required and define the entry vehicle configuration (based on the extendible flap aeroshell concept) starting with a 6,000 kg MAV mass. This case is an extension of Case 5 and the addition of flaps allowed deletion of the ballute since the additional flap drag was sufficient to slow thevehicle down to a safe main chute deployment condition at a reasonable altitude.

Figure II-15 shows a plan view of the extendible flap configuration. The flaps would be extended and locked into place during the interplanetary or approach phase prior to entry. The drag area is increased from 14.31 m^2 (153.9 ft^2) for the fixed aeroshell to 38.4 m^2 (412.9 ft^2) for the extended flap configuration. Although the extendible flaps add 660 kg of mass, there is a compensating savings in system mass obtained by deletion of the large ballute and reduction in ACS system, reduction of main parachute mass, and savings of structural mass in the fixed portion of the aeroshell configuration due to less severe (by about 1/2) entry loads and heating. The overall effect of the area and mass changes was to reduce the ballistic coefficient from 471 kg/m^2 (3.0 $slugs/ft^2$) for the fixed aeroshell (Case 5) down to 157 kg/m^2 (1.0 $slug/ft^2$) for the extendible aeroshell (Case 6).

Table II-7 presents the extendible aeroshell mass properties which can be compared to those of the fixed aeroshell in Table II-6.

The extendible aeroshell packaging concept is depicted in Figure II-16 and the arrangement is similar to that ofthe fixed aeroshell (Case 5). The relative center of gravity for this configuration is somewhat further aft than that of Viking, however the vehicle is still both statically and dynamically stable. Dynamic damping augmentation was still provided in the ACS system to assure adequate dynamic stability. For comparison purposes, Figure II-16 shows a further extension to the flaps which would be required to obtain the same relationship as Viking for the position of the center of gravity and the aeroshell maximum diameter interface. This diameter is 9.7 m (31.8 ft).

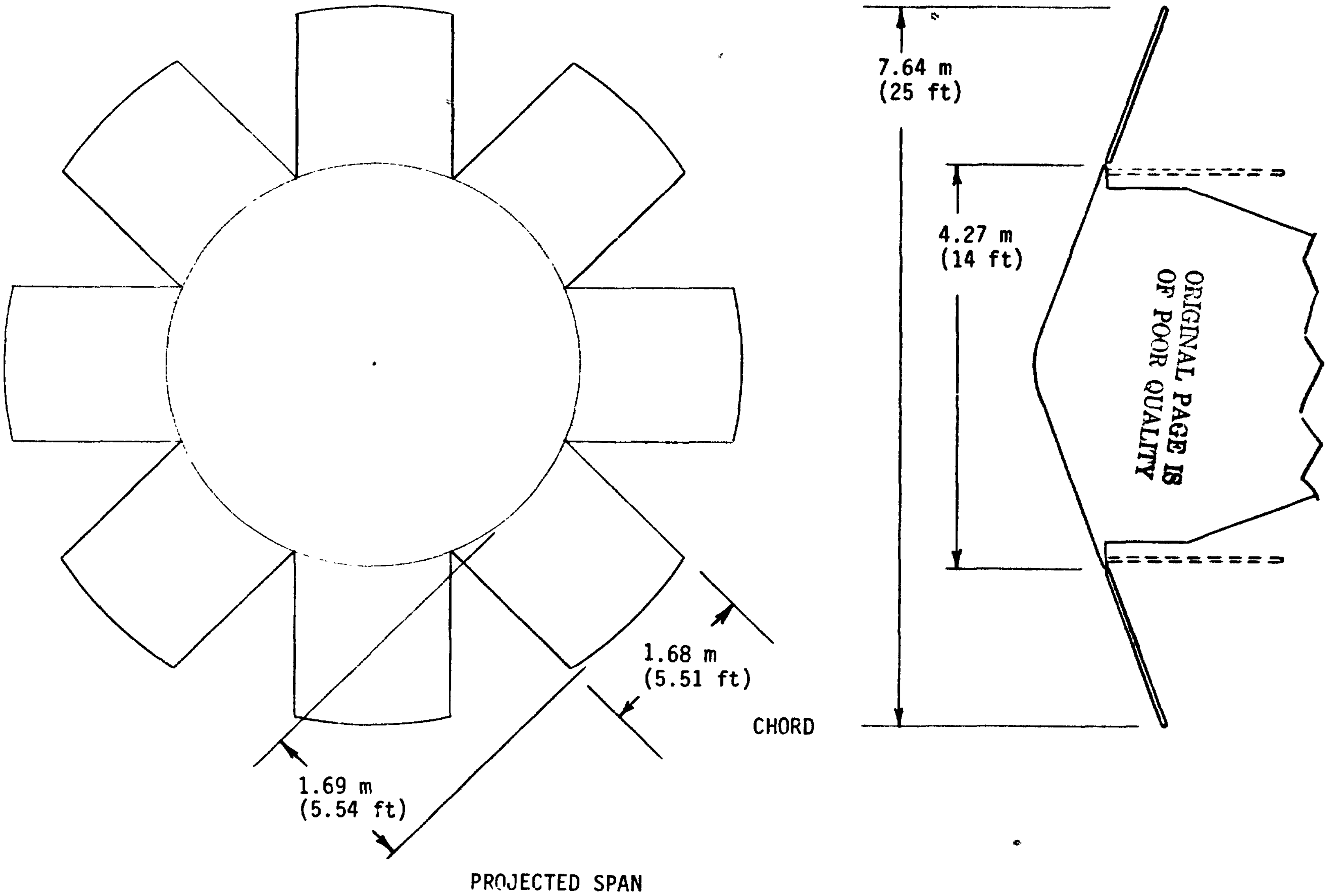


Figure II-15 Extendible Aeroshell Configuration (Case 6)

TABLE II-7

ENTRY MASS REQUIRED FOR EXTENDIBLE AEROSHELL CONCEPT (CASE NO. 6)

6,000 Kg MAV - DIRECT RETURN MSR

ENTRY CHARACTERISTICS

TYPE ENTRY	CONSTANT L/D (0.15)
BALLISTIC COEFFICIENT (Kg/m^2)	157 (1.0 $SLUG/FT^2$)

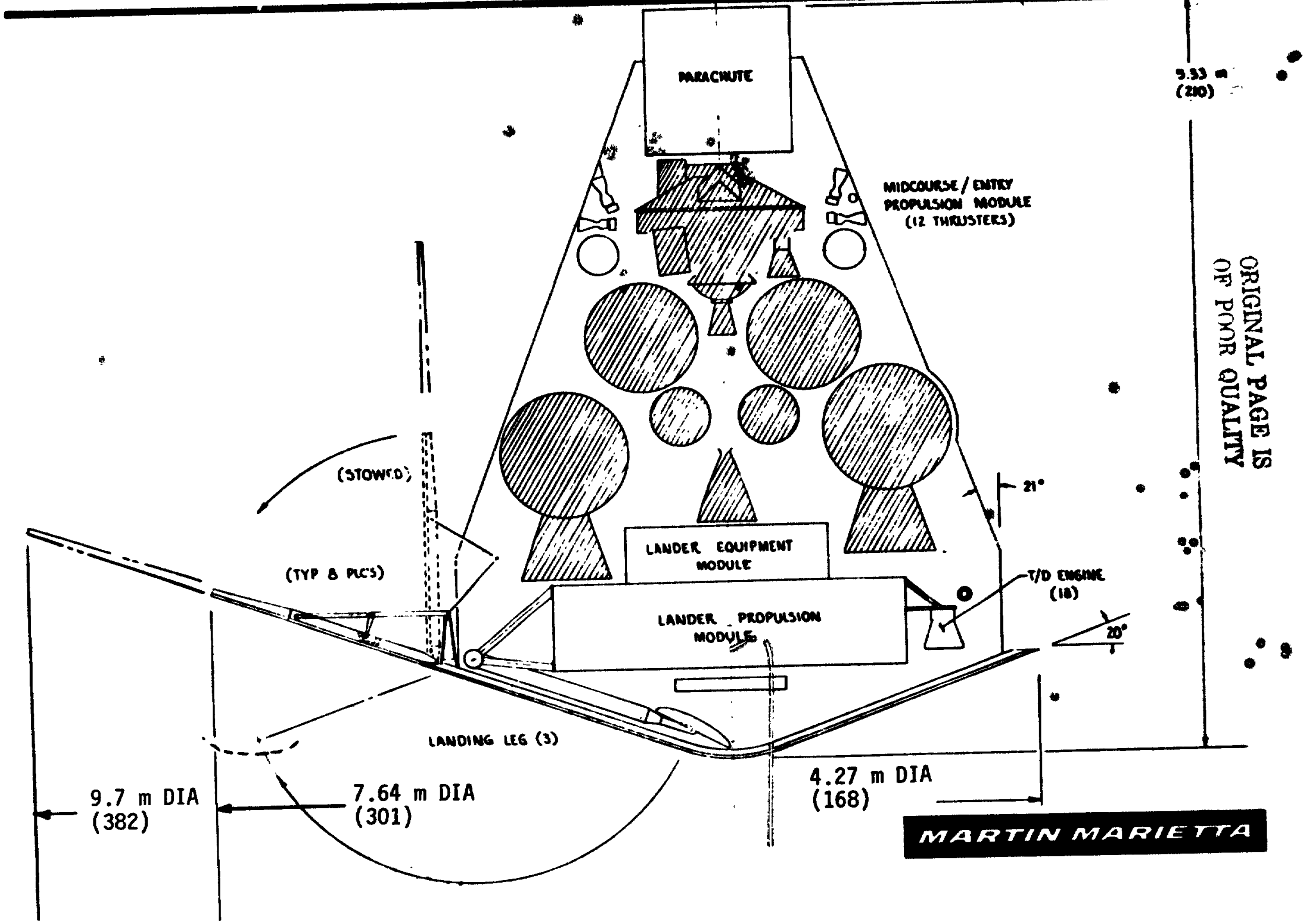
MASS SUMMARY (Kg)

BASIC AEROSHELL	290
AEROSHELL EXTENSION (FLAPS)	660
BASE COVER	158
ACS SYSTEM (EST.)	21
BALLUTE (MACH 5)	0
PARACHUTE	354
TERMINAL PROPULSION SYSTEM (SOLID + LIQUID)	958
LANDER	650
SUBTOTAL	3,091
ENTRY SYSTEM MARGIN	309
ENTRY/LANDING TOTAL	3,400
MAV (INCL. 400 Kg MARGIN)	6,000 (1)
TOTAL (ENTRY MASS)	9,400

(1) 6,000 Kg MAV PER JPL DESIGN FOR WORST YEAR IN 1980's (i.e., 1988)

FIGURE II-16

EXTENDIBLE AEROSHELL ENTRY SYSTEM (CASE NO. 6)



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In comparing the fixed (Case 5) and extendible (Case 6) aeroshell concepts for a direct return Mars sample return mission, the following design considerations were identified.

Design Considerations Common to Both Concepts

- Very large parachute required, 42 meters diameter (138 ft); deployment dynamic pressure two to three times Viking.
- Solid rocket motors required for terminal descent in addition to 18 Viking engines (site alteration and integration concern).
- Landing gear design difficult due to high c.g. and stowage space limitations.

Design Considerations - Fixed Aeroshell Concept (Case 5)

- Entry stability problem with long afterbody and aft c.g. - large uncertainty in ACS requirement.
- Significant afterbody heating due to shallow return angle and high angle of attack at L/D = 0.3.

Design Considerations - Extendible Aeroshell Concept (Case 6)

- Mechanisms for flap extension infringe on lander space.
- Uncertainties in performance and heating associated with gaps between flaps.
- Low L/D and low ballistic coefficient permitted by large drag area allows flight profile similar to Viking.

The following general conclusions were drawn relative to the fixed versus extendible aeroshell concepts for a 6,000 kg MAV Mars Sample Return mission:

- Either concept can probably be made to work.
- Entry and landing system mass required is similar for the concepts.
- Extendible flap concept appears to have fewer development uncertainties (program risk) and allows more direct application of Viking technology.

e. Four Stage IUS Payloads (Cases 7 and 8) - The objective of this task was to estimate the landed mass available when using the full four stage IUS launch vehicle. Table II-8 presents the mass breakdown starting with the launch payload injected into a trans-Mars trajectory for the 1988 and 1990 opportunities. These payload masses were obtained from Reference 1.

The 400 kg margin was assumed to include miscellaneous allocations for subsystems required for navigation, communication, thermal control, etc, during interplanetary flight and encounter at Mars.

The entry mass was then adjusted using ratios developed in Case 6 to estimate the masses required for the various entry system components such as aeroshell/heat shield, chute, base cover, and landing system structural and propulsion elements. The difference between the entry mass and the entry and landing subsystems gives the mass available for the MAV plus its margin. These available masses exceed the baseline MAV mass plus margin of 6,000 kg by 1,016 kg for 1990 and by 1,908 kg for 1988.

TABLE II-8

LANDED MASS AVAILABLE USING FOUR-STAGE IUS PAYLOAD

MASS IN KG

OPPORTUNITY	1990	1988
PAYLOAD	10,750	12,000
MARGIN	400	400
ENTRY WEIGHT	10,350	11,600
TOTAL AEROSHELL/HEAT SHIELD WEIGHT	1,067	1,230
WEIGHT ON CHUTE (ENTRY--A/S)	9,283	10,370
CHUTE WEIGHT	360	402
BASE COVER	180	214
LANDER WEIGHT AT IGNITION	8,743	9,754
LANDING SYSTEM		
STRUCTURAL/MECHANICAL	501	501
PROPULSION INERTS	273	303
OTHER SUBSYSTEMS (POWER, T/C, ETC)	150	150
PROPELLANT	803	892
TOTAL LANDING SYSTEM	1,727	1,846
MASS AVAILABLE FOR MAV + MARGIN	7,016	7,908

7. Alternate Entry Shapes

Several alternate entry shape concepts were evaluated and the results are summarized on Table II-9. Evaluation comments shown are relative to the Case 5 configuration. Potential entry vehicle shapes are constrained by the shuttle bay geometric characteristics and by the volume and high density requirements of the MAV vehicle. The aerodynamic characteristics of vehicles meeting these geometric constraints are not compatible with the high lift characteristic necessary to land very heavy payloads on the surface. Configurations such as the extendible flap concept appear to offer the best approach for delivering maximum volume/weight payloads to the surface of Mars.

TABLE II-9

ALTERNATE ENTRY SHAPES -- PART 1 OF 2

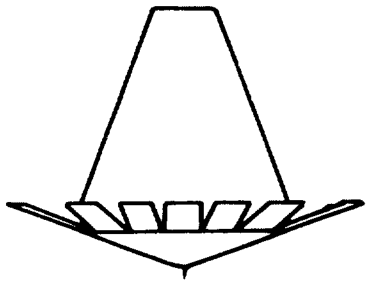
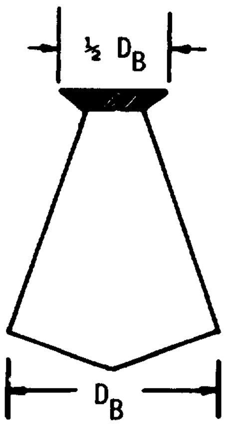
ENTRY SHAPE	DESCRIPTION	EVALUATION
	• BASIC SYSTEM FOR 6,000 Kg MAV (CASE NO. 6) • REDUCES B_E FROM 484 Kg/m^2 TO 157 Kg/m^2 • EIGHT PANELS 1.67 m SPAN;1.69 m CHORD	1. REDUCE ENTRY ENVIRONMENT. 2 RELIEVE CHUTE DEPLOYMENT CONDITION. 3. ELIMINATE BALLUTE. 4. REDUCE L/D REQUIRED TO 1.5 5. INCREASE TERRAIN HEIGHT CAPABILITY. 6. INCREASED MECHANICAL COMPLEXITY. 7. MORE COMPLEX HEAT PROTECTION.
	• BASIC SYSTEM FOR 6,000 Kg MAV (CASE NO. 5) • AFT FLARE IMPROVES STATIC AND DYNAMIC STABILITY • STAGE WITH BALLUTE	1. REDUCE REQUIRED RCS. 2. RETAINS HIGH DRAG AND L/D CAPABILITY. 3. FITS SHUTTLE ENVELOPE. 4. ADDED STRUCTURAL WEIGHT AND HEAT SHIELD.

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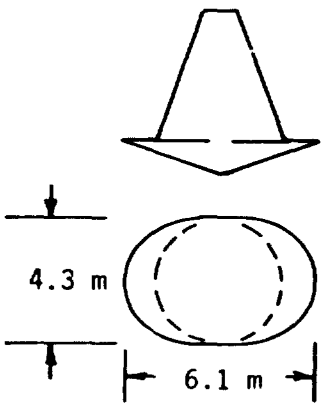
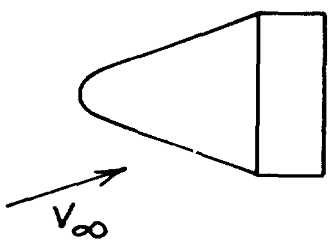
TABLE II-9 CONTINUED

ALTERNATE ENTRY SHAPES -- PART 2 OF 2

ENTRY SHAPE	DESCRIPTION	EVALUATION
	• ASYMMETRICAL BALLISTIC AERO-SHELL • $B_E \sim 94$ Kg/m^2	1. LOWER B_E. 2. APPEARS FEASIBLE AERO-DYNAMICALLY. 3. DIFFICULT TO STOW IN SHUTTLE WITH MAV CONFIGURATION.
	• BLUNTED CONE (REVERSED CASE 5 CONFIGURATION)	1. STATICALLY STABLE BUT REQUIRES TRIM DEVICE. 2. REASONABLE DYNAMIC CHARACTER-ISTICS. 3. FITS SHUTTLE PAYLOAD ENVEL. 4. L/D ABOUT 0.5. 5. LOW DRAG RESULTS IN $B_E \sim 780$ Kg/m^2. 6. HIGH ENTRY HEATING. 7. CHUTE DEPLOY ALTITUDE EXTREMELY LOW (~ MSL) 8. VEHICLE MUST ROTATE ON CHUTE.

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D. CONCLUSIONS - MARS SAMPLE RETURN SYSTEM TASK

The following general conclusions were drawn relative to the Mars Sample Return entry system evaluated within the ground rules of this task:

- An entry/landing system can be developed to deliver a landed-science/earth return system of the order of 6,000-7,000 kg.
- Such a system can be based on Viking technology but will require further development in areas of:
  - Extendible aeroshell segments;
  - Larger, higher dynamic pressure parachute systems;
  - Combined solid/liquid terminal descent system.
- Going beyond these mass limits is not practical due to the severe constraint on configuration packaging imposed by the Shuttle payload bay geometry.
- Payload (useful landed mass) will be of the order of 60 to 70% of total entry mass.
- Target elevation capability appears limited to 3 km or less -- site alteration may be a problem due to massive terminal propulsion system.

Table II-10 summarizes the direct return mission margins as a function of the entry and landing system mass requirements. The general conclusion is that a four-stage IUS launch vehicle configuration is required for the 1988 and 1990 launch opportunities although only a three-stage IUS is required in 1986.

TABLE II-10 Implications of Entry and Landing System Mass Requirements on Direct Return Mission Margin

	LAUNCH YEAR		
	1988	1986	1990
MAV (Including 230 Kg ERV and 400 Kg Margin)	6,000	5,458 (3)	5,232 (3)
Entry/Landing System	3,400 (1)	3,092 (4)	2,965 (4)
SUBTOTAL (Entry Mass)	9,400	8,550	8,197
Allocation for Midcourse Correction, Biocap, IUS Adapter, Level I Margin	400	400	400
TOTAL	9,800	8,950	8,597
Three-Stage IUS Payload Capability	9,000 (2)	9,970 (2)	7,900 (2)
Margin	- 800	+1,020	- 697
Portion of Margin Usable for Landed Ops.	N/A	+ 650 (5)	N/A
Four-Stage IUS Payload Capability	12,000 (2)	No Data Available	10,750 (2)
Margin	+2,200		+2,153
Portion of Margin Usable for Landed Ops.	1,430 (5)		+1,378 (5)

NOTES:

(1) BASED ON EXTENDIBLE AEROSHELL VERSION, CASE NO. 6

(2) PER NAGORSKI HANDOUT, APRIL 1978 MARS MISSION REVIEW

(3) ESTIMATED BY SCALING 6,000 kg MAV DOWN BY PRELIMINARY JPL MAV MASS vs OPPORTUNITY DATA

(4) ESTIMATED BASED ON MASS RATIOS FROM CASE NO.6

(5) ESTIMATED BASED ON MASS RATIOS FROM CASES 7 AND 8

III. TASK 2 - HARD LANDER ENTRY SYSTEM

A. TECHNICAL CONSIDERATIONS

The hard lander concept was conceived as a low cost, relatively simple lander which could survive a hard landing at impact velocities on the order of 20 m/s. By comparison, the Viking soft lander impacts at about 2 m/s. A goal of this task was to maintain the philosophy of a simple entry system with passive entry and descent subsystems where practical. However, a parachute was required to slow down the entry vehicle in the thin Mars atmosphere as well as means of sensing the deployment conditions and activating the system. The selected deceleration concepts utilized hardware components and techniques that are well within the state of the art.

B. OBJECTIVE OF TASK

The objective of this task was to establish an entry and descent concept for a hard lander having the general size, mass and impact velocity characteristics of the hard landers evolved in References 7 and 8.

C. ANALYSIS

1. Guidelines and Ground Rules

The hard lander entry system guidelines and ground rules for this study were as follows:

- Entry from approach.
- Use either JPL or Martin Marietta configuration from 1976 studies for lander definition:
  - Mass ~ 50 Kg;
  - Landing velocity ~ 20 m/s.
- Minimize complexity and size of entry system to greatest extent possible.

* Landing elevation as high as possible -- 5 or 6 km acceptable.

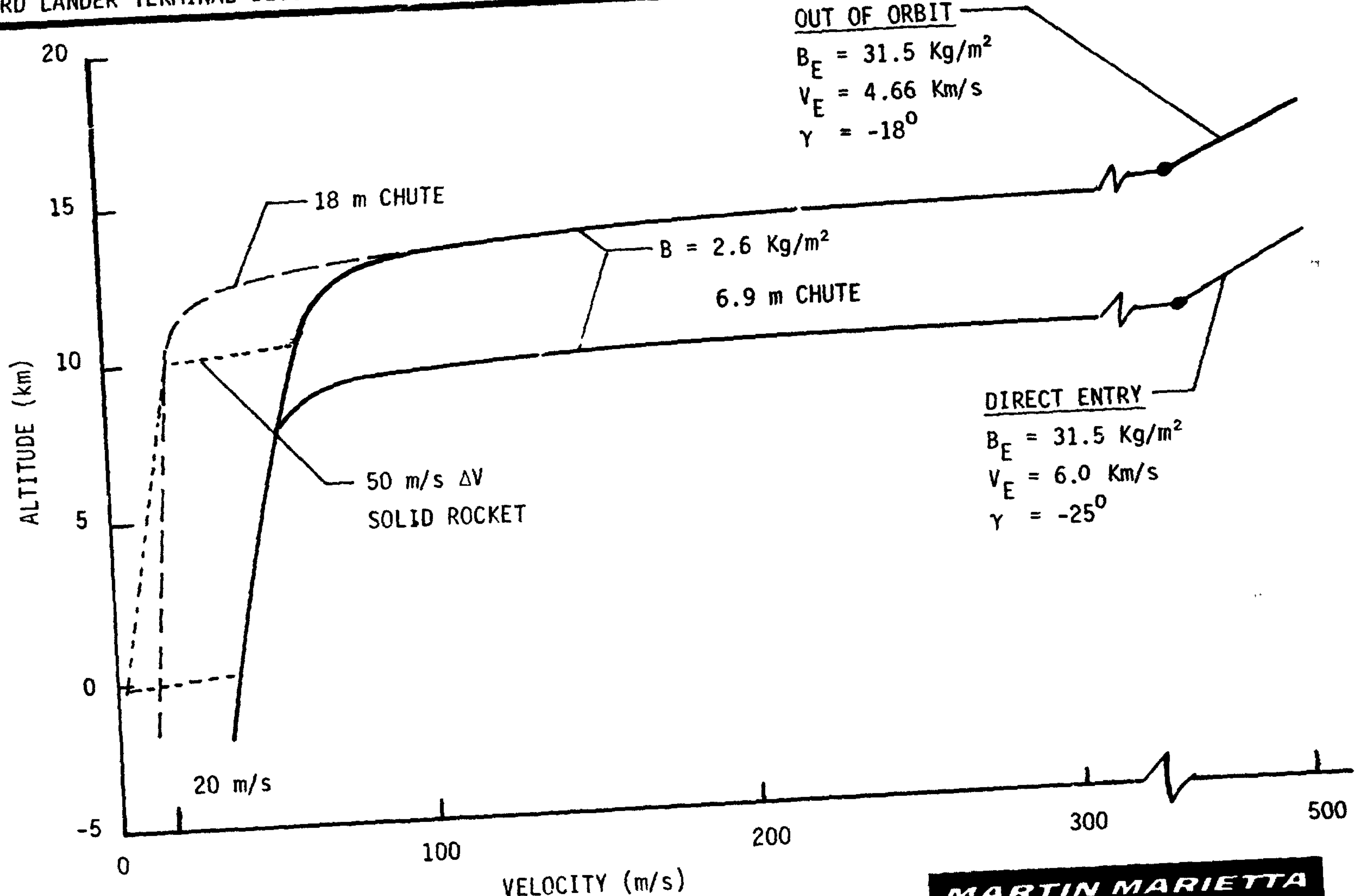
2. Entry and Descent

Entry was from direct approach at a velocity of 6.0 km/sec and a flight path angle of -25^{o}. The entry vehicle, which had a ballistic coefficient of 31.5 kg/m^2, and flew a nonlifting trajectory, decelerated to the chute deployment condition of Mach 2.0 at a velocity of 440 mps at an altitude of 10 km. The chute, which had a ballistic coefficient of 2.6 km/m^2, continued to decelerate the lander and turn the flight path angle until terminal velocity was reached in the vicinity of 6 km above the reference surface. From this altitude on down the velocity exceeded the desired 20 mps by 10 to 15 mps. To remove this excess velocity a solid rocket motor was fired just prior to impact. The ignition signal for the solid rocket motor was provided by a proximity radar. The descent profile is illustrated on Figure III-1.

An 18 m diameter second parachute which provides a terminal velocity of 20 m/s was briefly examined, however this approach was rejected because of its excessive mass of 50 kg.

FIGURE III-1

HARD LANDER TERMINAL DESCENT



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3. Configuration and Mass Properties

The hard lander entry system configuration is shown in Figure III- 2 with the 50 kg hard lander packaged within. A 45° half-angle cone aeroshell was selected because it minimized the diameter of the entry capsule and provided a very stable ballistic entry configuration. Spin-up/de-spin rockets were necessary to provide orientation stability during the trajectory deflection maneuver prior to entry. A 6.9 m diameter parachute, deployed at a Mach number of about 2.0, decelerated the vehicle to about 50 m/s at 7 km altitude. At this condition, either a terminal phase solid rocket or an 18 m diameter parachute could be used to reach the final design impact velocity of 20 m/s. As discussed in the previous section (paragraph III.C.2), the solid rocket was chosen because of its very light mass (<3 kg) as compared to 50 kg for the 18 m parachute.

The terminal phase solid rocket subsystem required a proximity radar altimeter to signal rocket ignition just prior to impact. Although not shown, the solid rocket could be mounted on the parachute harness lines using the same technique that the Russians' have used in the recovery of their manned spacecraft.

Table III-1 defines the hard lander entry system mass breakdown for the direct approach mission mode.

FIGURE III-2

HARD LANDER ENTRY CONFIGURATION

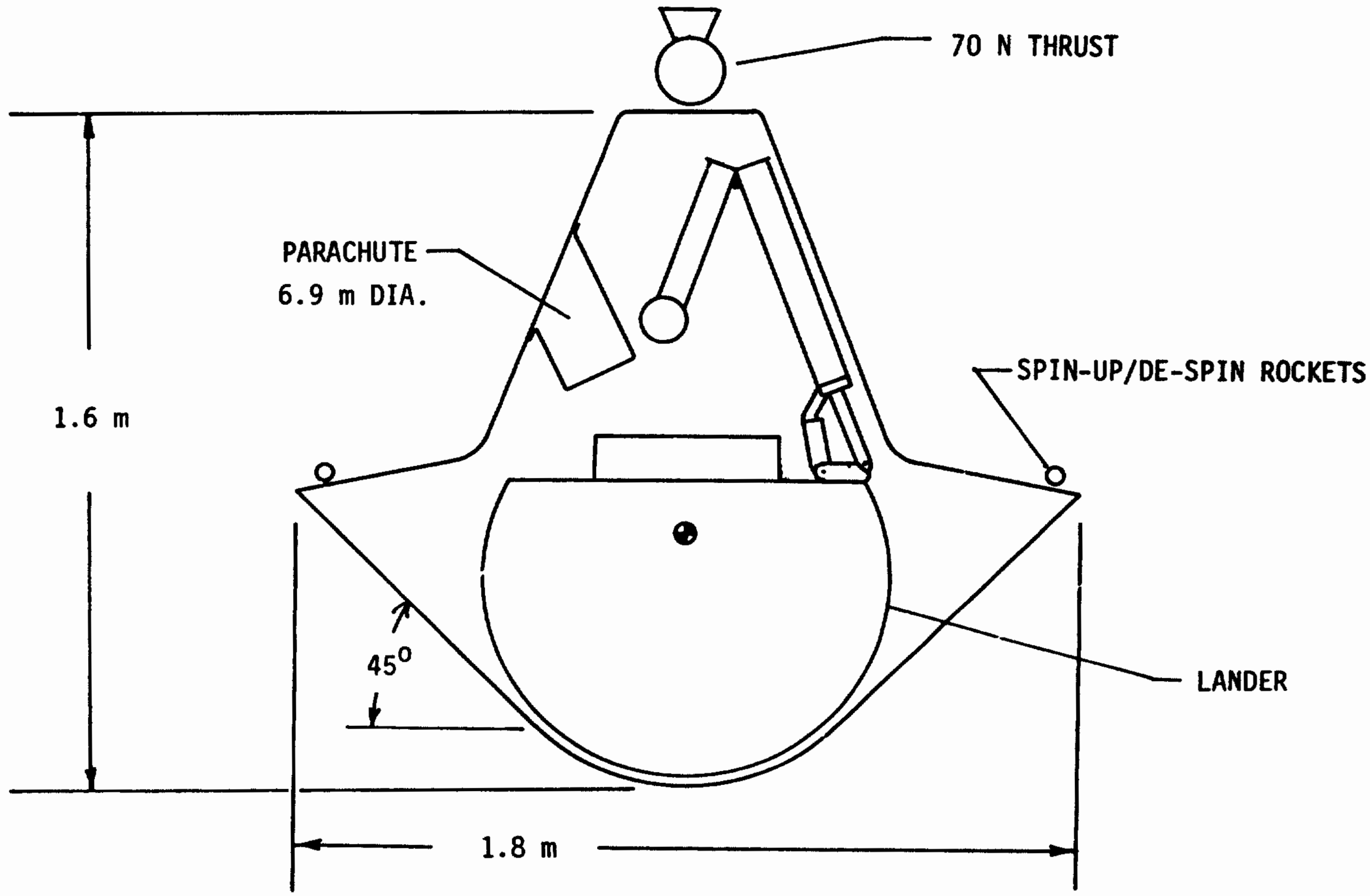


Table III-1 Hard Lander Entry System Mass Breakdown--Direct Entry

		Mass, Kg
Aeroshell		37.5
Structure	27.0	
Heat Shield	9.5	
Spin Rockets/Support	1.0	
Base Cover (A/S + H/S)		13.5
Decelerator		8.0
Parachute	7.5	
Mortar	0.5	
Terminal Phase Rocket and Support		3.0
Radar Altimeter		2.0
Subtotal		64.0
Lander		50.0
ENTRY MASS		114.0

D. CONCLUSIONS - HARD LANDER ENTRY SYSTEM

The following general conclusions were drawn relative to the Mars hard lander entry system evaluated within the ground rules of this task:

- Direct entry system mass of about 65 kg is required in support of a 50 kg lander.
- Either solid rocket or parachute can provide terminal descent and impact below 20 m/s velocity.
- Solid rocket is recommended terminal descent decelerator because of small mass required.
- Solid rocket requires proximity radar altimeter for ignition timing.

IV. TASK 3 - MARS AIRPLANE ENTRY SYSTEM

A. TECHNICAL CONSIDERATIONS

The primary technical concerns for the Mars airplane entry system were weight constraints and volume constraints imposed by the Shuttle payload bay geometry. The airplane stowed configuration characteristics and mass goal of 300 kg were initially specified by JPL. Three entry capsules with one airplane in each could be packaged together within the Shuttle payload diameter constraint of 4.27 m (14 ft) and, by stacking an additional cluster of three plus a single capsule on the end, a total of seven airplane/entry capsules could be packaged within a 9.1 m (30 ft) length. To accommodate this arrangement, the Viking type entry capsule was constrained to 3.81 m (12.5 ft) diameter.

A design goal of the study was to minimize the entry capsule/airplane mass by combining entry functions and hardware into the airplane system wherever practical. Such items as guidance, ACS control, and sequencing could use common computer equipment located on the airplane. These subsystems as well as a common power source were briefly evaluated in an attempt to minimize the combination system mass.

B. OBJECTIVE OF TASK

The objective of this task was to establish an entry system for delivering the Mars airplane concept defined in Reference 6. The airplane package (stowed configuration) size and mass were defined by JPL, and the required conditions (altitude and velocity) for deployment of the airplane were established by JPL. As part of this task the staging altitudes and decelerator configurations for second and third stage decelerators were to be evolved.

C. ANALYSIS

1. Guidelines and Ground Rules

The guidelines and ground rules for the design of the Mars airplane entry system are as follows:

- Entry from 500 km by 24-hour orbit - similar to "Mars '84" mission plan.
- Perform Viking-type entry, i.e., ACS system used for deorbit, entry orientation and to maintain $L/D \approx 0.2$ during atmospheric entry.
- Slow payload to 60 m/s at $\geq$ 7.5 km altitude (relative to areoid).
- Airplane released in stowed configuration - unfolds in free fall phase.
- Airplane computer controls entry and deployment sequence.
- If practical, airplane power system provides all entry system power requirements.
- Airplane mass is approximately 300 kg's.

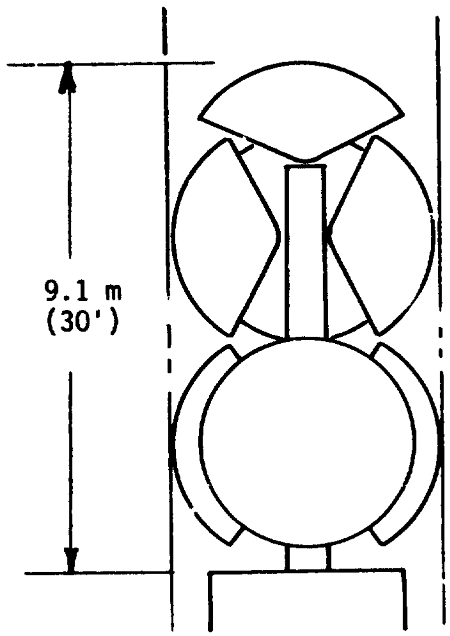
2. Design Loads

The design loads that will be experienced by the airplane/entry capsule combination are shown in Figure IV-1, along with the possible installation arrangement within the Shuttle bay. The Shuttle launch load factors, presented in the inset table, define a maximum Shuttle longitudinal load factor of 4.2 gs. Since the entry capsule may be installed nose down or on its side, it may experience a 4.2 g launch load on either the longitudinal axis or on a selected lateral axis as depicted in the diagram labled "Launch." During the Mars entry at the steepest entry angle of about -15.5°, the entry capsule will experience a longitudinal deceleration load of 7.14 gs. The lateral loads remain small during entry since it is a nonlifting ballistic entry with anticipated small oscillation angles.

The combination worst case design loads for both launch and entry are shown in the lower diagram with a 7.14 g maximum longitudinal load and a 4.2 g maximum lateral load selected by orientation on the Shuttle to coincide with the airplane fuselage longitudinal axis.

FIGURE IV-1

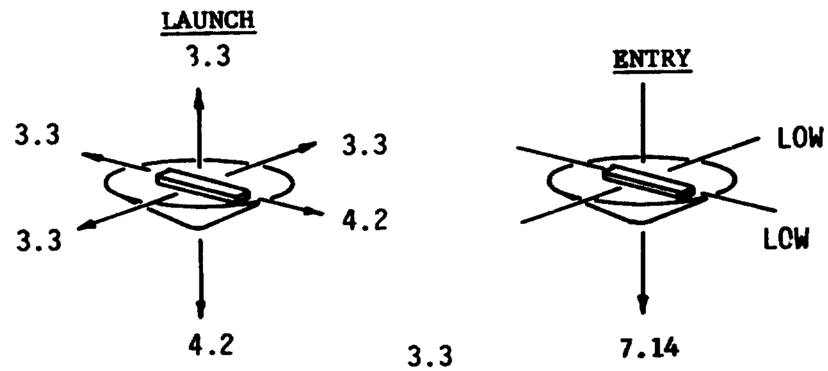
DESIGN LOADS SUMMARY -- LAUNCH AND ENTRY/DESCENT



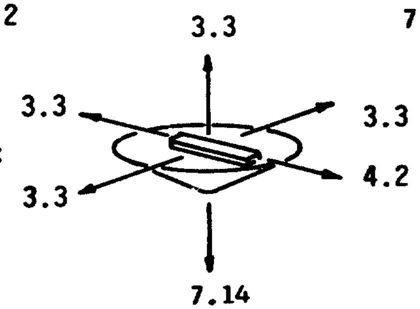
FROM VIKING III REPORT (MOBILE LANDER)

SHUTTLE LAUNCH LOAD FACTORS

LONG.	TRANSVERSE	
X	Y	Z
4.2	1.3	3.3
	1.7	2.0



∴ DESIGN FOR:



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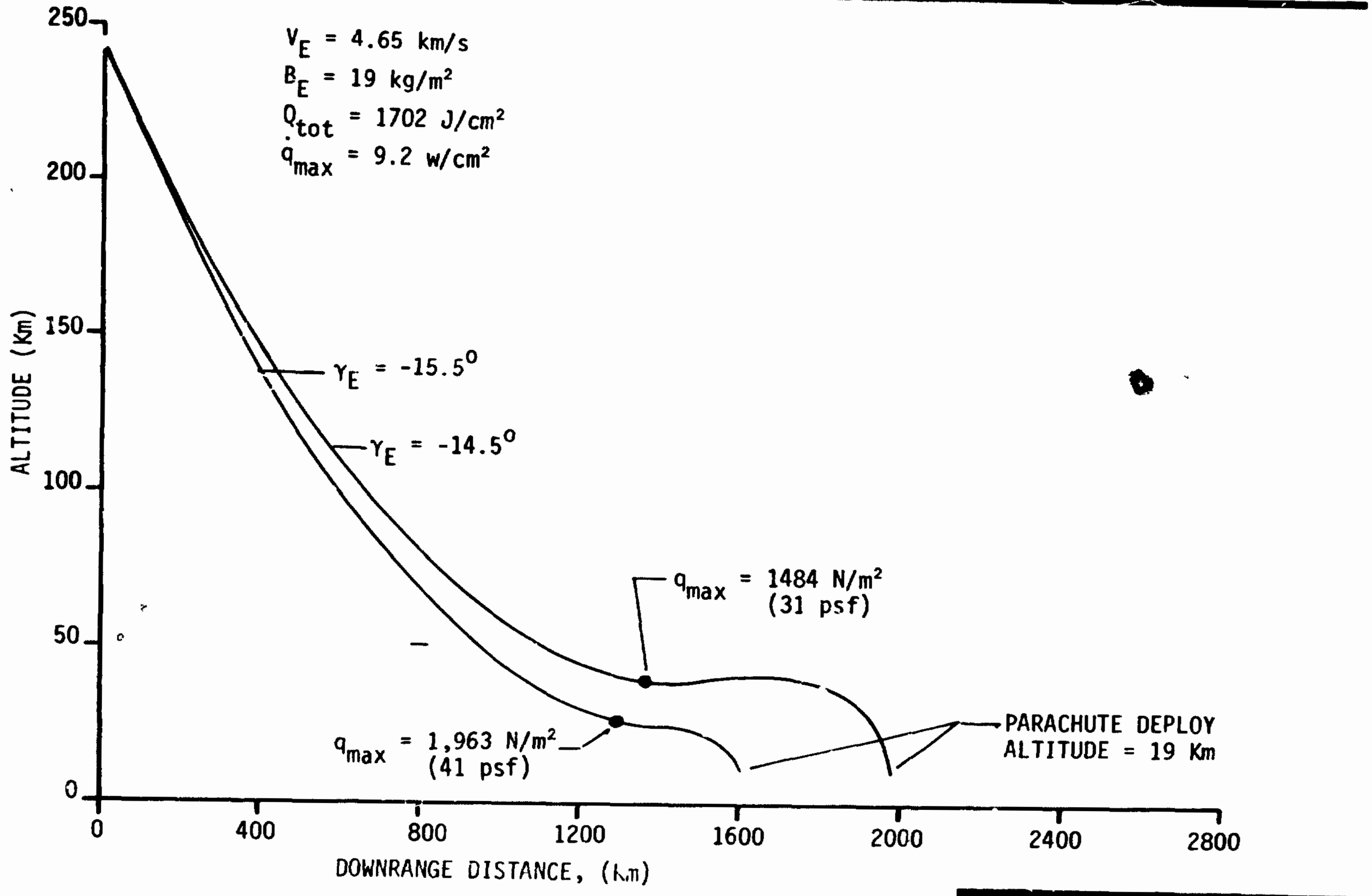
3. Entry and Descent Trajectories

Entry for the Mars airplane delivery system was from a 500 km altitude-24-hour period orbit. Entry velocity and flight path angle at 243 km altitude was 4.65 km/sec and 14.5° to 15.5° respectively. The 1.0° flight path uncertainty is compatible with uncertainties associated with entry from orbit. Entry aeroshell phase was ballistic (non lifting) down to an altitude of approximately 19 km where a 14.5 m chute was deployed at Mach 2.0. A few seconds later the aeroshell was staged and the vehicle continues to decelerate and turn the flight path angle. At 7.5 km altitude the vehicle had slowed to 60 mps and was descending vertically. Entry environment and altitude characteristics are shown in Figures IV-5 and IV-6.

Figure IV-2
ATMOSPHERIC ENTRY PROFILE - MARS AIRPLANE



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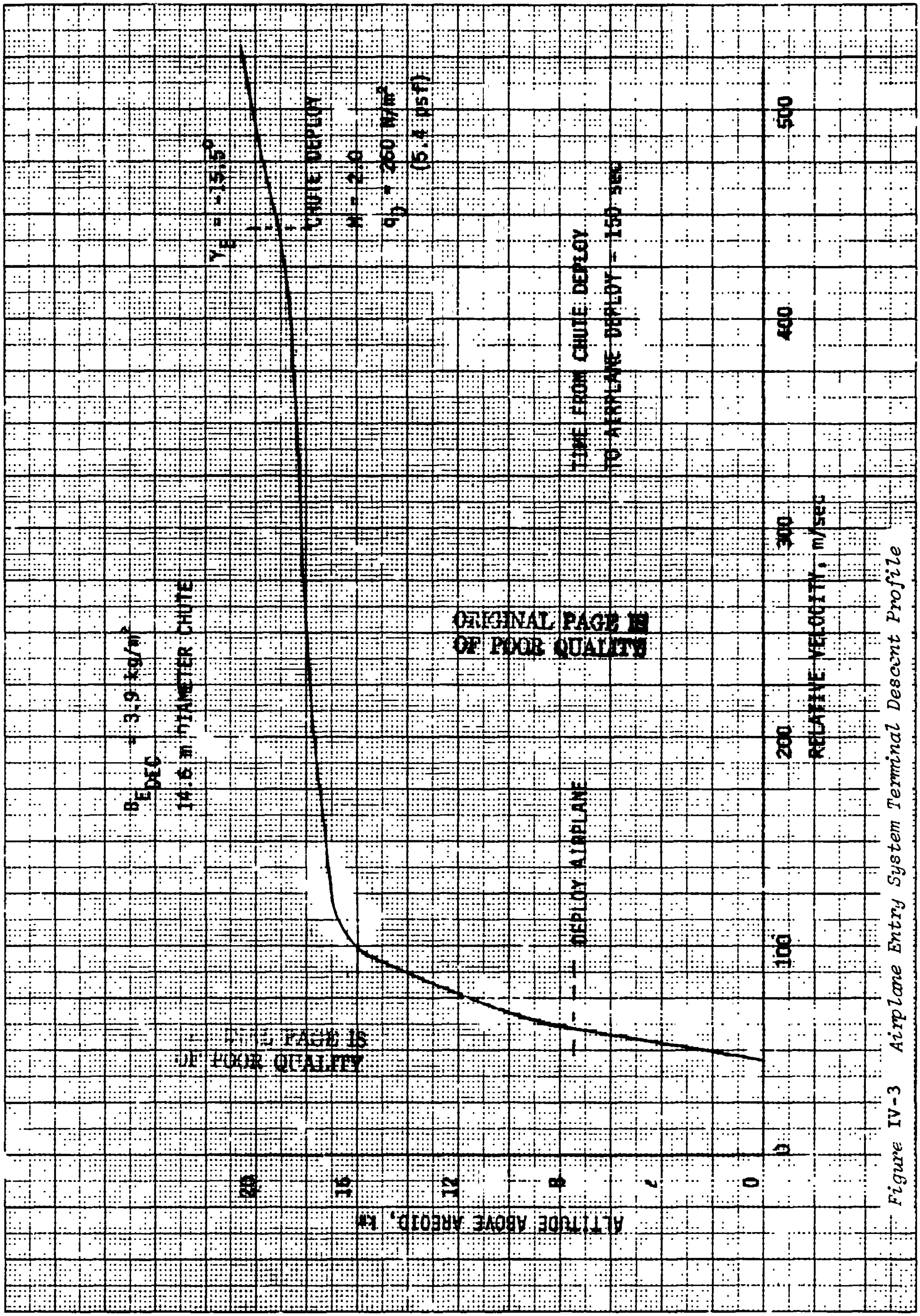


Figure IV-3 Airplane Entry System Terminal Descent Profile

4. Configuration and Mass Properties

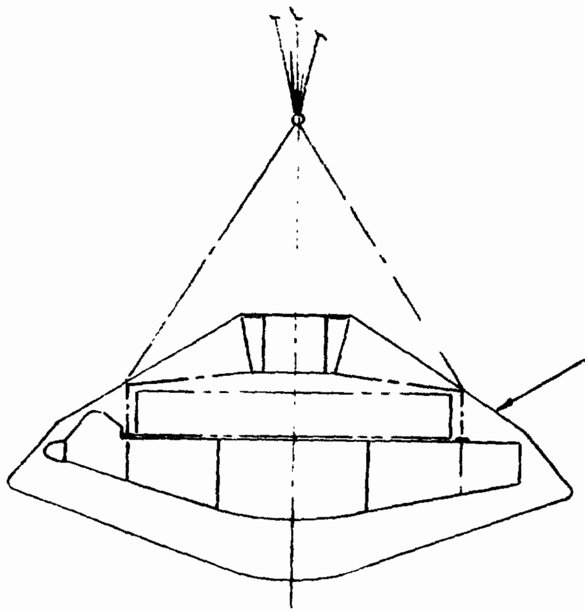
The entry system for the Mars airplane concept was required to provide initial entry protection with additional deceleration stages to slow the vehicle down to 60 m/s at or above 7.5 km altitude. As discussed in Section IV.C.3, the main parachute is deployed at a Mach number of 2 and the aeroshell is staged immediately afterward. This configuration, with the airplane exposed at the bottom, continues to decelerate down to the airplane terminal deployment condition. Two approaches to support the airplane and carry the parachute loads into the aeroshell were considered and are shown in Figure IV-4. The configuration on the left supports the airplane from above at two structural hard points which are attached to a base cover truss. The base cover truss also extends to the circumference of the aeroshell, thus carrying the aeroshell loads around the airplane. This configuration maximized the available envelope volume for the airplane tail end stowage, had a clean release of the airplane from the basecover truss, assembly was relatively easy, and the aeroshell could be released immediately after main parachute deployment. A disadvantage of this concept was the relatively heavy basecover truss assembly.

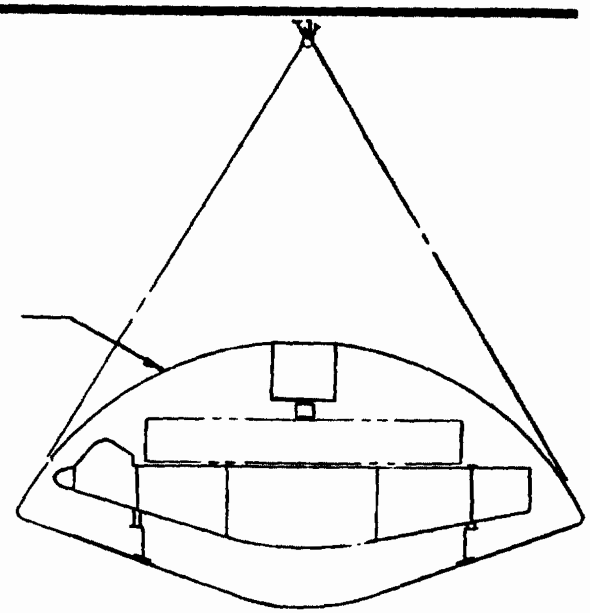
In contrast, the configuration on the right of Figure IV-4 supports the airplane from below and carries its weight through a ring frame into the aeroshell. Its basecover is a light structural shell or aerodynamic fairing. The parachute harness carries loads directly to the outside circumference of the aeroshell, which in turn supports the airplane weight. This concept has the potential for weight savings since it has a relatively light weight basecover. However, compared to the first concept, it presented a more difficult assembly, the airplane support hardware penetrated the available tail stowage envelope, and the airplane must be supported from the top during aeroshell staging.

Based on the advantages and disadvantages discussed above, the configuration on the left with the basecover truss was chosen as the recommended concept for further evaluation.

FIGURE IV-4

AIRPLANE/ENTRY SYSTEM INTEGRATION CONCEPTS





PRO

- MAXIMIZES ENVELOPE FOR TAIL END STOWAGE
- CLEAN RELEASE OF AIRPLANE
- EASE OF ASSEMBLY
- EARLY RELEASE OF AEROSHELL

CON

- WEIGHT OF BASECOVER ASSEMBLY

PRO

- POTENTIAL FOR WEIGHT SAVINGS

CON

- DIFFICULT ASSEMBLY
- AIRPLANE SUPPORTS PENETRATE TAIL STOWAGE ENVELOPE
- DIFFICULT RELEASE OF AIRPLANE

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For a more detailed definition of the selected configuration, the entry system functions were compared to the airplane system functions in order to identify common elements. Table IV-1 presents the entry system hardware, fun^tions, and key requirements for the Viking '75 entry vehicle which should be similar to those of the Mars airplane entry system. Table IV-2 defines the compatibility of the airplane and entry capsule systems, and based on this data, the entry capsule will require an ACS system with tanks, a separate radar altimeter antenna, and a separate remotely activated Ag Zn battery.

Figure IV-5 shows a more detailed inboard profile of the airplane/entry capsule integration concept and Figure IV- 6 illustrates the plan view of the basecover truss relative to the airplane wing envelope. The ACS require two propellant tanks with a nominal inside diameter of 29.2 cm (11.5 in.). Off-the-shelf tanks with an I.D. of 32.7 cm (12.88 in.) manufactured by Propulsion Systems, Inc. have flown on the Canadian Technology Satellite and these tanks weigh 2.5 kg each. The tank mass includes a pedestal mount which, when replaced by a conventional bracket, could probably reduce the mass somewhat.

Table IV-3 presents a mass breakdown for the Mars airplane entry system. For an airplane mass of 300 kg and an aeroshell diameter of 3.8 m (12.5 ft) the entry system mass was 201.0 kg and the total airplane/entry capsule mass was 501.0 kg. The aeroshell design used Viking-type aluminum skin-stringer construction and Viking heat shield ablative material. A potential savings in aeroshell mass may be possible by usir. ~raphite epoxy composite materials.

TABLE IV-1

VIKING LANDER ENTRY SUBSYSTEMS OPERATIONAL HARDWARE

FUNCTION	KEY REQUIREMENTS
COMPUTER	GCSC - 18K MEMORY, 8μ SEC ADD, 123μ SEC DIVIDE
GYROS	IRU - 100 DEG/SEC MEASUREMENT, 0.3 DEG/HR RANDOM DRIFT
ACCELEROMETERS	IRU - 50 μG RANDOM BIAS
ACS ENGINE DRIVE	VDA - 12 RCS ENGINES
RADAR ELECTRONICS	RA - 1 GHz, 135 W, 450K FT to 135 FT
COMMUNICATIONS	UHF - 1W, 10W, 4 KBPS
ELECTRICAL POWER	3 BATTERIES, 24 AH (1 BATTERY OUT DESIGN)

MARTIN MARIETTA

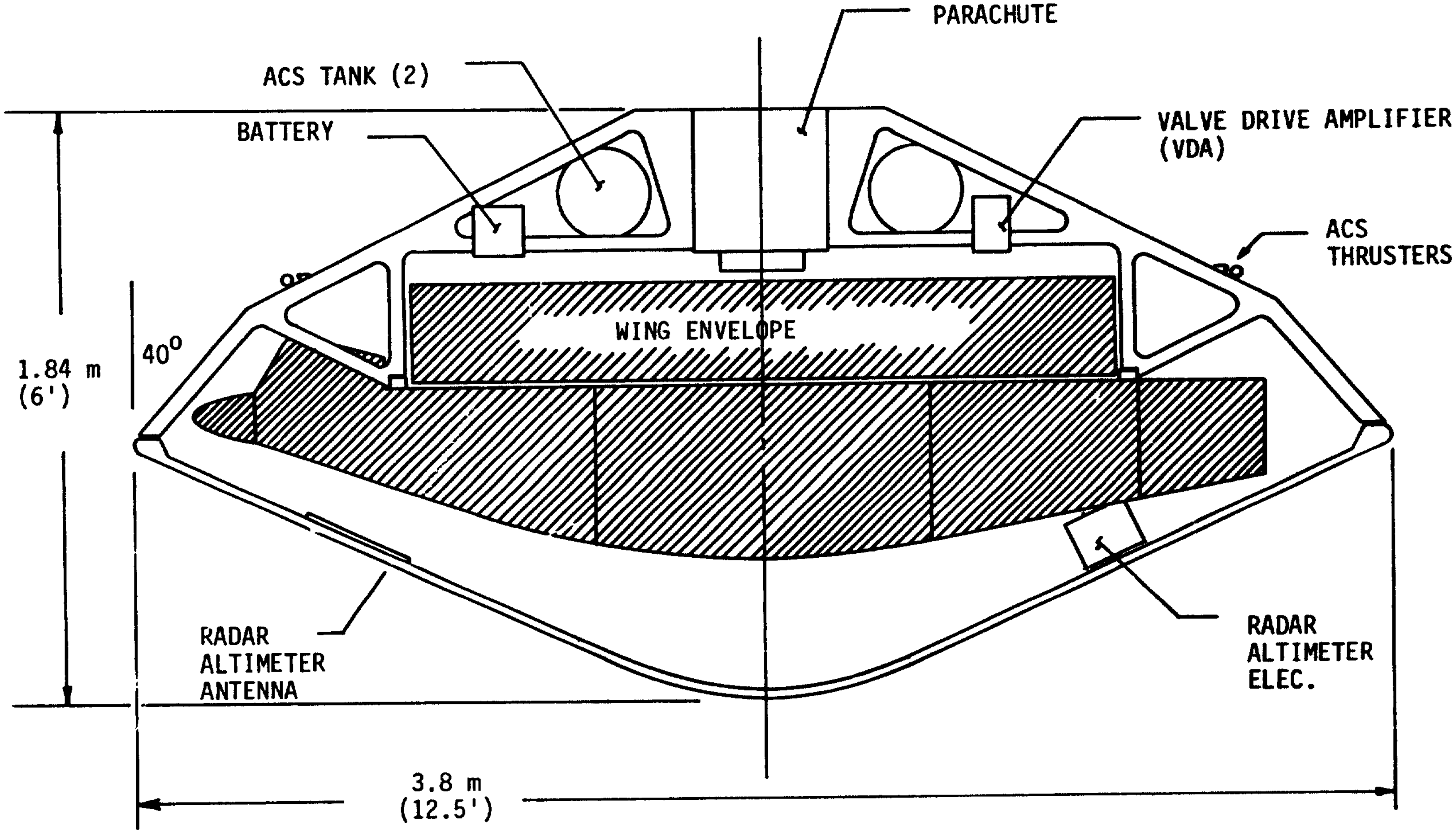
TABLE IV-2

AIRPLANE/ENTRY SYSTEM COMPATIBILITY

FUNCTION	A/P	E/S
COMPUTER	YES	N/R
IRU	YES	N/R
RADAR ELECTRONICS	YES	RA (1 ONLY)
RADAR ANTENNA	YES	A/S ANTENNA
RCS ELECTRONICS DRIVE	YES	VDA (MODIFIED)
ORDNANCE FIRING	YES	N/R
POWER	YES	REMOTELY ACTUATED AgZn (15 AH)
TELEMETRY	YES	SENSORS
UHF COMMUNICATIONS	YES	N/R

FIGURE IV-5

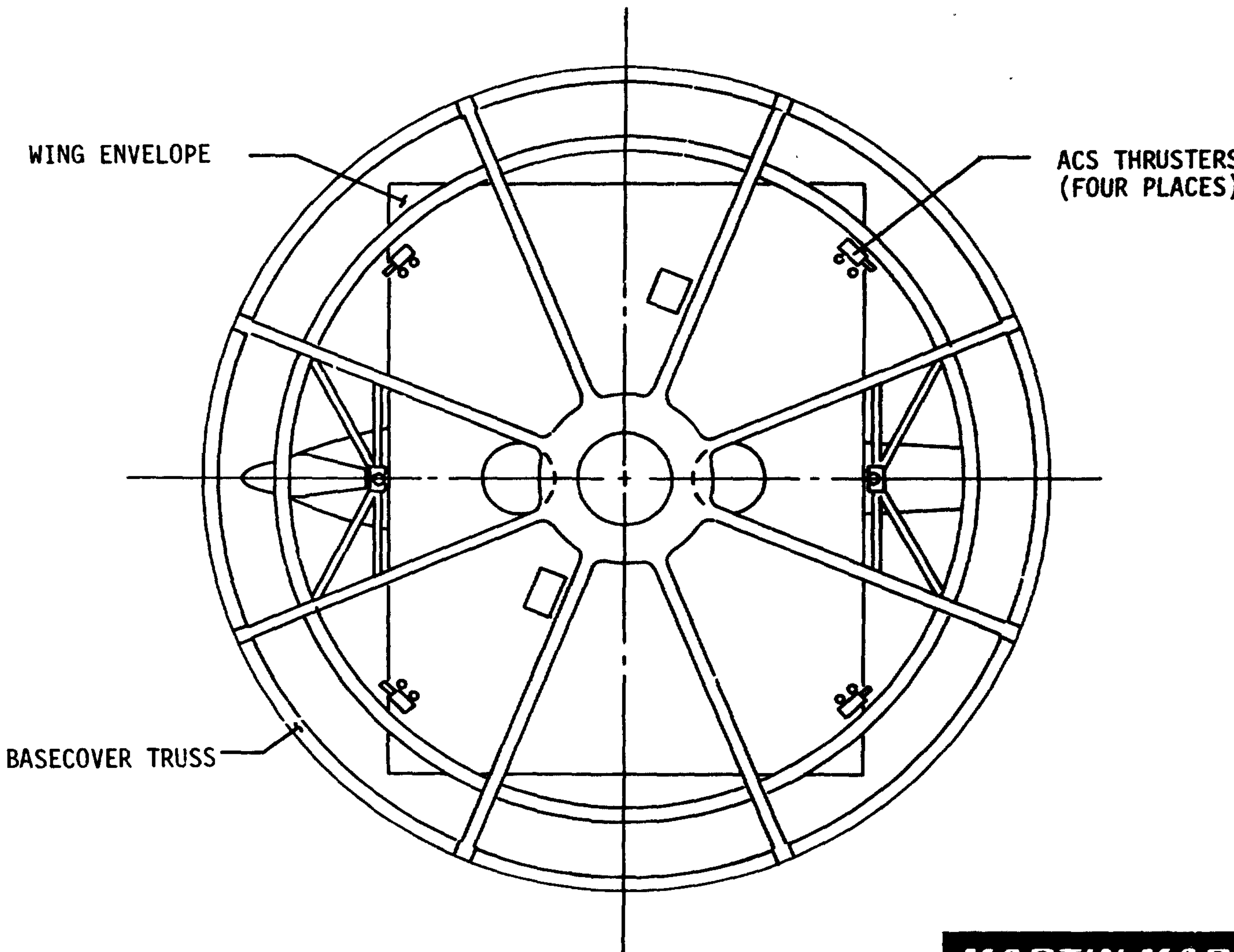
AIRPLANE/AEROSHELL INTEGRATION



MARTIN MARIETTA

FIGURE IV-6

AIRPLANE/BASECOVER INTEGRATION



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MASS STATEMENT -- OUT OF ORBIT ENTRY SYSTEM

ASSUMPTIONS:
- AIRPLANE MASS OF 300 Kg
- $\gamma_E = -14.5^o$ TO -15.5^o
- A/S DIAMETER OF 3.8 m

ITEM	MASS, Kg		COMMENTS
ENTRY SYSTEM AT SEPARATION		201.0	ADD 300 Kg FOR AIRPLANE FOR W_E
DEORBIT + MANEUVER PROPELLANT	12.5		DEORBIT BURN TO ENTER AT γ_E
ENTRY SYSTEM AT ENTRY		188.5	
RATE DAMPING PROPELLANT	0.5		
RELEASE AEROSHELL	82.5		ALUMINUM SKIN-STRINGER CONSTRUCTION, VLC HEAT SHIELD MATERIAL
ENTRY SYSTEM ON PARACHUTE		105.5	
PARACHUTE SYSTEM	27.9		
BASE COVER STRUCTURE	28.1		
PARACHUTE TRUSS + MORTAR ASSEMBLY	16.1		
RCS HARDWARE + UNUSABLE CONSUM.	10.6		
RA + AA + VDA + BATTERY + PYRO DEVICES	16.4		VLC RADAR ALTIMETER ELECT. + ANTENNA, MODIFIED VALVE DRIVE AMPLIFIER, FOUR EXPLOSIVE NUTS, TWO PIN PULLERS
CABLING + STAGING CONNECTOR	6.4		ELECTRICALLY ACTUATED STAGING CONNECTOR

D. CONCLUSIONS - MARS AIRPLANE ENTRY SYSTEM

The following general conclusions were drawn relative to the Mars airplane entry system evaluated within the ground rules of this task:

- Base cover truss supported airplane is preferred concept.
- 200 kg approach from orbit entry system will support 300 kg airplane entry and provide proper deployment.
- Shuttle payload envelope is compatible with launch of four to seven airplane entry systems depending on support spacecraft size.
- Entry command and control functions can be incorporated into the airplane.
- Entry system must retain radar altimeter, RCS subsystem, valve drive circuitry, and supplemental battery power.

V. REFERENCES

-1 JPL Presentation Data, Mission Analysis, by Russ Nagorski, Dated April 13, 1978.

-2 "Mars '84 Landing System Definition", MMC Report MCR-77-78, April 1977.

-3 "Automated Mars Surface Sample Return Mission Concepts", NASA Technical Memorandum, NASA TM X-3184, June 1975.

-4 JPL Memo, Large Mars Lander Guided Entry Trajectories, by M. I. Cruz, dated February 6, 1978.

-5 NASA CR-66663, Final Report Study of Direct vs Orbital Entry for Mars Missions, Volume V, Appendix C - Entry Configuration Analysis, August 1968.

-6 "Mars Airplane Presentation Material", JPL Report No. 760-198, November 28, 1977.

-7 "Alternate Planetary Lander Study Report", JPL Report No. 760-149, July 27, 1976.

-8 "Advanced Conceptual Study of an Alternate Hard Lander", Martin-Marietta Corporation Report No. MCR-76-368, July 1976.

Interoffice Memo

Date: 1 February 1978

Refer To: 78-APP-003

To: S. J. Ducsai, A. J. Butts, R. E. Frank, J. R. Mellin, L. D. Friedman (JPL), J. R. French (JPL), R. P. Nagorski (JPL)

From: C. E. French

Subject: Landed Weight on Mars

Reference:
1) Mars '84 Landing System Definition MMC Report MCR-77-78, April 1977.
2) Automated Mars Surface Sample Return Mission Concepts, NASA Technical Memorandum, NASA TM X-3184, June 1975.

- -

A brief study was conducted to determine the maximum mass which can be landed on the surface of Mars using Viking system concepts. Two methods of extending the Viking capability were investigated. Concept 1 is consistent with the Viking 75 design entry environment limits of dynamic pressure, heating rate and heat load. The landed weight performance was assessed as a function of aeroshell diameter, with a consistent increase in parachute diameter and in terminal phase propulsion thrust. These results are an extension of the work presented in Reference 1. In addition, Concept 2 allows increased entry dynamic pressure, heat environment and a small increase in parachute deployment conditions. Parachute deployment is by pilot chute rather than by mortar. This analysis is based on the work reported in Reference 2.

Study results for Concept 1 and 2 are presented on Figure 1. Landed dry mass is plotted as a function of mass at entry, for aeroshell diameters from 11.5 to 15.0 ft, and for three landing site terrain heights. It can be seen that Viking designs, or residuals, i.e. 11.5 ft aeroshell, and a landing site terrain height of 1.5 km, would result in a landed dry mass of 790 kg (1742 lbs). By increasing entry and descent vehicle sizes, and also environment severity, a range of landed dry masses up to 2400 kg (5291 lbs) could be attained.

For Concept 1, the following constraints and assumptions were made.

1. Entry is from Mars orbit.

2. Viking entry flight path corridor is maintained (-15.9° to -17.6°).

3. Entry ballistic coefficient ($0.50\ sl/ft^2$) is held constant as A/S diameter is increased.

APPENDIX A--LANDED WEIGHT ON MARS - OUT OF ORBIT ENTRY

This appendix contains the memorandum which served as the basis for the landed performance and weight estimates for Cases 1, 2, and 3.

4. Viking parachute deploy conditions and ballistic coefficient maintained.

5. Terrain height of 1.5 km assumes 1.5 km uncertainty for a site at MSL. This is 1/2 the Viking design uncertainty.

6. Terminal phase propulsion system is regulated to a constant pressure feed. T/W ratio is held constant as landed weight is increased.

For Concept 2, based on the analysis presented in Reference 2, the following assumptions and qualifications are made.

1. Entry is from Mars orbit.
2. Ballistic coefficient is increased (0.92 sl/ft^2).
3. Lift to Drag ratio is increased (0.28).
4. Entry flight path corridor is steepened (-20° to -22°).
5. Aeroshell structural weight is increased to compensate for the higher dynamic pressure.
6. Heat shield weight (and material) is changed for the more severe heating environment.
7. Terminal phase T/W is increased.

A comparison of the two concepts at the basic 11.5 ft aeroshell diameter is shown on Table I, attached. Table II presents the weight variation of major subsystems as aeroshell diameter is increased.

The landed weights achieved by Concept 2 are felt to represent the upper limit of those achievable with a Viking system concept for the following reasons:

1. Aeroshell diameter is limited by Space Shuttle bay constraints.
2. L/D ratio is constrained by maximum trim angle of attack of approximately 20°.
3. Ballistic coefficient is constrained by parachute deployment conditions.
4. The analysis of Reference 2 did not allow for dispersions in the atmosphere model and assumed a wind profile with zero velocity at the surface.

C. E. French

Enclosure

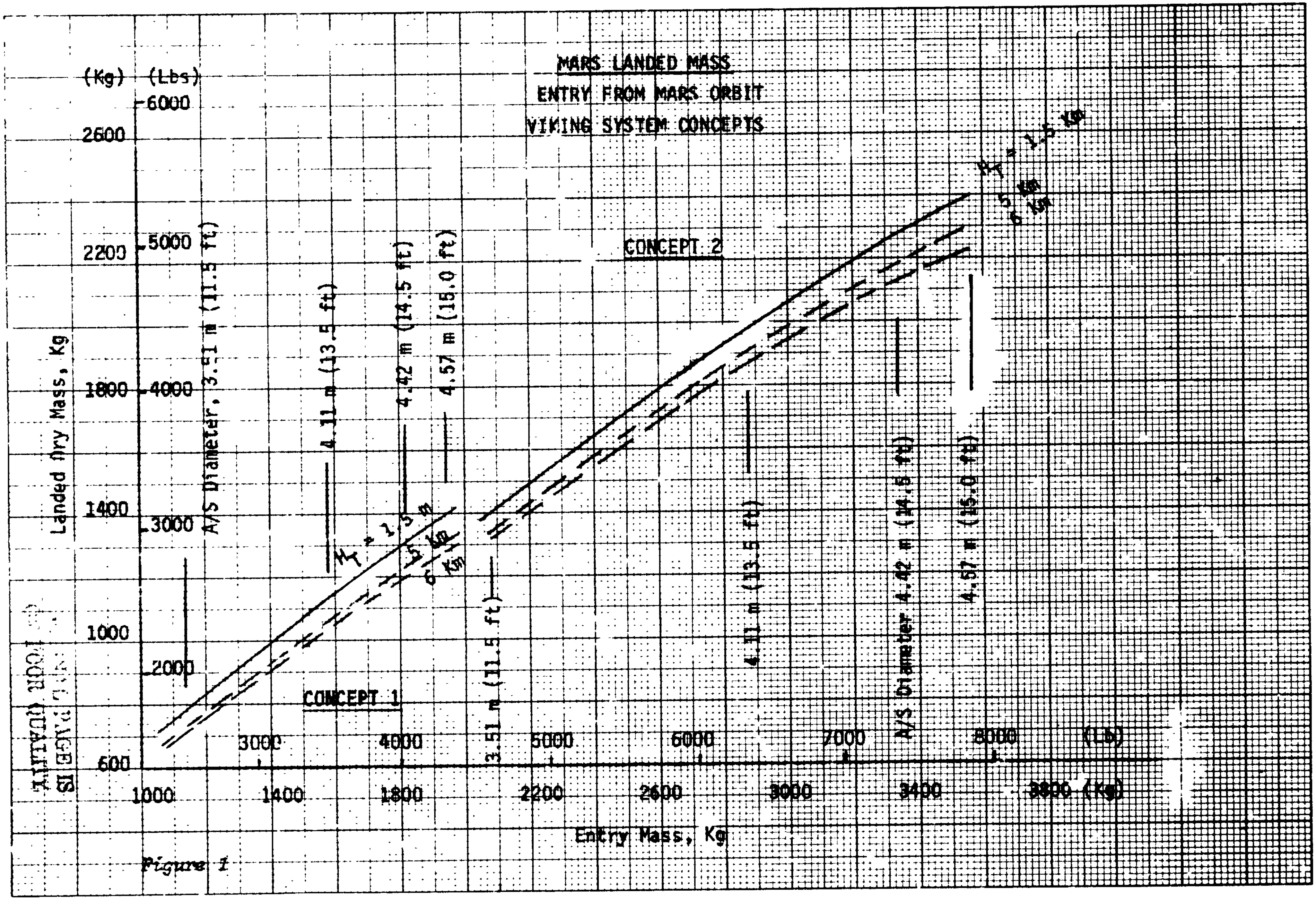

Figure 1

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TABLE I

MISSION CONCEPT COMPARISON

	CONCEPT 1	CONCEPT 2
Entry Weight, lb	2525	4600
A/S Diameter, ft	11.5	11.5
L/D	0.18	0.28
γ_E, Corridor, deg	$-15.9 \rightarrow -17.6$	$-20 \rightarrow -22^{o}$
V_E, ft/sec	15,275	15,091
B_E, sl/ft^2	0.50	0.92
Aeroshell:		
Structure	207	373**
Thermal Protection	56	117
Propulsion inerts & residual	63	154
q_{max}, psf	144	334
$\dot{q}_{max}$, BTU/ft^2-sec	26*	53
Q_{total}, BTU/ft^2	1510*	1762
<u>Weight at Chute Deploy, lb</u>	2131	3938
Parachute + Backface		
Chute Diameter, ft	53	75
B_{E_D} sl/ft^2	.049	.047
Chute Structure, lb	83	269
Mortar + misc	41	11
B/F Struct	101	146
Heat Protection	-	66

*Design Values

**This weight is increased by 140 lbs over that of Reference 2

TABLE I (Continued)

MISSION COMPARISON

	CONCEPT 1	CONCEPT 2
Therminal Phase:		
Descent Propellant, lb	132	315
Landed Weight, Dry	1732	3109
$W/L / W_E$	.686	.708
Propulsion Inerts + Residuals	137	243
Landed Less Propulsion	1595 lb	2867 lb

TABLE II

INCREASED AEROSHELL DIAMETER WEIGHTS

Concept 1			
A/S Diameter, ft	11.5	13.5	14.5
A/S Area, sq ft	103.9	143.1	165.1
W_E , lb	2525	3479	4013
A/S Weight, lb (Structure & Heat Shield)	263.5	363.1	418.9
Chute Diameter, ft	53	63	67
Chute Area, sq ft	2210	3142	3572
Chute Weight, lb	124	159	175
Backface Weight	115.8	301	332
Terminal Phase Propellant:			
Vel Contour Fuel, lb	132	187	219
Concept 2			
A/S Diameter, ft	11.5	13.5	14.5
W_E , lb	4600	6344	7349
A/S Weight, lb (Structure + Heat Shield)	490*	675	779
Chute Diameter, ft	75	90	95
Chute Area, sq ft	4417	6280	7139
Chute Weight, lb	300	427	485
Backface Weight	212	293	339
Terminal Phase Propellant:			
Vel Contour Fuel, lb	315	444	501

*This weight is increased by 140 lbs over that of Reference 2

APPENDIX B -- VEHICLE MASS SCALING FACTORS

Major entry vehicle subsystem mass estimates for MSR are based on actual Viking hardware masses. Scaling factors used in estimating mass properties for MSR are described below.

Aeroshell Structure:

$$W_{STR_{MRS}} = W_{VIK}\ (K) \left(\frac{P_{S_{MRS}}}{P_{S_{VIK}}}\right)^{0.33} \left(\frac{AREA_{MRS}}{AREA_{VIK}}\right)$$

Where:

W_{VIK} = 86.2 Kg, Viking aeroshell structural mass;

K = 1.1, factor to account for increased diameter;

P_S = Surface pressure at maximum dynamic pressure;

AREA = Projected aeroshell area.

For cases 6, 7, and 8 extended flap weights were estimated by use of Report NASA CR-66663, Reference 5 , which presents flap weight as a function of surface pressure and total aeroshell (flaps extended) radius. Flap weights are added to the fixed aeroshell weight, which is estimated as above, to give total aeroshell structural mass.

Aeroshell Heat Shield:

Heat shield weights for Cases 4 and 5 are calculated as discussed in Section II.C.5 in the main report. Heat shield weights for Case 6 with extended flaps are estimated by:

$$W_{HS} = W_{HS_{CASE\ 5}} \left(\frac{AREA}{AREA_{CASE\ 5}}\right) \left(\frac{TOTAL\ HEAT}{TOTAL\ HEAT_{CASE\ 5}}\right) .$$

As cases 7 and 8 have the same entry heating environment their heat shield weights are ratioed by area (flaps + fixed aeroshell).

Base Cover Structure:

Base cover structural weights are ratioed on the basis of surface area to the weight presented in the LRC study, Reference 2. The LRC structural mass is 66 Kg.

Base Cover Heat Shield:

It is assumed the heat shield material on the base cover cannot be applied at thicknesses less than 0.1 m. Heat shield weight is therefore a function of surface area only.

Parachute System Mass:

Parachute area is sized by assuming the same ballistic coefficient on the chute (10.44 Kg/m^2) as optimized in the LRC study, Reference 2. Parachute system weight is estimated as follows.

$$\text{Canopy Weight} = \text{VIKING}_{WT} \left(\frac{\text{AREA}_{MSR}}{\text{AREA}_{VIK}}\right)\left(\frac{q_{D_{MRS}}}{q_{D_{VIK}}}\right)$$

$$\text{Lines} = \text{VIKING} \left(\frac{\text{A/S DIAM}_{MSR}}{\text{A/S DIAM}_{VIK}}\right)\left(\frac{q_{D_{MRS}}}{q_{D_{VIK}}}\right)$$

$$\text{Swivel} = \text{VIKING} \left(\frac{q_{D_{MRS}}}{q_{D_{VIK}}}\right)$$

Drogue Diam. 2X A/S diameter = 11 Kg (Based on Pioneer Venus Chute)

Mortar + Truss = 5 Kg

Total Drogue System, 16 Kg

APPENDIX C - CONSTANT L/D RATIO PARAMETRIC DATA

This appendix presents parametric entry characteristic curves and parachute deployment conditions as a function of entry flight path angle, ballistic coefficient, and L/D ratio. These data are for constant L/D ratio (not roll modulated) and are for a concept on which analysis was terminated early in the study.

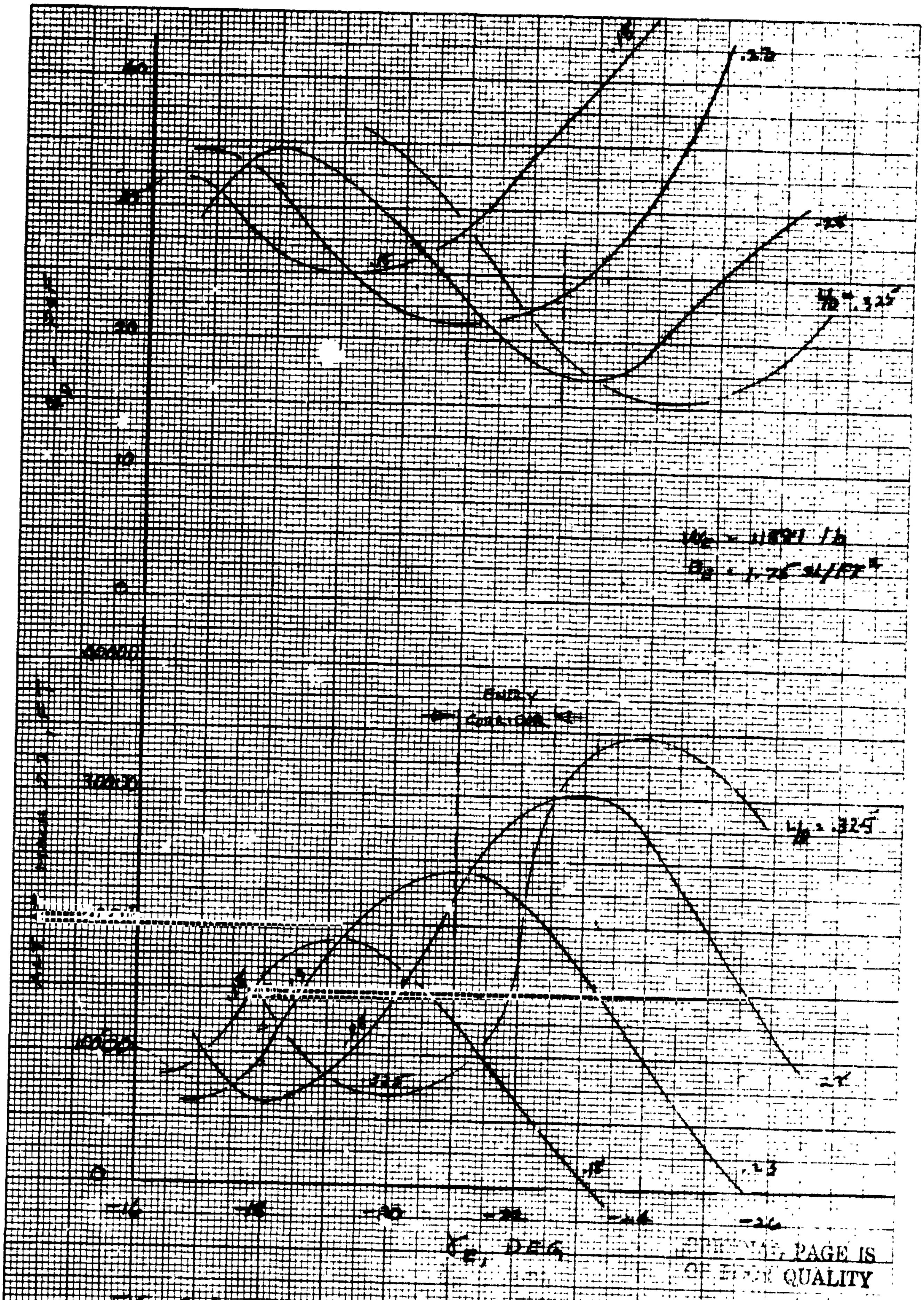

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FIG. C-1 ALTITUDE AND q_D AT M 2.2 VS γ_E & L/D RATIO

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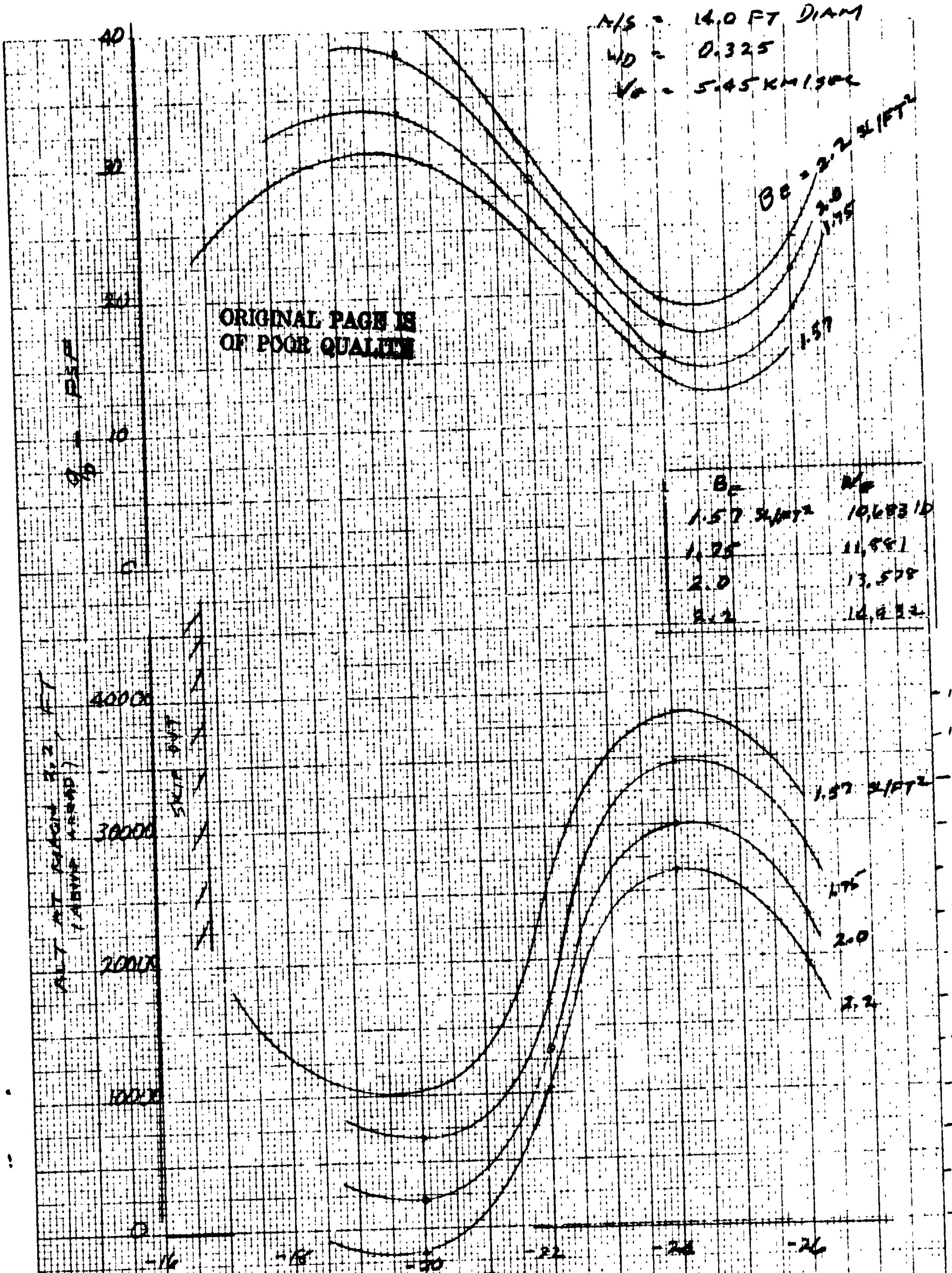

FIG C-2 ALTITUDE AND q/D AT M 2.2 VS δE & BE

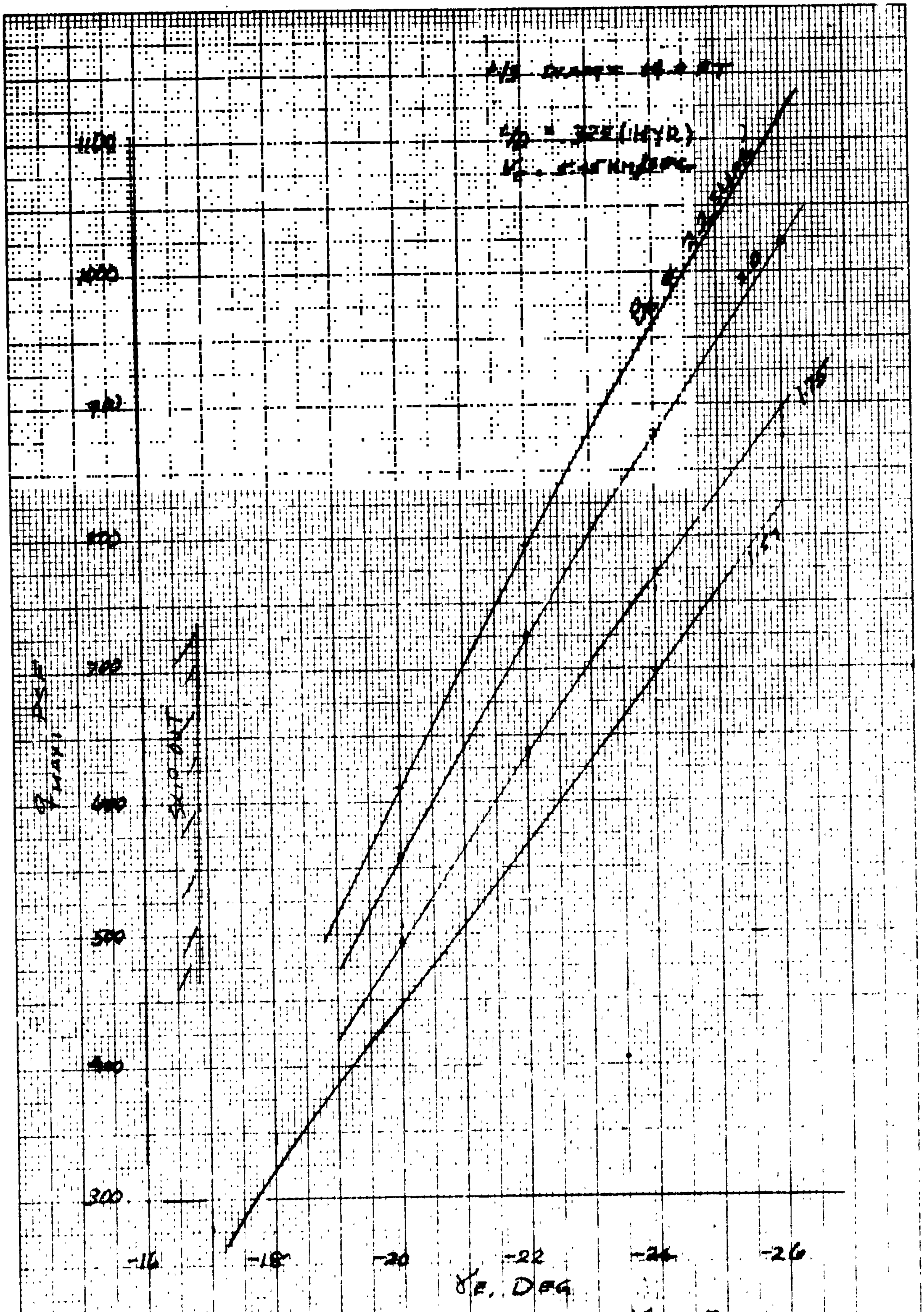

FIG C-3 MAX. DYNAMIC PRESS VS γ_E & B_E

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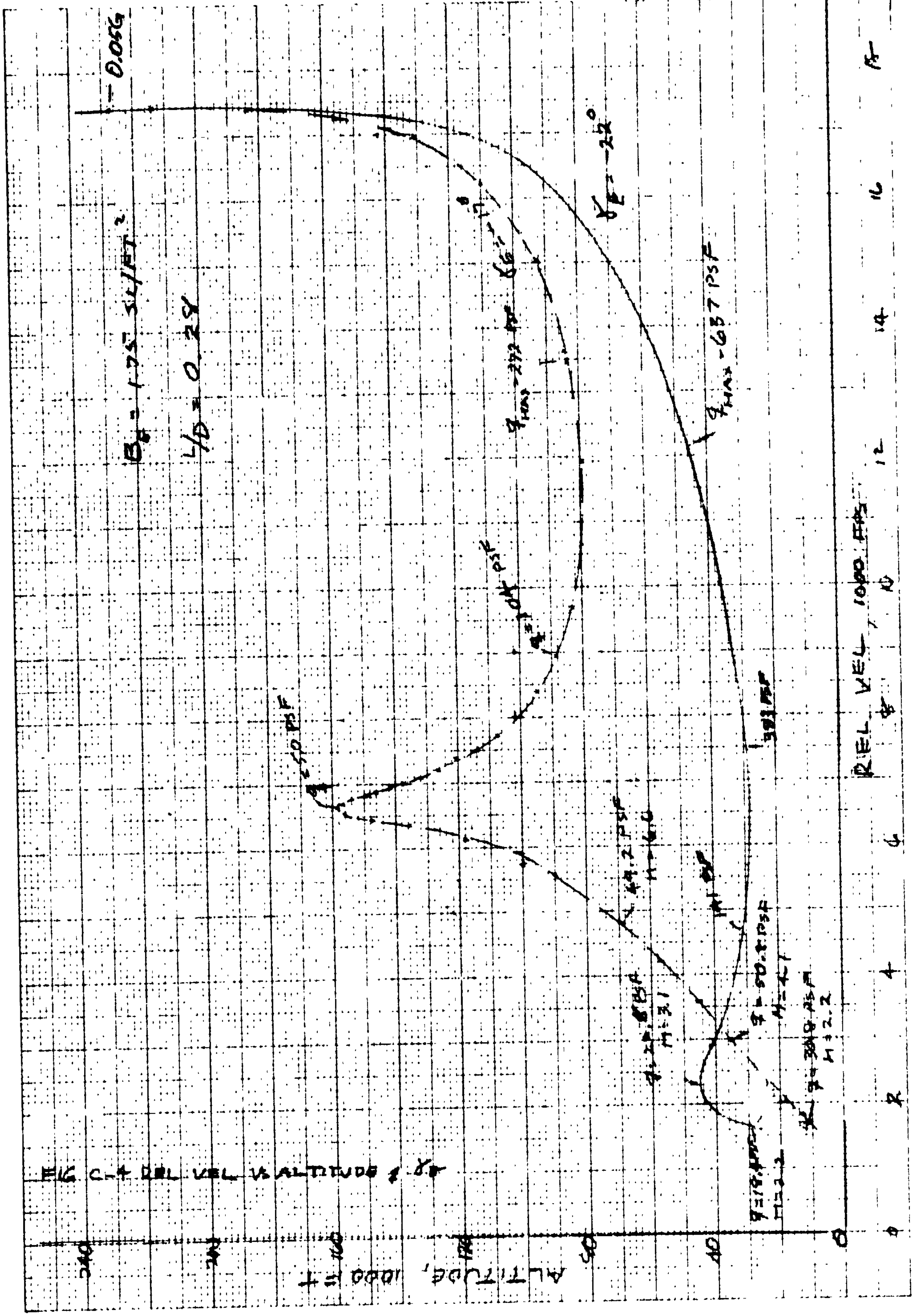

FIG C-4 REL VEL VS ALTITUDE & γ_E