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MARS HARD LANDER CAPSULE STUDY

Volume III Capsule Parametric Study

Prepared by

GENERAL ELECTRIC
RE-ENTRY SYSTEMS

for Langley Research Center



31 JULY 1968

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION-WASHINGTON, D. C.

MARS HARD LANDER CAPSULE STUDY

Volume III
(Book #1)

CAPSULE PARAMETRIC STUDY

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FOR
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FOREWORD

The Mars "Hard Lander" Study Final Report is divided into four volumes and bound in eight books. The titles of the volumes and a brief description of the contents of each book are presented below.

VOLUME I - SUMMARY (CR-66678-1)

Volume I contains a summary of the study activity, the conclusions reached, and a description of a possible design implementation suggested by the study results.

This study indicates that meaningful scientific payloads of approximately 1500 pounds can be placed on the Mars surface, survive for several months, and transmit more than a hundred million bits of data to Earth.

In addition, the study provided data which shows that a smaller Capsule of 700 to 900 pounds has the ability to transmit approximately 10 million bits of imagery and additional scientific surface data.

VOLUME II - MISSION AND SCIENCE DEFINITION (CR-66678-2)

Volume II contains a description of the 'reference' mission plans, both direct entry and out-of-entry, the mission analyses conducted to define the reference plans, the assumed Mars models considered, and the science definition tasks accomplished to select entry and surface science packages/measurement sequences specifically designed to satisfy LRC's scientific goals.

VOLUME III - CAPSULE PARAMETRIC STUDY (CR-66678-3, -4)

A discussion of the analysis and results derived in determining the Capsule subsystems' design characteristics parametrically is provided for the range of assumed Mars Models and the reference mission plans. The synthesis of these subsystems into complete Capsule systems is presented in terms of Capsule performance, total imagery data obtainable, and surface lifetime.

CR-66678-3 presents the Capsule System Parametric Synthesis and Entry and Retardation Subsystem Studies. CR-66678-4 presents both studies of the Lander and Re-entry Subsystems and Appendices associated with the Parametric Study.

VOLUME IV- CAPSULE POINT DESIGNS AND SUPPORTING ANALYSES (CR-66678-5, -6, -7, -8)

Volume IV contains a presentation of the detailed Capsule 'Point Designs', and their supporting analyses, derived to identify specific hardware approaches, weights, and system configurations; and confirm the correctness of the parametric results. In addition to the Capsule's engineering and design details, the results include development status, probability of success, and constraints imposed on the Orbiter by the Capsule mission.

CR-66678-5 contains a definition of the Capsule Point Design Requirements and descriptions of Point Designs 1 and 2. CR-66678-6 contains descriptions of Point Designs 3 and 4 and CR-66678-7 of Point Designs 5 and 6. CR-66678-8 provides additional information on Impact Attenuation, Surface Environment Definition, Effects on Point Designs due to Variations in Assumed Design Parameters as well as the Effects of a Lander on the Mariner Orbiter.

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1. INTRODUCTION

1. INTRODUCTION

1.1 STUDY OBJECTIVES

The objective of the parametric study was to provide accurate parametric design data for each element of the Hard Lander Capsule. The parametric data is to be used by designers to synthesize Lander designs and determine Capsule performance capabilities. Capsule performance parameters such as total data return and surface lifetime should be readily accessible in terms of weights, aeroshell diameter, and entry mode. The parametric data should provide the capability to readily extrapolate point design concepts to increasing or decreasing Capsule capabilities. Three broad missions have been defined for which these study objectives are to apply; these missions are:

- Minimum science (defined) and minimum life (1 day)
- Minimum science (defined) and extended life (90 days or more)
- Extended science (defined) and extended life (90 days or more)

The Capsule parametric design data presented in this volume are completely compliant with the study objectives and mission guidelines for both direct and out-of-orbit entry modes.

1.2 STUDY SCOPE

The Capsule parametric studies provide the basic information from which Hard Lander design concepts may be selected and performance estimates derived. Reference missions described in Volume II of the study, together with the environmental specifications contained in refs. 2-1 and 2-2 of Volume II, are the basis for the parametric studies. Direct and out-of-orbit entry Capsules are investigated over the range of encounter conditions defined by the mission analyses in Volume II for type I and II trajectories in the 1974 time frame.

Complete parametric information is provided for design evaluation and performance estimates of ballistic entry Capsules with conventional parachute configurations, as well as lifting, inflatable afterbodies, and extendable flap aeroshell applications to the direct entry mode. This data was presented for the purpose of providing direct design comparisons with hammerhead applications of the Titan booster, where shroud diameters exceed the desired design goal of 10 ft.

The guidelines and constraints that were provided by NASA Langley Research Center and which form the basis for this parametric study are:

Overall

- MM '73 opportunity of prime interest
- Dry heat sterilization required. ETO compatibility not required. Sterile insertion of propellants permitted.

Interplanetary, Entry, and Retardation

- Type I and Type II transfers
- $C_3 = 25 \text{ to } 40 \text{ km}^2/\text{sec}^2$
- Direct and out-of-orbit entry
- VM-1 thru VM-10 atmospheric models
- Parachute deployment not over Mach 2.0
- Maximum shroud diameter 16 ft - 10 ft desired

Planetary Impact

- Voyager Environmental Standard of September 1967
- Landing site: near equator (e.g., Syrtis Major), 30° from either terminator

Data Return

- Orbiter in synchronous orbit for a few days after landing, then Orbiter asynchronous to enhance mapping mission, inclination 60° .
- 10^7 bits imagery minimum, 10^8 bits or more desired.
- Surface science consists of imagery, pressure, temperature, wind velocity, and moisture; soil composition desired.
- Communication mode may be relay to Orbiter, direct to Earth or combination of these.

- DSN capability is either one 210-ft antenna and two 85-ft antennas or three 210-ft antennas.
- Imagery mission performed immediately after landing before Orbiter flies over horizon.

Surface Lifetime

- One day minimum, several months desired.

Several power sources are parametrically described for Hard Landers in this study; these include batteries, solar cells, fuel cells, and RTG's. The emphasis has been placed on the battery powered systems for minimum life and on solar array/battery systems for extended life missions. Complete parametric data for telecommunication design of relay or direct link applications for extended life is provided, and the variations due to landing site dispersions, Sun angles, and Earth viewing for different encounter geometries and encounter dates are readily available. To insure the accessibility and use of the parametric data, numerous examples are provided for the reader.

This document consists of two basic parts. The first part (Section 2) has three subsections which consist of: (1) results of parametric trade-offs for various Capsule concepts in the form of "Mission Lander Maps", (2) variations in performance due to perturbations in the reference missions, and (3) performance comparisons between parametric and point design data. The second and major part of the document (Sections 3, 4, and 5), contain the basic subsystem parametric studies which provide the data to construct the Mission Lander Maps.

The Mission Lander Maps summarize the principal trade-offs for the retardation, Lander, and landed Capsule components into one useful plot. The Capsule designer or science experimenter is then presented with one piece of data which permits him to manipulate data return and lifetime in terms of Capsule weight and performance.

1.3 STUDY APPROACH

The approach taken in the study provides accurate parametric data for each subsystem in the Capsule, using a format of variables and variable ranges which permits a design build-up of a Capsule concept. The Mission Lander Maps are merely products of the synthesis of subsystem parametric data and a presentation of final results in a format which provides performance estimates of capability for the most significant trade-offs. The reader may use the basic data to construct additional mission Lander Maps for performance estimates other than those detailed in this document. To aid in constructing mission Lander Maps, a complete example of the use of the data is provided in each subsystem section.

The approach to providing accurate parametric data requires that we first generate and organize a meaningful quantity of parametric data for each Capsule subsystem. This is provided in Sections 3, 4, and 5. Second, we are required to fully understand the major inter-relationships occurring between the mission parameters, the Capsule/Lander design, and the Orbiter/Lander interplay as they influence the Lander's useful lifetime and imagery data capability; this is accomplished in Section 2 with the aid of the mission Lander map.

The reference missions described in Volume II were developed simultaneously with the parametric data of this volume. Therefore, the initially assumed range of parametric variables was large to avoid limitations in the use of the data. The final reference missions described by nominal trajectories and standard deviations about the nominal due to separation and deflection maneuvers during entry and by injection, trimming, and orbit period errors subsequent to landing, are shown to be well within the assumed parametric range of variables.

1.4 SUMMARY OF STUDY RESULTS

The results of the parametric study show that there is a fixed weight of landed subsystems required to perform a minimum science 90-day extended life mission with 10 bits. This data return can be achieved with 400-lb landed subsystems (including container structure) for out-of-orbit entry and with 500 lbs for direct entry. The two reference missions used in the parametric studies have arrival dates of 10 January 1974 for direct entry, and 30 April 1974 for out-of-orbit entry. These arrival dates provide significantly different results for the conduct of the landed operations.

A decelerator load of 850 to 1150 lbs is obtainable for either out-of-orbit or direct entry. The ballistic out-of-orbit mission may be accomplished within a 10-foot shroud for decelerator loads up to a 1150-lb limit. This means that possible landed subsystems of 750 lbs are obtainable (well above the 400-lb minimum requirements for extended life). The direct entry ballistic mission requires a 16-ft hammerhead shroud to package the equivalent out-of-orbit payload. However, lift or inflatable afterbodies are also shown to offer a design alternative for staying in the 10-ft shroud with comparable payload capability. The additional weight and packaging penalties for lifting and inflatable concepts have not proven to be prohibitive. Development costs and schedules for the alternative lift and afterbody concepts would have to be compared with the hammerhead booster before a judgement could be made on the preferred direct entry configuration.

Study results are very conclusive in showing that a Hard Lander designed for postulated worst case combinations of touchdown and terrain constraints initially

contacts the surface multi-directionally on the bottom side. A second and subsequent impact may take place with attendant crushing of attenuation materials but these effects are readily identifiable in terms of change of shape and additional absorption requirements. Effects of slopes, winds, gusts, rocks, g-levels, and soil bearing strength are all readily identifiable in terms of mission performance.

Maximum data is returned for the 30 April reference (out-of-orbit) extended life mission if a simple antenna scheme is used. If a pointed antenna is used (with Sun sensors or rf Earth tracking capability), then the early 10 January encounter provides maximum data return (since the narrow beam and shorter range compensate for difficulties associated with a wider beam antenna.

Large footprint errors associated with the direct entry reference mission, which are primarily longitude errors, do not have a marked influence on extended life. If a direct entry high inclination orbit were possible (Type II only) then the parametric studies clearly show that the high latitude missions (27 deg N.) with 24 May 1974 arrival date does yield useful data return.

Maximum data return for the 30 April out-of-orbit encounter occurs near 10° N latitude with the simple antenna concept. Footprint errors for the out-of-orbit case or different landing sites from 10° N slightly degrade the data return.

For the extended life experiments with a 30 April encounter, the command link will be used to schedule various imagery sequencing and format. For a 10 January encounter, a limited imagery mission can be accommodated by the Lander with a simple antenna. For both direct and out-of-orbit entry extended life missions, the data return is improved by an order of magnitude with a medium gain pointed antenna.

On paper, the RTG is clearly the most efficient power source; however, there is no baseline RTG to use for the study and reasonable estimates of hardened performance were based on extended capability of present systems.

Small increases in weight beyond the critical mass of 400 to 500 lbs for the landed subsystems provide marked increases in system performance. With an assumed upper limit of 750 lbs available for the out-of-orbit or direct entry modes, many possibilities are apparent. Additional weight may be used to harden components particularly science instruments. There does not appear to be a satisfactory approach to shock mounting individual instruments below the Lander rigid body g-level constraint, because of the deflections involved. It appears, therefore, the candidate science instruments for the Hard Lander application will require shock hardening to the Lander level. The parametric studies of the impact limiter suggest many approaches to specifying Lander requirements for the most efficient use of the structure.

2. CAPSULE SYSTEM PARAMETRIC SYNTHESIS

2. CAPSULE SYSTEM PARAMETRIC SYNTHESIS

This section examines the total Capsule system synthesis and presents summary information on key subsystem performance. The buildup of subsystems and synthesis of a Capsule is accomplished by connecting the major portions of the Capsule together on one plot as shown in fig. 2-1. The mission Lander map of fig. 2-1 (hereafter referred to as the "Map"), is based on a 30 April out-of-orbit mission. The mission was constructed so that the Orbiter deflects the Lander Capsule from a 60° inclined synchronous orbit on a trajectory which impacts at 10° North latitude at approximately 14° true anomaly from the periapsis passage of the Orbiter. The Orbiter cooperates for 3 days with the Lander and thus provides a relay link for transmission of photo-imaging data, after which the orbit period is changed slightly and the Orbiter becomes asynchronous for an Orbiter mapping mission. After the first 3 days the landed Capsule conducts a direct link mission back to Earth, transmitting data at a reduced level for meteorology. The three blocks of the map permit three major performance trade-offs which are: (1) the aeroshell and aerodecelerator (entitled retardation), (2) the impact structure design (entitled Lander), and (3) the power, communications, environmental control, reference science, and container structure (entitled landed subsystems). In addition, the weight of the pre-entry subsystems (attitude control, deorbit propulsion, separation, structure and canister/adaptor, can be shown on the retardation block, providing an all-up Capsule weight. This information has been purposely left off the maps for clarity but can easily be added by the reader by reference to Section 5. The map of fig. 2-1 may be used for performance trade-offs and to determine useful design trends based on defined mission or program related design constraints. One such constraint is a desired 10-ft booster shroud goal which reduces the aeroshell diameter to 8.34 ft.

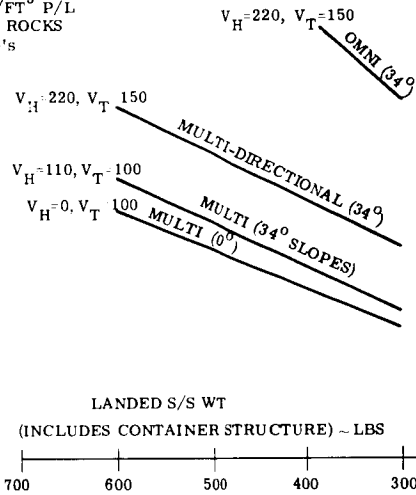
In Section 2.1 there are presented nine retardation blocks, four Lander blocks, and five landed subsystem blocks summarizing the top level parametric results which could theoretically provide 180 possible maps. Only special cases are of interest here and, therefore, six selected maps are discussed (Section 2.1). The maps described provide broad performance comparisons, e.g., solar arrays versus RTG's, weight variations due to different arrival times, etc. Section 2.2 describes the performance sensitivity resulting from changes in the reference missions or environmental specifications, e.g., separation errors, maneuver errors, entry errors, atmosphere errors, changes in terrain features, etc. Many of the significant effects described in Section 2.2 occur at a fixed landed subsystem weight and are variables into the plane of the map, e.g., variation in data return due to encounter date and landing latitude tend to occur at a fixed or negligible weight variation and for constant solar array area.

In Section 2.3, six maps are described which contain the six point designs. The variations in the point designs from the parametric data are then investigated, for both magnitude and cause.

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK - SOLAR ARRAY (1-90 DAYS)
- 30 APRIL ARRIVAL

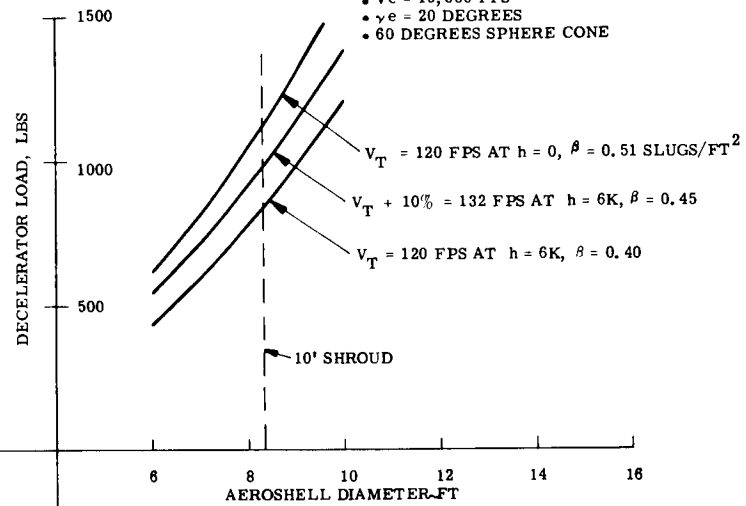
LANDER

- 40 LB/FT³ P/L
- 5 INCH ROCKS
- 1000 G's



RETARDATION (OUT-OF-ORBIT)

- $V_e = 15,300$ FPS
- $\gamma_e = 20$ DEGREES
- 60 DEGREES SPHERE CONE



LANDED SUBSYSTEMS

- 10° N LATITUDE
- VERTICAL ANTENNA
- 3 DAY BATTERY

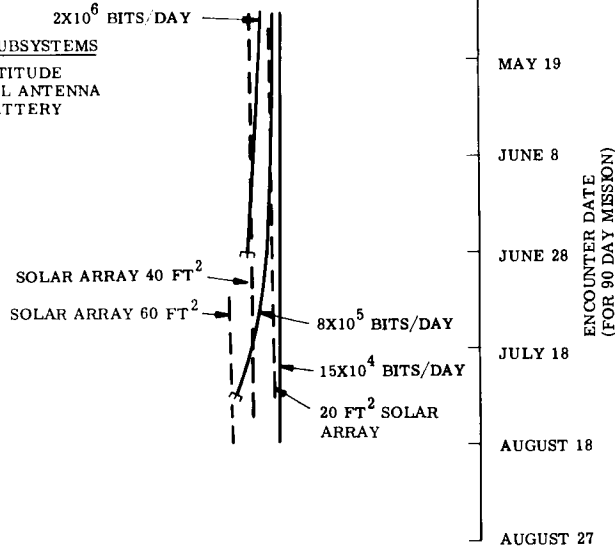


Figure 2-1. Hard Lander Map 1

2.1 MISSION LANDER MAPS

The six Maps described in this section are summarized in table 2.1-1. Map 1 was described briefly in fig. 2-1. The retardation section of the map contains constant ballistic coefficient (β), curves for direct and out-of-orbit entry capsules designed to a range of parametric variables, (entry velocity, path angle, aeroshell configuration, etc.), which are broadly defined to cover all possible reference '73 missions. The structure is designed to maximum loading conditions or to a constant minimum gage unit weight (whichever is desired). The heat shield is sized to the worst total heating environment. The constraint on the aerodecelerator is made according to a design terminal velocity being reached at a design altitude in a worst combination of atmospheres. The limiting aeroshell diameter constraint is placed on the curve in the form of booster shroud diameter.

The Lander portion of the map provides trade-offs of landed subsystem size, shape, and weight in terms of packaging density and container geometry. The flat-pack container for the Lander parametric studies is a baseline design shape and is based on previous results described in detail in Section 3.5. The toroidal Lander with wraparound crush-up material is also a baseline for the parametric study but should not be construed as an optimum shape or configuration. The symmetric shape lends itself to computer analysis of the many design variables and can be related to other possibly more efficient shapes. The material properties used in the baseline parametric studies of the impact limiter (attenuation crush-up) are for phenolic glass honeycomb. This material is well characterized for parametric studies. Both the materials and the Lander geometry are merely references for the parametric studies. Balsa wood and other materials such as polyimides can be treated parametrically in the same fashion. The Lander block portion of the map considers trade-offs of variable winds, descent velocities, ground slopes, g-levels, and other terrain features such as surface bearing and rock size.

In the landed subsystem portion of the map, the approach to preparation of data was to synthesize a payload for the container structure using the reference science packages (described as "black boxes" in Section 4.1), together with the various subsystem support parametric data. The reference missions are used to establish Earth - Sun - Mars relationships for communications and variations in solar energy. A baseline mission of 90 days is used with a data return of 10^7 and 10^8 bits as a goal. A partial listing of mission trade-off parameters, relay link assumptions and direct link assumptions are listed below.

Mission Trade-Offs

- Encounter date
- Mission duration
- Sun elevation angle variation

TABLE 2.1-1. MAP TRADE-OFF SUMMARY

Map	Entry Mode	Power Source	Lander Type	G-Level	Wind Velocity	Descent Velocity	Arrival Date	Landing Latitude	Slopes	Link	Retardation	Science
No 1	0/0	Solar Cells (Array area is variable)	Variable • Omni • Multi	1000	110	100	30 Apr	10° N	Variable • 0° • 34°	Variable • Bit Rate • Xmit Time	Ballistic	Minimum
No 2	0/0	Variable • Solar • RTG • Fuel Cell	Multi	Variable • 750 • 1000	Variable • 0 • 220	100 150	30 Apr	25° N	Variable • 0° • 34°	Variable • Bit Rate	Ballistic	Minimum
No 3	0/0	Variable • Solar • RTG	Multi	1000	110	Variable • 100 • 150 • 200	Variable 1 Jan - 1 mos to 1 Sep	10° N	34°	Fixed	Ballistic	Minimum
No 4	D/E	Solar Cells (Array area is variable)	Variable • Multi	1000	110	100	10 Jan	10° N	Variable • 0° • 34°	Variable • Bit Rate • Xmit Time	Variable • Lift • Inflatables	Minimum
No 5	D/E	Battery	Multi	1000	110	Variable • 100 • 150 • 200	Variable 10 Jan 30 Apr	10° N	34°	Relay	Ballistic	Minimum
No 6	D/E	Solar Cells (Array area is variable)	Variable • Omni • Multi	1000	Variable • 0 • 110 • 220	Variable • 100 • 150	10 Jan	10° N	Variable • 0° • 34°	Variable • Bit Rate • Xmit Time	Ballistic	Minimum

- Earth - Mars range variation
- Earth - Mars mutual visibility time
- Landing latitude

Relay Link Assumptions

- Orbiter cooperation for 3 days (4 periapsis passes)
- Command system for flexibility in photo imaging
- VHF, 50 watt transmitter (195,000 bps)

Direct Link Assumptions

- Varying antenna type and beamwidth
- 34° slopes and 10° radio horizon

Power Sources

- Solar cell/battery
- RTG/Battery
- Fuel Cells/Battery

2.1.1 MAP 1

Map 1, which was briefly described in fig. 2-1, is a composite of the Type I relatively high launch energy mission, which has very favorable arrival dates and geometries, and enters from the out-of-orbit mode. The type of mission considered is a 3 day relay link to the Orbiter (photo imaging), followed by reduced transmission of meteorology on a direct link for 90 days or more. Power is supplied by a solar cell-battery system to a vertically oriented antenna operating on a 50 watt S-band transmitter.

There are several significant trade-offs which may be made on Map 1. However, there are two features of this map which should be recognized before discussing trade-offs: (1) a minimum landed subsystems weight of 450 lbs is required for the mission, and (2) all trade-offs are based on a 8.34 ft aeroshell which fits in a 10 ft shroud.

2.1.1.1 Map 1 (Lander Design)

The Lander block describes two trade-offs: (1) wind variation from 0 to 220 fps (110 fps represents the maximum for the new NASA/LRC environmental specification), and (2) a weight comparison is made of omni- and multi-directional Landers. All trade-offs are based on rigid body 1000 g loads, 5 in. rocks, and 34° ground slopes. The omni-directional Lander may impact on any surface bottom down, in an otherwise random manner within the reasonable parachute sway limits.

For the limiting entry constraints placed on the aeroshell, these weights are well outside the capability of a 10-ft shroud for a 450 lb landed weight. The multi-directional Lander initially may impact in any direction up to 34° from the body roll axis, bounce and come to rest in any direction. The new maximum wind specification of 110 fps does not require significant attenuation material beyond that required to land vertically with no slopes. The old specification of 220 fps penalizes the Lander system considerably, but not nearly as much as that of selecting an omni-directional Lander. The variation between omni- and multi-directional Lander designs is 500 to 700 lbs, whereas the variation between the old and new wind specification is 200 lbs. Reducing the maximum new wind specification of 110 fps to zero produces only a small saving on impact material.

Therefore, it can be concluded from the map that a modest weight penalty would be attractive in controlling the Lander attitude to stay below some figure such as 40° . It would also be desirable to design missions which have low maximum horizontal velocities of 110 fps.

2.1.1.2 Map 1 (Aeroshell Design)

The retardation block is based on a path angle of 20° maximum, which is the value that was used in the parametric studies. It will be shown in Sections 2.2 and 2.3 that this is very conservative. If the worst case terminal conditions are chosen (terminal velocity, 120 fps, and terminal altitude, 6000 ft), then the decelerator load is minimum (850 lbs). However, this is far too conservative and 10 percent above terminal at the terminal design altitude was used, which increases the decelerator load by 150 lbs to 1000 lbs. If the design altitude could be lowered below 6000 ft, further improvement would be possible. This could be accomplished by knowledge of the planet radius uncertainties from the Mariner '69 and '71 missions.

2.1.1.3 Map 1 (Landed Subsystem Operations)

The landed subsystem block describes the trade-offs for a solar cell power source with varying solar array areas. From the landed subsystems portion of the map, the trade-offs for three direct links are shown at constant bit rates. At 1.5×10^5 bits/day, a 20 ft^2 solar array is shown to be adequate and would provide 1.5×10^7 bits in 100 days. This data, which is solely surface meteorology, is added to the 4.3×10^7 bits per relay link pass, and for four passes provides 1.87×10^8 total bits for the example mission. If we examine the 8×10^5 bits/day link, at 35 days, the power demand exceeds the 20 ft^2 array capability due to increasing Earth/Mars ranges and changing Sun angles. At 75 days into the mission, the capability of the 40 ft^2 solar array is exceeded and at 90 days antenna breakdown is predicted. For the 2×10^6 bits/day, link capability is exceeded and power demands coincide at about 40 ft^2 solar array capability at 60 days, however, more data is returned on this link at 60 days than on the previous links at 90 days or more. It is apparent that a variable bit rate link with a command link would be the next logical investigation and this is precisely the design trend or direction desired as a result of examining this map.

2.1.1.4 Map 1 (Mission Assurance and Contingency Planning)

From Map 1, it is seen that a capsule design based upon these limiting entry conditions with a terminal design velocity, $V_T + 10$ percent at 6000 feet altitude, provides a decelerator load of approximately 1000 lbs.

2.1.2 MAP 2

Map 2 (fig. 2.1-1) is also an out-of-orbit mission having the same synchronous orbit and relay link geometry at periapsis as Map 1. The purpose of the map is to show the trade-offs between fuel cells, solar cells, and RTG's as well as variations caused by the extremes in surface winds and landing g-levels on the payload. The fuel cell is seen to be heavy and have limited lifetime for the missions under consideration. Two encounter dates of 10 January and 30 April are selected for comparison of the power sources. Fixed data rates are picked for the missions and power and weights are allowed to increase with lifetime. The landing site of 25° N latitude accentuates the differences in power sources because of Sun angle and its effect on the solar cell array.

2.1.2.1 Map 2 (Lander Design)

The Lander block is based on a multi-directional Lander for all cases and two trade-offs are described: (1) the variation between landing g-levels, and (2) the influence of winds. It can be seen from this data that a very large penalty is paid for lowering the g-level below 1000 g's at high horizontal velocities, whereas the effect is negligible at small velocities. This conclusion supports the Map 1 discussion, which suggests some minimal control of Lander attitude and a simple means for reducing the effect of winds. This could be achieved by incorporating an all-flexible parawing in lieu of the ringsail parachute used in the study.

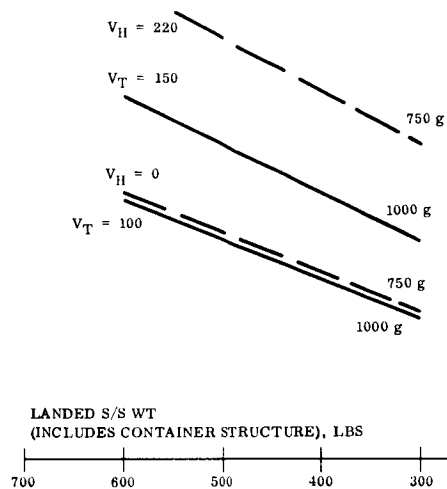
2.1.3 MAP 3

Map 3 (fig. 2.1-2) is an out-of-orbit mission similar to those previously described. The data on Map 3 is constructed for 90 day missions only and with varying arrival dates. The arrival dates vary from 10 January to 10 August '74. Only one landing site is shown at 10° N. Both the solar array and the RTG are evaluated as power for the telecommunications and science equipments. For the different arrival dates considered, the telecommunications subsystem provides maximum data return with a 50 watt transmitter and a minimum antenna gain of 14 dB. The power subsystem was varied to provide the required power. The maximum data obtained is 7.3×10^7 bits (direct link) for the minimum landed weight of 405 lbs (RTG/battery). The comparable minimum weight solar/battery system weighs 440 lbs. Although the range is least at the 10 January encounter, the communication geometry is poor at 25° N and requires a wide bandwidth (82 deg) for the early portion of the mission. This severely penalizes the Capsule system 30 days later where, although the range has increased greatly, the visibility is much better. By 10 April the maximum data rate has fallen to 18 bps with an antenna gain of 6.4 dB and beamwidth of 82° . Initially, at the 10 January date, the data rate was 500 bps. Thus, the constant data rate is a poor way to conduct a mission

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK - SOLAR ARRAY (1-90 DAYS)
- 30 APRIL ARRIVAL

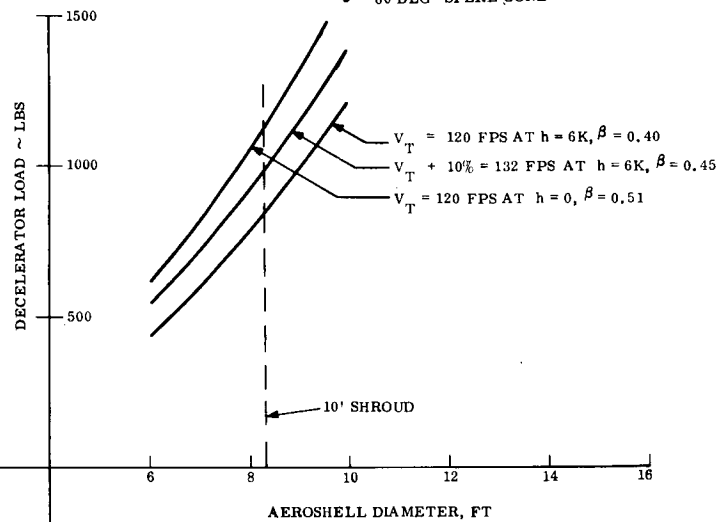
LANDER

- MULTI-DIRECTIONAL
- 40 LB/FT³ P/L
- 5 INCH ROCKS



RETARDATION (OUT-OF-ORBIT)

- $V_a = 15,300$ FPS
- $\gamma_a = 30$ DEG
- 60 DEG SPHERE CONE



LANDED SUBSYSTEMS

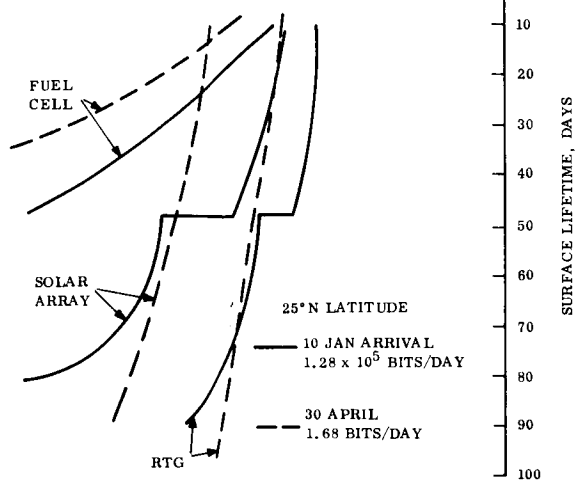


Figure 2.1-1. Hard Lander Map 2

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK AFTER 3 DAYS

LANDER

- MULTI-DIRECTIONAL
- 40 LB/FT³ P/L
- $V_H = 110$ FPS
- 1000 G's

RETARDATION (OUT-OF-ORBIT)

- $V_e = 15,300$ FPS
- $\gamma_e = 20$ DEG
- 60 DEG SPHERE CONE

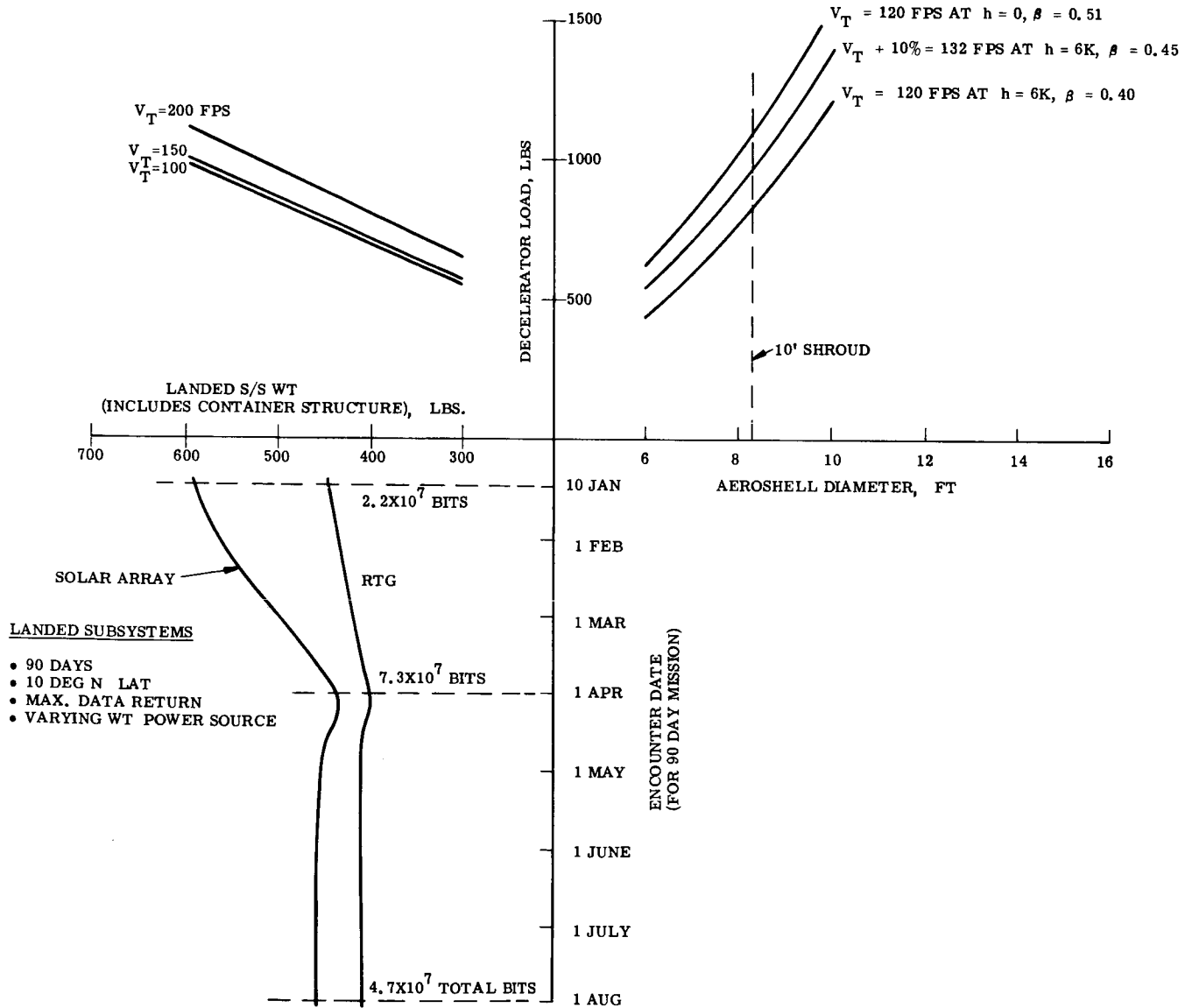


Figure 2.1-2. Hard Lander Map 3

if useful data is required over the entire period. Furthermore, there may be trade-offs on length of mission, bit rate scheduling, and arrival geometry which are not evident in this simple mission format.

2.1.3.1 Map 3 (Lander Design)

The Lander block describes a trade-off for three terminal descent velocities of 100, 150, and 200 fps at a wind speed of 100 fps and for a single deceleration level of 1000 g's. The results show that there is a non-linear increase in attenuation weight as terminal velocity is increased. However, from the retardation block it can be seen that the increase in terminal velocity tends to have the opposite effect on decelerator performance. The conclusion is that there exists some optimum terminal velocity for minimum Capsule system weight. This is shown much more effectively in Map 5 and proven in the body of the report.

2.1.4 MAP 4

Map 4 (fig. 2.1-3) is for direct entry and provides information on lifting and inflatable afterbodies which would overcome the difficulties of hammer-heading a Titan-Centaur booster. Map 3 shows that for an 8.34-ft aeroshell which fits the 10 foot shroud, both the 60 deg sphere-cone lifting body or the inflatable-attached afterbody provide attractive solutions to meeting decelerator load requirements for a mission similar to the out-of-orbit cases just discussed with solar cell - battery power. The lifting sphere-cone configuration requires only a small trim to achieve the necessary lift. The trim condition is achieved by slightly offsetting the entire Lander. The Capsule would be roll stabilized with center-of-gravity offset. The alternate inflatable afterbody is extended prior to entry and rigidized. The Lander block provides the same trade-offs for omni- and multi-directional impact as was shown in Map 1. A trim $L/D = 0.27$, with the limiting case terminal retardation constraints of $V_T = 120$ fps at 6000 ft provides 850 lbs decelerator load. The same decelerator load is obtained for a 13-ft inflatable afterbody. An omni-directional Lander (for the 850 lbs decelerator load) only delivers 350 lbs to the surface. The minimum required is approximately 450 lbs critical landed payload. The minimum payload provides 10^8 bits; however, it leaves a very marginal design for the extended life meteorology (20 ft^2 solar array). A small improvement in weight makes a marked improvement in the overall extended life mission. A very large improvement occurs when the 850 lb decelerator load is translated into landed weight by the multi-directional 1000 g Lander curve. It should be noted in Map 4 that the direct entry conditions are the limiting conditions for the parametric study ($\gamma_e = 30^\circ \pm 10^\circ$), and Section 2.3 will show in the point design comparisons that the actual reference requirements ($\gamma_e = 25^\circ \pm 8^\circ$) will significantly improve the Capsule system performance. Such effects are also evident in the other blocks of the maps.

2.1.5 MAP 5

Map 5 (fig. 2.1-4) provides the information necessary to establish the minimum type of direct entry mission. A minimum science payload is used together with a 120 lb equipment container, a 3 day battery, and associated support equipments to perform a 3 day relay link mission. The total landed subsystems weigh 270 lbs, and the requirement is to

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK - SOLAR ARRAY (1 - 90 DAYS)
- 10 JANUARY ARRIVAL

LANDER

- 40 LB/FT³ P/L
- 5 INCH ROCKS
- 1000 g's

RETARDATION (DIRECT)

- $V_e = 20,800$ FPS
- $\gamma_e = 40$ DEG (MAX)
- 8.34 FT AEROSHELL
- 60° DEG SPHERE-CONE
- $V_T = 120$ FPS AT $h_T = 6000$ FT

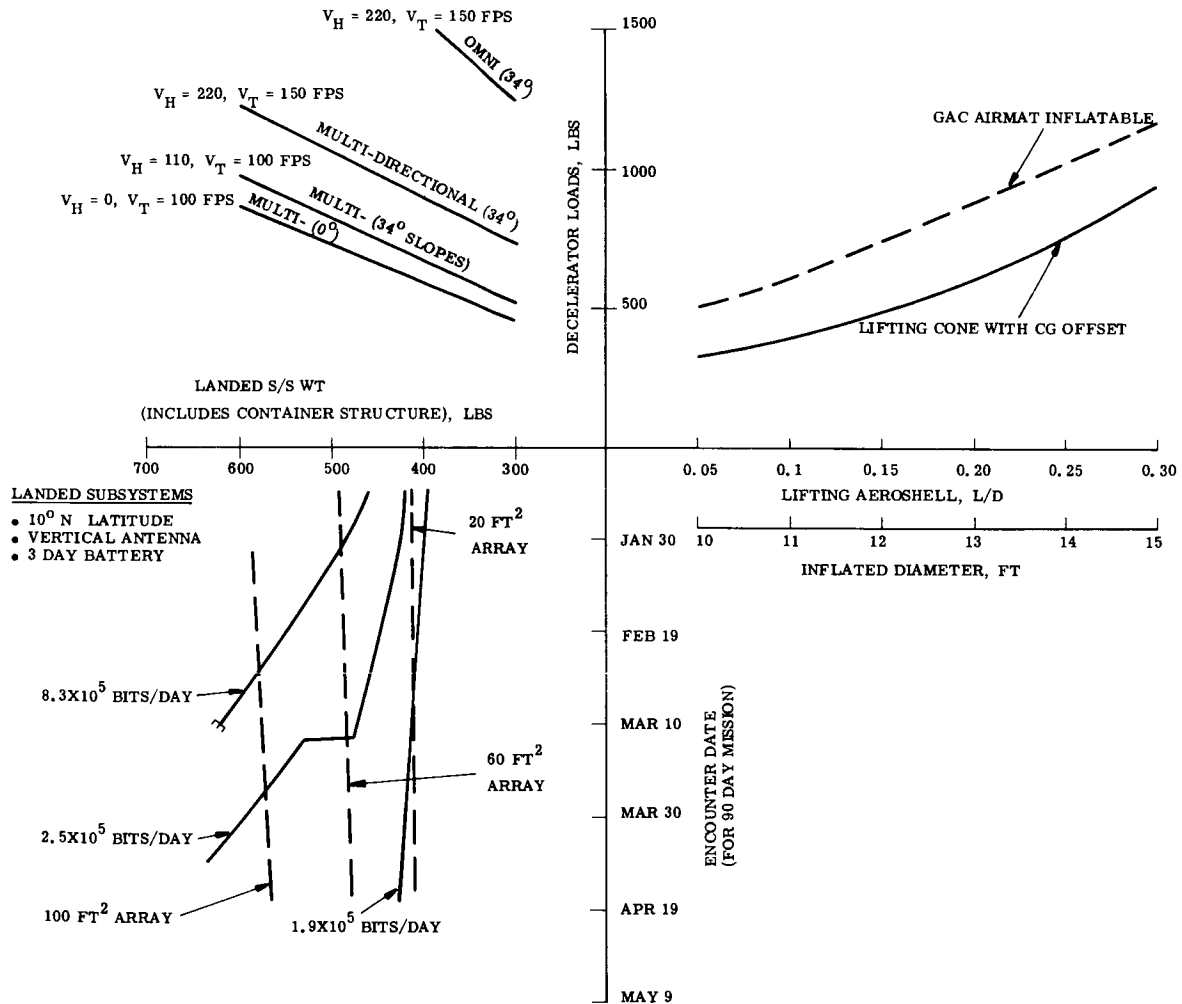


Figure 2.1-3. Hard Lander Map 4

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT

LANDER

- MULTI-DIRECTIONAL
- 40 LB/FT^3 P/L
- $V_H = 110 \text{ FPS}$
- 1000 G's

RETARDATION (DIRECT)

- $V_e = 20,800 \text{ FPS}$
- VM 8 ATMOS
- 60° DEG SPHERE-CONE

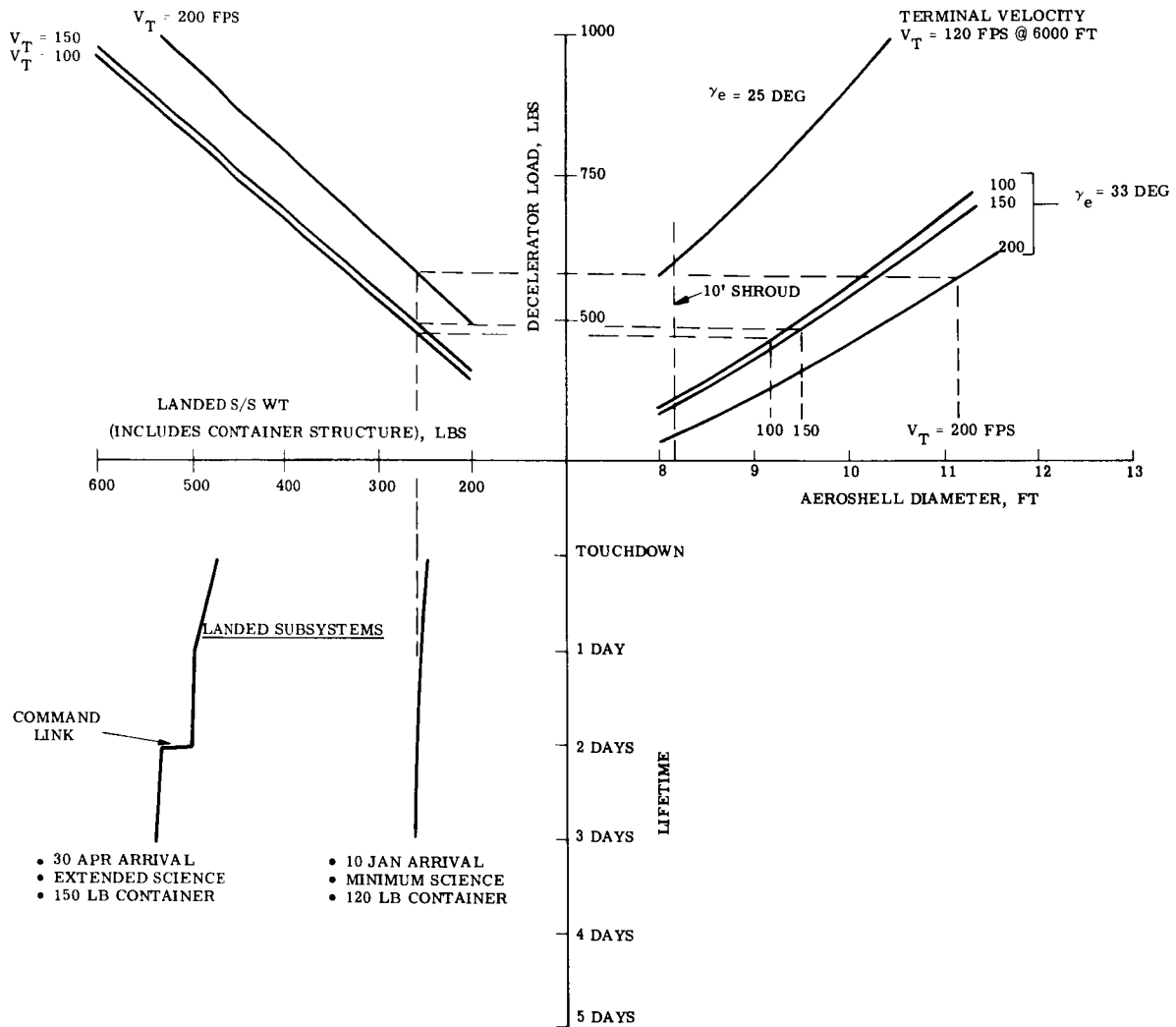


Figure 2.1-4. Hard Lander Map 5

find a suitable Lander aeroshell design which will package this container within a 10 foot shroud diameter. For comparison purposes, an extended science package (described in Section 4.1) is included, which has the additional battery power for more instruments, command link, and regenerative power required after the first three days.

2.1.5.1 Map 5 (Retardation)

In the retardation block, the variation of terminal descent velocity corresponds to the variation in the Lander block. Lines have been drawn on Map 5 which connect the corresponding points on each block for a minimum weight Lander sufficient to survive 1 to 3 days (Note: the penalty of the second and third days is negligible). The near minimum decelerator load is seen to be 470 lbs and is attained at a terminal velocity of 100 fps. However, the aeroshell diameter must be increased beyond the 10 ft shroud.

The conclusion reached for this type of mission is that the maximum path angle must be decreased till the shroud diameter constraint is achieved. Map 5 shows a ballistic coefficient for a $\gamma_e = 25^\circ$ at $V_T = 120$ fps and $h_T = 6000$ ft, which easily meets the shroud constraint. Thus, there is a very important coupling between the Lander mission map and the requirements of the reference mission. As shown in Volume II and discussed in the following paragraphs, the path angle is sensitive to the impact parameter error. A maximum path angle of 25° would be difficult to achieve but not impractical. The constraints used in defining the multi-directional Lander of Map 5 are:

- 1000 g impact
- 110 fps maximum horizontal wind
- 100 fps vertical descent velocity
- 40 lb/ft³ packaging density
- 2.32 aspect ratio for the container disk
- 5 in rocks
- 34° slopes

2.1.6 MAP 6

Map 6 (fig. 2.1-5) describes the general performance characteristics of a direct entry extended 90 day life mission. Because the landed subsystems require 450 lbs to perform a minimum level of extended life, the decelerator load is too high for an 8.34 ft aeroshell and, thus, a larger diameter hammerhead shroud is required. For the very conservative maximum path angle of 40° , it is not possible to package an extended life omni-directional Lander for extended life in a 16 ft shroud. However, a multi-directional Lander based on the new wind constraint can attain the extended life capability.

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK - SOLAR ARRAY (1 - 90 DAYS)
- 10 JAN ARRIVAL

LANDER

- 40 LB/FT³ P/L
- 5 IN. ROCKS
- 1000 G's

RETARDATION (DIRECT)

- $V_e = 20,800$ FPS
- $\gamma_e = 40$ DEG (MAX)
- 60 DEG SPHERE-CONE

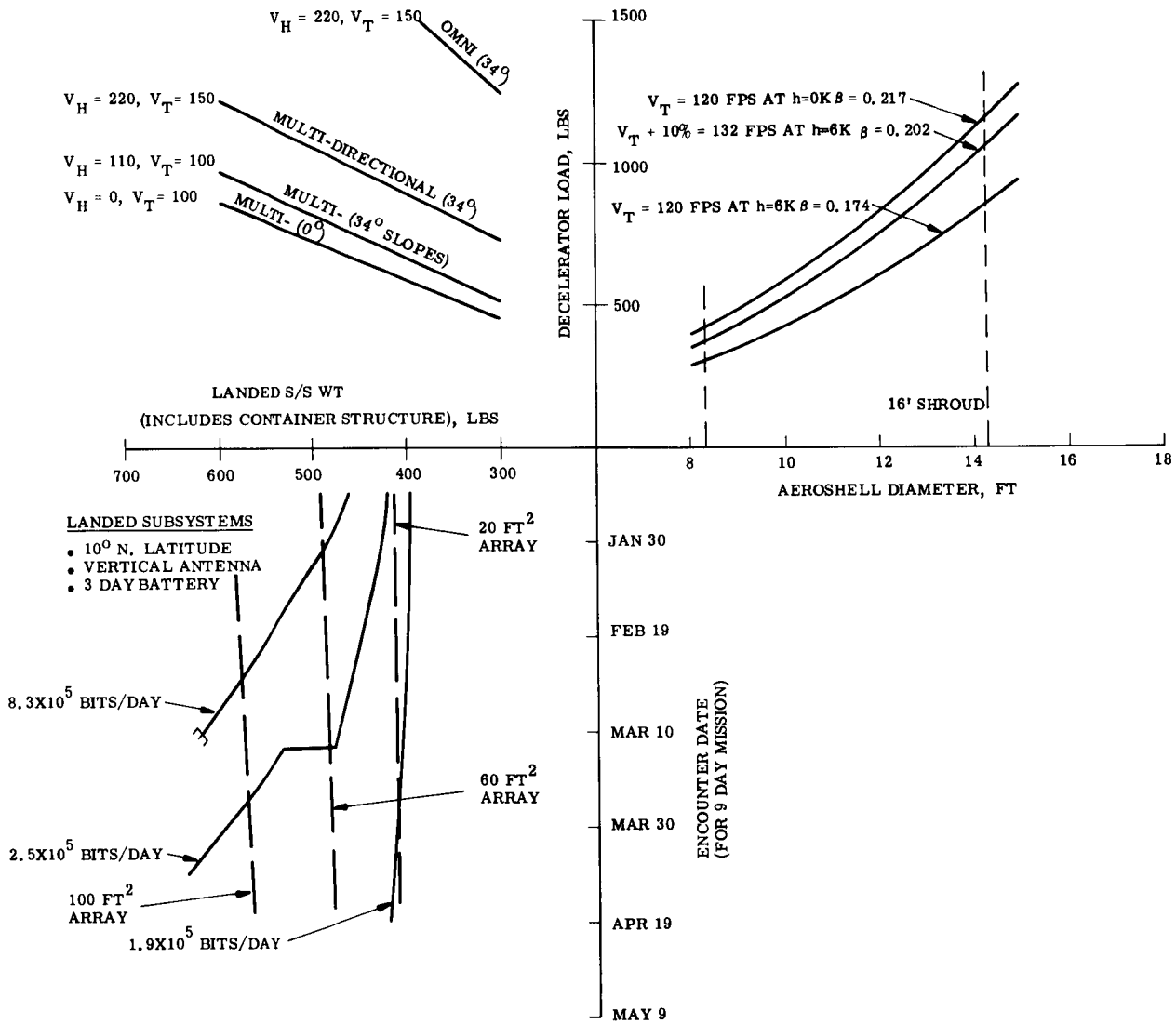


Figure 2.1-5. Hard Lander Map 6

2.2 MISSION MAP SENSITIVITY

The reference missions defined in Volume II are based on specific arrival dates, arrival geometry, and landing sites. The performance capability estimates from the Mission Lander Maps of the previous section were based on the limiting case for the range of parametric variables. In general, this range of parametric variables, e.g., entry velocity, entry path angle, and atmospheric variables, is far greater than the range of variables established for the reference missions. The reference mission contains nominal trajectory motion together with all perturbations about the nominal value due to major error sources. Because of the conservatism in choosing the parametric range of variables, the Mission Lander Maps are, in general, conservative and therefore this section deals with this conservatism in terms of "mission map sensitivity".

Section 2.2.1 describes the range of variations in entry conditions which can be anticipated due to standard errors in the separation and deflection maneuvers, and the resultant landing site footprint dispersions which may be expected. Errors or variations in the orbit period are discussed as they effect post landed relay and direct link communications.

Section 2.2.2 describes the sensitivity of the aeroshell and aero-decelerator to variations in entry conditions caused by errors and off-nominal conditions. Section 2.2.3 describes the sensitivity of the Lander parametric performance due to variations in touchdown conditions caused by descent velocity, winds and gusts, terrain features, and variations in impact limiter materials. Section 2.2.4 describes the variations in landed system performance which occur due to errors in landing site locations as a result of separation, entry, and atmospheric errors. Variations in relay and direct link performance due to orbit period errors and Earth-Mars-Sun range and visibility are described.

2.2.1 CAPSULE PERFORMANCE VARIATIONS DUE TO TRAJECTORY AND ORBIT ERRORS

Trajectory dispersions are due to separation errors and orbit or approach trajectory errors. These errors have, in all cases, only a very small effect on the entry velocity, so that they are felt mainly in the entry path angle and the location of entry (with respect to the desired or nominal location). The combination of entry path angle and the dispersion of entry location into one of a range of postulated atmospheres, causes the dispersion of the landing point, or footprint. There is, in addition to this, the dispersion of trajectory time from entry to landing, caused by path angle dispersion and the uncertainty of the atmosphere. This effect causes the Orbiter to be at a different location with respect to the Lander (and the Lander horizons) at the time of landing and has an important effect on the success of relay communication during entry and immediately following impact.

2.2.1.1 Out-of-Orbit Mission

In the out-of-orbit mission, the effect of the orbit errors, as specified in the mission constraints, are small compared to the effect of the separation errors. Based

on a separation velocity error of 1.5 percent (3σ) and a misalignment error of 1.8° (3σ), the path angle dispersion is about $\pm 1^\circ$ (3σ) on a nominal value of 16° . Considering also the effect of the range of atmospheres, the resulting footprint length is about 610 km. In this case, the separation errors were specified such that a footprint of this length was obtained; the 1° path angle error was a secondary result. Otherwise, they could be specified in relation to the entry path angle error, such that minimum path angle would be greater than the skip limit. This would result in a greater path angle dispersion (about 14° to 18°), with correspondingly greater (about twice as much) downrange dispersion. The effects of dispersions on the relay communications are shown in detail in Volume II, Section 3.2.

2.2.1.2 Direct Entry Mission

The entry path angle dispersion in the direct entry mission depends on the impact parameter of the approach trajectory and on the separation errors. For an impact parameter error of 300 km (3σ) this error source is the most significant and the entry path angle error is so large that it actually determines the choice of the nominal path angle to insure that the minimum path angle is above the skip limit. For separation errors of 1.5 percent (3σ) in magnitude and 1.8° (3σ) in direction and an impact parameter error equal to 300 km (3σ), the path angle dispersion is from 16.5° to 31.4° , with a nominal value of 25° . The corresponding footprint is 2400 km long, over the entire range of atmospheres. The effect of these dispersions on the relay communication link is shown in detail in Volume II, Section 3.2. The path angle dispersion is to 21.0° to 28.4° , if the impact parameter error is reduced to 150 km (3σ) with separation errors as quoted above. The dispersion error is then reduced to the order of 1400 km.

2.2.1.3 Relay Communication After Landing

In the out-of-orbit mission, succeeding relay communication passes are very nearly identical to the first pass due to the accuracy with which the synchronous orbit is established before lander separation. In the direct entry mission, on the other hand, the success of the succeeding orbital relay link communication pass is dependent on the orbit error; this error may be several hours. This means that, in the case of highly inclined orbits, the periapse communication link is likely to be lost after several passes. In the case of a near-equatorial landing site and orbit, the relay communication link is not lost but is likely to occur a considerable distance away from periapse, with resultant longer ranges. Typical results are shown in Volume II, Section 3.4.

2.2.2 CAPSULE PERFORMANCE VARIATIONS DUE TO SENSITIVITIES IN RETARDATION CONDITIONS

Section 2.2.1 describes the effect of various entry variables which influence the retardation concept trade-offs in the Lander Maps. Section 3 of this volume presents in considerable detail the derivation of the decelerator load as a function of aeroshell and aerodecelerator trade-offs. The results of Section 3 have been incorporated into the retardation blocks of the Lander Maps. For these Lander Maps, specific entry

conditions are assumed. The sensitivity of variations in entry conditions upon Capsule-aeroshell and aerodecelerators is discussed in this section.

For the parametric study of this volume, a set of entry conditions for both out-of-orbit and direct entry were assumed based on past studies of Mars atmospheric entry. The assumed conditions were:

Direct Entry

$$V_e = 20,800 \text{ fps}$$

$$\gamma_e = 30^\circ \pm 10^\circ$$

and

$$\gamma_e = 22^\circ \pm 3^\circ$$

Out-of-Orbit Entry

$$V_e = 15,300 \text{ fps}$$

$$\gamma_e = 16^\circ \pm 4^\circ$$

The most significant parameter in the retardation block, with respect to the effect on landed weight performance, is the entry flight path angle. This is due to the significant variations in allowable entry vehicle ballistic coefficient (β) with path angle (γ). The allowable β is associated with the maximum entry flight path angle (i. e., γ_e nominal + γ_e tolerance), such as $\gamma_e = 16 + 4 = 20^\circ$ for the assumed out-of-orbit entry. Since the maximum path angle translates into a maximum β , which relates entry weight and decelerator load to vehicle diameter, the decelerator load can be developed as a function of aeroshell diameter and maximum entry path angle. Such a variation is shown in fig. 2.2.2-1 for the out-of-orbit mode. Note that terminal conditions for the parachute are listed which when combined with flight path angle, define entry vehicle β , as will be described in Section 3.2. A similar curve for the direct entry mode is presented in fig. 2.2.2-2. The significance of an increase in path angle is to substantially reduce decelerator load capability for a given aeroshell diameter.

The reference mission studies described in Volume II result in the following entry conditions.

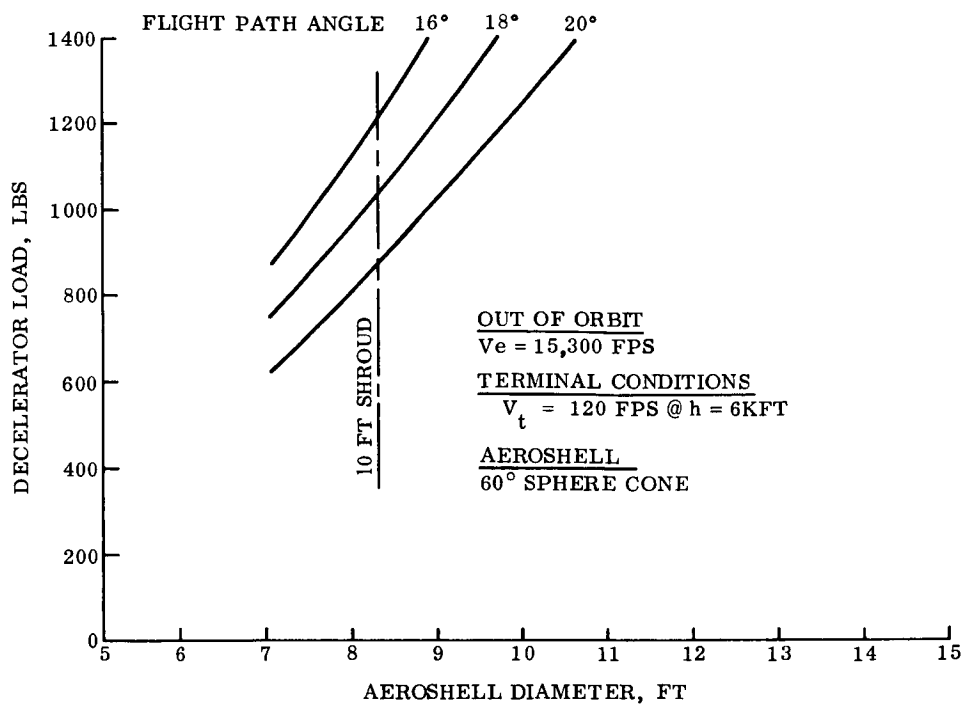


Figure 2.2.2-1. Effect of Flight Path Angle on Decelerator Load (Out-of-Orbit)

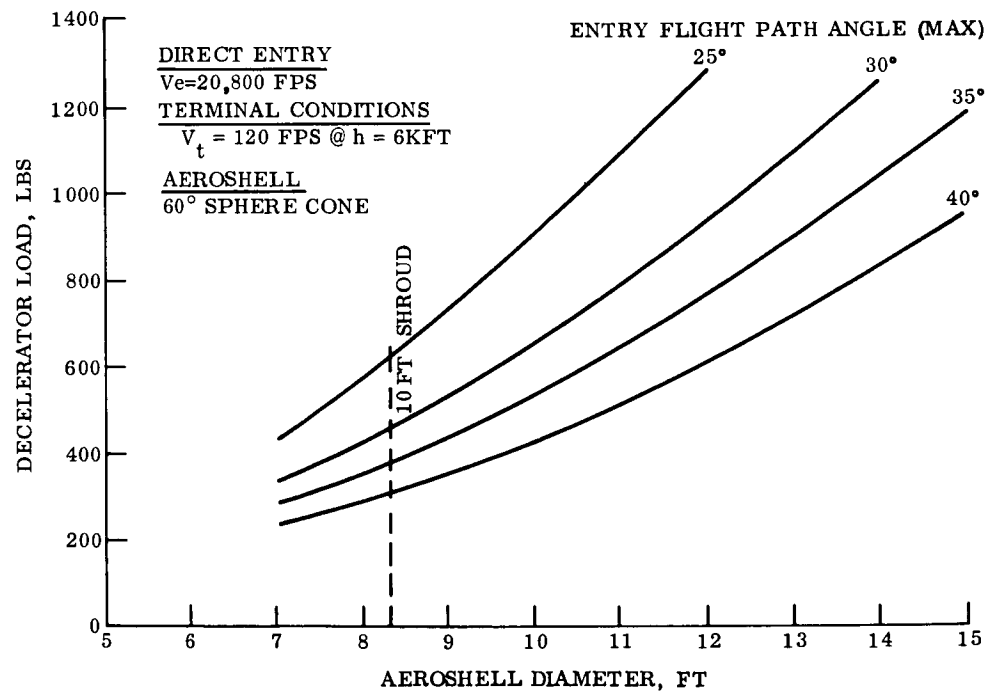


Figure 2.2.2-2. Effect of Flight Path Angle on Decelerator Load (Direct Entry)

Direct Entry

$$V_e = 20,800 \text{ fps}$$

$$\gamma_e = \begin{matrix} & + 6.4^\circ \\ 25^\circ & \\ & - 8.5^\circ \end{matrix}$$

Out-of-Orbit Entry

$$V_e = 15,300 \text{ fps}$$

$$\gamma_e = 16^\circ \pm 1^\circ$$

Thus, the assumed velocity was identical to the reference mission value; however, the flight path angle was larger (hence conservative). It is desirable to know what effect these changes in entry conditions would have upon the entry and retardation requirements and thus upon the mission maps. From the entry and retardation studies discussed in Section 3, it is found that, depending upon the exact terminal parachute conditions desired (descent velocity and altitude), the entry vehicle ballistic coefficient for the assumed direct and out-of-orbit entry conditions are 0.175 slugs/ft² and 0.40 slugs/ft², respectively. Examination of the reference mission entry conditions results in vehicle ballistic coefficients of 0.25 slugs/ft² for direct entry (fig. 2.2.2-2) and 0.53 slugs/ft² for out-of-orbit (fig. 2.2.2-1). The increase in ballistic coefficient and hence, weight is considerable for the reference missions.

This increase could be further improved by smaller direct entry path angles, associated with the 50 km (1 σ) impact parameter error, $(25^\circ +^{3.4}_{-4.0})$. This decrease of maximum path angle from 31.4^o to 28.4^o represents approximately 100 lbs increase in the decelerator load for a direct entry mission with a 10 ft shroud constraint (fig. 2.2.2-2).

2.2.3 CAPSULE PERFORMANCE VARIATIONS DUE TO SENSITIVITIES IN TOUCHDOWN AND TERRAIN CONDITIONS

The capsule impact attenuation system trade-offs described in the mission maps and presented as parametric studies (Section 4.5), are necessarily based on nominal values for various Lander design parameters and surface conditions. This section indicates the effects on these trade-offs due to variations in the reference conditions and identifies those areas where perturbations are especially significant. It is important to recognize that the variations are truly of statistical character due to uncertainties in the problem. Therefore, the following discussion assumes that these variations have occurred when, in fact, their probability of occurrence is exceedingly small.

2.2.3.1 Martian Surface Definitions and Touchdown Conditions

This first group of variables which produce design perturbations are described in fig. 2.2.3-1, and summarized below:

- Ground wind velocity
- Descent velocity
- Surface slopes
- Surface bearing strength
- Rock outcrop definition
- Parachute sway
- Rigid body g-tolerance
- Payload density and system weight.

In general, if the ground wind velocity and/or the descent velocity are other than the values assumed in a particular design, the crush-up stroke provided will be incorrect. If actual wind velocities are less than the design values, the stroke is on the conservative side and the deceleration load is attenuated below the specified g-loading. If the actual velocities are in excess of those assumed in design, the resulting insufficient stroke will allow higher g-loads to be transmitted to the container structure and its contents. Since the required stroke varies as the square of the velocity component normal to the surface slope, the perturbations introduced by this effect can be substantial. The velocity-squared effect also influences the kinetic energy to be absorbed in the attenuator. This implies that impact velocities greater or less than those assumed will require larger or smaller volumes of crush-up material for the toroidal Lander configuration. This aspect of the problem is included in the stroke discussion since it is the stroke length which, in conjunction with rock protection requirements, determines the volume of energy absorbent material required.

Variations in ground surface slopes enter the Capsule trade-offs in two ways. First, the surface slopes interact with the horizontal wind and Capsule descent velocity to produce the normal velocity component perpendicular to the surface. This normal velocity vector affects the length of the stroke and the size of the attenuator. The effect of slope angle variations cannot be generalized since they must be considered in conjunction with particular wind and descent velocity components. If, for a particular landing situation, the wind velocity is more than the descent velocity, a decrease in slope angle (flatter slope) will decrease the normal velocity component and hence, the stroke required for attenuation. An increase in slope angle will have the opposite effect. However, if descent velocity predominates over the wind velocity, a steeper slope than the reference condition will decrease the normal velocity component and thereby decrease the required stroke. A variation toward a flatter slope would produce the opposite effect.

The second effect of slope angle perturbations is on the configurations of the toroidal Landers. For the omni-directional Lander this second effect is eliminated

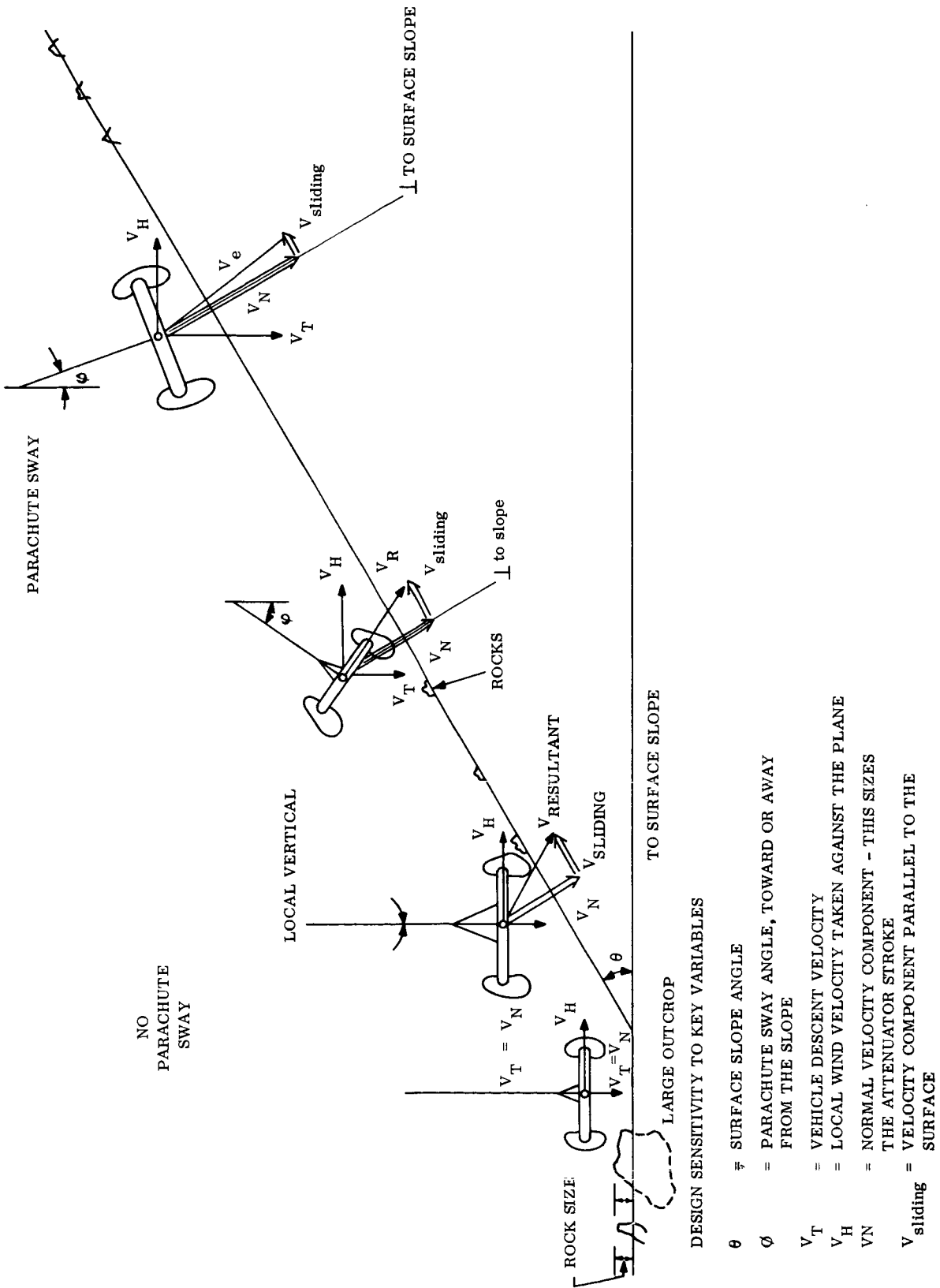


Figure 2.2.3-1. Lander Design Sensitivities

by definition. For multi-directional analysis this effect is important since the steeper slopes require added capability over and above some reference conditions. If a multi-directional Lander is designed for a certain surface but actually impacts on a flatter slope than anticipated, there will be a positive margin in the impact attenuator. If, however, a steeper slope is encountered, then insufficient stroke will be available to adequately stop the vehicle within acceptable g-levels. This is a direct consequence of the vehicle directionality as reflected in its geometrical shape.

Surface bearing strength influences the mission trade-offs and parametric studies as follows. All of the computations for a crushable honeycomb design are based on an infinitely rigid non-yielding surface. Consequently, a yielding (softer) surface would produce enhanced vehicle capability, since the stroke will be comprised of the honeycomb crush-up and the penetration of the yielding surface. If a soft yielding surface can be assured, a smaller amount of honeycomb is required for crushing.

Variations in the rock size encountered by a Lander also influence the design trade-offs. If a Lander is sized for rocks of a given size, it has the requisite capability for all lesser sizes. Larger individual rocks will conceivably penetrate the attenuator and reach the container structure. However, all such larger rocks do not necessarily have the angularity to be of potential danger. The long slender slivers that could be envisaged would no doubt snap before fully penetrating the impact attenuator. Moreover, larger rocks tend to be larger in all respects and hence, flatter. As a result, the net effect is that an isolated hard surface presents no special difficulties. In this situation larger variations are more beneficial than small variations.

Parachute sway variations do not influence the omni-directional trade-offs and analyses. For the multi-directional Lander all sways up to and including a given design value can be safely tolerated. For sway variations in excess of the design value, the directionality inherent with a multi-directional geometry renders the vehicle protection inadequate from a stroke and crush-up area viewpoint. In essence, a variation which increases the sway angle necessitates more material for a multi-directional Lander if sufficient stroke and crush-up areas are to be furnished.

Two obvious points will now be mentioned. If the component g-tolerance is lower than that assumed in design, larger strokes are required if damage is to be avoided. If the components can tolerate more severe g-loadings than planned in the design, a positive margin exists with the design stroke. If the Lander system weighs more than anticipated in the design (as for example when a subsystem does not separate), the kinetic energy of impact will increase; this necessitates an increment in crush-up material volume. A second effect is that the material density will increase because the required crushing stress (for a greater Lander weight) is increased. If the Lander system weighs less than anticipated, the reverse of the above argument is true. If the weight difference is a substantial negative variation, a less dense material of reduced crushing stress ought to be used to insure crush-up behavior within allowable g-levels.

2.2.3.2 Material Performance Variations

The second category of variables, which when perturbed influence the design, may be identified as material performance properties. These include:

- Crushing stress
- Density
- Specific energy
- Stroke efficiency
- Crush-up behavior
- Effective areas.

The honeycomb material crushing stress is a prime design quantity. If for some reason the crushing stress is higher than anticipated, the attenuator will function but the g-level will be exceeded since the g-level is directly related to the crushing stress. Should the crushing stress be lower than anticipated, the operating g-level will be lower than the design value; this means that the required stroke must be higher than anticipated in the design. A similar argument can be constructed in terms of specific energy. If the material specific energy is lower than expected for the same impact conditions, more material is required. This means an increased weight. For a constant crushing stress and effective area, the operating g-level decreases and an increased stroke is indicated. The reverse argument holds for a more efficient energy absorber. The system will operate at a higher g-level with a decreased stroke. A material density variation will by itself affect the system weight as previously discussed.

In sizing a crush-up impact attenuator, provision is made for stroke efficiency. In simplest terms this is the effect whereby a given thickness of honeycomb is not completely crushed when exposed to the design impact conditions. If the material is more efficient than assumed, no difficulty is encountered. If the material in a particular design is less than assumed, and there are no rocks, the design will be inadequate. However, for certain combinations of efficiency and rock dimensions the attenuator thickness determined by rock protection considerations is greater than that determined by a negative variation in stroke efficiency. As a result, variations in material stroke efficiency will not produce marked design perturbations.

If geometric variations from the nominal Lander geometry are introduced, the effective cross-sectional area which crushes up will be affected. If this area is decreased compared to the design value, the operating g-level will be decreased, requiring an increased stroke for adequate attenuation. If an excessive area is effective, the operating g-level will be higher than desired and the components will be damaged. This would imply that the design strokes as supplied were excessive.

Integrity of the crush-up system is essential for this mode of impact attenuation. Gross failure of the material by fracture of massive fragments or by other non-crushing behavior will severely compromise the design.

2.2.4 CAPSULE PERFORMANCE VARIATIONS DUE TO SENSITIVITIES IN LANDED SYSTEMS OPERATIONS

It has been found in the parametric studies that many of the major trade-offs associated with maximizing the data return are not highly weight sensitive in the landed subsystems area. Once a critical landed weight is delivered to the surface, small improvements in weight provide tremendous possibilities for landed operations. These types of trade-offs are described in this section and are difficult to show on the Lander maps because of their negligible weight variances.

2.2.4.1 Data Return vs Landing Site Dispersion

The effect of landing site dispersion has a significant effect on data return as seen in fig. 2.2.4-1. For the latitude variations shown, significant variations in data return occur because of the variations in Sun elevation angles on the solar array power output. In addition, depending on the arrival date and latitude geometry variation with time, there is a significant change in Earth communication look angles. In a lesser manner, the environmental control system also is sensitive to the changing solar energy levels.

For the data return trade-offs of fig. 2.2.4-1, an S-band transmitter of 50 watts with a simple helix antenna (14 dB gain) was assumed. A solar array/battery power system with 40 ft² area was used. The data reflects encounter dates of 10 January and 30 April in which landing latitudes of 0° to 50° were investigated. The maximum data return is obtained for a 30 April encounter at a landing site of 10°N latitude. The 10 January arrival is shown to provide substantially reduced data and at 50°N there is negligible data for a direct link mission.

As is shown in fig. 2.2.4-1 for the two encounter dates identified, the maximum data return occurs at 10°N latitude for the 30 April encounter with a reduction of data return for 0°, 25°, and 50°, in that order. This indicates that the best conditions (namely declination angle, range to Earth, and Sun elevation angle) are optimum at this latitude.

The optimum conditions for the 10 January encounter occur at 0°N latitude. However, the data return is 10 percent more than that obtained for the 30 April encounter. The data return reduces as a function of latitude to 50°N latitude where the data return is 1 percent of that realized for the best condition obtained at 10°N for the 30 April encounter.

When comparing min-max values of data return for the different latitudes a variation of 10:1 is observed for each encounter date.

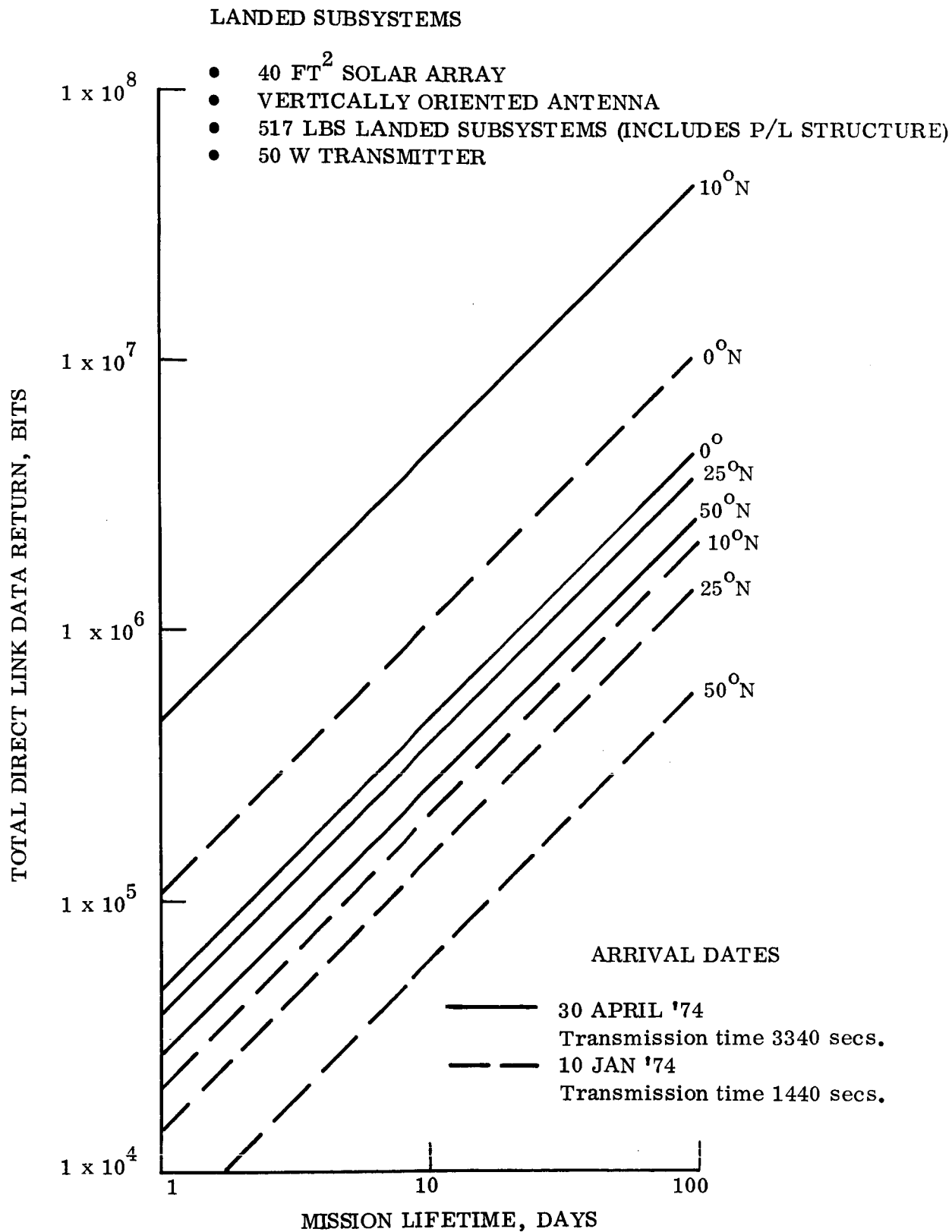


Figure 2.2.4-1. Data Return vs Landing Site Latitude Variation

When considering a landing latitude of 25°N , which could result with presently defined missions, an interesting phenomena is observed in fig. 2.2.4-6, whereby maximum data return occurs at this latitude with a 1 June encounter. Although the data return is still less than that obtained for the 10°N at the 3 April encounter, the difference in data return between the two latitudes amounts to about 4 percent with the solar array power systems. Also shown in fig. 2.2.4-6, is the maximum data return for all conditions which is achieved with an RTG power system.

2.2.4.2 Data Return vs Antenna Configuration

A variation in data return of 800:1 is realized when antenna gains, transmission times, fixed transmitted power and 15 percent variation in solar radiation are evaluated for variations of: (1) a simple vertically oriented helix antenna, and (2) a steered multiple helix array. Three antenna orientation schemes are considered (1) simple locally sensed vertical (1.56 hours transmission time), (2) pointed array (4 hours transmission), and (3) steered (8 hours transmission). Fig. 2.2.4-2 presents the results of trade-offs using a solar cell power source for two encounter dates. Fig. 2.2.4-3 describes the trade-offs associated with a 30 April arrival for both solar array and RTG's. The comparable results of a 10 January arrival are shown in fig. 2.2.4-4. The trade-offs for RTG power only at the two varying encounter dates is shown in fig. 2.2.4-5. Pointing is accomplished by either Sun sensors and/or homing on an r-f signal from Earth. The steered array tracks Earth continuously. The data return is based on landing at 10°N . The maximum data is obtained with a 10 January encounter and is primarily a capability for a steered antenna to overcome poor mutual visibility and solar power degradation with very high gain low beamwidth. For lower gain, wider beamwidth systems, the 30 April encounter has the highest data return. Since 10^8 bits are program objectives, the lower gain and less complex antenna is attractive.

2.2.4.3 Data Return vs Encounter Date

The encounter date has a significant effect on maximizing the data return for a fixed weight power system and this is shown in fig. 2.2.4-6. A 40 ft^2 solar array system with an average power output of 14 watts is compared to a comparable weight RTG system. Maximum data is obtained for a 1 May encounter with landing sites at 10°N latitude. The total data returned (based on 90 day missions), is 1.3×10^8 bits with the RTG. The solar array returns 5×10^7 bits. The 25°N landing site degrades the data to 10^7 bits.

10°N LAT LANDING SITE
 50W TRANSMITTER
 SOLAR CELL POWER

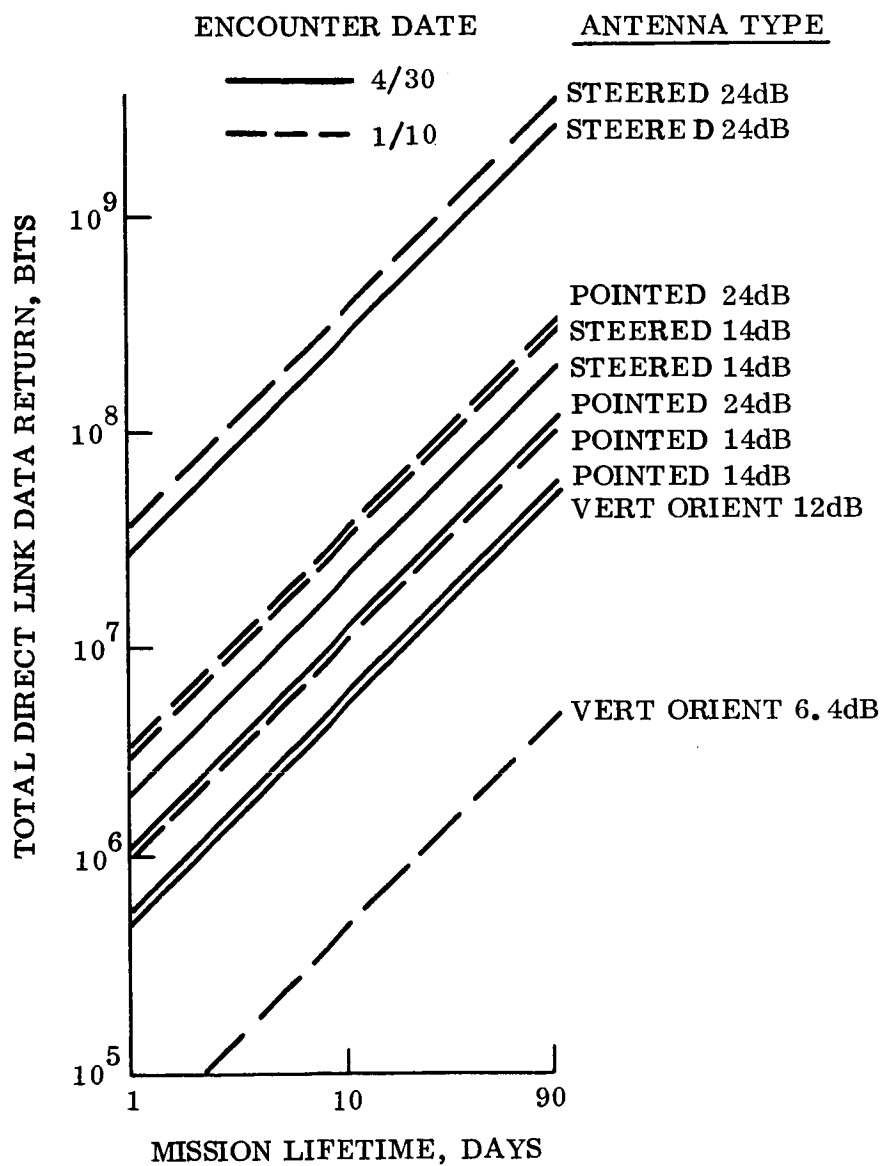


Figure 2.2.4-2. Data Return vs Antenna Configuration

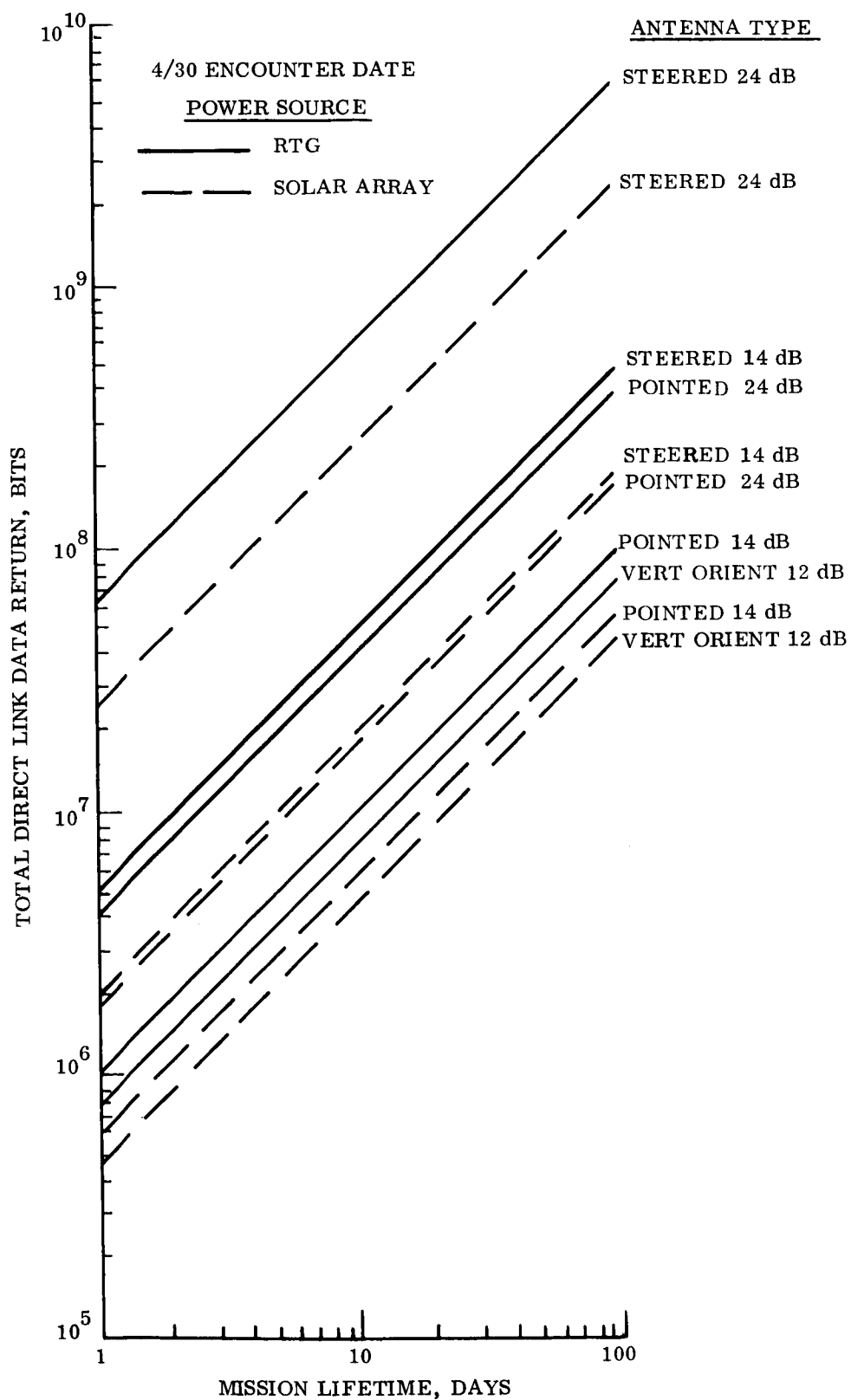


Figure 2.2.4-3. Data Return vs Antenna Configuration

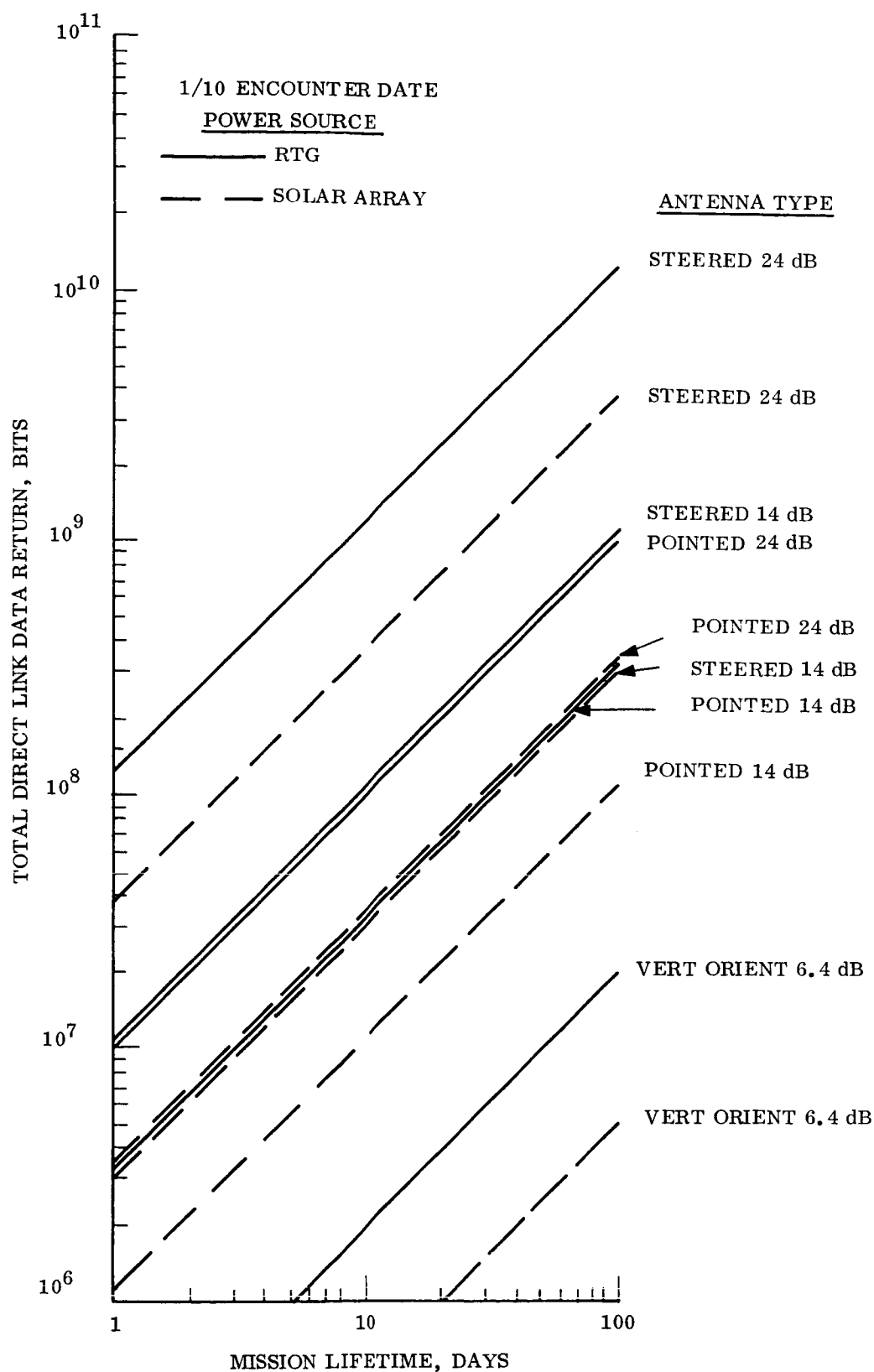


Figure 2.2.4-4. Data Return vs Antenna Configuration

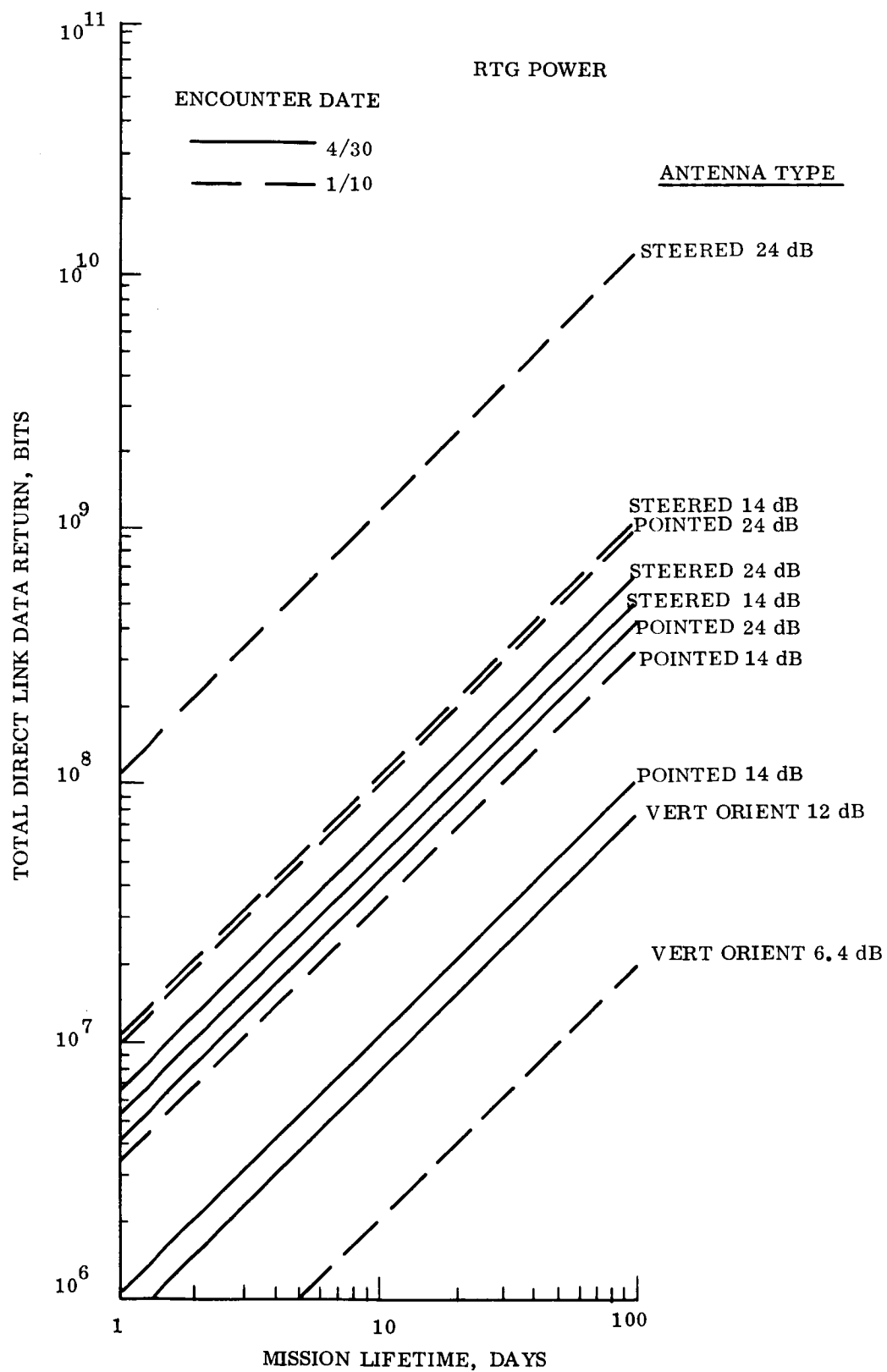


Figure 2.2.4-5. Data Return vs Antenna Configuration

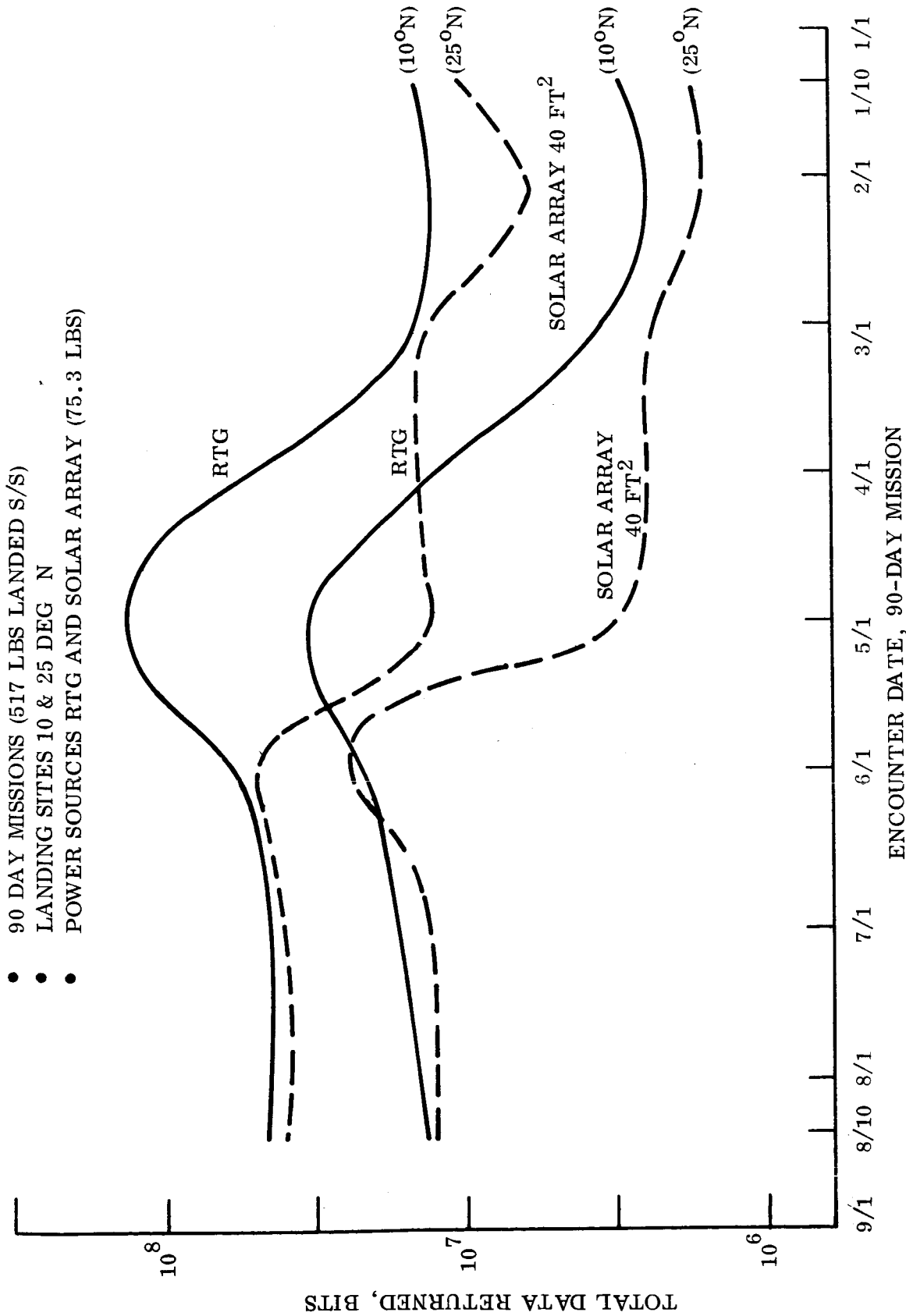


Figure 2.2.4-6. Data Return vs Encounter Date

2.3 COMPARISON OF CAPSULE PARAMETRIC AND POINT DESIGNS

In this section six point designs described in Volume IV of the Hard Lander Study are compared with the parametric studies of this Volume. This comparison is intended to calibrate the parametric data to a high degree of accuracy for extrapolation to other missions. Since the Lander Maps and point designs are composed of four specific weight breakdowns - entry weight, decelerator load, Lander weight, and landed weight - a comparison of the point design with the parametric data can occur in each quadrant of the Lander Map.

The six specified point designs reflect three broad mission objectives, which are given as:

1. Minimum science and lifetime
 - Basic science package (ref. Section 4.1)
 - Battery powered
 - Relay link only
 - Surface lifetime through one diurnal cycle
2. Minimum science and extended lifetime
 - Basic science package
 - Solar cells plus battery power
 - Relay and direct link
 - Surface lifetime 90 days or more
3. Extended science and extended lifetime
 - Increased science package (ref. Section 4.1)
 - Solar cells plus battery power
 - Relay and direct link
 - Surface lifetime 90 days or more

The six specific design cases are presented on five Lander Maps which follow (Lander Maps 7, 9, 10, 11 and 12). The three missions outlined above present point design solutions for both direct and out-of-orbit entry modes.

Table 2.3-1 lists the point designs and establishes a nomenclature for each design case. A qualitative summary statement is included about each major area of comparison. The Lander structure and attenuation estimates are found to be the most susceptible to variation between parametric and point designs. However, this variation is apparently different by only 10 percent.

2.3.1 MINIMUM SCIENCE AND MINIMUM LIFETIME CAPSULE SYSTEMS (POINT DESIGNS 1, 2a, AND 2b)

Point Designs 1, 2a, and 2b are plotted on Lander Map 7, fig. 2.3.1-1. Map 7 describes the minimum science/minimum lifetime mission. Adopting the nomenclature of table 2.3-1 for each design case, there is one out-of-orbit (O/O) entry with an omni-Lander and one direct entry (D/E) with both an omni- and a multi-Lander. The decelerator load and the aeroshell diameter are used as the basis for comparison between point design and parametric data.

The comparison begins by examining the difference between the design weight statement and the parametric data. Table 2.3.1-1 contains the weights for both design and parametric, based on the design weight statement decelerator load. For the 0/0-1-omni design, the entry weight is 1246 lbs for a 100 in. aeroshell. The parametric data predicts 1320 lbs using the Configuration Selection Program (CONSEP) data presented in Section 3.1.2.2. A detailed examination of the design weight statements and the CONSEP aeroshell will show that the 20-30 lb variation in weight is due to different estimates of attitude control and miscellaneous items other than basic structure and thermal protection.

In all three design comparisons, the difference in decelerator load between the parametric curve and the point design is not greater than 30 lbs out of an approximate decelerator load of 1200 lbs, or 2.5 percent. Path angle and entry velocity have a small effect on aeroshell structure because the structure is minimum gage over the range of interest. The same results are obtained for the remaining four point designs in the following discussion and, therefore, no further discussion will be given the retardation block. Once the path angle and terminal deceleration constraints are specified (i. e., $V_T = 100$ fps and $h_T = 6000$ ft), the ballistic coefficient is fixed.

2.3.1.1 Map 7 (Lander Block)

The minimum science payload, component support, and container structure is weight listed as 408 lbs for all three design points. The dashed lines connecting the decelerator load and subsystems is the Lander design and is denoted point A_D on Map 7.

In actuality, the design specification on 0/0-1 and D/E-2a was for an omni-Lander, 1000 g-load, 220 fps winds, 5 in. rocks, 39.5 lb/ft^3 packaging density, flat pack design aspect ratio of 2.84, and a terminal velocity of 100 fps which is denoted A_p . The approximate parametric curve having $V_T = 100$ fps is dashed on the Lander block.

TABLE 2.3-1. POINT DESIGN NOMENCLATURE FOR PARAMETRIC AND DESIGN COMPARISON

Entry Mode	O/O	D/E	D/E	O/O	D/E	O/O	D/E
Point Design	1	2a	2b	3	4	5	6
Lander Mode	OMNI	OMNI	MULTI	MULTI	MULTI	MULTI	MULTI
Lifetime	ML	ML	ML	EL	EL	EL	EL
Science Mode	X	X	X	X	X	Z	Z
Designation	0/0-MLX-1-OMNI	D/E-MLX-2a-OMNI	D/E-MLX-2b-MULTI	0/0-ELX-3-MULTI	D/E-ELX-4-MULTI	0/0-ELZ-S-MULTI	D/E-ELZ-6-MULTI
Qualitative Comparison Results							
..Landed S/S 40 lb	Good correlation between battery powered relay link. Difference represents improved parametric power subsystem						
..Retardation	Negligible variation in performance of the aeroshell and parachute						
..Lander	Parametric data over-estimates OMNI attenuation, correctly estimates MULTI						
ML - Min Life EL - Extended Life X - Min Science Z - Extended Science							

HARD LANDER MAP 7

- RELAY LINK AT PERIAPSIS

LANDER

- 40 LB/FT³ P/L
- 5 IN. ROCKS
- 1000 G's

RETARDATION (60 DEG SPHERE-CONE)

	O/O	D/E
V_E (FPS)	15,300	20,800
γ_E (DEG)	17	33
$h_{\text{TERMINAL}} = 6000 \text{ FT}$		

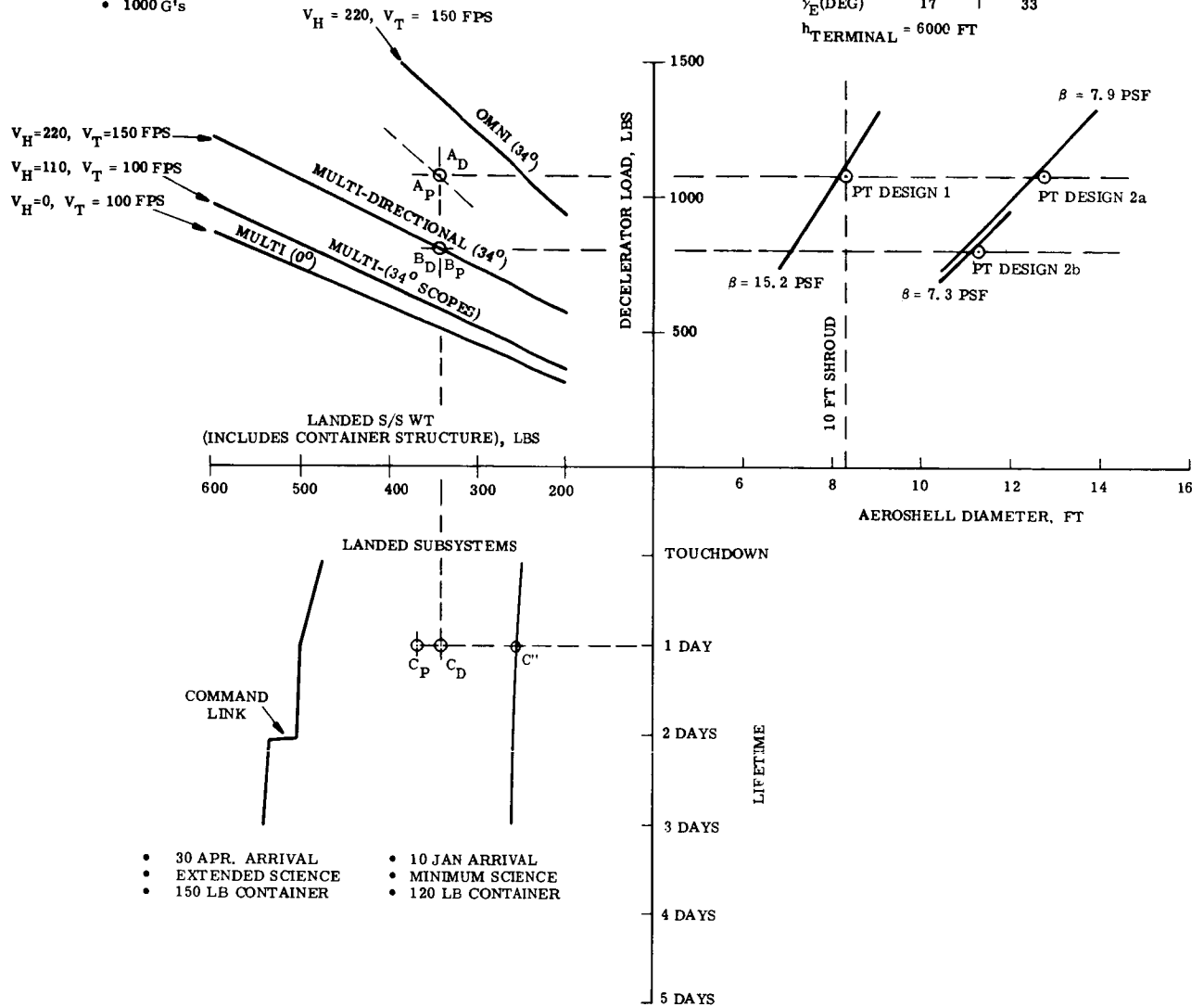


Figure 2.3.1-1. Hard Lander Map 7

TABLE 2.3.1-1. PARAMETRIC AND DESIGN COMPARISON FOR
MINIMUM SCIENCE/MINIMUM LIFE

Entry	O/O		D/E		D/E	
Lander Mode	OMNI		OMNI		MULTI	
Aeroshell Diameter (inches)	100		152.4		135.6	
Decelerator Load (lbs)	1086		1086		795	
Design Point	Para	1	Para	2	Para	2b
Ballistic Coefficient (psf)	15.1	15.2	7.8	7.9	7.2	7.3
Capsule Wt (lbs)		1626		1882		1463
Entry Wt (lbs)	1240	1246	1240	1480	1095	1127
Lander Wt (lbs)	865	858	865	858	637	623
Landed Wt (lbs)	350	336	350	336	350	336
Data Return ($\times 10^7$)	16.2	16.2	5.0	5.0	5.0	5.0
Lifetime (days)	1	1	1	1	1	1

The intersection with the decelerator load is denoted A_P and indicates that the parametric data is only in variance by 5 percent in estimating A_D for the 336 lbs of subsystems. The aspect ratio error and variation in packaging density for this design is tabulated in table 2.3.1-2. The variations are small; however, they may contribute to the A_D-A_P variation. Figs. 2.3.1-2 and -3 show the variation in weight fraction that occur for packaging density and aspect ratio.

For the D/E-multi-Lander, the points B_D and B_P appear to correlate well (at least within the accuracy of these curves for the $V = 100$ fps constraint). The conclusions are that the Lander block as well as the retardation block accurately predict the performance of the Hard Lander Capsule System for the various types of touch-down and terrain constraints imposed.

TABLE 2.3.1-2. SUMMARY OF LANDED CONTAINER ASPECT RATIO AND PACKAGING DENSITY VARIATIONS

Design Point	1	2a	2b	3	4	5	6
Container Aspect Ratio, (Parametric)	← 2.32 →						
Container Aspect Ratio, (Design)	2.84	2.84	2.84	2.5	2.5	2.47	2.47
Aspect Ratio Error (%)	+2.0	+2.0	-2.5	-1.0	-1.0	-0.8	-0.8
Packaging Density (lb/ft) (Parametric)	← 40 →						
Packaging Density (Design)	← 40 ± .5 →						

REFERENCE CONDITIONS

$W_{\text{LANDED}} = 400 \text{ LBS}$

$V_H = 220 \text{ FPS}$

$V_D = 100 \text{ FPS}$

5 IN. ROCK

$2 b/h = 2.32$

1000 G CONDITIONS

34° SLOPE

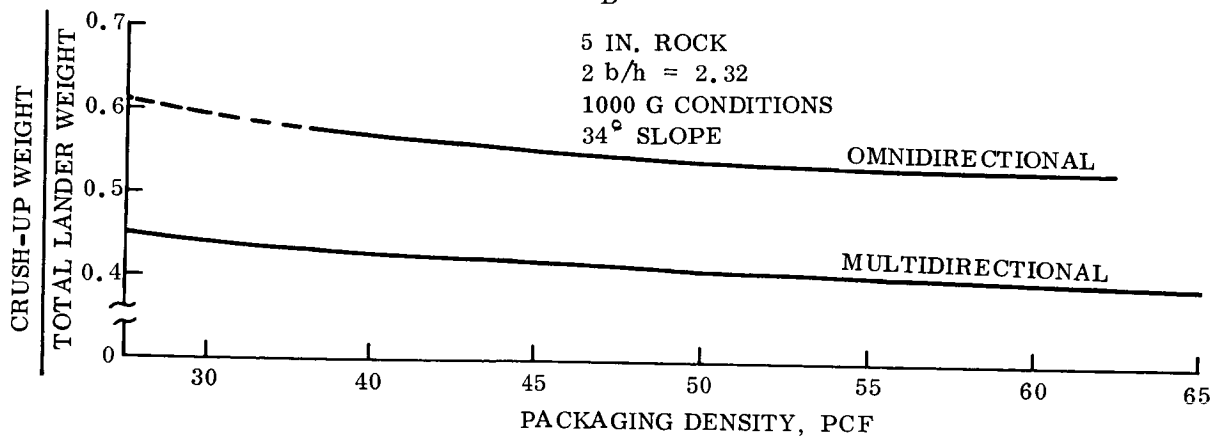


Figure 2.3.1-2. Effect of Packaging Density Variation on the Fraction Crush-up Weight/Total Lander Weight

REFERENCE CONDITIONS

$$W_{\text{LANDED}} = 400 \text{ LB}$$

$$V_H = 220 \text{ FPS}$$

$$V_D = 100 \text{ FPS}$$

5 IN. ROCK

1000 G CONDITIONS

40 PCF PACKAGING

DENSITY

34° SLOPE

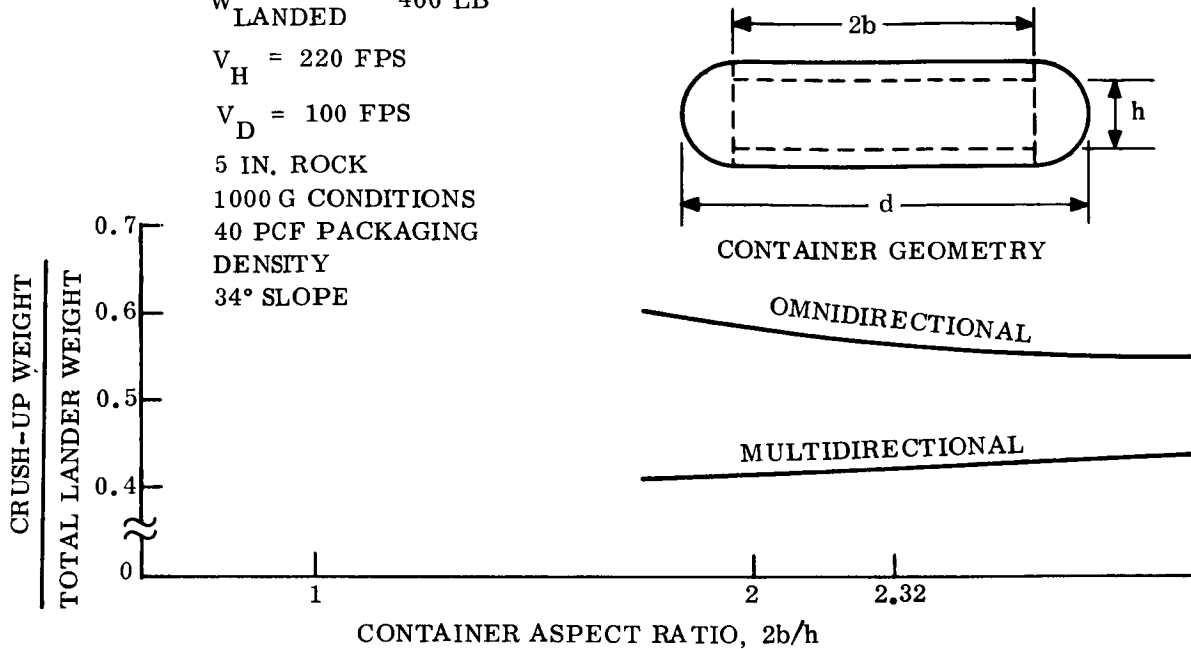


Figure 2.3.1-3. Effect of Aspect Ratio Variation on the Fraction Crush-up Weight/Total Lander Weight

2.3.1.2 Map 7 (Landed Operations)

For the minimum science/minimum lifetime mission, the relay link optimization for out-of-orbit and direct entry represents one of the most significant areas of consideration. To effect maximum data return from the planet, the periapsis pass represents the most efficient method of obtaining data since it represents the minimum Orbiter/Lander communication range with its high data rate transmission capability, providing maximum data with the minimum weight Lander. To provide a design of high reliability with a high confidence factor, the landed subsystem design incorporated a fixed low-gain broad beamwidth antenna to assure communication with the Orbiter.

Antenna gain, effects of slopes, radio horizon and asymmetry in crush-up material must be considered to arrive at the optimum value of transmission time and data rate to maximize to data return from the communication relay link. For the purpose of design for the 0/0 case, the communication time was set at 8.5 minutes, and the range at 1500 km. This results in the design of a communication link whose bit rate was nominally 160,000 bits per sec using a 50 watt transmitter. The total data obtained under these conditions was 8.1×10^7 bits. With this quantity of data, the next consideration is the formulation of a transmission sequence during the period that the

Orbiter would be overhead to receive the data. The formulated transmission sequence was based on a 4 minute transmission period. During this 4 minute transmission time, stored entry data, surface meteorological data, and diagnostic data would be transmitted initially. This would be followed by the transmission of imagery data of about 1 picture (10^7 bits) per minute. Between pictures 2 and 3, the data that was transmitted initially would be retransmitted since it would have been placed in the memory unit after which real time imagery would again be transmitted to the Orbiter. At the completion of the transmission of the additional pictures, the previously defined science data would be again transmitted. This sequence can be repeated or additional data could be factored in the transmission period in the form of another scene of pictures with possible inclusion of stereo and/or even color pictures for transmission at this time. The initiation of the 4 minute transmission sequence would occur approximately 2-1/2 minutes after impact of the Lander on the surface. This period would be used to allow the Lander to stabilize and deploy appropriate equipment, such as cameras. The data rate would be switched from the low entry data rate to the high data rate and transmission would be initiated. This transmission would continue until the Orbiter passed overhead, whereby the loss of signal from the Orbiter would be detected by the UHF receiver on board.

For the direct entry relay link, a 25° entry path angle and the VM-8 atmosphere were used. The cone angle of transmission, based on factors previously discussed, was set at $\pm 45^\circ$. Range for the direct entry case was set at 3000 km, which results in a transmission time of 900 sec. In performing the link analysis with a 50 watt transmitter, the bit rate was defined as 31,000 bits per sec. This resulted in total data of 2.5×10^7 bits during the initial pass of the Orbiter. The recommended sequence of events would be comparable to that identified for the out-of-orbit case as stated above. Stored entry would be transmitted initially followed by a period of image taking, then transmitting the science data, again followed by another imagery taking period, then another transmission of science data. This sequence would be repeated continuously for the period that the Orbiter is determined to be in range of the Lander. Since this Lander has been designed for a 1 day lifetime on the planet, subsequent transmission times would be initiated by sensing the presence of the Orbiter via the receiver and/or with a stored hardware program on board. When considering the capability of four transmission periods for a 3 day mission life (range assumed for parametric studies), the total data that can be obtained for the out-of-orbit case represents 32.4×10^7 bits. For the case of the direct entry mission a total data of 5.3×10^7 bits can be realized. The present concept reflects obtaining imagery data real time during the passage of the Orbiter over the Lander. Point C_D represents the design weight statement. C_P represents the parametric data for similar capability. The conclusion is that the results should be very close to that shown, given the same antenna and power system. The 14 lb difference is due to the parametric electrical power system.

2.3.1.3 Map 7 (Capability Estimates)

The landed subsystems block shows two limiting science payloads, between which lies the NASA/LRC specified minimum science payload, C_D . The real utility of the map is apparent when we consider the trade-offs associated with eliminating certain science

measurements, e.g., α scattering, and reducing the power requirements to a minimum compatible with some point such as C", which performs a reduced photo imaging and surface meteorology mission. Various aeroshell and parachute combinations can be investigated in conjunction with the preferred approach to the Lander design. Used in this context, the Lander Map provides the mission planner, the Capsule System designer, and the experimenter, the facility to lay out a mutually acceptable set of specifications for the Capsule design.

2.3.2 MINIMUM SCIENCE AND EXTENDED LIFETIME CAPSULE SYSTEMS (POINT DESIGNS 3 AND 4)

Point Designs 3 and 4 are described on Maps 8 and 9 (figs. 2.3.2-1 and -2). Table 2.3.2-1 summarizes the weight comparisons between point and parametric data. In the retardation block of Map 8, there are negligible weight variations between the point design weight list and the predicted parametric aeroshell weight.

The landed subsystems of 0/0-3-multi on Map 8 are weight listed as 528 lbs and denoted point C_D. The intersection of point C_D and the decelerator load specifies the design Lander case, denoted A_D. The corresponding parametric Lander point is located at point A_P. Based on the design data, the parametric data underestimates the attenuation requirements by 10 percent.

2.3.2.1 New NASA Wind Specification

From Map 8, it may be seen that for the lower maximum wind specification of 110 fps ($V_T = 100$ fps), there is a point B_P which has considerably greater capability than A_D. The possible approaches to utilizing this capability are very extensive. First, to take advantage of the new wind specification would mean a complete redesign in order to use the additional weight provided. Part of the weight could be budgeted to the recalibration of A_P to A_D which would place B_P at approximately A_D. An additional 100 lbs of decelerator load or 74 lbs of landed subsystems is available for such things as instrument hardening, deployment, and mission assurance.

2.3.2.2 Map 8 (Landed Block)

The 20 lb difference between design point C_D and parametric point C_P is due to the margin and redundancy which was assigned to the landed subsystems. This weight variation is due to an improved parametric EP&D subsystem.

In Lander Map 8 a plot of solar array weight versus surface lifetime is provided for different bits per day from the telecommunications system. This map was based on the concept of maximizing the landed subsystem for a specific mission design lifetime. Based on other studies which have been previously discussed for a specific mission design lifetime, worst case conditions are assumed for the subsystems. As an illustration, the Earth-Mars range and look angles for the communication system design would be based on the worst case condition occurring during the mission lifetime. In this map, three mission lifetimes were considered: 60, 90, and 120 days. In reviewing the curve, it can be seen that the maximum quantity of bits per day are

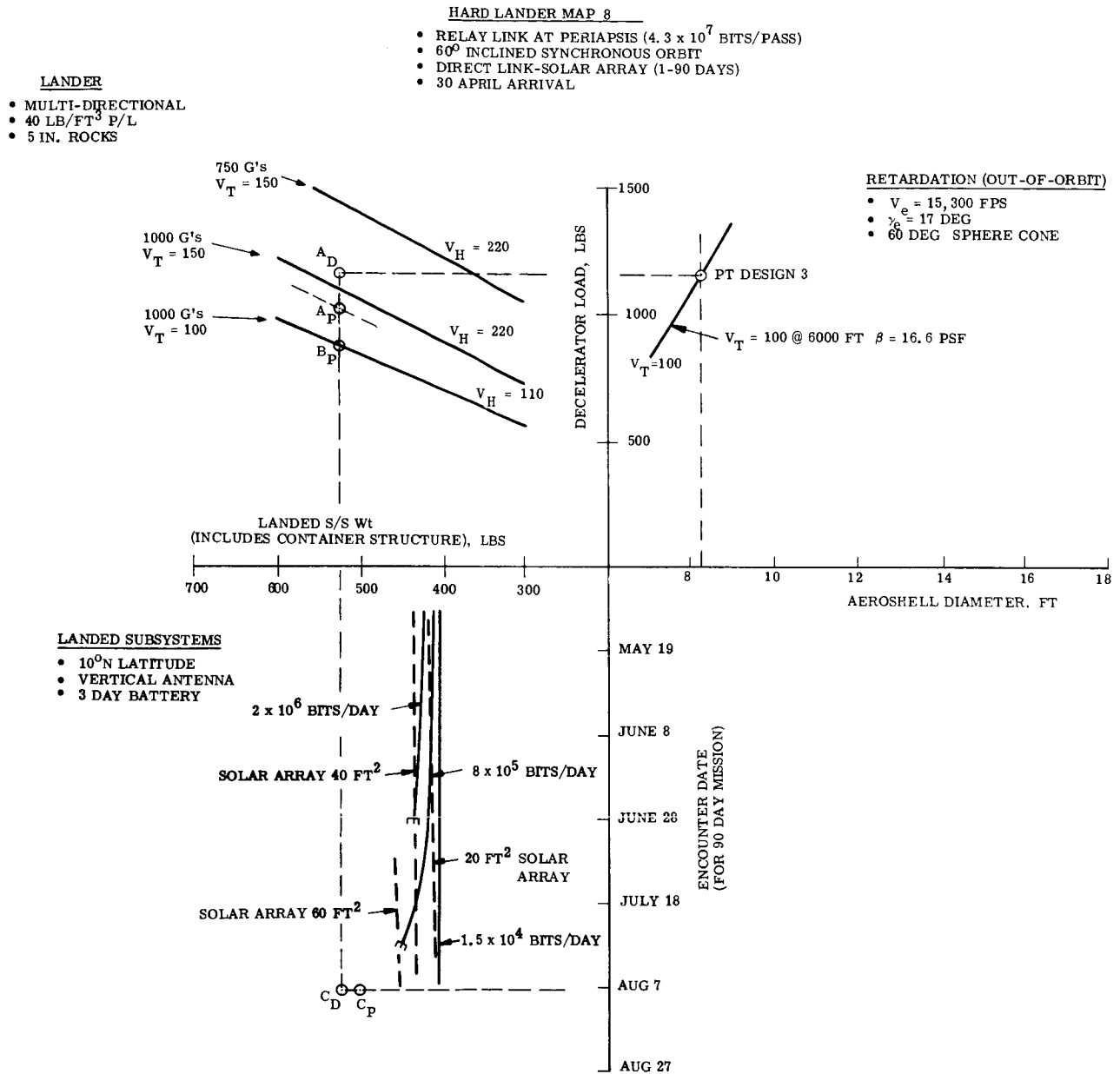


Figure 2.3.2-1. Hard Lander Map 8

HARD LANDER MAP 9

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60° INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK - SOLAR ARRAY (1-90 DAYS)
- 10 JAN ARRIVAL

RETARDATION (DIRECT)

- $V_E = 20,800$ FPS
- $\gamma_E = 33$ DEG
- 60 DEG SPHERE CONE

- LANDER
- MULTI-DIRECTIONAL
 - 40 LB/FT³ P/L
 - 5 IN. ROCKS

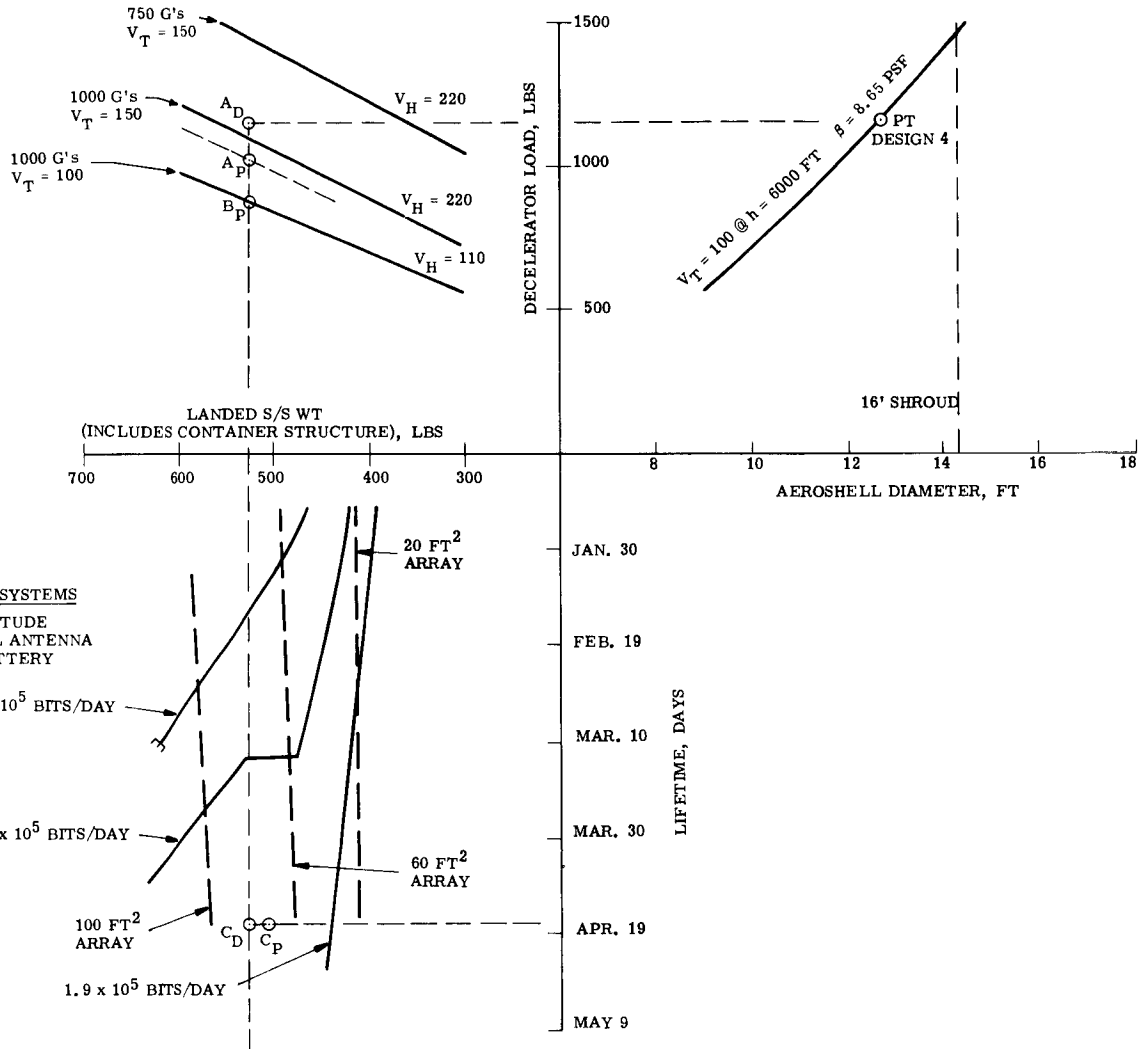


Figure 2.3.2-2. Hard Lander Map 9

TABLE 2.3.2-1. PARAMETRIC AND DESIGN COMPARISON FOR
MINIMUM SCIENCE/EXTENDED LIFE

Entry Mode	O/O		D/E	
Lander Mode	MULTI		MULTI	
Aeroshell Diam. (in.)	100		152.4	
Decelerator Wt (lbs)	1162		1162	
Design Point	Para	3	Para	4
Ballistic Coefficient	16.3	16.6	8.4	8.65
Capsule Wt (lbs)		1846		1979
Entry Wt (lbs)	1310	1324	1540	1647
Lander Wt (lbs)	900	934	900	934
Landed Wt (lbs)	508	528	508	528
Data Return ($\times 10^7$) Bits	2.3 ($P_T = 50w$)	0.38 ($P_T = 20w$)	0.25 (50w)	0.16 (20w)
Lifetime (Days)	90	90	90	90

obtained for the shortest mission duration whereas the minimum bits per day are reflected in the longest term mission design lifetime. These variations are due to the change in Earth-Mars range and "look angles" as well as the change in sun elevation angle with respect to the planet. In relating this parametric data to the 0/0-3-multi design, the solar array identified for this design consists of a 25 sq ft array. This point occurs on the curve at the intersection of the 90 day and 20 sq ft array intersection point. This difference is due to the fact that the map was based on a horizontally oriented solar array system. However, the 0/0-3-multi design, packaging was such that a 25 sq ft array could be handled within the 100 in. diameter aeroshell. Therefore, to minimize the complexity of the design, a non-oriented solar array system was used. Since the array is not pointed at the Sun, a degradation of about 25 percent is incurred and the array size was increased to 25 sq ft.

In considering the data return, the telecommunication subsystem design was based on a data return of 40,000 bits per day. This provides a total data return for a 90 day mission of approximately 3.6×10^6 bits.

2.3.2.3 Map 8 (Data Return Capability)

This data provides the capability for the meteorological data during the complete diurnal cycle at the rate of one sample per hour. In addition to the science, capability is provided for diagnostic as well as memory status. Using the command system, some form of degraded imagery capability is provided towards the end of the mission.

In addition to the capability provided by the direct link identified herein, additional capability is provided via the relay link during periapse passage whereby a total of 3.2×10^8 bits are obtained in four transmission periods extending over a 3 day mission for the out-of-orbit entry condition. The details pertaining to the relay link are included in Section 4.2.

2.3.2.4 Map 9 (D/E - MLX - 4-MULTI)

Table 2.3.2-1 summarizes the comparison results for D/E-4-MULTI design and parametric data. The 1162 lb decelerator load packages in a 152.4 in. aeroshell which compares well with the parametric data. For the multi-Lander constraints selected, the design is unconservative in the same sense as previously mentioned. However, for a D/E mission with a 10 June arrival date, the landed performance and data return is more strongly affected by mission lifetime. As lifetime increases to 90 days and beyond, the landed subsystems support is greater.

2.3.2.5 Map 9 (Landed Operations)

Map 9 presents the same type of data return information identified for Map 8, i. e., the landed subsystem block reflects subsystem design for specific mission design lifetimes. If Map 8 and Map 9 are compared, it is apparent that for 10 January encounter date the landed S/S weight is significantly greater by about 100 lbs. Since the only difference between the maps lies in the encounter date, the reasons are those previously cited in Section 2.2.4 where landed subsystem weight is presented as a function of encounter date.

Since the data rate curves are based on an optimum design link for a specific mission design lifetime, the sharp increase in weight encountered about 10 March reflects the transition from coded to uncoded data. At the reduced data rate that follows, an increase in solar array size and subsequent weight is required to sustain the same data rate. D/E-multi-4 is identified on this curve at the intersection of the 20 sq ft array curve and midway between 30 March and 19 April.

Although the 20 sq ft array is identified parametrically, the 25 sq ft was used in the design to eliminate the need for a horizontally oriented solar array which was the basis for the 20 sq ft array.

For D/E-4 multi Design 3 the available data rate is 11 bits/sec. With this data rate the total data return for 90 day mission with the direct link is 1.7×10^6 bits. The maximum daily data return based on array size is about 11000 bits/day. Approximately 6,000 bits would be required by science, diagnostic, and memory status. The additional capability can be used to extend the defined science capability of one sample

per hr. It is apparent from a mission standpoint that the data return for a 10 January encounter is significantly less than that of the 30 April encounter. Additionally, 6.0×10^7 bits are provided by the relay link for the periapse passages during support by the Orbiter.

2.3.3 EXTENDED SCIENCE AND EXTENDED LIFETIME CAPSULE SYSTEMS (POINT DESIGNS 5 AND 6)

The two designs investigated for the extended science and extended lifetime mission are summarized in table 2.3.3-1 and described in Maps 10 and 11 (figs. 2.3.3-1 and -2). The 0/0-5-multi design does not package in a 10-ft shroud for the 1299 lb specified decelerator load. Map 10 describes the trade-offs for comparison of parametric and design data.

2.3.3.1 Maps 10 and 11 (Landed Subsystem Block)

The landed subsystem block portion of Maps 10 and 11 presents the various weight penalties associated with the different science instruments. The weight of the individual science instrument consists of the following items: the apportioned weight of the power system and container, plus the weight of the deployment device required, plus the instrument's own weight. The power apportionment is based on the battery weight required for its operation where a device operates during the time the system is on batteries (nominally three days) or where single measurements are made (such as from a scatter device, subsurface water detector, etc.).

Where an instrument is operated over the 90 day mission lifetime, the array power penalty is charged to the instrument in addition to the power penalty associated with transmitting its data to Earth. This is based on an apportionment of its data compared to the total data transmitted for a single day.

Where a device is operated during the early stages of the mission, as well as for 90 days, the initial installed weight is considered in the first day mission. It is assumed that the basic capability of the Lander exists for this exercise; accordingly, with the removal of all the science payload, the landed subsystem weight of the Lander represents a fixed capability that has been provided for in the design.

One area of defined science instruments not covered by this curve is the life detection experiments. From the results of some general studies performed on presently defined instruments, the incorporation of these devices in a Hard Lander poses certain problems. The presently identified units are capable of surviving shock loads of up to 50 g's with an estimated capability of 100 g's to provide the local shock absorption capability. Increased stroke to limit the g-loads transferred to these instruments would not be feasible with the present packaging concept as discussed in the Lander Sensitivity Section, 2.2.3.

The subsystem block of Maps 10 and 11 provide the capability to select any specific instrument or combination of instruments and determine their effect on the Capsule performance. Due to the high data rate required of the imagery equipment,

this unit would obviously have the greatest influence on the solar array, since for a specific data rate the transmission time will be greater with the subsequent increase in power consumption and need for additional array capability.

TABLE 2.3.3-1. PARAMETRIC AND DESIGN COMPARISON FOR
EXTENDED SCIENCE/EXTENDED LIFE

Entry Mode	O/O		D/E	
Lander Mode	MULTI		MULTI	
Aeroshell Diameter (in.)	120		168	
Decelerator Wt (lbs)	1299		1299	
Design Point	Para	5	Para	6
Ballistic Coefficient	13.1	13.5	7.9	7.94
Capsule Wt (lbs)		2114		2227
Entry Wt (lbs)	1510	1530	1760	1766
Lander Wt (lbs)	1030	1053	1030	1053
Landed Wt (lbs)	625	623	625	623
Data Return ($\times 10^7$) Bits				
Lifetime (days)	90	90	90	90

2.3.3.2 Map 10 (Retardation and Lander Design)

The Capsule-aeroshell weights correlate well with the parametric results for the same reasons previously enumerated. The interesting aspect of Map 10 is that, based on the point design evaluation, the decelerator load would not package in the 10 ft shroud. However, the parametric data indicates that the design is conservative to the extent that with the new wind specification a smaller decelerator load is required and therefore the 10 ft shroud constraint is very close to attainable. Redesign of the Lander to account for the reduced attenuation requirements would probably permit the payload container structure to package in the 100 in. aeroshell. This is a significant result and permits the point design to be designed within a 10 ft shroud and still carry all of the expanded science.

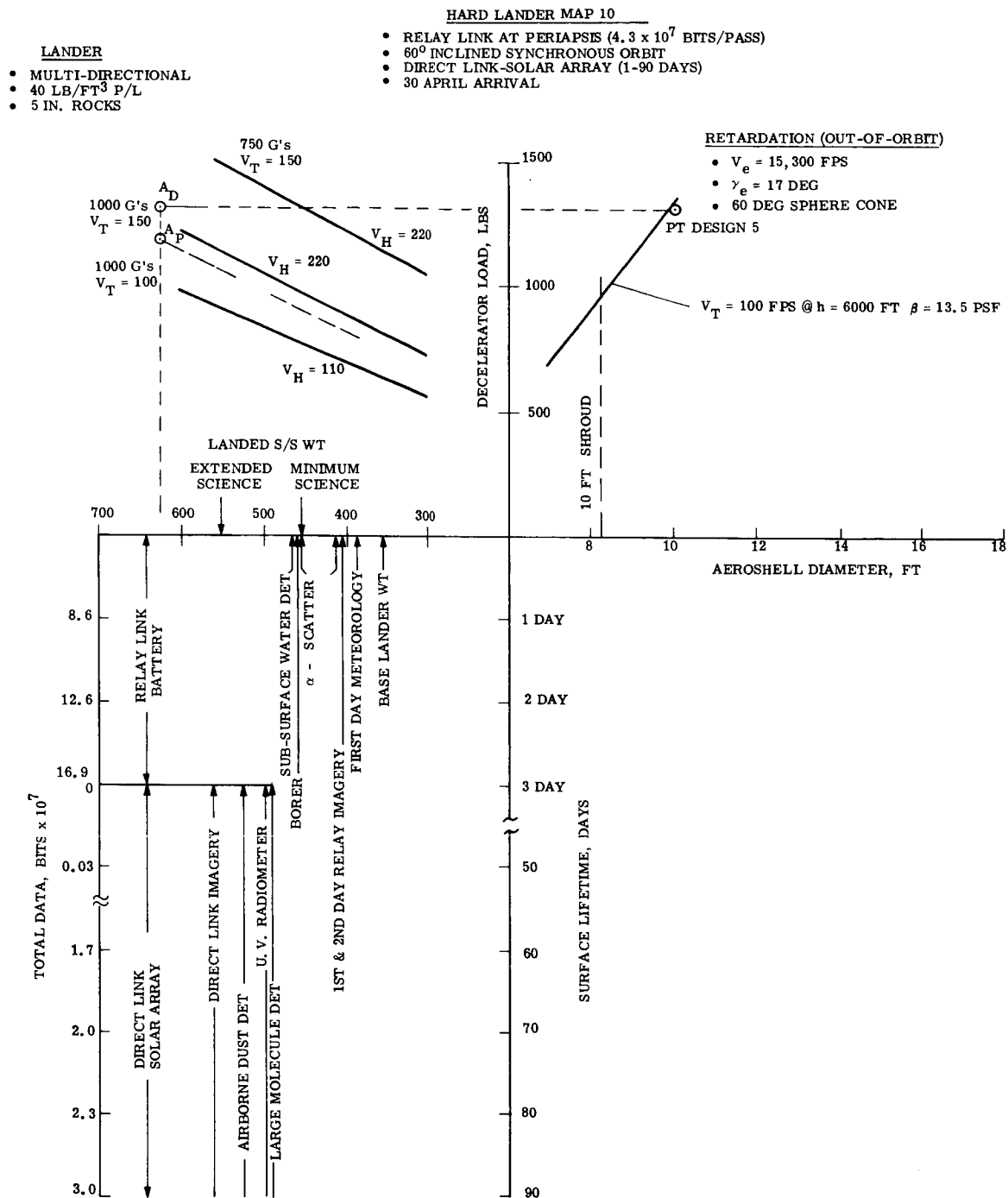


Figure 2.3.3-1. Hard Lander Map 10

HARD LANDER MAP 11

- RELAY LINK AT PERIAPSIS (4.3×10^7 BITS/PASS)
- 60 DEG INCLINED SYNCHRONOUS ORBIT
- DIRECT LINK - SOLAR ARRAY (1-90 DAYS)
- 10 JAN ARRIVAL

LANDER

- MULTI-DIRECTIONAL
- 40 LB/FT³ P/L
- 5 IN. ROCKS

RETARDATION (DIRECT)

- $V_E = 20,800$ FPS
- $\gamma_E = 33$ DEG
- 60 DEG SPHERE CONE

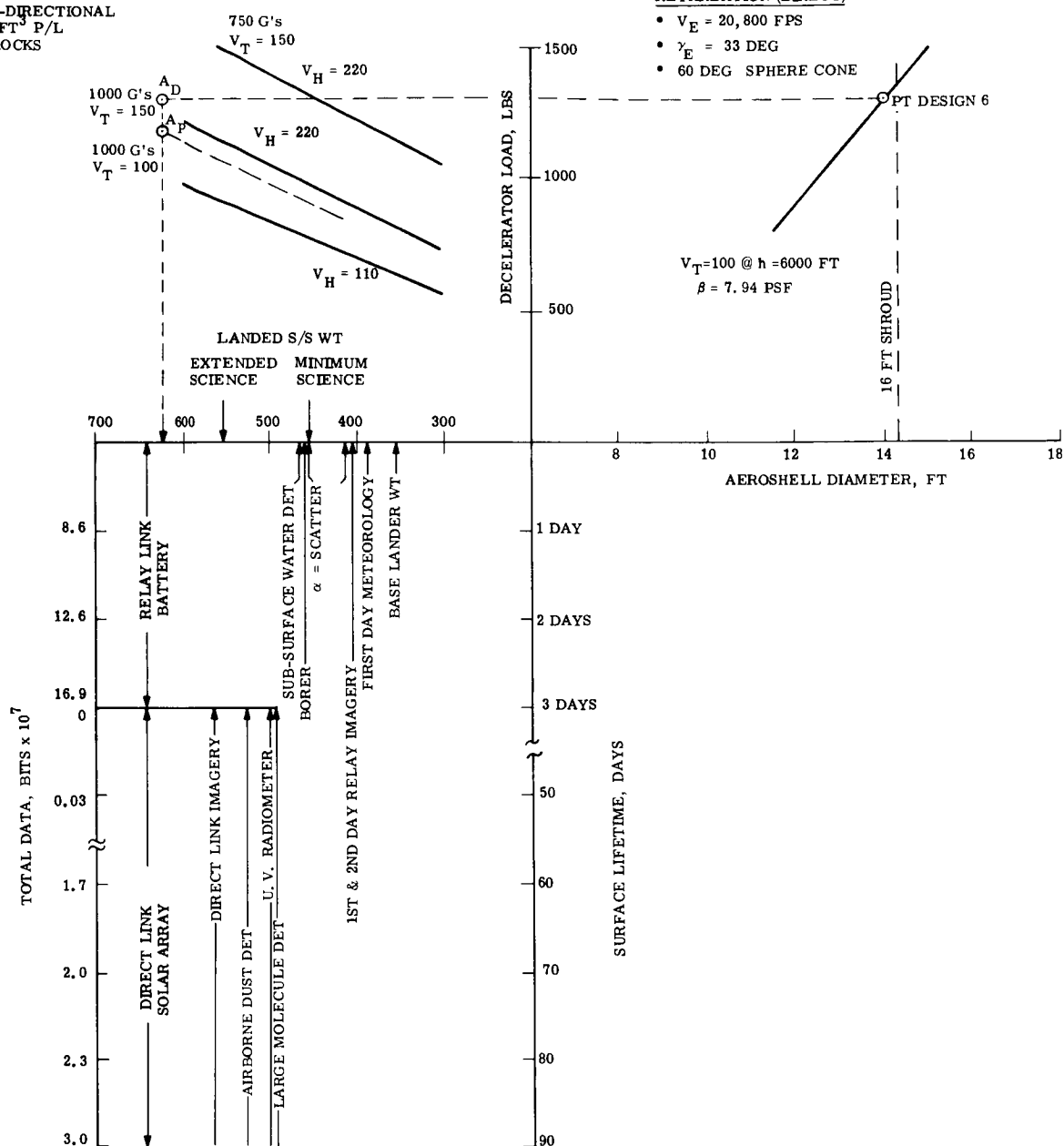


Figure 2.3.3-2. Hard Lander Map 11

2.3.3.3 Map 11

Map 11 (fig. 2.3.3-2) shows, for a ballistic mission, that the decelerator load exceeds the 10 ft shroud capability and requires hammerheading. However, the use of the lifting or inflatable aeroshell concepts described in Map 4 of Section 2.1 in conjunction with a 10 ft shroud would have merit for the direct mission with extended science. The concept is being investigated, and it does appear that such a mission would be feasible to consider for this class of payload.

3. ENTRY AND RETARDATION SUBSYSTEMS

3. ENTRY AND RETARDATION SUBSYSTEMS

3.1 ENTRY VEHICLE STUDIES

3.1.1 GENERAL CONFIGURATION PERFORMANCE CONSIDERATIONS

The design and development of an entry aeroshell and Retardation Subsystem for a Mars Hard Lander is well within the current state-of-the art. In the design of the aeroshell several design approaches and materials are available which will perform the job; however, trade-offs must be performed to select the optimum approach. A honeycomb structure of either aluminum or stainless steel combined with an elastomeric heat shield is the recommended approach for the aeroshell. The Parachute Retardation Subsystem design is a straight forward application of current technology employing the results of the NASA - LRC PEPP program. Dacron is selected as the textile material for the Parachute Subsystem both to satisfy the sterilization requirement and because yarn is available for weaving into specific material. The unique features of the parachute design are the large size requirement, low deployment dynamic pressure, and supersonic deployment Mach number. A parachute design satisfying such requirements has been demonstrated in the NASA/LRC PEPP program. However, optimization of a particular design and subsequent development testing for a specific Mars Lander must still be performed, as in any Parachute Subsystem application.

These comments apply to the use of a ballistic aeroshell combined with a parachute aerodecelerator, which has been the primary approach for the Mars Hard Lander study. This concept is entirely adequate for the out-of-orbit mode of atmospheric entry. For the direct entry mode in the limiting case (40°), the entry vehicle has a low ballistic coefficient (on the order of 0.175 slugs/ft^2) to satisfy parachute deployment requirements. Combining the low ballistic coefficient value with the desired 10-ft shroud limit (aeroshell diameter of 8.34 ft), results in a decelerator load lower than that required for the minimum science payload. Consequently, alternate means of increasing the decelerator load were studied in an attempt to determine what modifications would be required to the aeroshell or to the booster to carry larger payloads in the direct entry mode.

Fig. 3.1.1-1 presents the entry concepts that have been investigated during this study. As just mentioned, the ballistic aeroshell was considered for both the direct and out-of-orbit entry modes. The basic 8.34 foot aeroshell diameter limit in the 10 foot shroud was increased in an iteration in which bulbous shrouds up to 16 feet were assumed permissible for the direct entry mode. For ballistic entry, two other concepts were considered that might increase the payload for direct entry while not exceeding the 10-ft shroud constraint: the use of an inflatable after body (second sketch, fig. 3.1.1-1) and extendable flaps. Both devices would be deployed after separation from the Orbiter, but before entry.

Another concept for increasing direct entry vehicle payload, to compare with values obtained for out-of-orbit entry, is the use of aerodynamic lift. Since only a small amount of lift is required when combined with reasonably low ballistic coefficients, the blunt lifting configurations shown in fig. 3.1.1-1 were selected as a set of shapes that provided both lift and reasonable packaging efficiencies. Offset centers of gravity were used for the blunt cone and Apollo Command Module-shaped configurations to obtain lift through balancing out vehicle aerodynamic moments at a trim angle of attack. The raked-cone shape obtains its lift at a zero angle-of-attack trim condition in a similar balancing of the vehicle moments.

The detailed discussions of this section present the results of the studies just outlined. These results include the parametric data prepared for the ballistic aeroshell entry concept and for the alternate concepts. The parametric data for the aerodecelerator are presented in the trade-offs associated with optimization of the combined entry vehicle and aerodecelerator. Since the lifting vehicle studies are a continuing effort, preliminary results are presented herein with results of the further efforts to be separately presented.

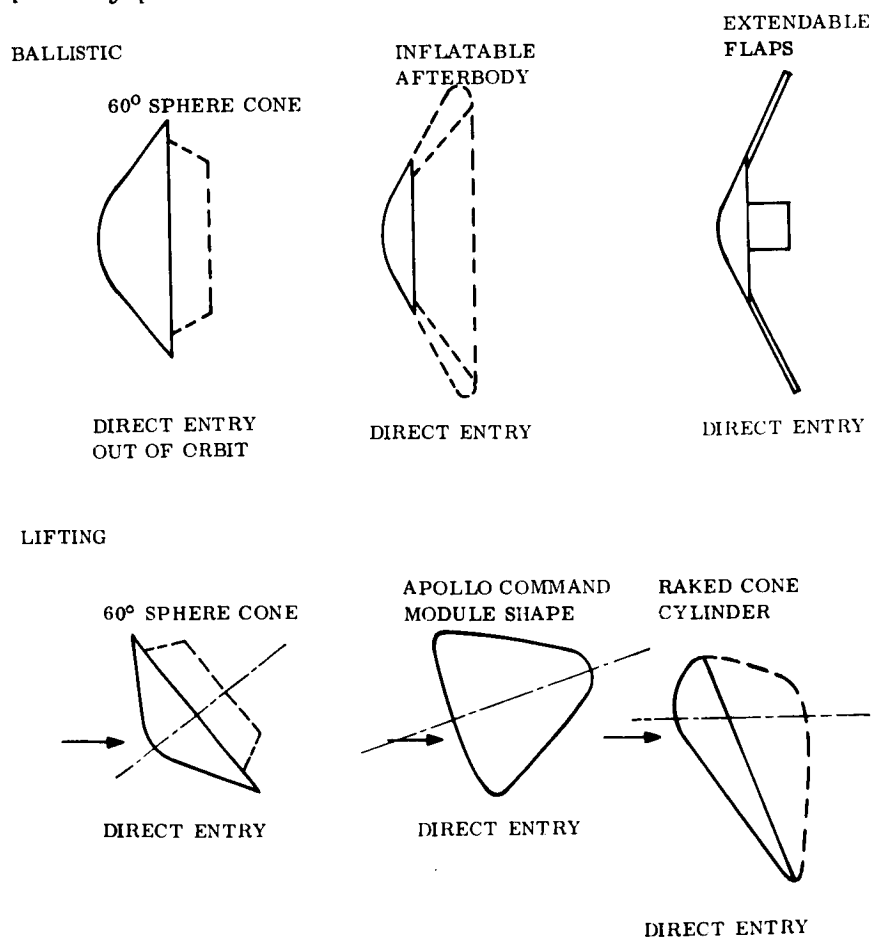


Figure 3.1.1-1. Entry Concepts

3.1.2 BALLISTIC ENTRY CONFIGURATIONS

3.1.2.1 Requirements and Technology

The trade-offs associated with the selection of an aeroshell configuration involve the aeroshell performance, weight, and payload packaging capability. The aeroshell parametric data used as input to these trade-offs was based on a set of specific requirements and constraints, as well as on current entry vehicle technology. A discussion of these requirements, constraints, and technology follows.

A goal of this study was that the aeroshell, along with associated separation hardware and canister, fit within the Titan 10-ft shroud limit for both the direct and out-of-orbit entry modes. Bulbous shroud configurations up to 16 feet could be considered, but with attendant penalties in booster lift-off capability and in program cost. The parametric data shows results for aeroshell diameters up to 15 ft with the two (10 feet and 16 feet) shroud limits indicated on appropriate trade-off curves.

Entry trajectory calculations were made at one out-of-orbit condition ($V_e = 15,300$ ft/sec, $\gamma_e = 16^\circ \pm 4^\circ$) and at two direct entry conditions, ($V_e = 20,800$ ft/sec, $\gamma_e = 30 \pm 10^\circ$ and $V_e = 20,800$ ft/sec, $\gamma_e = 22^\circ \pm 3^\circ$). These entry conditions were selected as a conservative basis for the parametric tasks and were not changed to reflect less stringent requirements identified in the reference mission results of Volume II. Sufficient data exists, however, to determine overall entry vehicle performance for the specific entry conditions derived from these mission analysis results.

The aeroshell configurations used in the parametric tasks provide the high drag and low weight required for entry into the tenuous Mars atmosphere. The configurations are sphere cones with cone semi-vertex angles ranging from 50 to 70° and bluntness ratios (R_N/R_B) of 0.2 to 0.8 . These configurations easily accommodate the deepdish Hard Lander design with considerable static stability at small angles-of-attack. The trade-off within this family of sphere cones is essentially between the desire for high drag, low weight, and the need for dynamic stability augmentation. While the need for stability augmentation is difficult to ascertain parametrically, without a rather concise description of the vehicle configuration, mass properties, aerodynamics and entry trajectory, an attempt has been made to indicate the type of stability problems that might be encountered and the relative effectiveness of the configurations in passively overcoming such difficulties.

3.1.2.1.1 Trajectory Performance Trade-Offs

The ballistic vehicle trade-off studies described here required 150 trajectories to cover the necessary range of atmospheres, ballistic coefficients, and entry conditions. The bulk of the work utilized the VM-3 and VM-8 atmospheres since these represent the

worst heating and loads cases, respectively; however, some trajectories were also compiled in each of the other eight Voyager atmospheres (ref. 3-1). The spread in the Voyager atmospheres brackets the maximum and minimum atmospheres of ref. 3-2. Ballistic coefficients (β) from 3 to 30 psf (0.1 to 1 slug/ft²) were considered with the bulk of the work centered at the lower β 's. Fig. 3.1.2-1 shows the entry velocity-flight path angle ($V_e - \gamma_e$) map which was studied. The entry altitude is defined here as 220 km (722 kft). The atmosphere model above this altitude has a negligible effect on the trajectories. Fig. 3.1.2-2 may be used to compare these results with other trajectory data compiled for a re-entry altitude of 244 km (800 kft).

Summary plots and tables of the computed trajectories are presented together with trajectory parameter plots for sample runs in the interest of conciseness. The low entry velocity - shallow angle trajectories of interest in the out-of-orbit missions will be presented separately from the direct entry data (high velocity - high angle). The thermal environment results will be discussed in another section and are not duplicated here. Therefore, the parameters of most interest in this section are those related to loads and parachute deployment. The parameters most affecting the loads determination are the peak dynamic pressure and peak axial acceleration. The parameters for parachute deployment include the altitude at $M = 2$; the altitude at the beginning of terminal fall; flight path angles at $M = 2$ and at 20 kft; and the dynamic pressures at $M = 2$, at 20 kft, and at 15 kft. The integrated dynamic pressure is useful in sizing the control system, and the time from entry to impact is factored into the Communication Subsystem analysis. These 11 parameters are tabulated in table 3.1.2-1 for an out-of-orbit entry into the 10 Voyager atmospheres given in ref. 3-1. These trajectories were initiated at an entry velocity of 15,300 ft/sec with an entry path angle of 16° for an entry altitude of 722 kft. The vehicle is a 60° sphere cone with a ballistic coefficient of 0.5 slug/ft². These same 11 parameters are given in tables 3.1.2-2 and -3 for direct entries at path angles of 20° and 40° , respectively. The velocity for direct entry in this case is 20,800 ft/sec at 722 kft altitude and the ballistic coefficient is 0.155 slug/ft².

Parametric data will be shown in the VM-8 atmosphere since the worst loading and parachute deployment conditions occur there. Figs. 3.1.2-3 through -7 show parametric data for out-of-orbit entries. Fig. 3.1.2-3 shows the altitude at $M = 2$ as a function of the ballistic coefficient (β) for entry path angles (γ_e) of 16° and 20° at a $V_e = 15,300$ ft/sec and for a $\gamma_e = 16^\circ$ at a $V_e = 14,000$ ft/sec. Fig. 3.1.2-4 shows the altitude at Mach 2 as a function of entry path angle for vehicles with β 's of 0.311, 0.465, and 0.560 slug/ft². The entry velocity in this case is 15,300 ft/sec. Figs. 3.1.2-5, -6, and -7 show the peak dynamic pressure, peak axial load, and integrated dynamic pressure, respectively, as a function of β for the same V - γ conditions shown in fig. 3.1.2-3. Figs. 3.1.2-8 through -15 show parametric data for a direct entry mission. Direct entry missions all appear to have entry velocities of about 20,800 ft/sec, the velocity selected for the presented parametric data. Trajectories

were also computed for velocities from 14,000 to 24,000 ft/sec but presentation of that data does not seem warranted. Fig. 3.1.2-8 shows the altitude at $M = 2$ as a function of β for entry path angles of 20° , 25° , 30° , and 40° into the VM-8 atmosphere, the assumed worst case. Fig. 3.1.2-9 shows the same data cross-plotted as a function of entry path angle for β 's of 0.155, 0.217, 0.311, 0.466 and 0.620 slug/ft². It is conceivable that for steep re-entries the thermal shock load might be significant when compared to the aerodynamic load. Thus, the loading data is shown for both the VM-3 and VM-8 atmospheres. Figs. 3.1.2-10 through -15 show the loading information (peak q_∞ , peak A_x and integrated q_∞) as a function of β for γ 's of 20° , 25° , 30° , and 40° .

Figs. 3.1.2-16 through -31 show the point mass trajectory parameters (altitude, velocity, flight path angle, Mach number, dynamic pressure, and axial load) for eight trajectories. These trajectories are nominal and design limit cases (worst heating and worst loads) for both direct and out-of-orbit entries of a 60° sphere-cone body. The axial load is given in Earth's 'g's (as opposed to martian 'g's). The worst heating case results from a shallow entry into the VM-3 atmosphere. The worst loading case occurs for a steep entry in the VM-8 atmosphere.

The out-of-orbit sample trajectories were initiated at an entry velocity of 15,300 ft/sec with an entry path angle of $16^\circ \pm 2^\circ$ at an entry altitude of 244 km (800 kft). A ballistic coefficient of 14 psf (0.435 slug/ft²) was used which is near the maximum allowable. Figs. 3.1.2-16 and -17 show the trajectory parameters for the out-of-orbit design heating case ($\gamma_e = 14^\circ$ into VM-3) while figs. 3.1.2-18 and -19 show the design loading trajectory ($\gamma_e = 18^\circ$ into VM-8). Figs. 3.1.2-20 through -23 present the parameters for nominal trajectories ($\gamma_e = 16^\circ$) in both VM-3 and VM-8.

The direct entry sample trajectories were initiated at an entry altitude of 220 km (722 kft) with a velocity of 20,800 ft/sec and a path angle of $25 \pm 9^\circ$. The vehicle in this case has a ballistic coefficient of 7 psf (0.217 slug/ft²). Figs. 3.1.2-21 through -25 show the direct entry design heating trajectory ($\gamma_e = 16^\circ$ into VM-3). Figs. 3.1.2-26 and -27 show the trajectory parameters for the design loading trajectory ($\gamma_e = 34^\circ$ into VM-8). Figs. 3.1.2-28 through -31 show the nominal ($\gamma_e = 25^\circ$) entry into the VM-3 and VM-8 atmospheres.

TABLE 3.1.2-1. OUT-OF-ORBIT TRAJECTORY SUMMARY

$V_e = 15,300 \text{ fps}, \gamma_e = 16 \text{ deg}, m/C_{DA} = 0.5 \text{ slug/ft}^2$											
	VM-1	VM-2	VM-3	VM-4	VM-5	VM-6	VM-7	VM-8	VM-9	VM-10	
Peak Ax ~	5.8	14.5	5.8	14.2	5.9	12.4	5.7	13.8	6.0	11.1	
Peak q_∞ ~	92	232	93	227	94	198	91	222	96	178	
h at M = 2 ~	48.0	24.8	65.0	33.7	80.3	47.8	31.6	15.2	96.9	64.7	
h: Terminal Fall ($G_X = 0.45$) ~	15.0	9.0	20.5	14.3	34.5	27.8	-	-	45.5	90.0	
Time to 20 kft	288.5	255.3	308.2	269.3	324.8	291.0	272.0	242.8	345.7	311.2	
Path angle at M = 2 ~	23.1	21.5	23.6	21.2	23.5	21.2	22.9	21.7	23.5	21.5	
Path angle at ~ 20 kft	39.3	26.7	53.1	39.8	63.6	56.8	28.5	18.3	73.9	67.2	
$\int q_\infty dt$ ~	7300	7400	8900	7900	8300	7700	7500	8300	8900	9100	
Dynamic Pressure at M = 2	17	17	18	18	18	18	17	17	18	18	
Dynamic Pressure at 20 kft	10	13	9	9	9	8	13	22	9	12	
Dynamic Pressure at 15 kft	9	11	9	9	9	8	12	16	9	12	

TABLE 3.1.2-2. DIRECT ENTRY TRAJECTORY SUMMARY

		$V_e = 20,800 \text{ fps}, \gamma_e = 20 \text{ deg}, m/C_{DA} = 0.155 \text{ slug/ft}^2$									
		VM-1	VM-2	VM-3	VM-4	VM-5	VM-6	VM-7	VM-8	VM-9	VM-10
Peak Ax	~	13.1	33.2	13.0	32.4	13.5	28.3	12.9	33	13.6	25.4
Peak q_∞	~	66	166	67	162	62	142	64	165	68	127
h at M = 2	~	92	46	109.5	52.4	124.8	67.3	76	38.7	141	86.3
h: Terminal Fall	~	49.5	31.1	60.2	34.9	59.2	53.3	37.0	22.8	62.8	92.2
Time to 20 kft		268	207	298	224	329	250	245	192	371	304
Path angle at M = 2	~	23	21	23	20.5	22.8	21.2	22.7	20.9	23.2	21.5
Path angle at 20 kft	~	75.2	65.5	83.4	76.8	87.3	85.2	75	51.7	89.4	89
$\int q_\infty dt$	~	3800	3400	3700	3400	3300	3600	3400	3300	3900	3500
Dynamic Pressure at M = 2		7	7	7	7	7	8	7	7	7	7
Dynamic Pressure at 20 kft		3	3	3	3	3	3	3	3	3	3
Dynamic Pressure at 15 kft		3	3	3	3	3	3	3	3	3	3

TABLE 3.1.2-3. DIRECT ENTRY TRAJECTORY SUMMARY

$V_e = 20, 800 \text{ fps}, \gamma_e = 40 \text{ deg}, m/C_{DA} = 0.155 \text{ slug/ft}^2$											
		VM-1	VM-2	VM-3	VM-4	VM-5	VM-6	VM-7	VM-8	VM-9	VM-10
Peak Ax	~	32.3	91.9	32.0	89.5	32.3	75.6	32.3	87.5	32.2	66.7
Peak q_∞	~	162	459	162	447	161	378	161	432	161	333
h at M = 2	~	53.3	25	71.3	33.2	86.7	46.8	36	16	104	65.9
H = terminal fall	~	11.3	4.3	33.1	15.6	48.5	29.3	-	-	61.7	46.1
Time to 20 kft		116	73	152	92	186	128	90	64	231	174
Path angle at M = 2	~	40.5	39.9	40.7	40	40.8	40	40.5	39.8	40.8	40
Path angle at 20 kft	~	61.4	42.9	77.5	58.4	85.1	79	46.4	38.5	88.9	87.8
$\int q_\infty dt$	~	3100	3200	3600	3100	3700	3700	3300	3100	4000	3200
Dynamic Pressure at M = 2		16	18	16	17	16	18	16	17	16	18
Dynamic Pressure at 20 kft		3	8	3	3	3	3	6	30	3	3
Dynamic Pressure at 15 kft		3	5	3	3	3	3	5	15	3	3

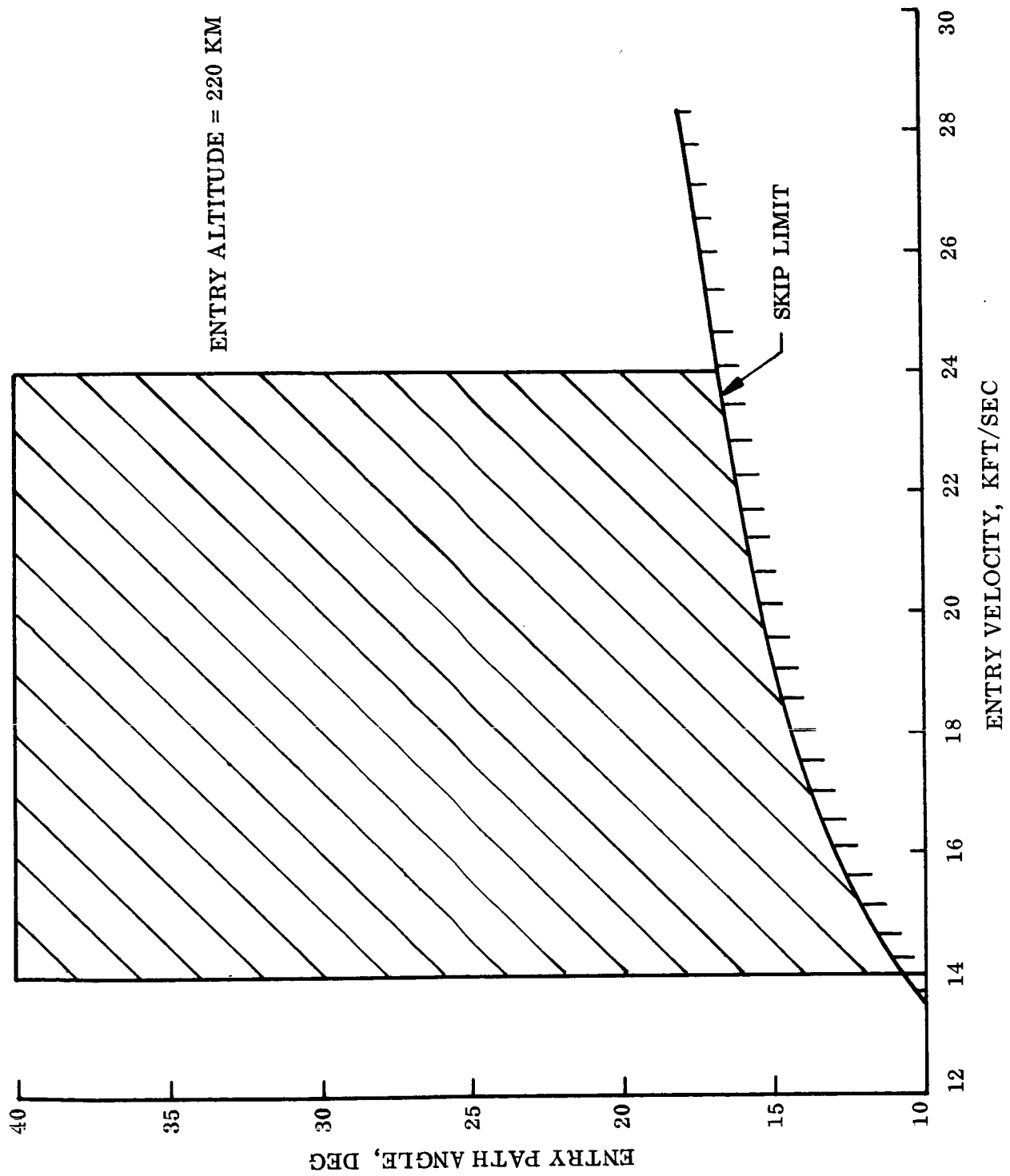


Figure 3.1.2-1. Entry V- γ Map

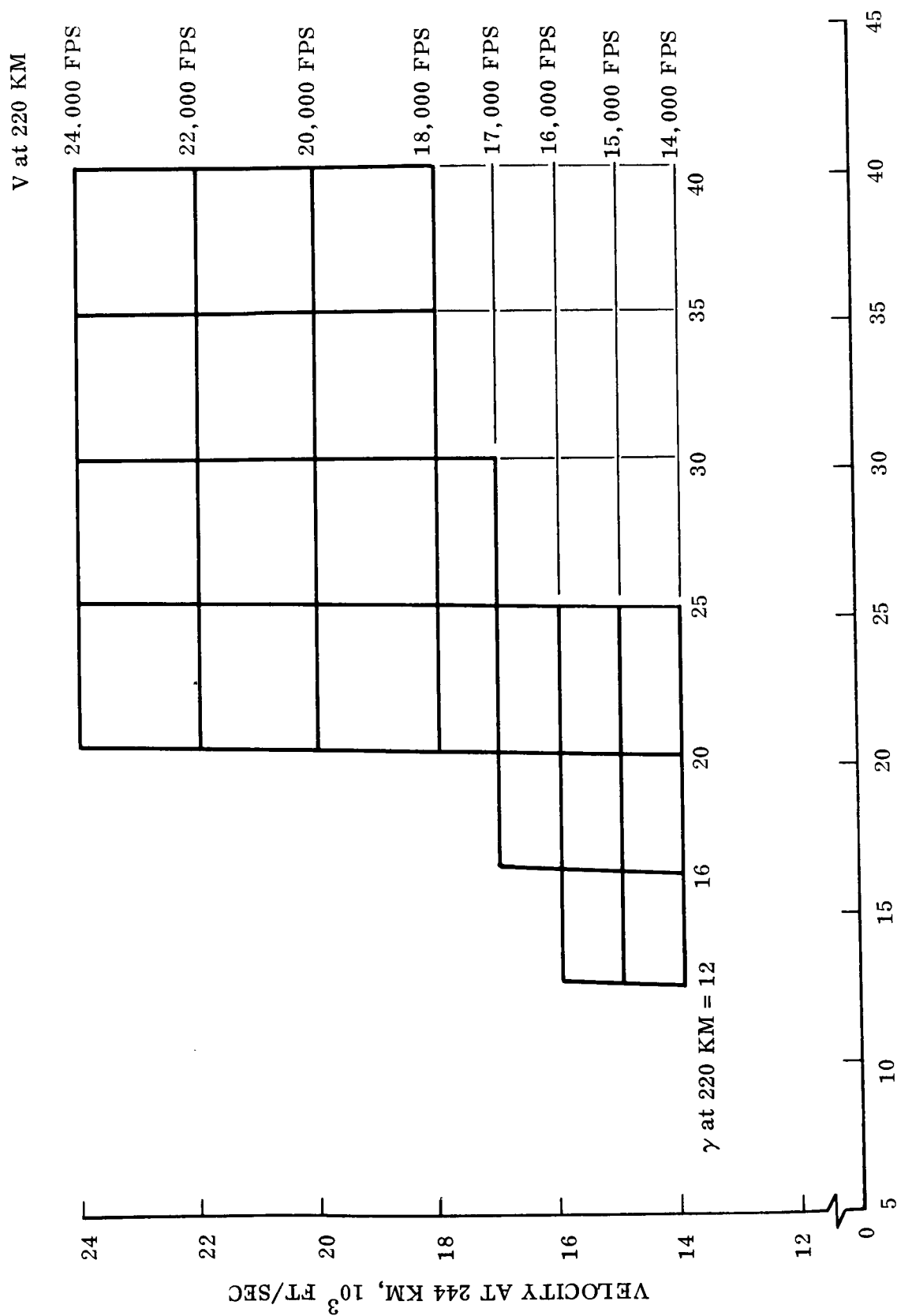


Figure 3.1.2-2. Entry Condition Conversion Chart

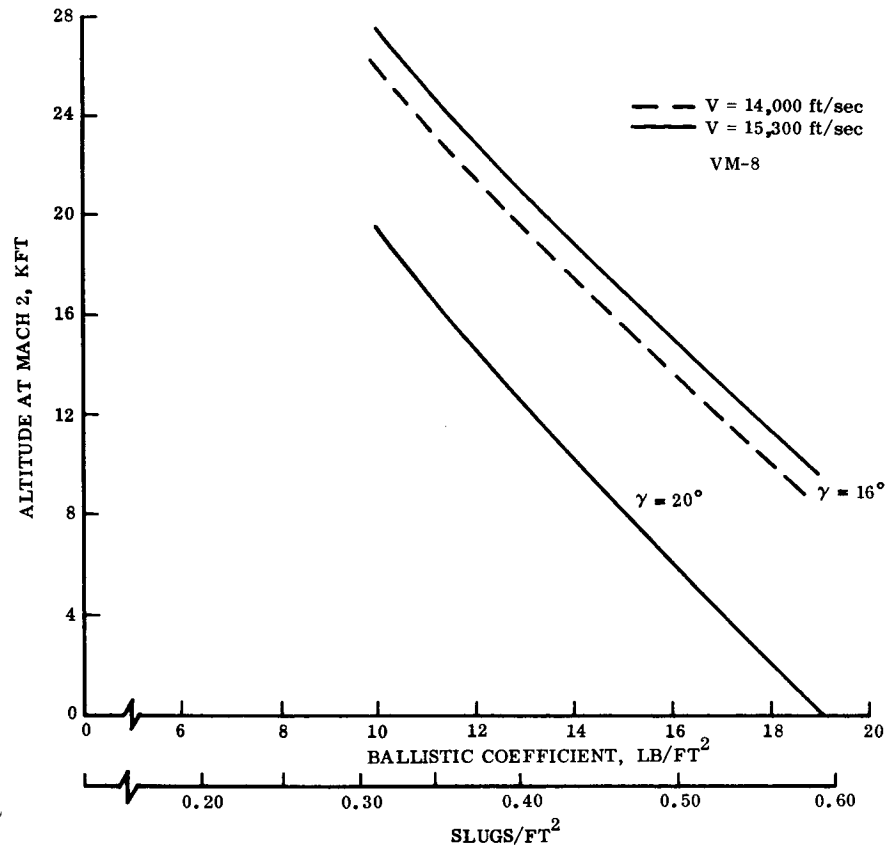


Figure 3.1.2-3. Variation of Altitude at Mach 2 With Ballistic Coefficient, Out-of-Orbit Entry

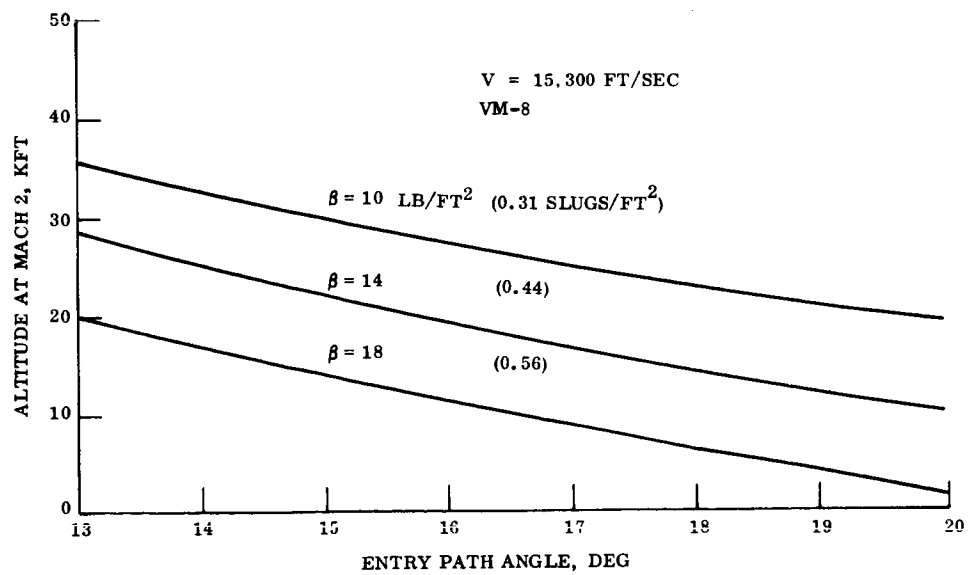


Figure 3.1.2-4. Variation of Altitude at Mach 2 With Entry Angle, Out-of-Orbit Entry

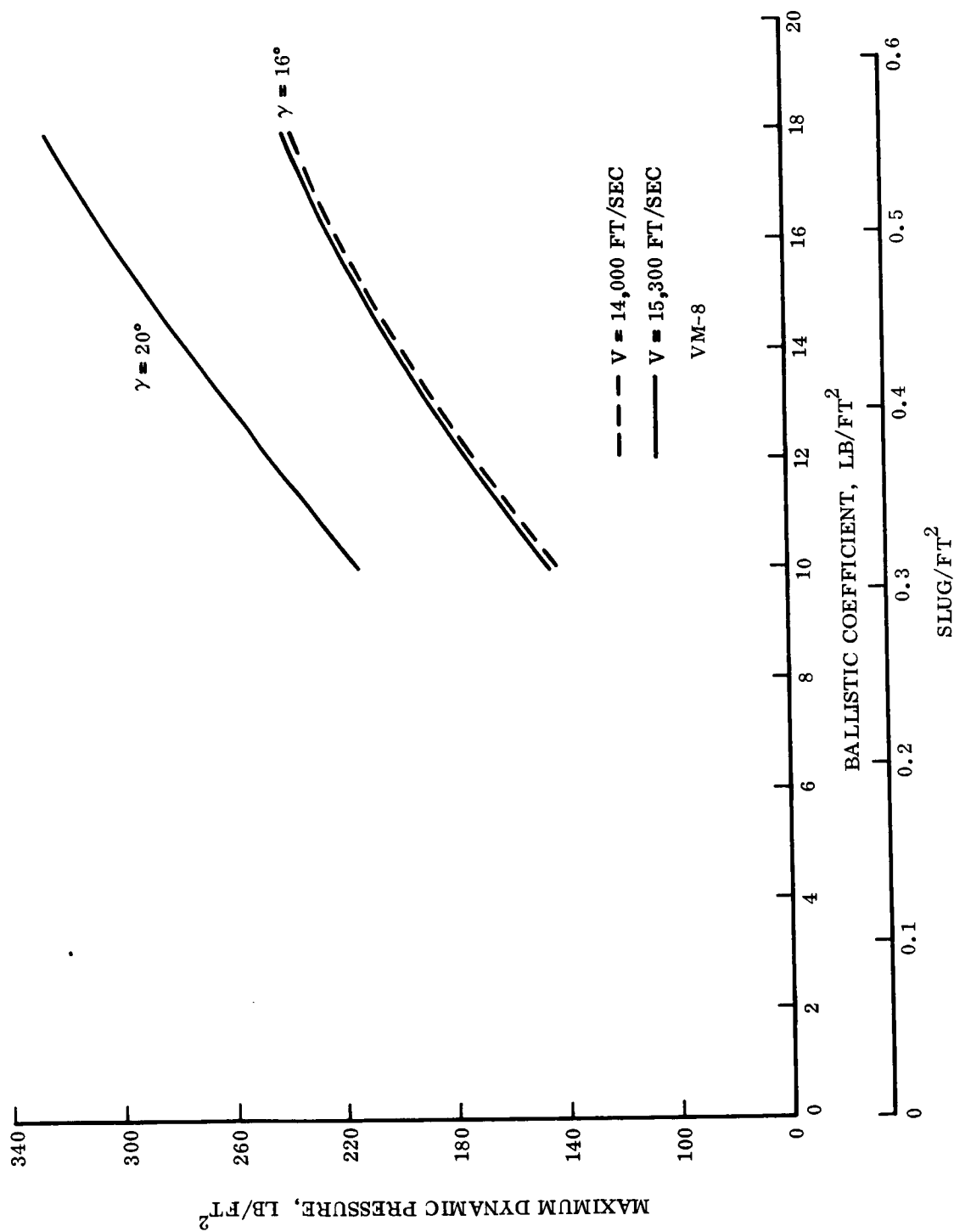


Figure 3.1.2-5. Maximum Dynamic Pressure, Out-of-Orbit Entry

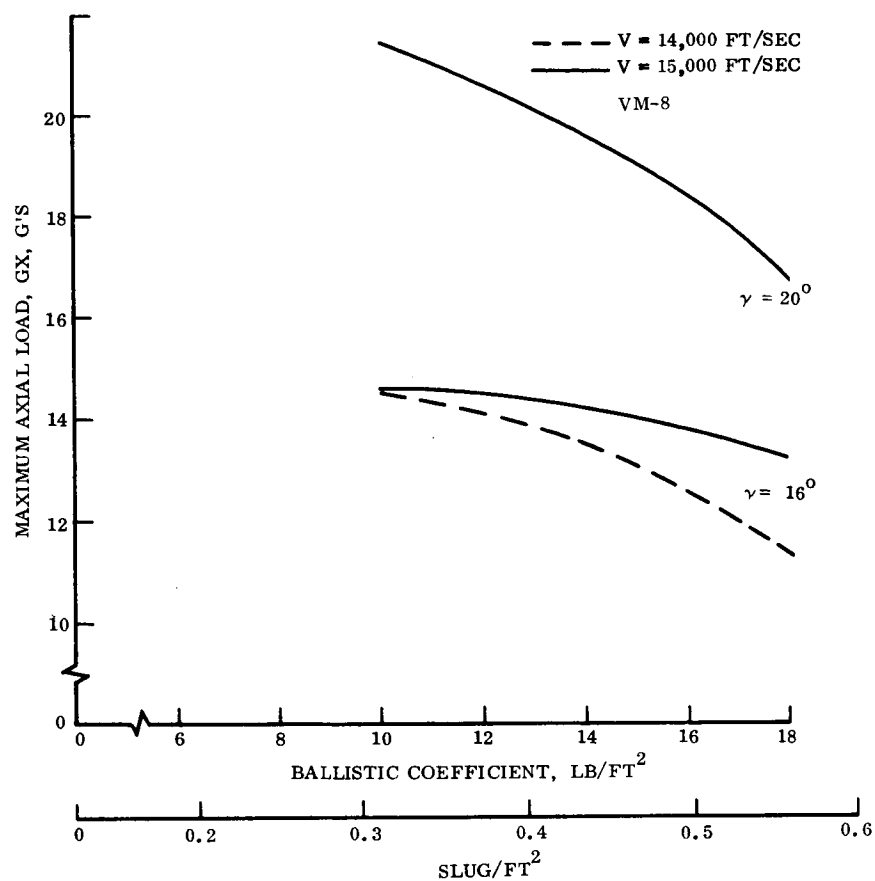


Figure 3.1.2-6. Maximum Axial Load, Out-of-Orbit Entry

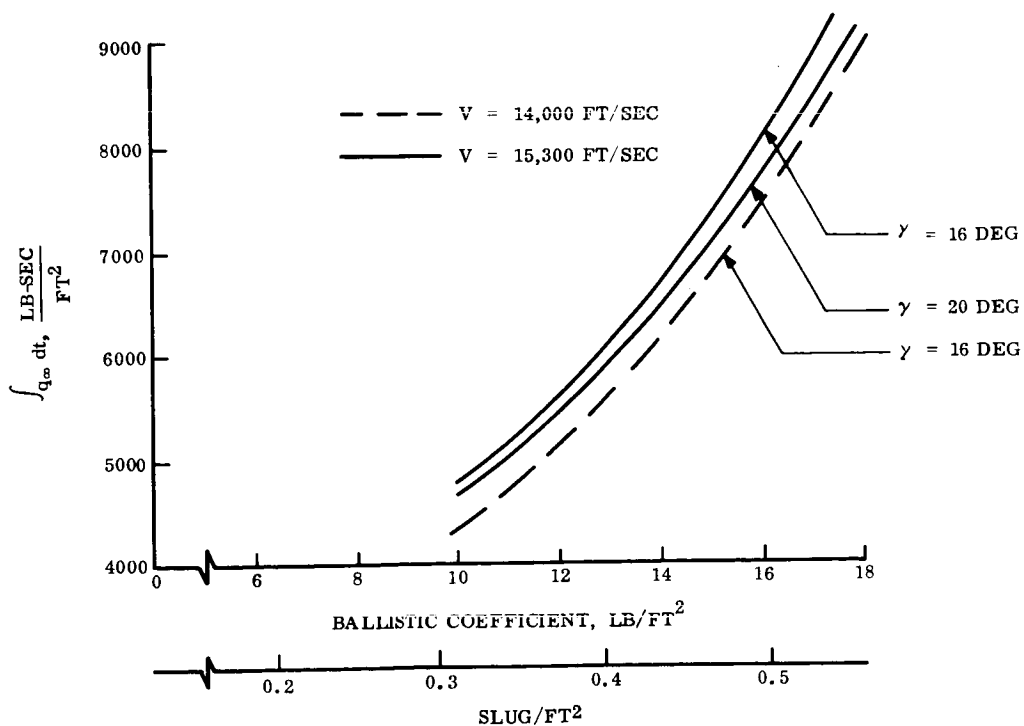


Figure 3.1.2-7. Integrated Dynamic Pressure, Out-of-Orbit Entry

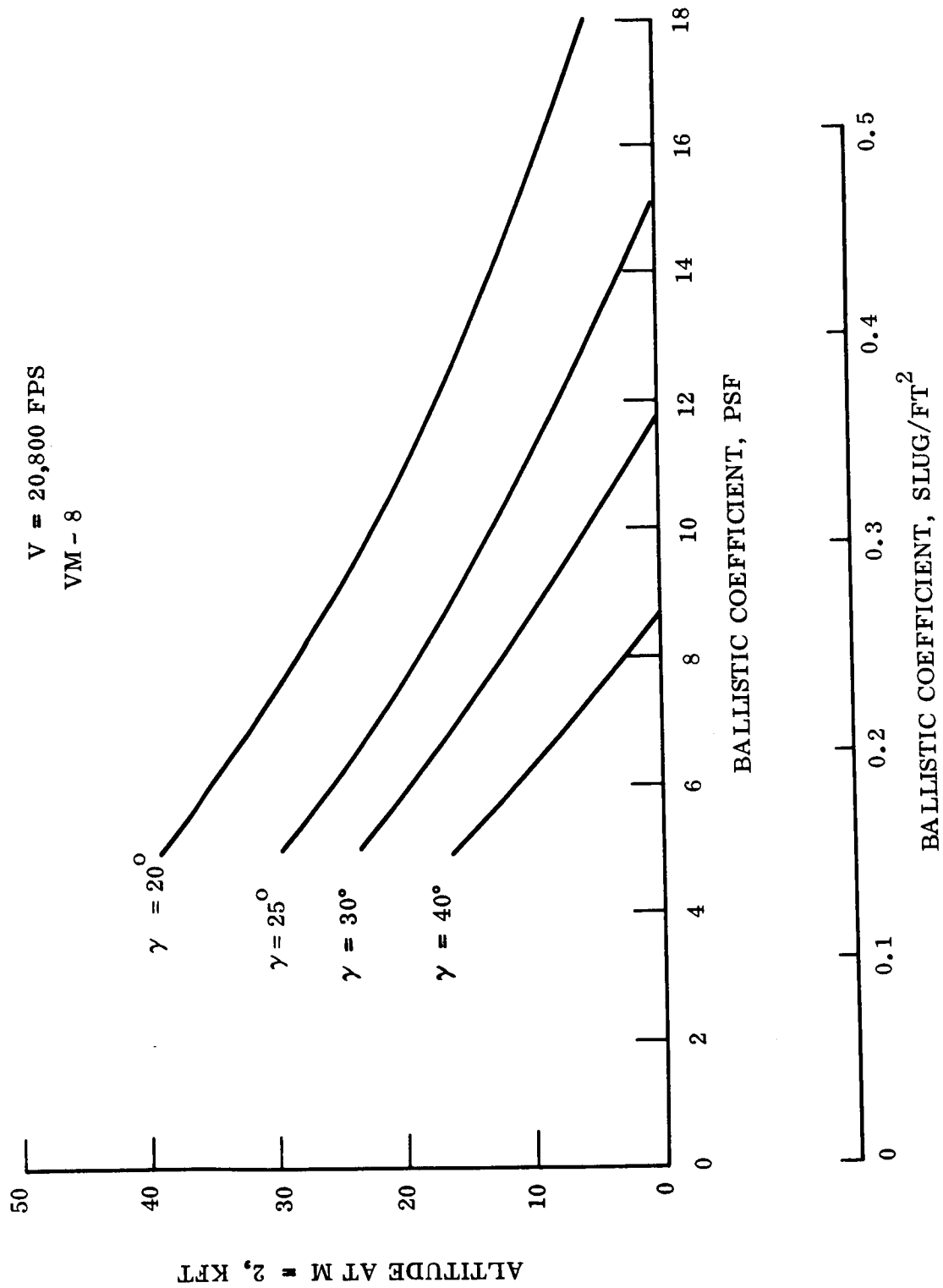


Figure 3.1.2-8. Variation of Altitude at Mach 2 With Ballistic Coefficient, Direct Entry

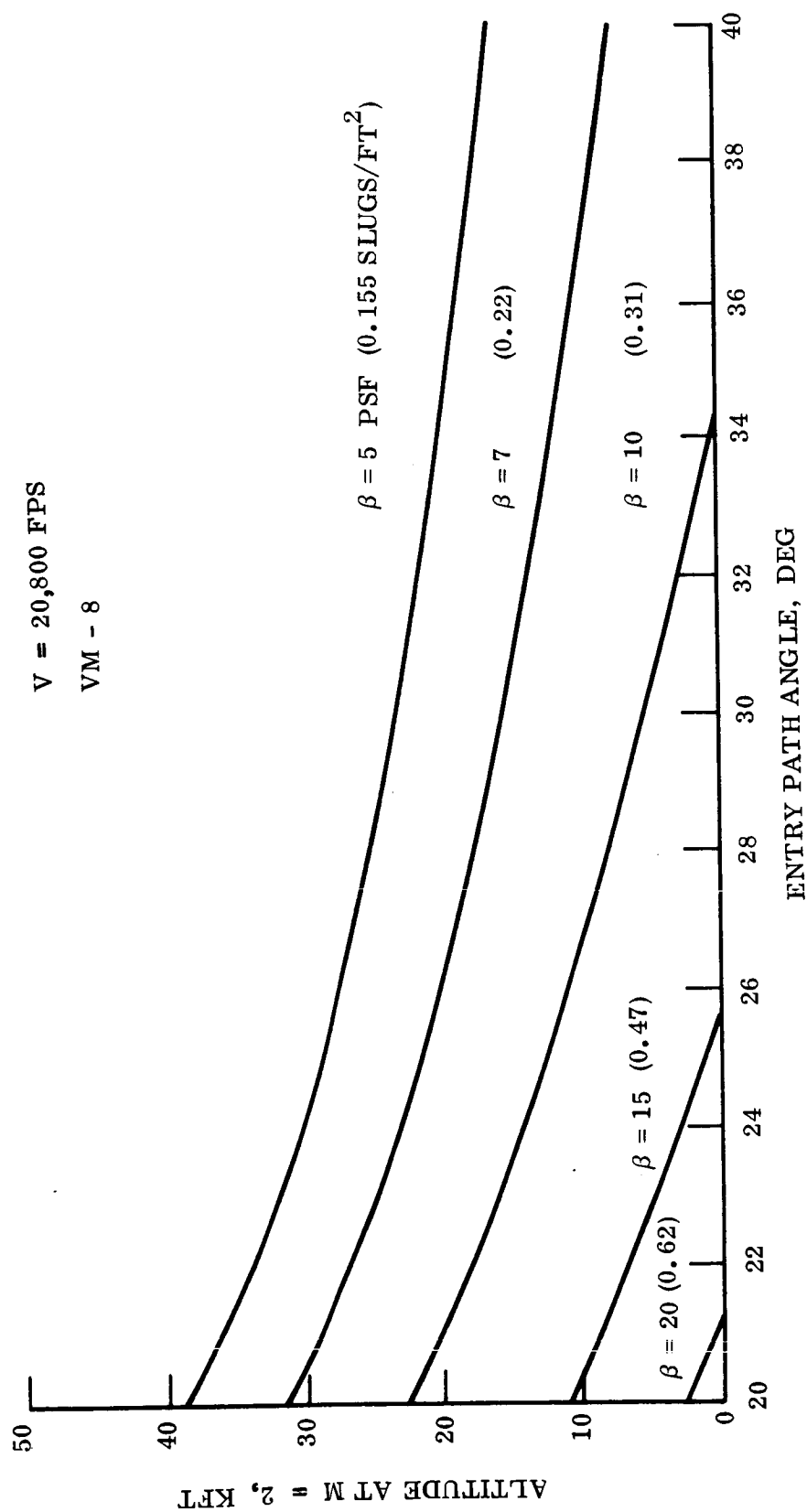


Figure 3.1.2-9. Variation of Altitude at Mach 2 With Entry Angle, Direct Entry

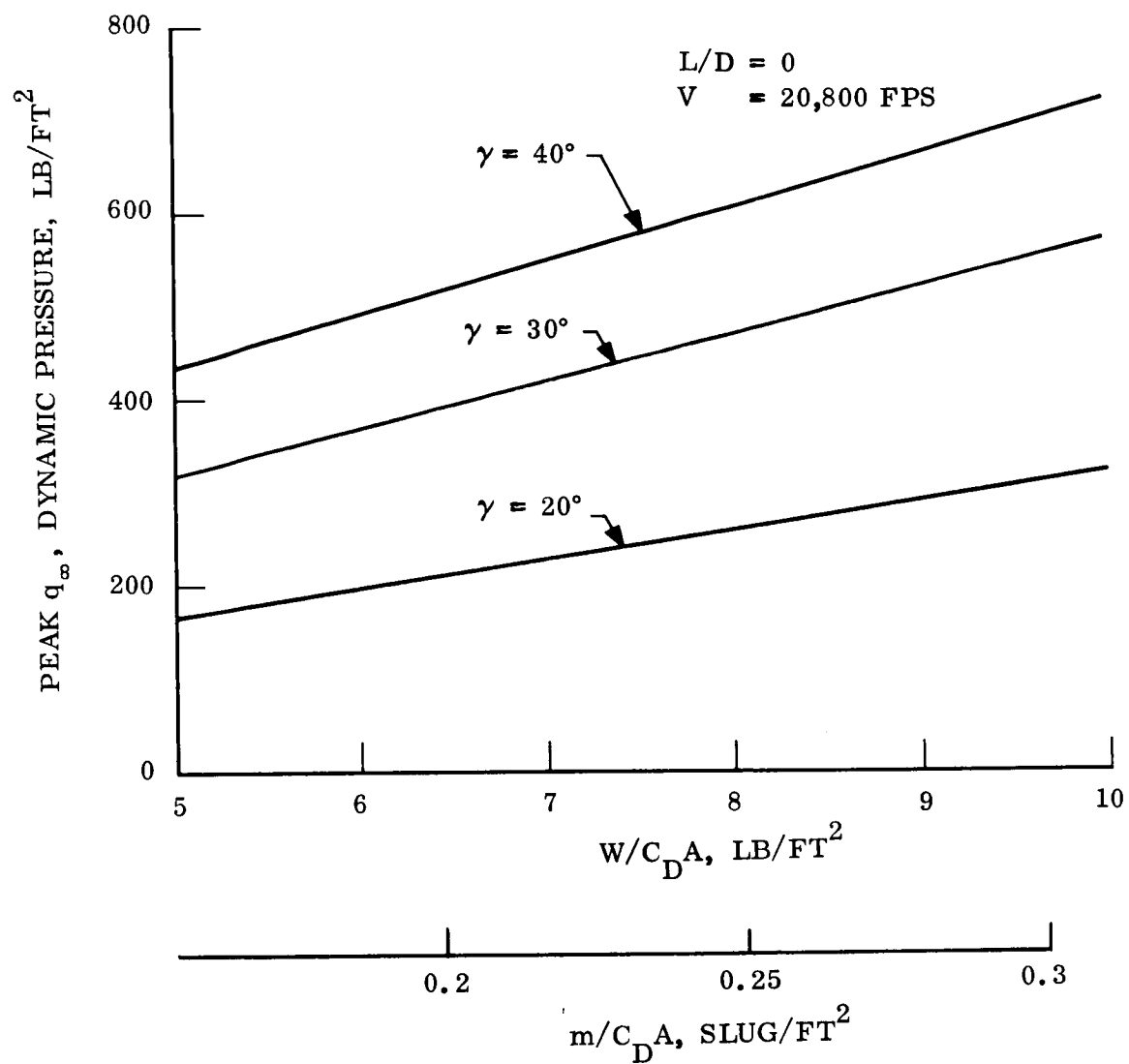


Figure 3.1.2-10. Peak Dynamic Pressure Map, Direct Entry into VM-8

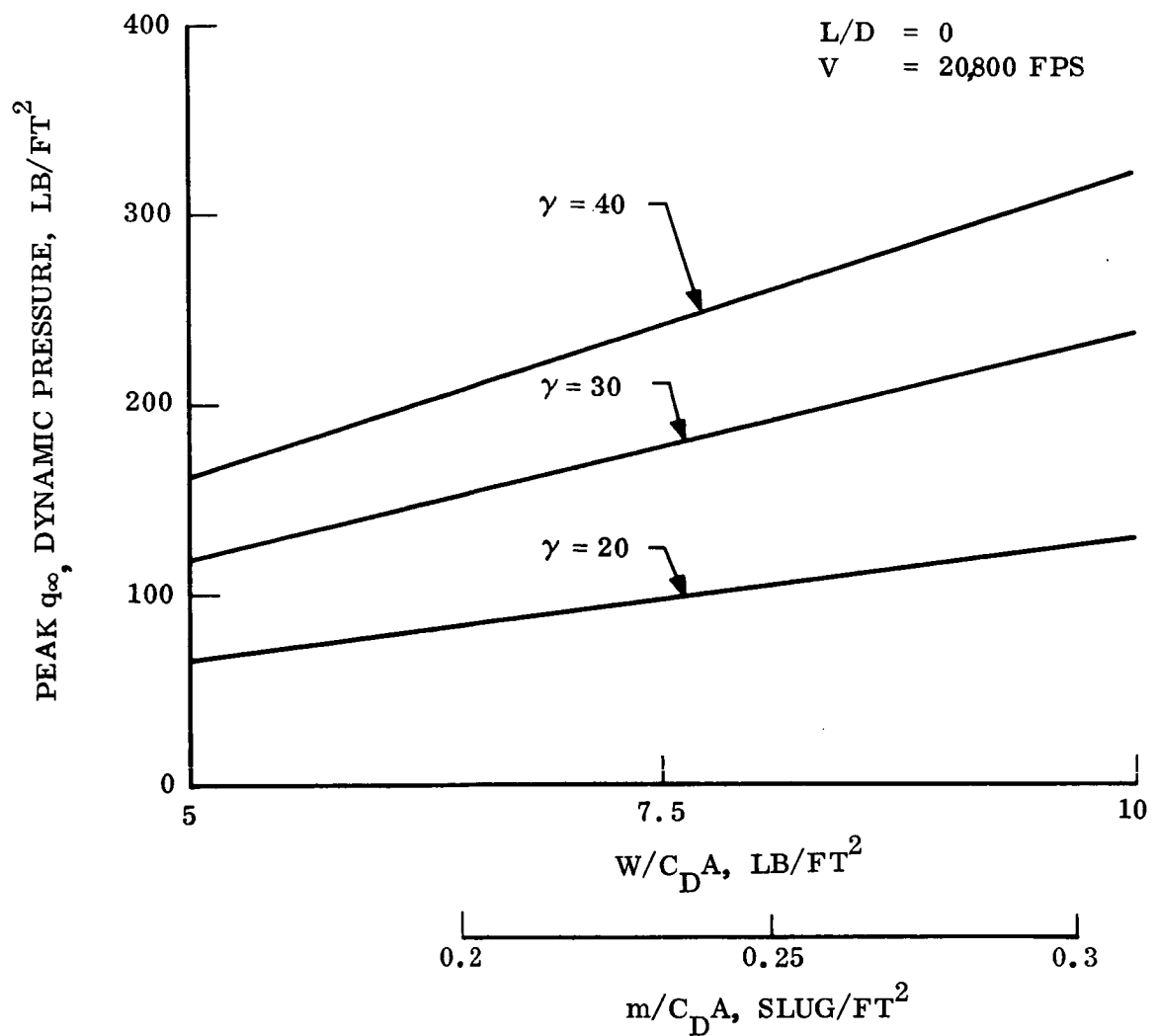


Figure 3.1.2-11. Peak Dynamic Pressure Map, Direct Entry into VM-3

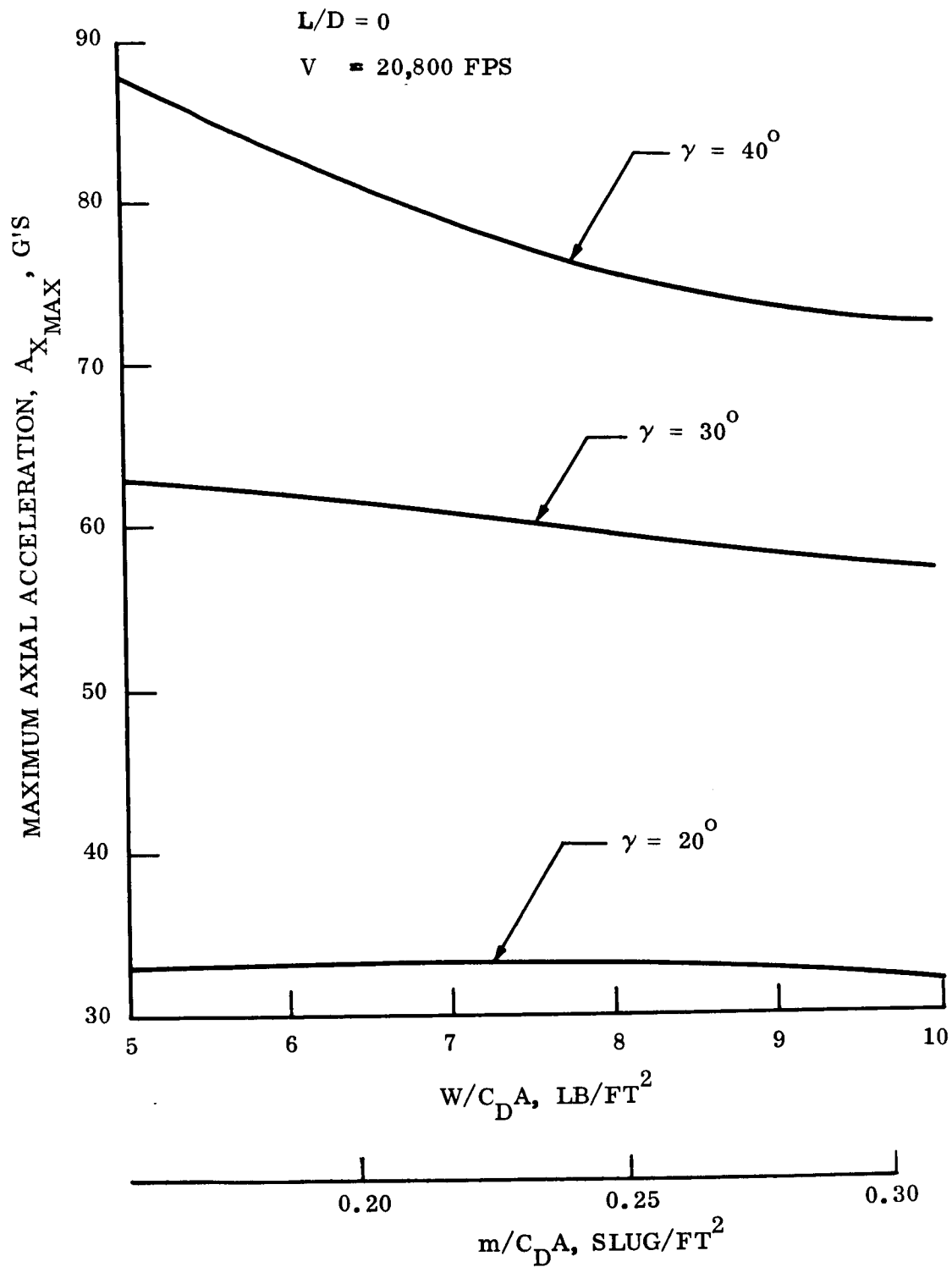


Figure 3.1.2-12. Maximum Axial Acceleration Map, Direct Entry into VM-8

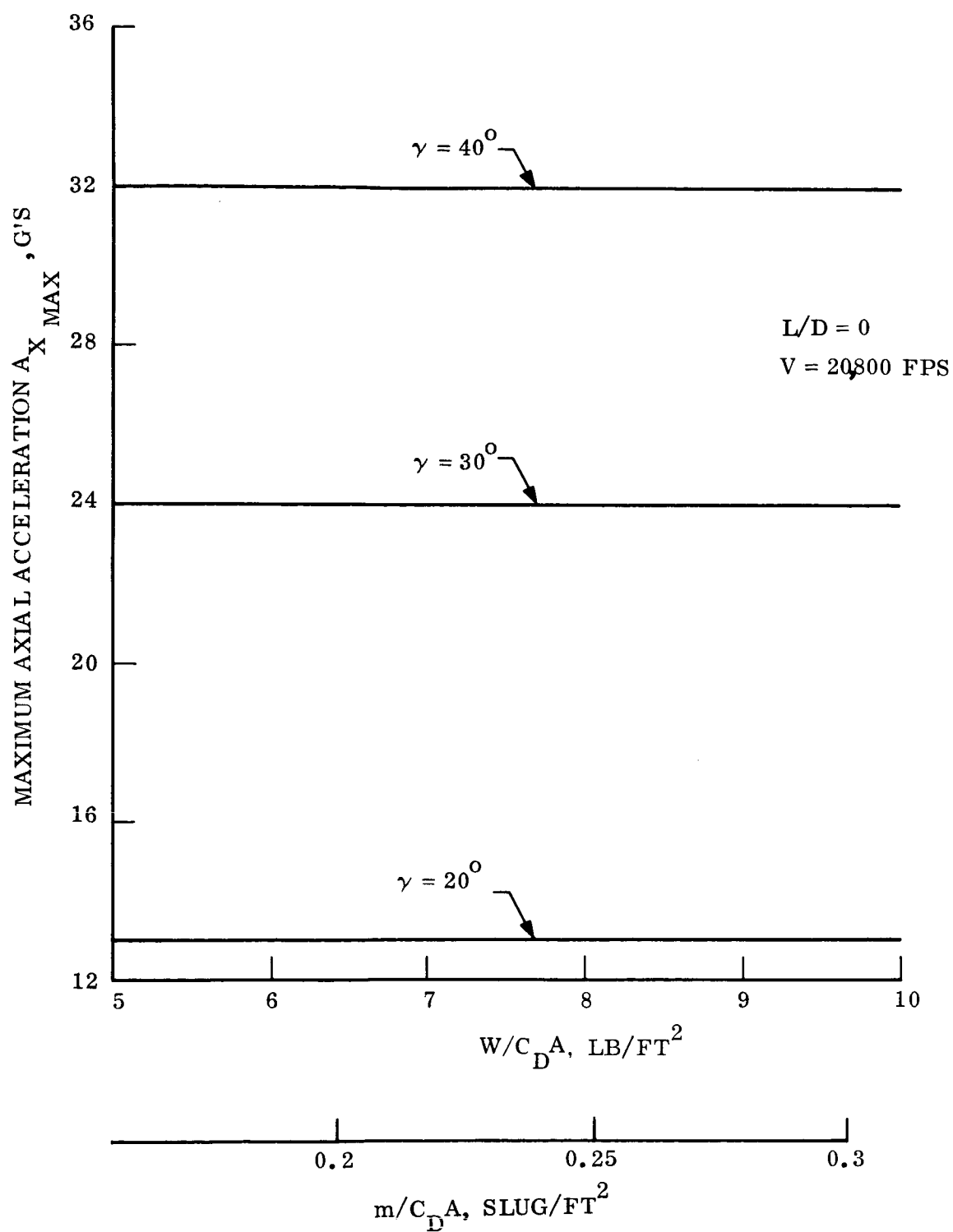


Figure 3.1.2-13. Maximum Axial Acceleration Map, Direct Entry into VM-3

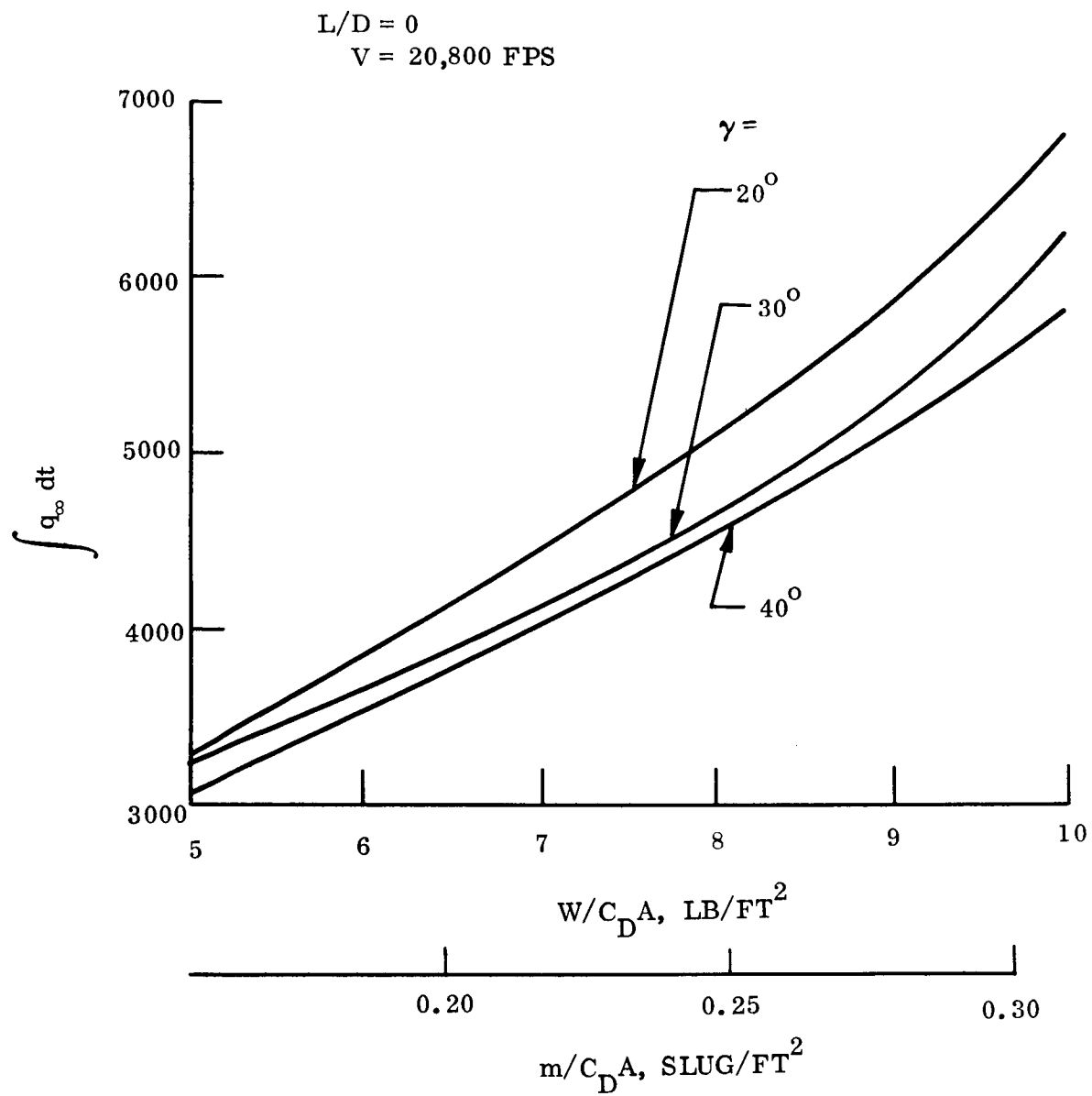


Figure 3.1.2-14. Integrated Dynamic Pressure, Direct Entry into VM-8

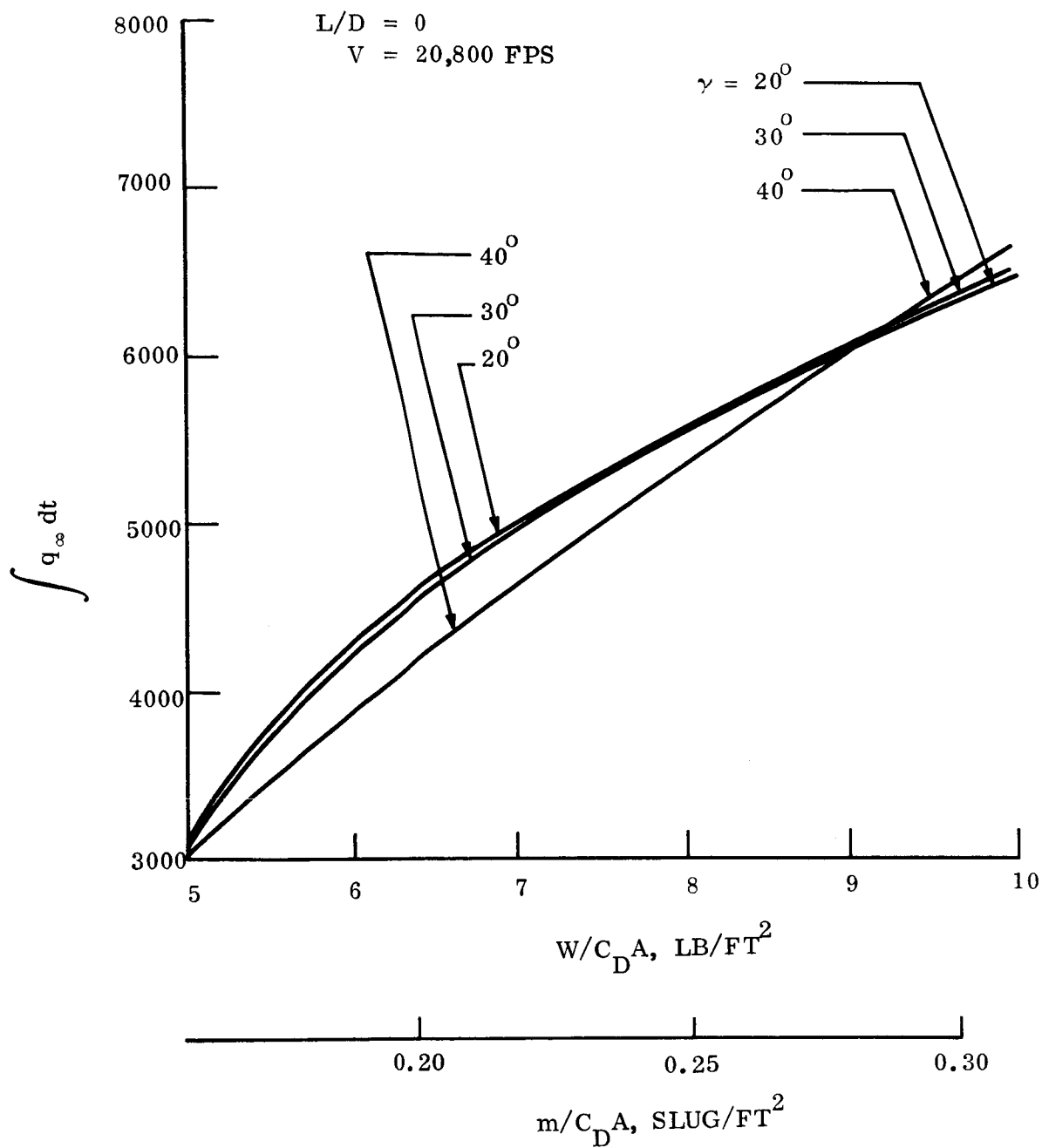


Figure 3.1.2-15. Integrated Dynamic Pressure, Direct Entry into VM-3

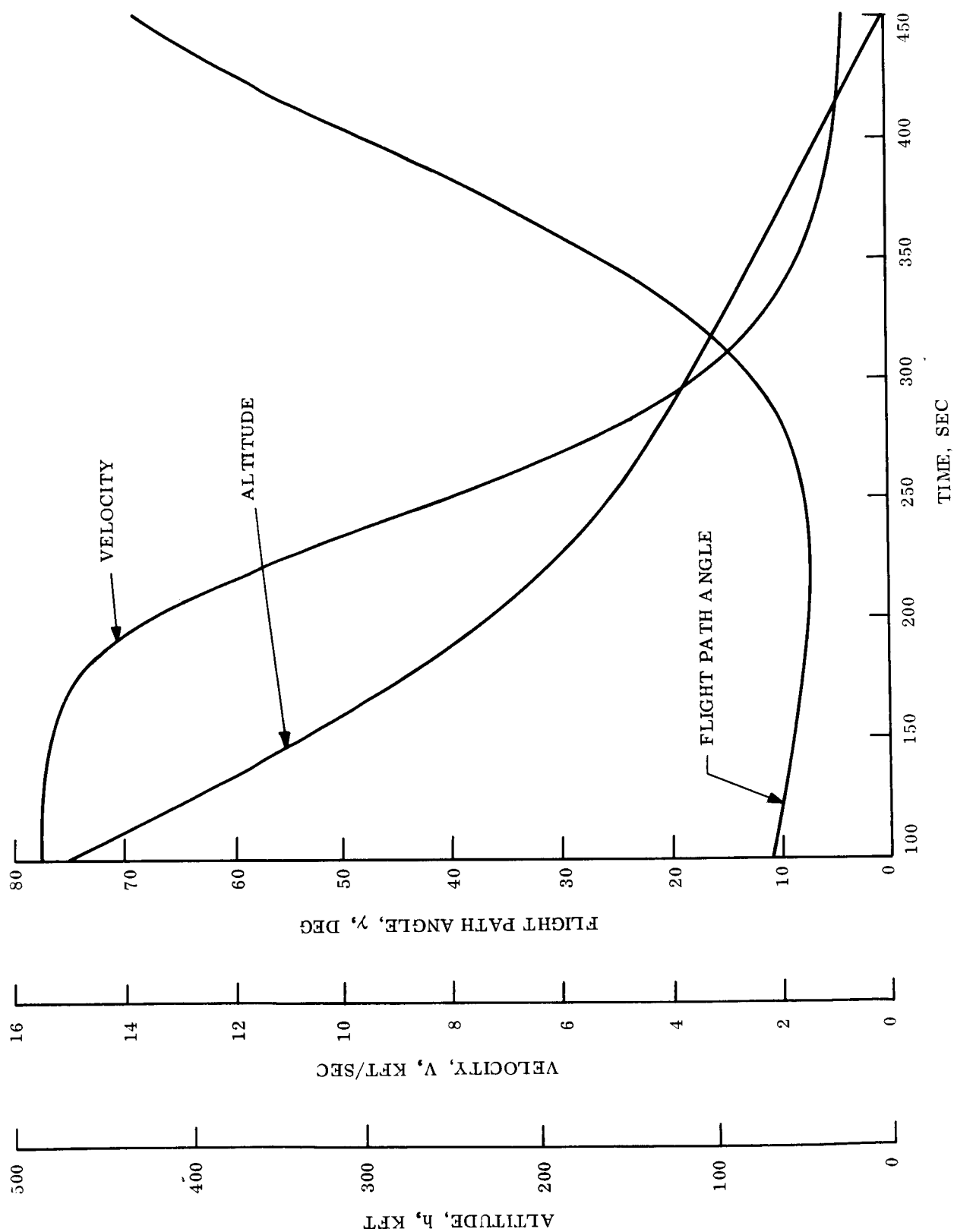


Figure 3.1.2-16. Entry Time History for h, V and γ at $V_E = 15.3$ kfps, $\gamma_E = 14^\circ$ and $\beta = 0.435$ Slug/ft²
Out-of-Orbit Entry, VM-3

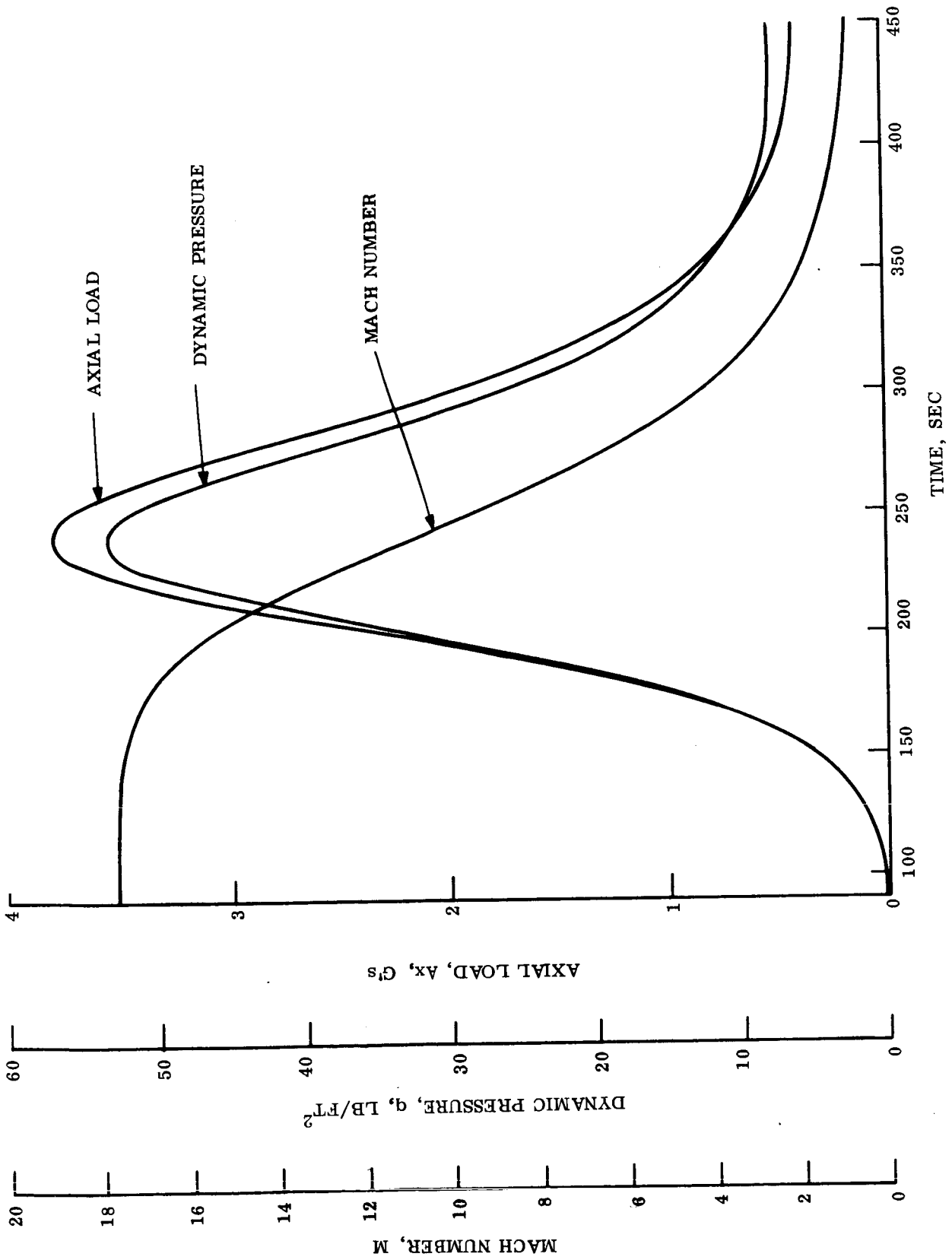


Figure 3.1.2-17. Entry Time History for M, q, and Ax at $V_E = 15.3$ kfps, $\gamma_E = 14^\circ$, and $\beta = 0.435$ Slug/ft²
Out-of-Orbit Entry, VM-3

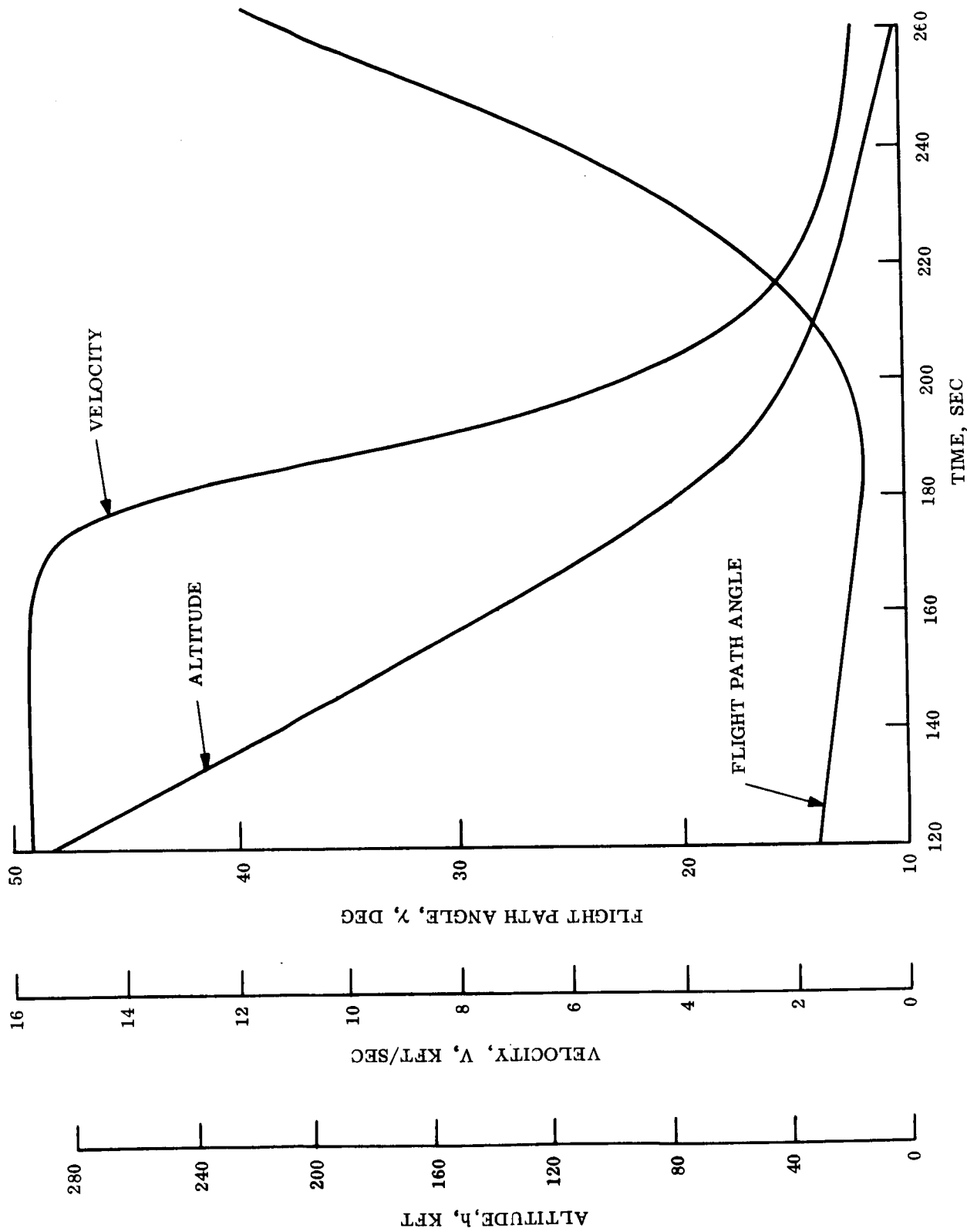


Figure 3.1.2-18. Entry Time History for h , V , and γ , at $V_E = 15.3$ kfps, $\gamma_E = 18^\circ$, and $\beta = 0.435$ Slug/ft²
Out-of-Orbit Entry, VM-8

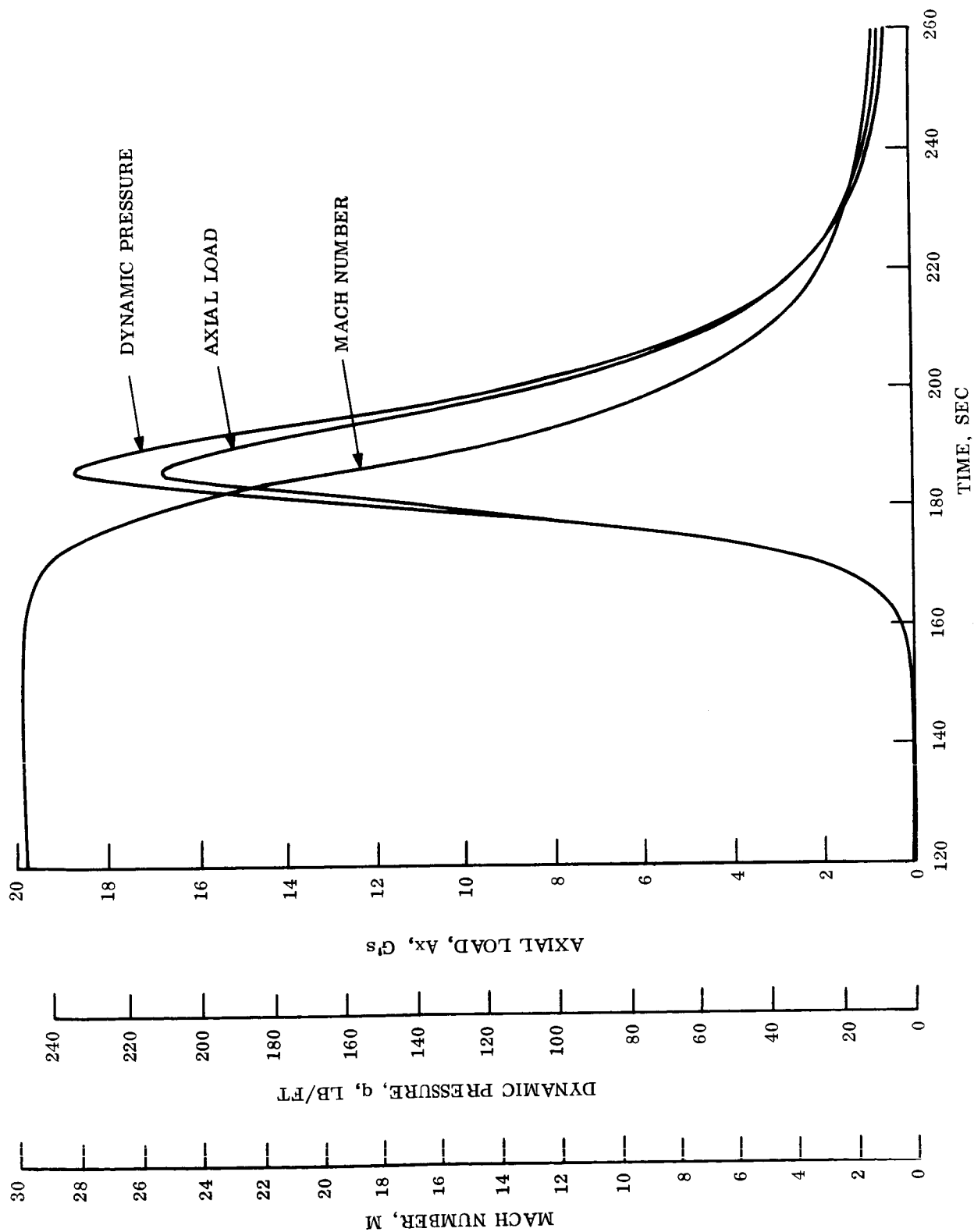


Figure 3.1.2-19. Entry Time History for M , q , and A_x at $V_E = 15.3$ kfps, $\gamma_E = 18^\circ$, and $\beta = 0.435$ Slugs/ft²
Out-of-Orbit Entry, VM-8

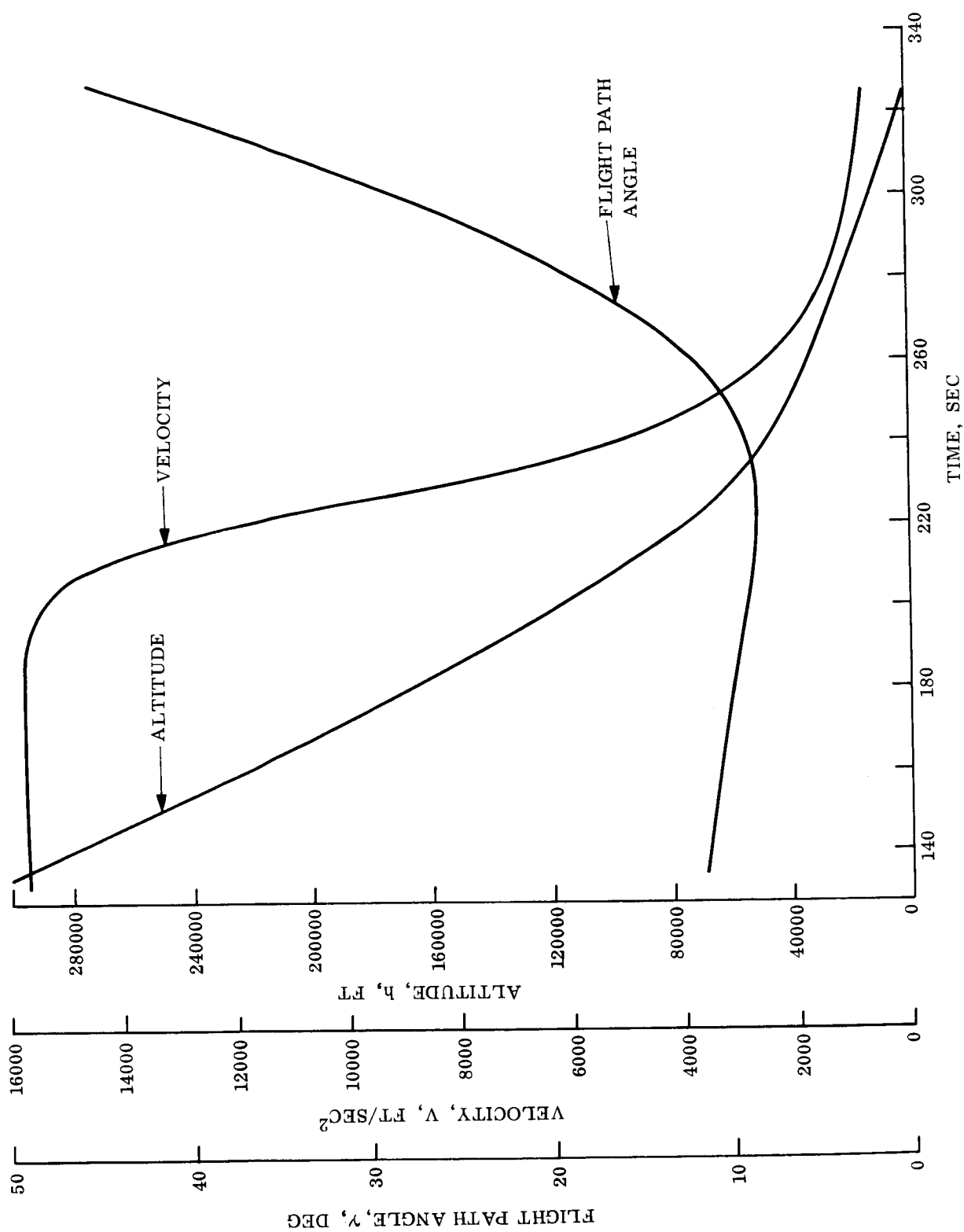


Figure 3.1.2-20. Entry Time History for γ , V , and h at $V_E = 15.3$ kfps, $\gamma_E = 16^\circ$, and $\beta = 0.435$ Slug/ft²
Out-of Orbit Entry, VM-8

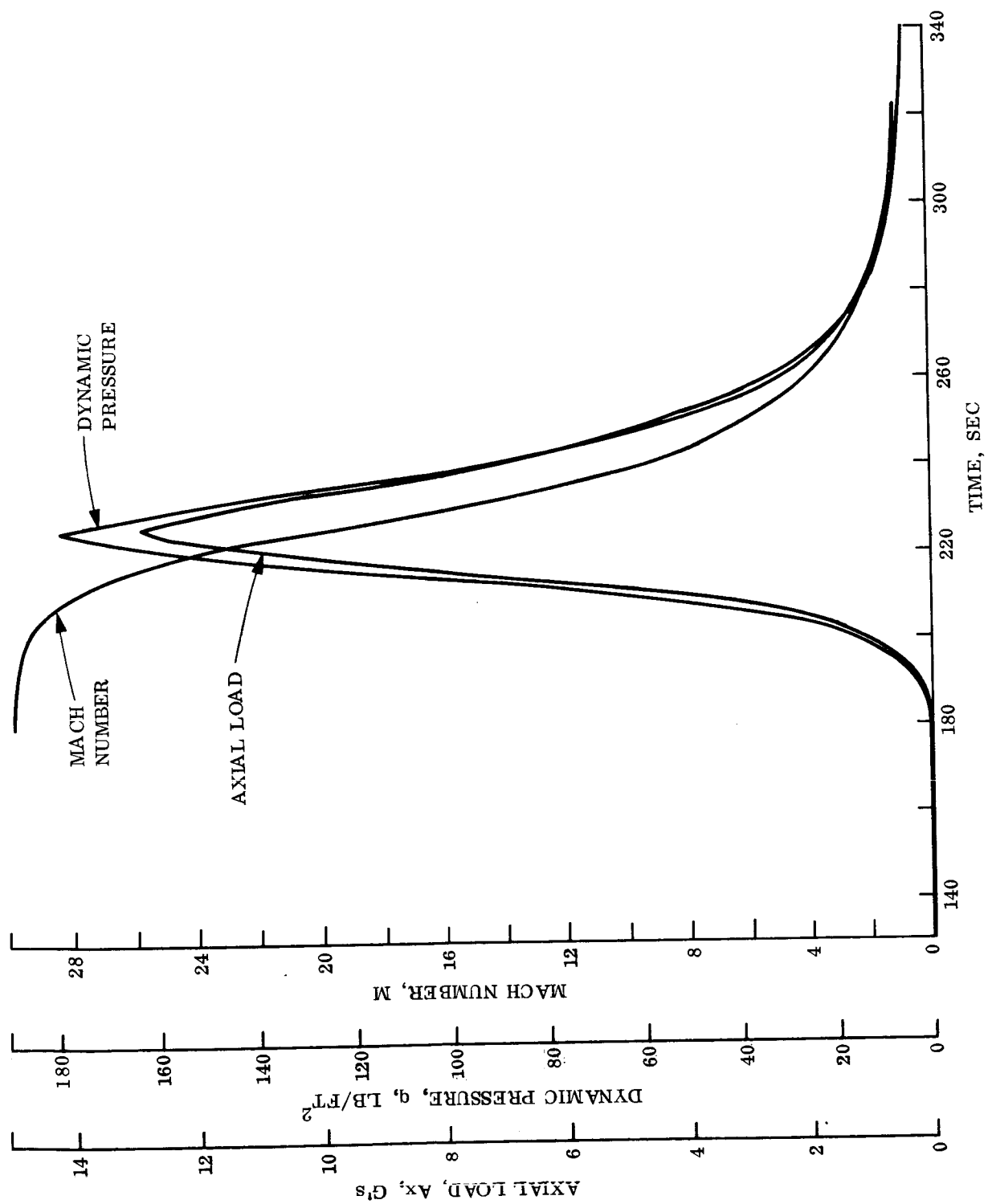


Figure 3.1.2-21. Entry Time History for A_x , q , and M at $V_E = 15.3$ kfps, $\gamma_E = 16^\circ$, and $\beta = 0.435$ Slug/ft²
Out-of-Orbit Entry, VM-8

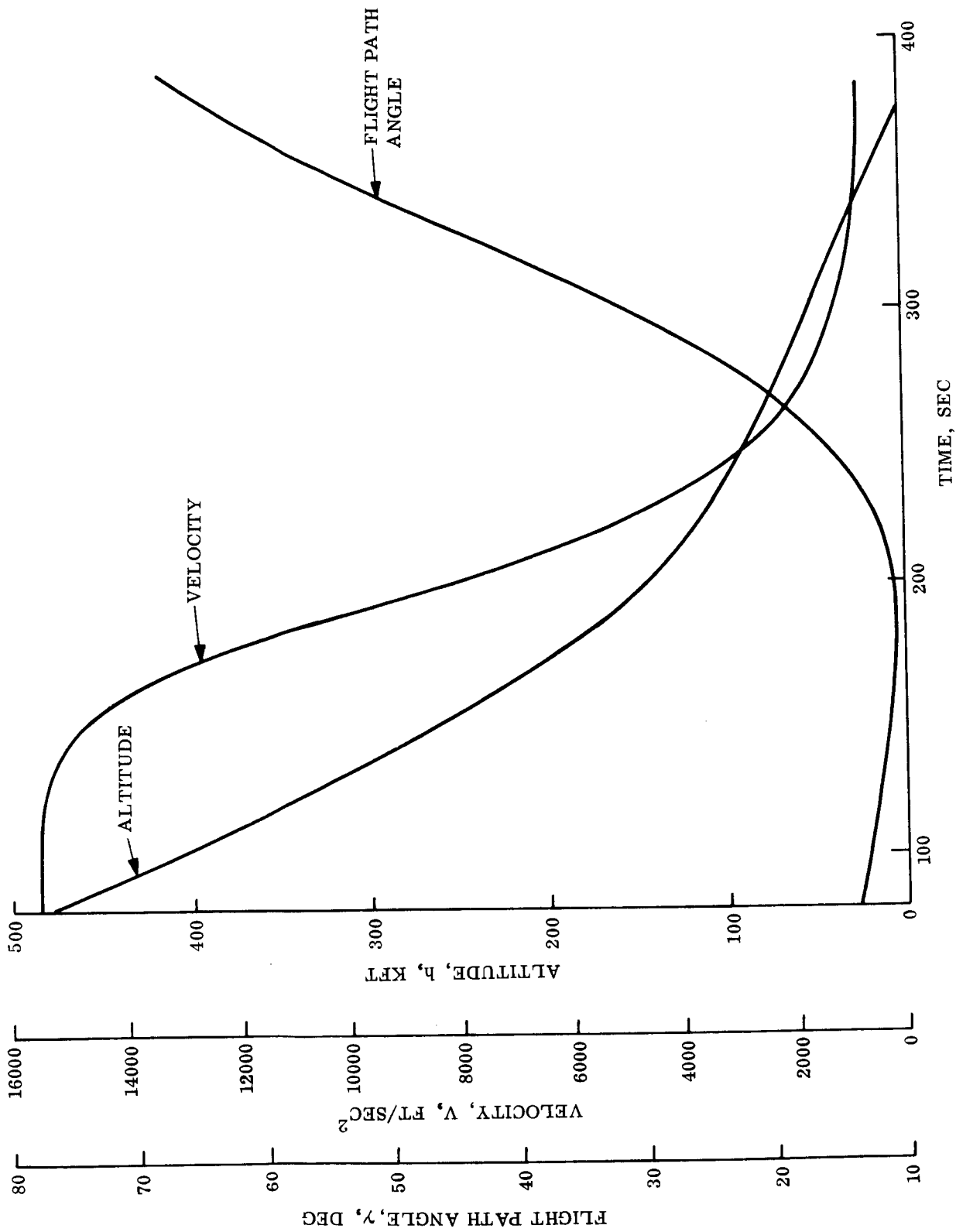


Figure 3.1.2-22. Entry Time History for γ , V , and h at $V_E = 15.3$ kfps, $\gamma_E = 16^\circ$ and $\beta = 0.435$ Slug/ft²
Out-of-Orbit Entry, VM-3

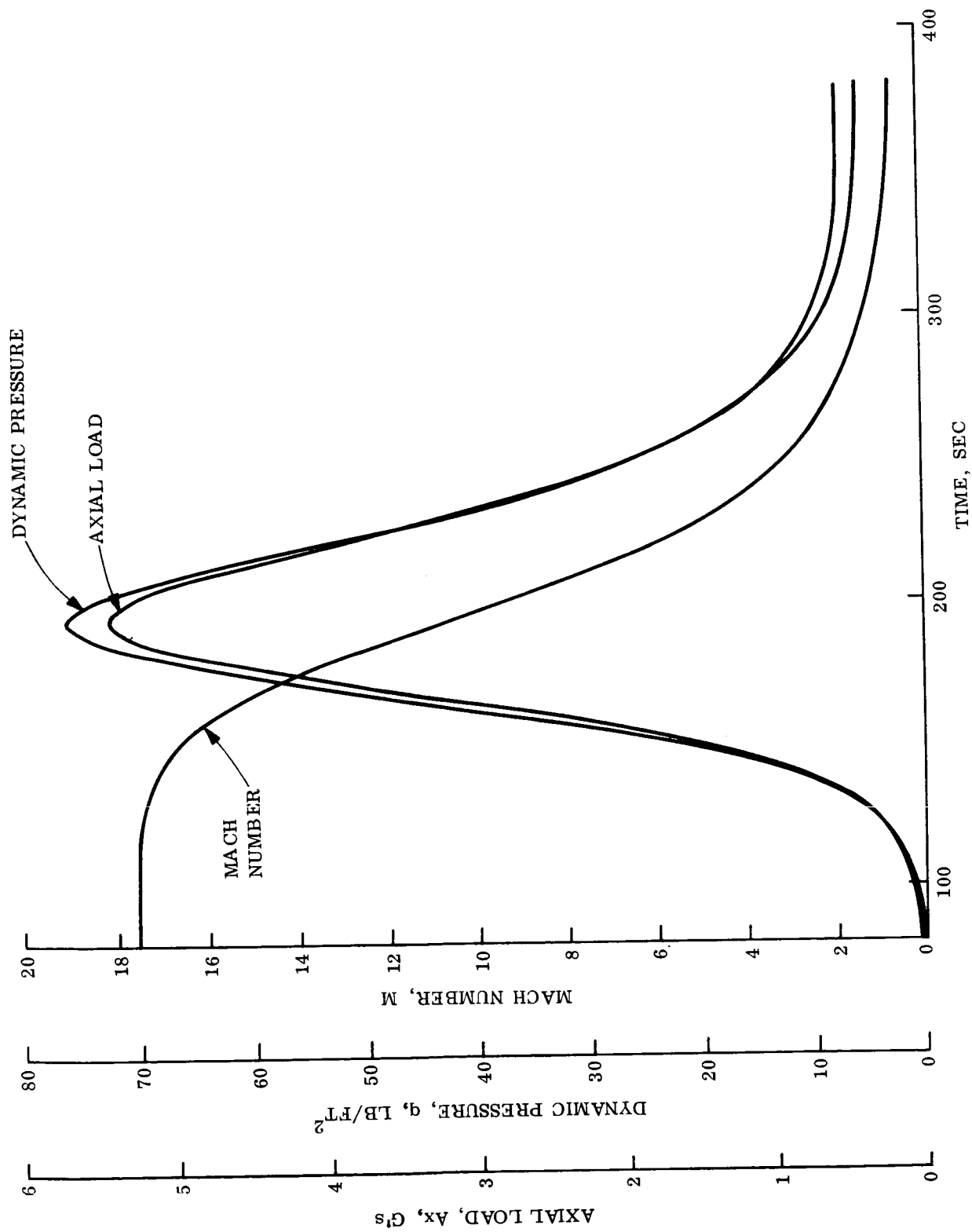


Figure 3.1.2-23. Entry Time History for A_x , q and M at $V_E = 15.3$ kfps, $\gamma_E = 16^\circ$, and $\beta = 0.435$ Slug/ft²
Out-of-Orbit Entry, VM-3

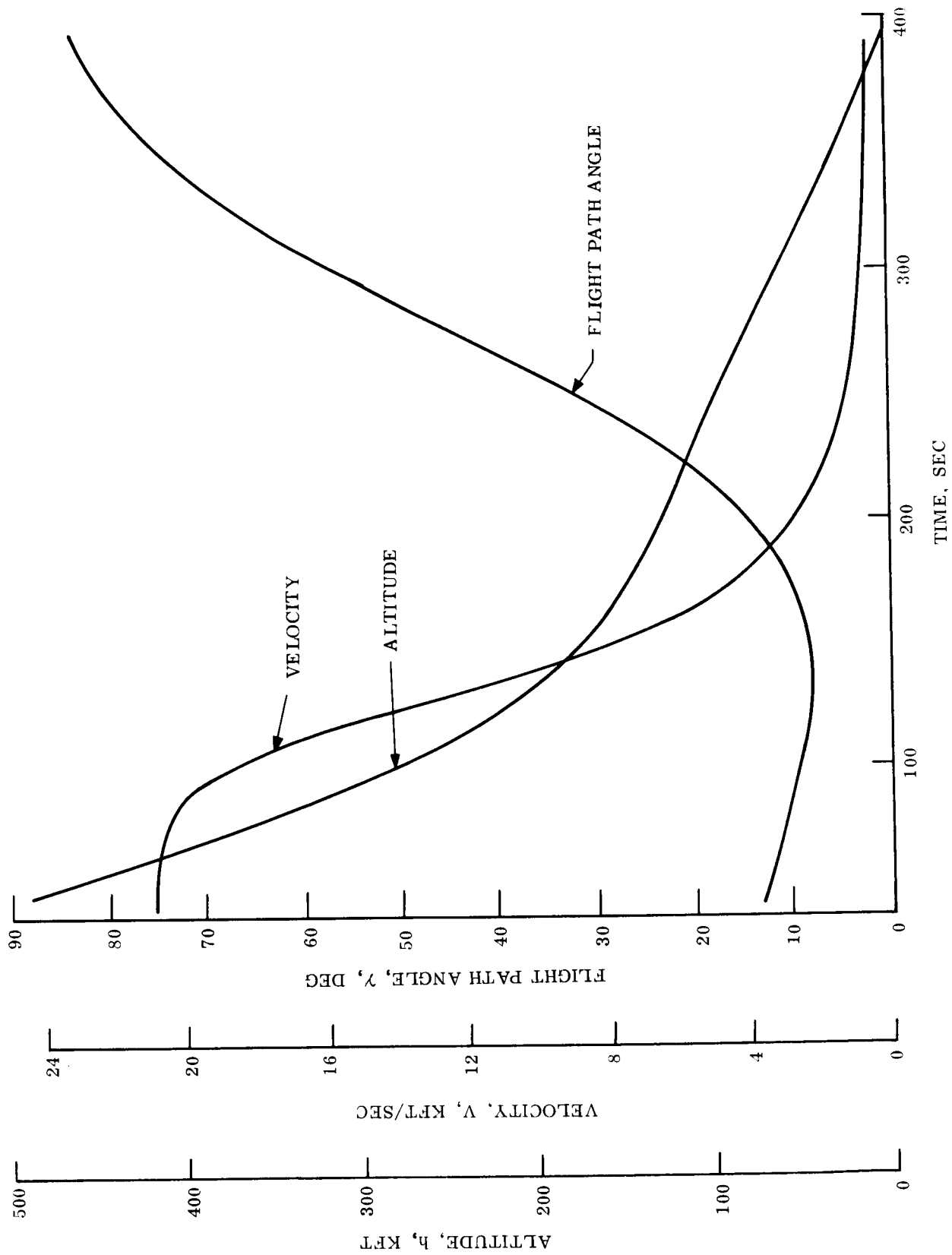


Figure 3.1.2-24. Entry Time History for h , V , and γ , at $V_E = 20.8$ kfps, $\gamma_E = 16^\circ$, and $\beta = 0.217$ Slug/ft²
Direct Entry, VM-3

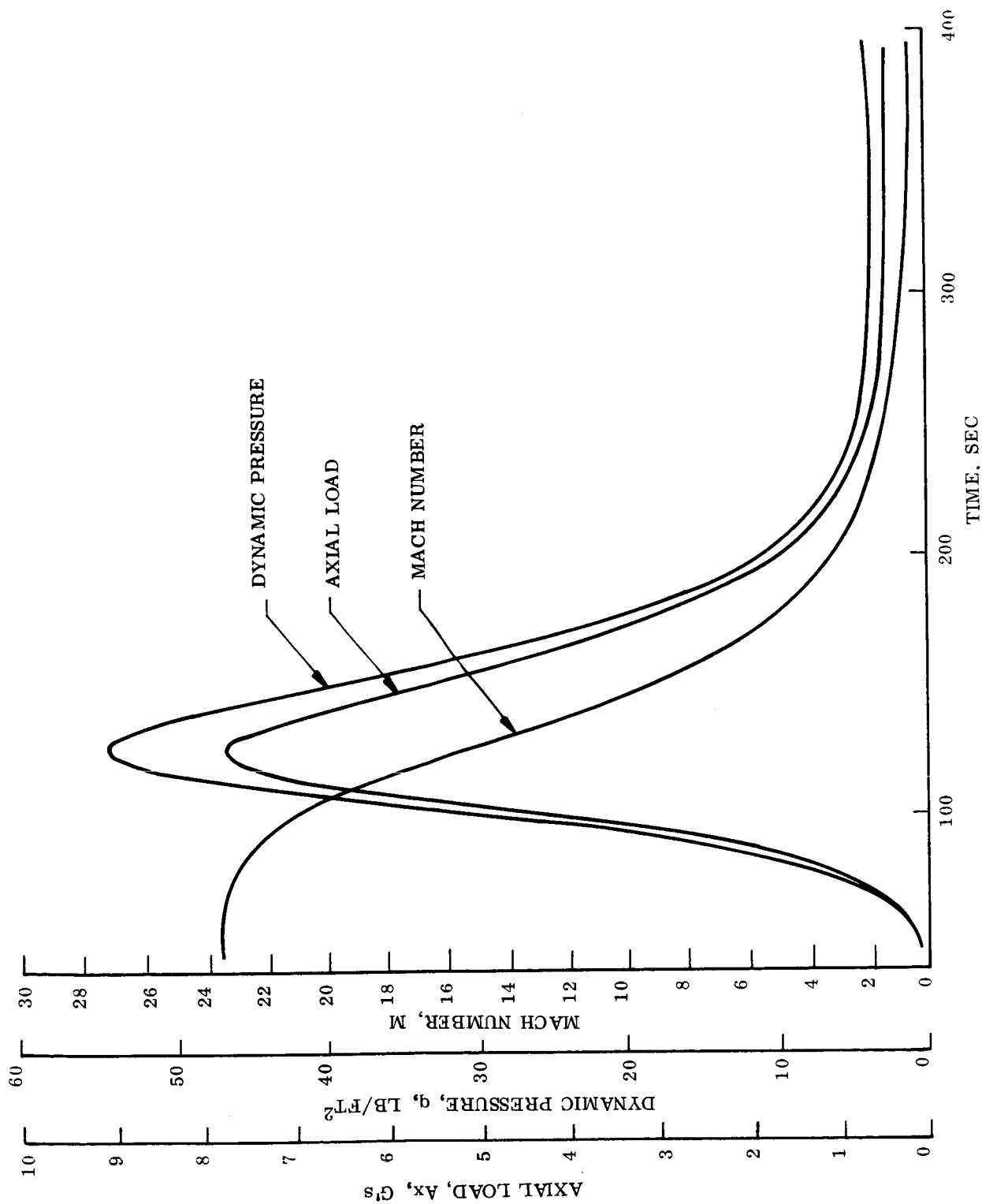


Figure 3.1.2-25. Entry Time History for A_x , q , and M at $V_E = 20.8$ kfps, $\gamma_E = 16^\circ$, and $\beta = 0.217$ Slug/ft²
Direct Entry, VM-3

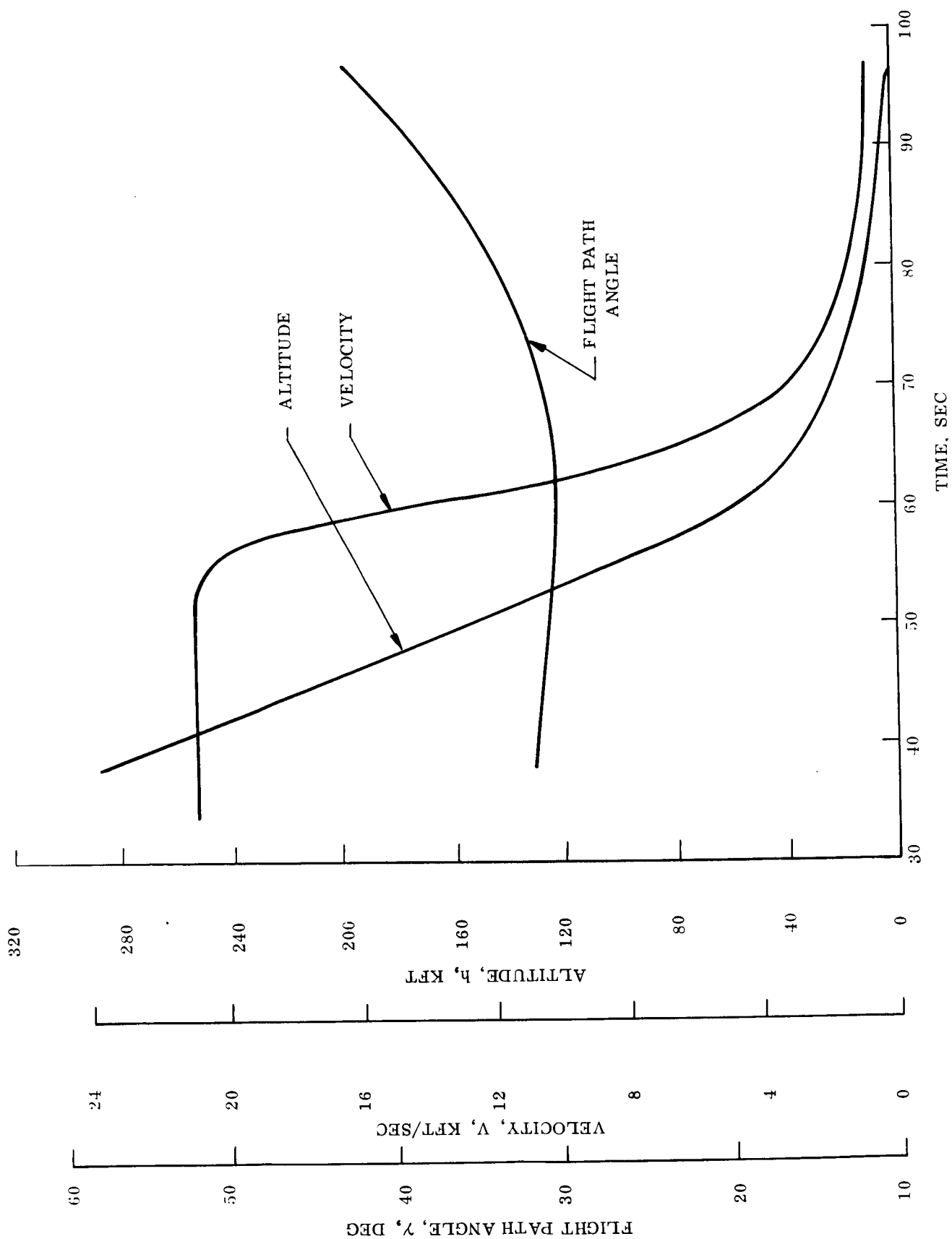


Figure 3.1.2-26. Entry Time History for γ , V , and h at $V_E = 20.8$ kfps, $\gamma_E = 34^\circ$, and $\beta = 0.217$ Slug/ft²
Direct Entry, VM-8

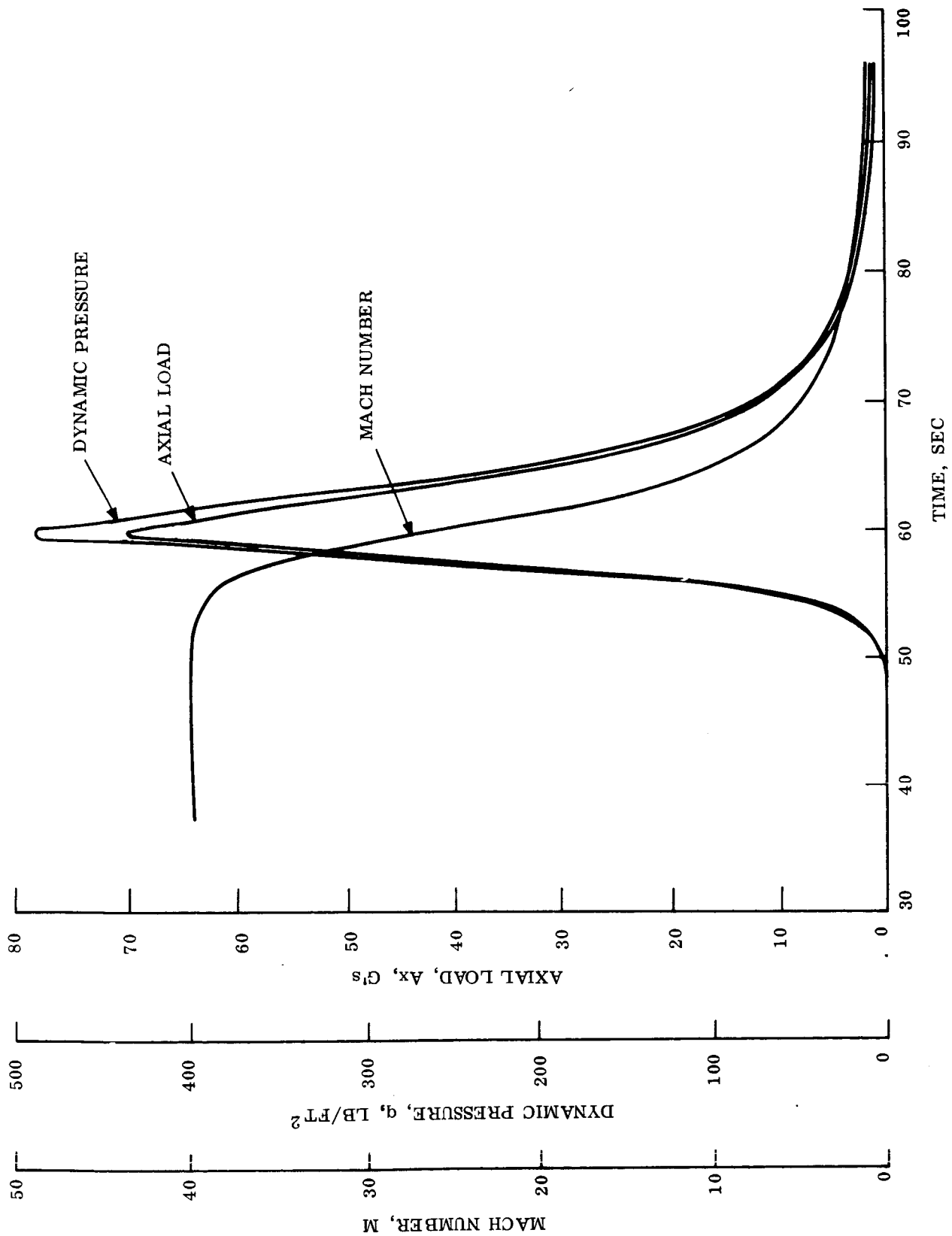


Figure 3.1.2-27. Entry Time History for M, q, and Ax at $V_E = 20.8$ kfps, $\gamma_E = 34^\circ$, and $\beta = 0.217$ Slug/ft²
Direct Entry, VM-8

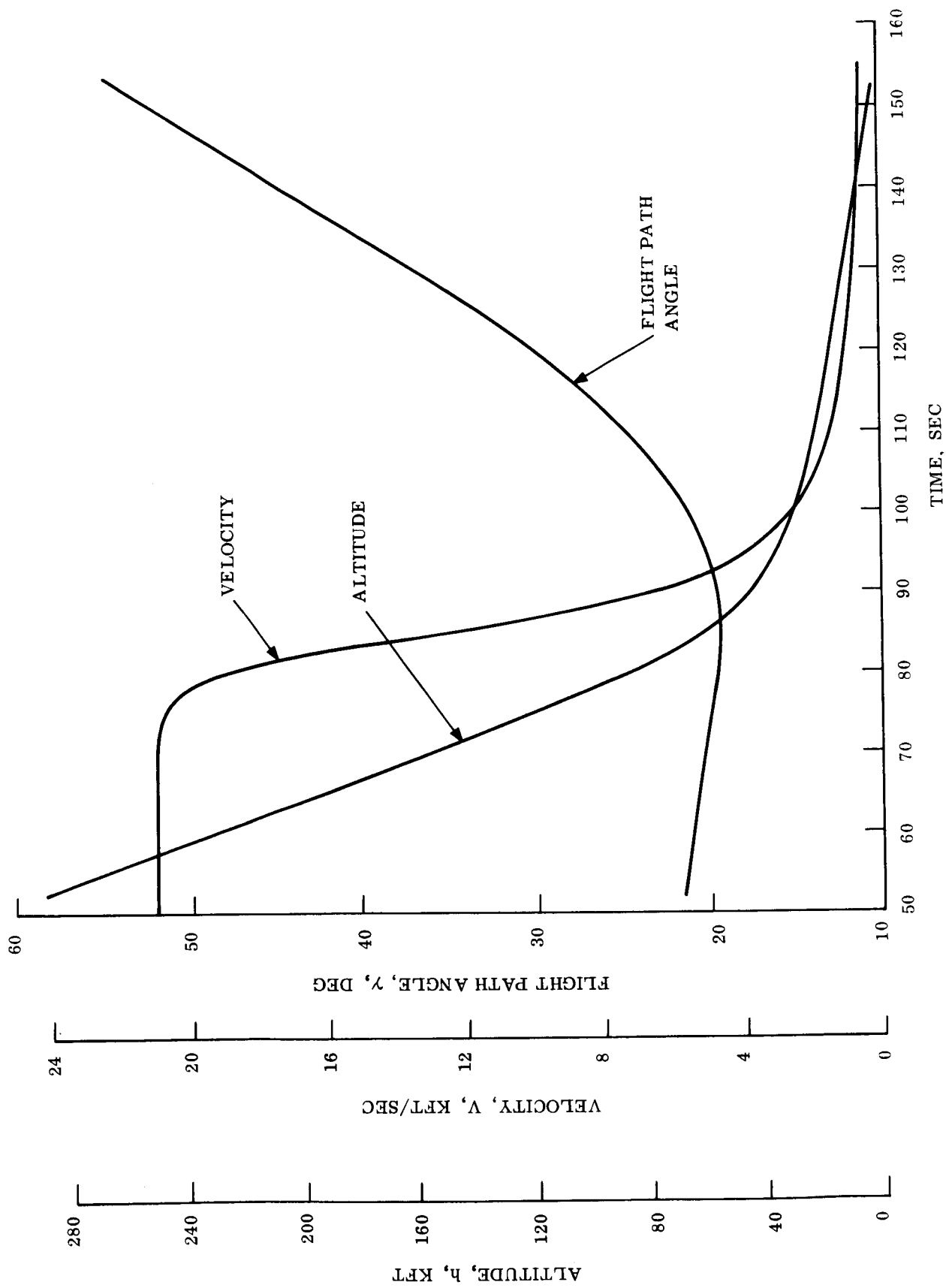


Figure 3.1.2-28. Entry Time History for h , V , and γ at $V_E = 20.8$ kfps, $\gamma_E = 25^\circ$, and $\beta = 0.217$ Slug/ft²
Direct Entry, VM-8

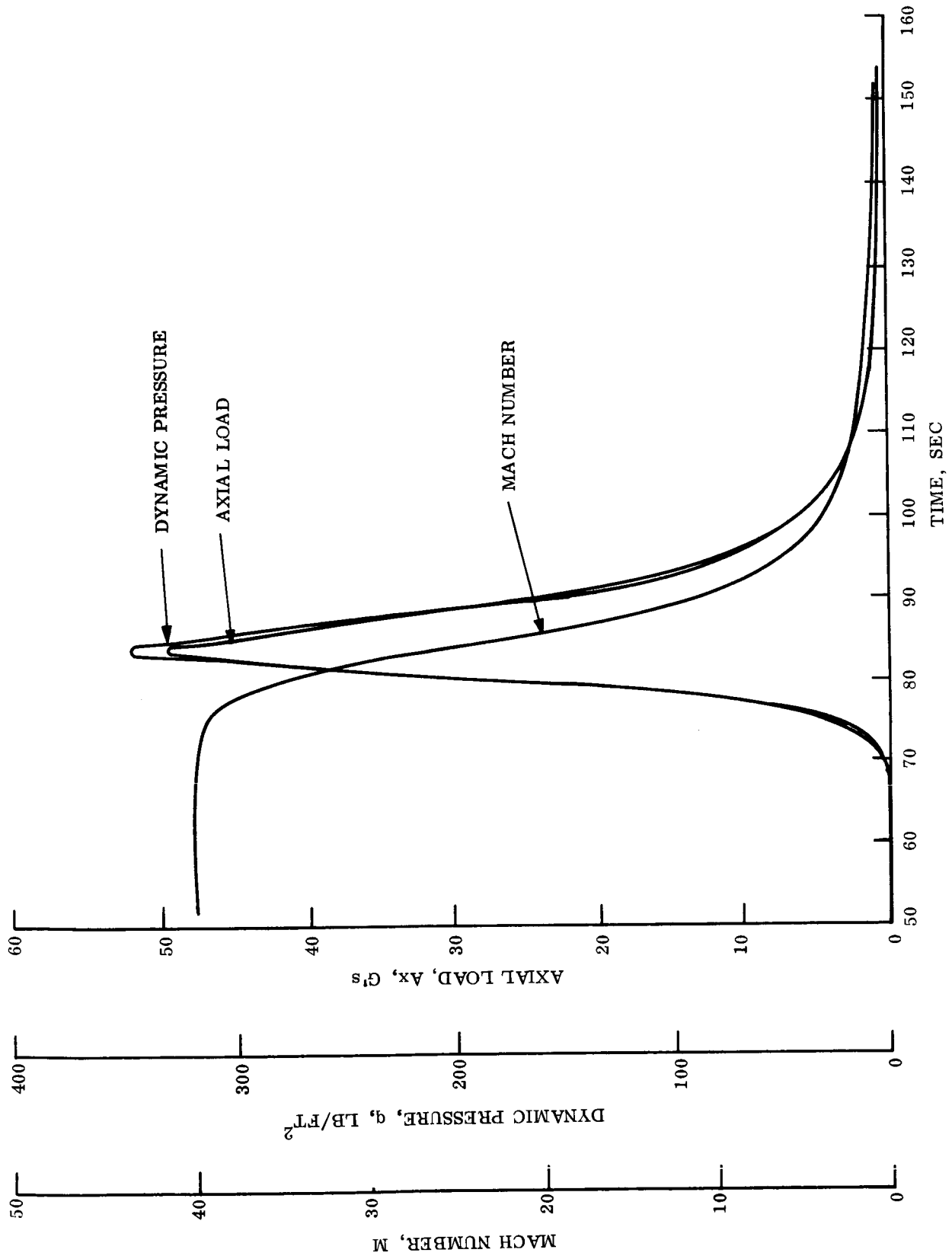


Figure 3.1.2-29. Entry Time History for M, q, and Ax at $V_E = 20.8$ kfps, $\gamma = 25^\circ$, and $\beta = 0.217$ Slug/ft²

Direct Entry, VM-8

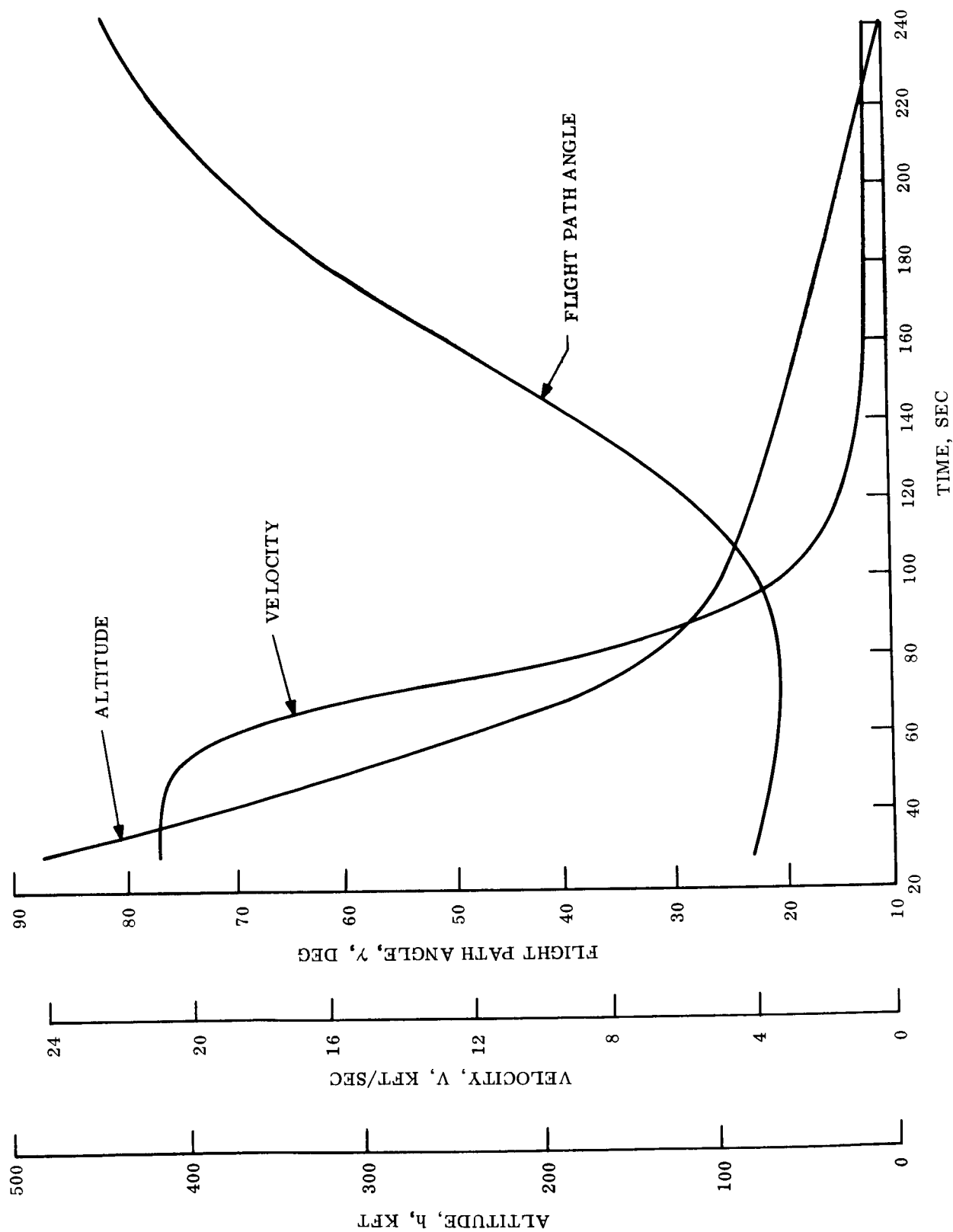


Figure 3.1.2-30. Entry Time History for h , V , and γ at $V_E = 20.8$ kfps, $\gamma_E = 25^\circ$, and $\beta = 0.217$ Slug/ft²
Direct Entry, VM-3

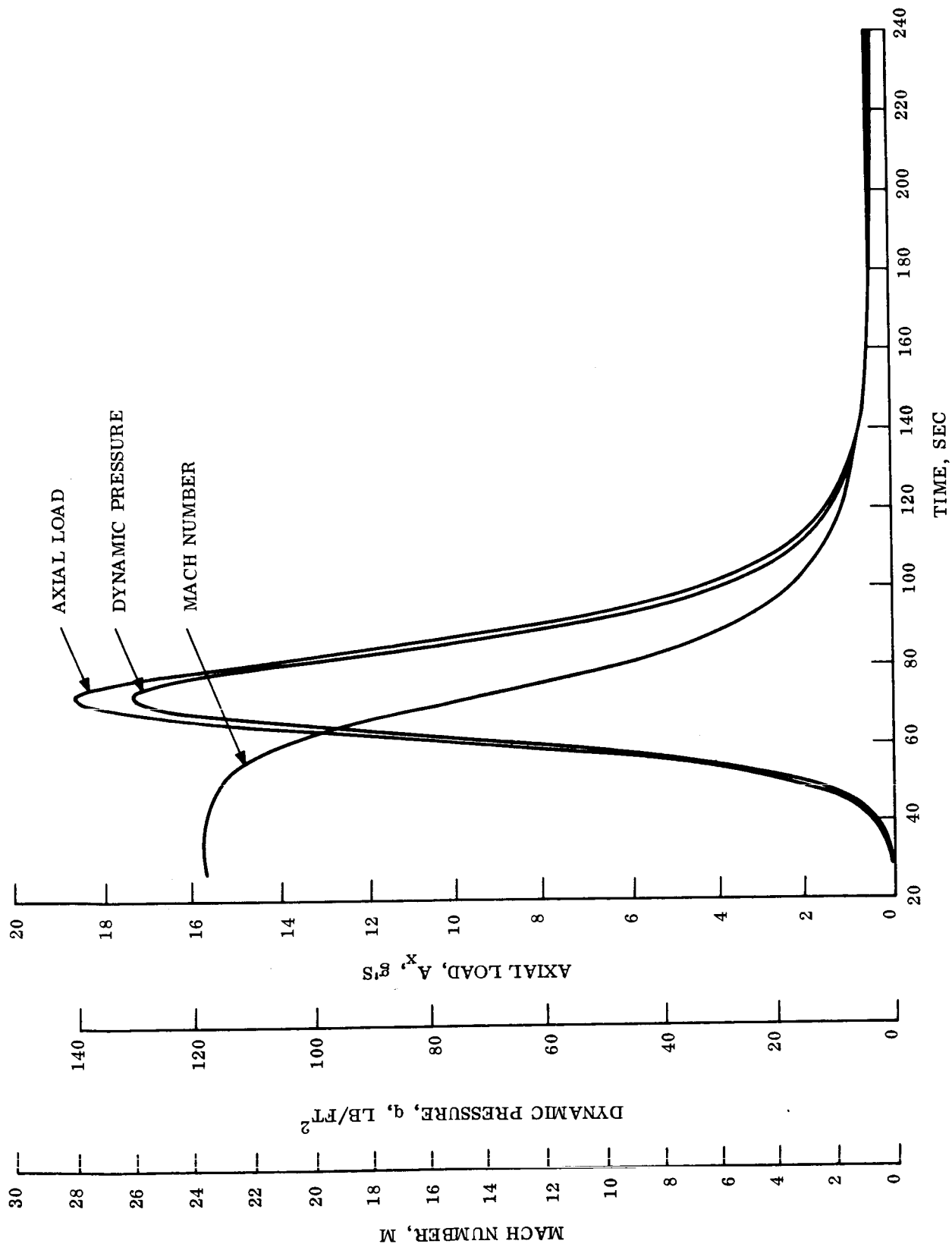


Figure 3.1. 2-31. Entry Time History for M, q and Ax at $V_E = 20.8$ kfps, $\gamma = 25^\circ$, and $\beta = 0.217$ Slug/ft², Direct Entry, VM-3

3.1.2.1.2 Aerodynamic Performance

The aerodynamic characteristics of large cone angle configurations are presented in this section. These data cover a range of cone angles between 40° and 90° with particular emphasis in the 50° to 70° region; a Mach number range between 0.2 and 20; and angles-of-attack up to 100° . These data were prepared from a compilation of ground test experimental data found in ref. 3-3 through 3-14

The presented data include aerodynamic axial and normal force coefficients, centers-of-pressure, and aerodynamic stability coefficients (dynamic damping). Effects of nose blunting and both afterbody and shoulder radii on the aerodynamic performance are also presented.

The investigated configurations are presented in fig. 3.1.2-32 with details of the afterbody shape and shoulder rounding shown in fig. 3.1.2-33.

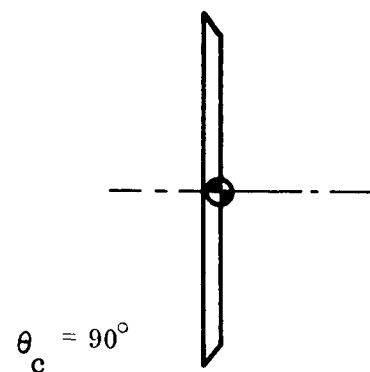
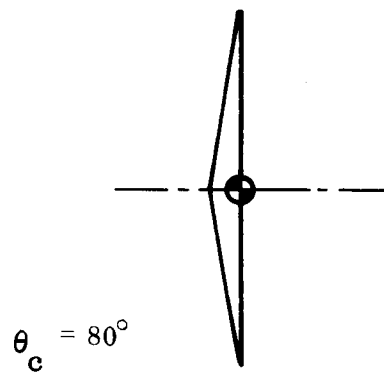
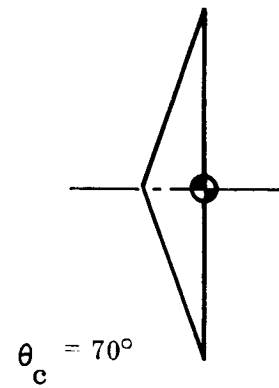
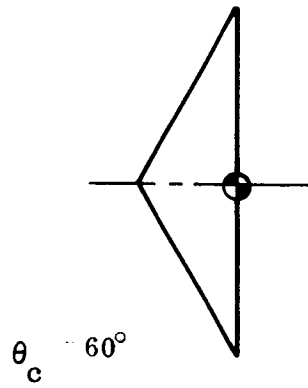
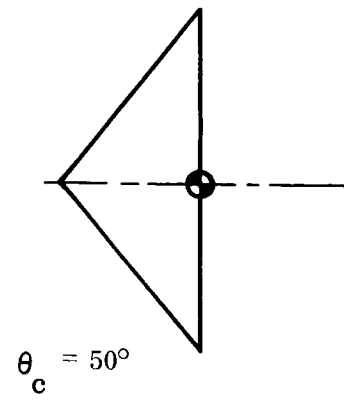
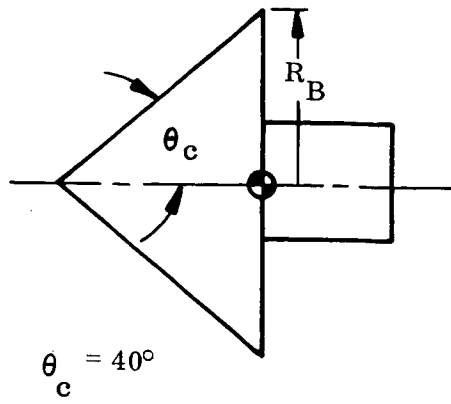
A. Axial Force Coefficient - The axial force coefficients for the various configurations ($40 \leq \theta_c \leq 90^\circ$) are presented in figs. 3.1.2-34 through -44. These data have been corrected to free stream pressure on the base. In order to account for base pressure effects on drag the correlation technique of Martellucci, ref. 3.1-16 was used. This correlation was prepared from sharp conical body test data where the cone semi-vertex angles were in the range of 5 to 15° , and the bluntness ratio did not exceed 0.30. Use of the correlation requires knowledge of the edge Mach number and edge Reynolds number based on wetted length for any specific shape, such as the configurations of figs. 3.1.2-32 and -33. For blunted cones, the edge correlation requires use of a detailed boundary layer program. Fig. 3.1.2-45 presents the base pressure ratio as a function of local Mach number and Reynolds number just prior to the base. The base drag coefficient then becomes

$$C_{D_B} = - \left[\frac{\frac{P_B}{P_\infty} - 1}{\frac{\gamma}{2} M^2} \right]$$

Figs. 3.1.2-34 through -39 present C_A as a function of angle-of-attack and Mach number for the various cone angles. Note that as the cone angle increases such that the sonic point moves to the shoulder, the angle-of-attack effects become less pronounced.

Fig. 3.1.2-40 presents data which show that the effects of bluntness (nose to base radius ratio) on axial force appear to be insignificant for the 60° cone.

Figs. 3.1.2-41 through -44 present cross plots of C_A as a function of Mach number for several angle-of-attack values. The effects of an afterbody dome are shown in fig. 3.1.2-44 for small angles-of-attack. There is little effect above a Mach number of 5.0.



ALL MOMENTS ARE ABOUT THE BASE

Figure 3.1.2-32. Configuration Sketches

$$\theta_c = 60^\circ$$

$$R_N/R_B = 0.25$$

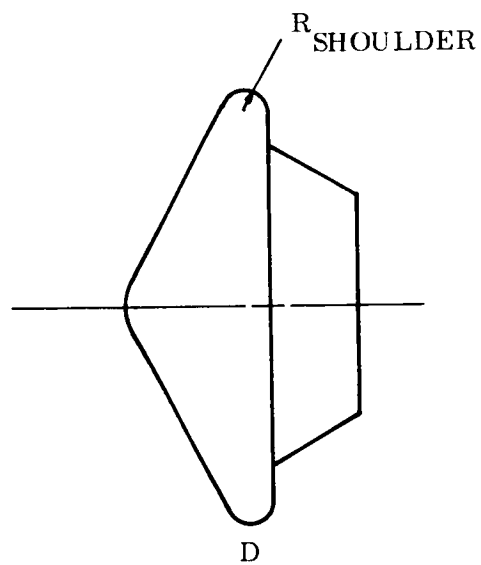
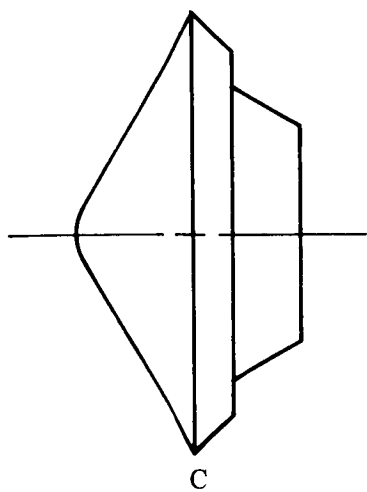
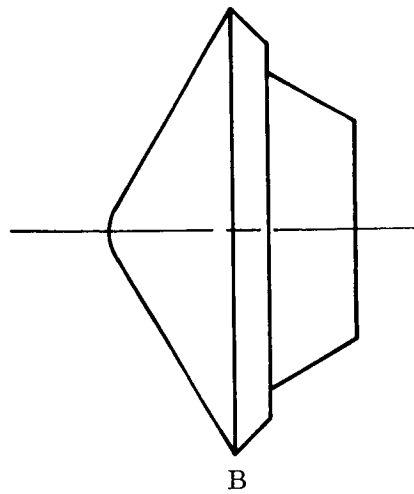
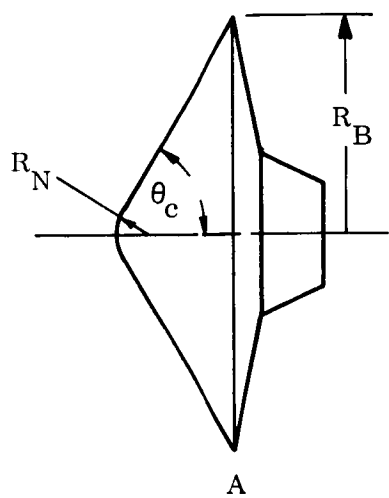


Figure 3.1.2-33. Afterbody Configurations

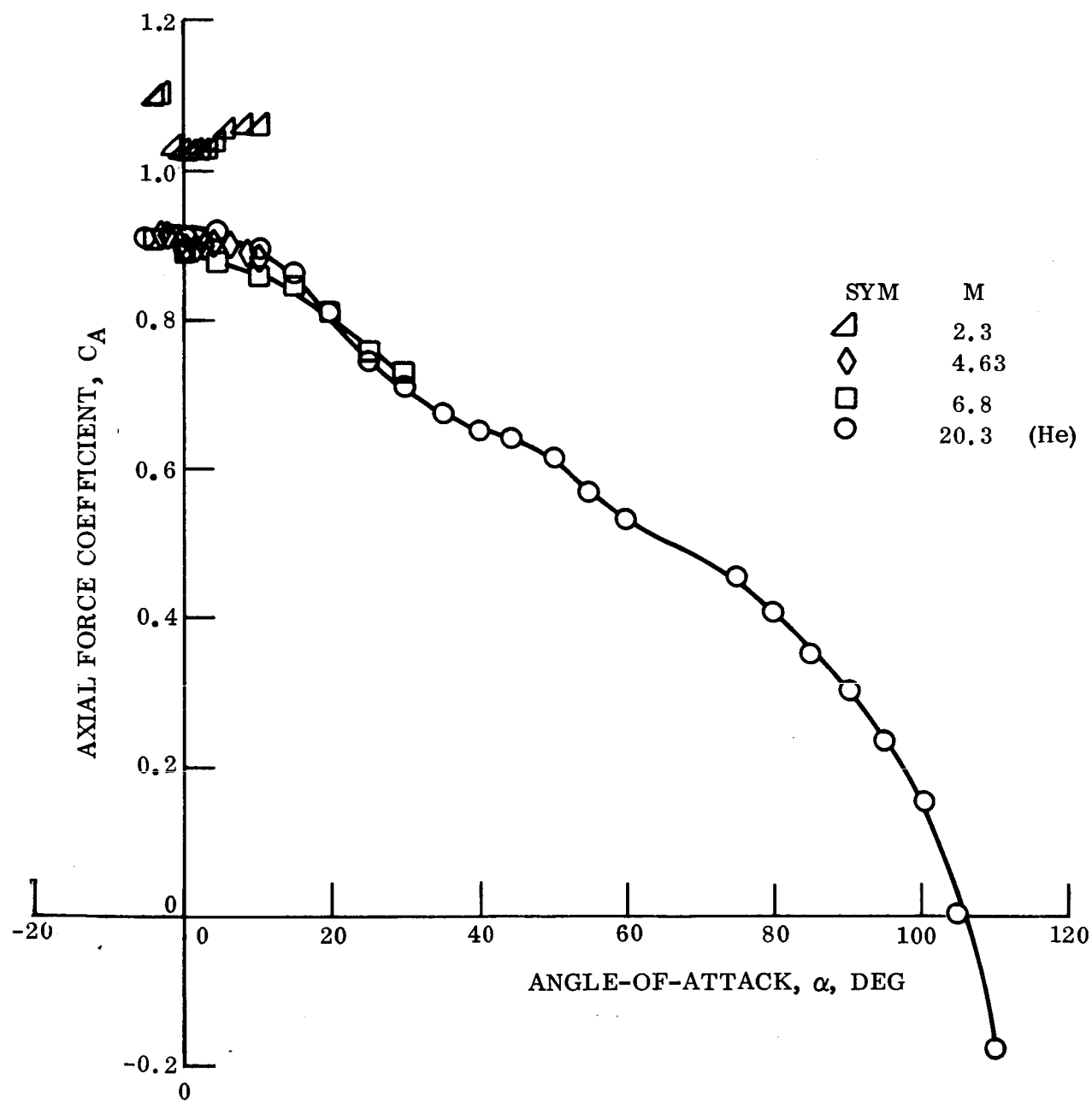


Figure 3.1.2-34. Axial Force Coefficient vs Angle-of-Attack
 $\theta = 40^\circ$ and $R_N/R_B = 0$

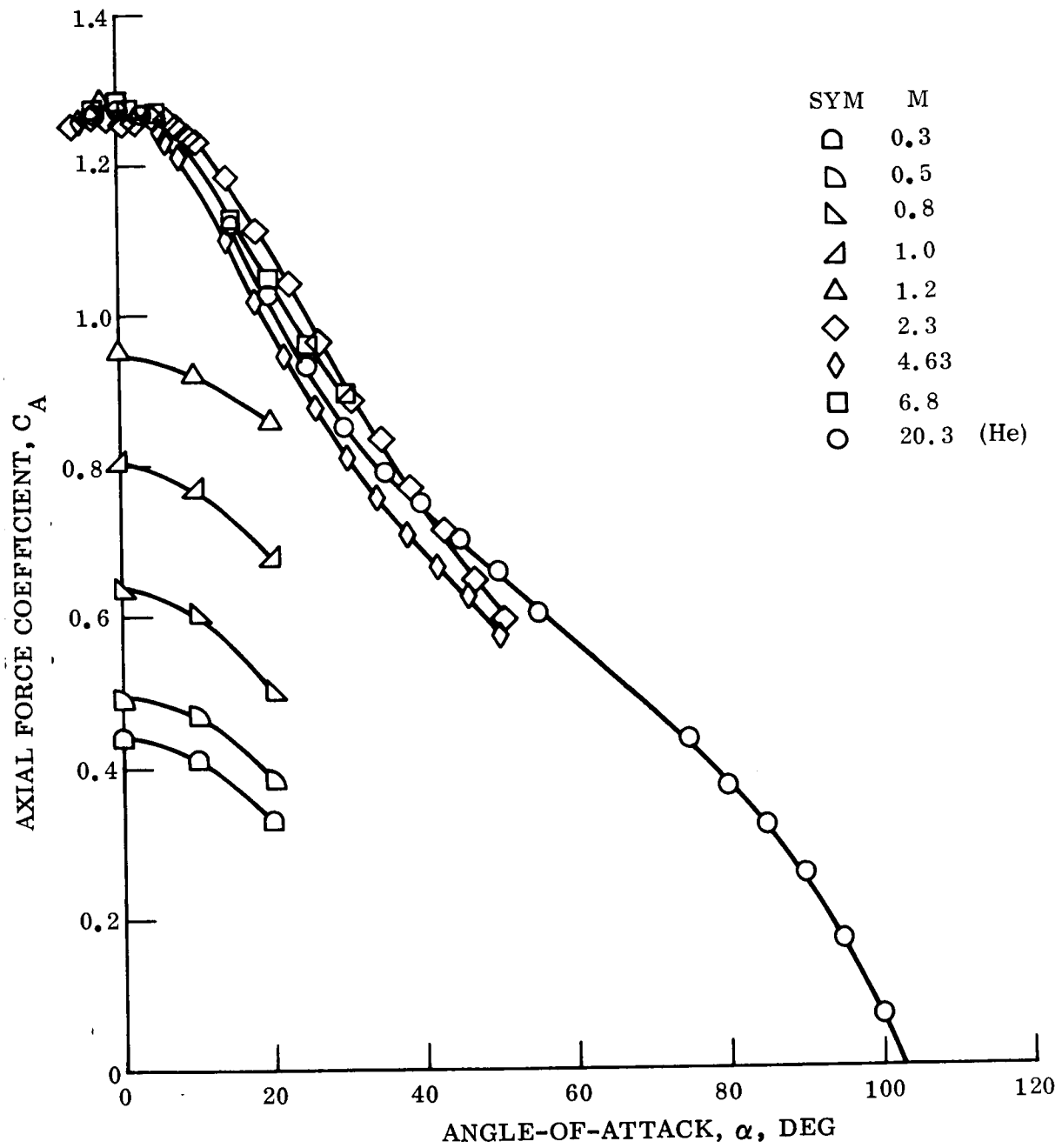


Figure 3.1.2-35. Axial Force Coefficient vs Angle-of-Attack
 $\theta = 50^\circ$ and $R_N/R_B = 0$

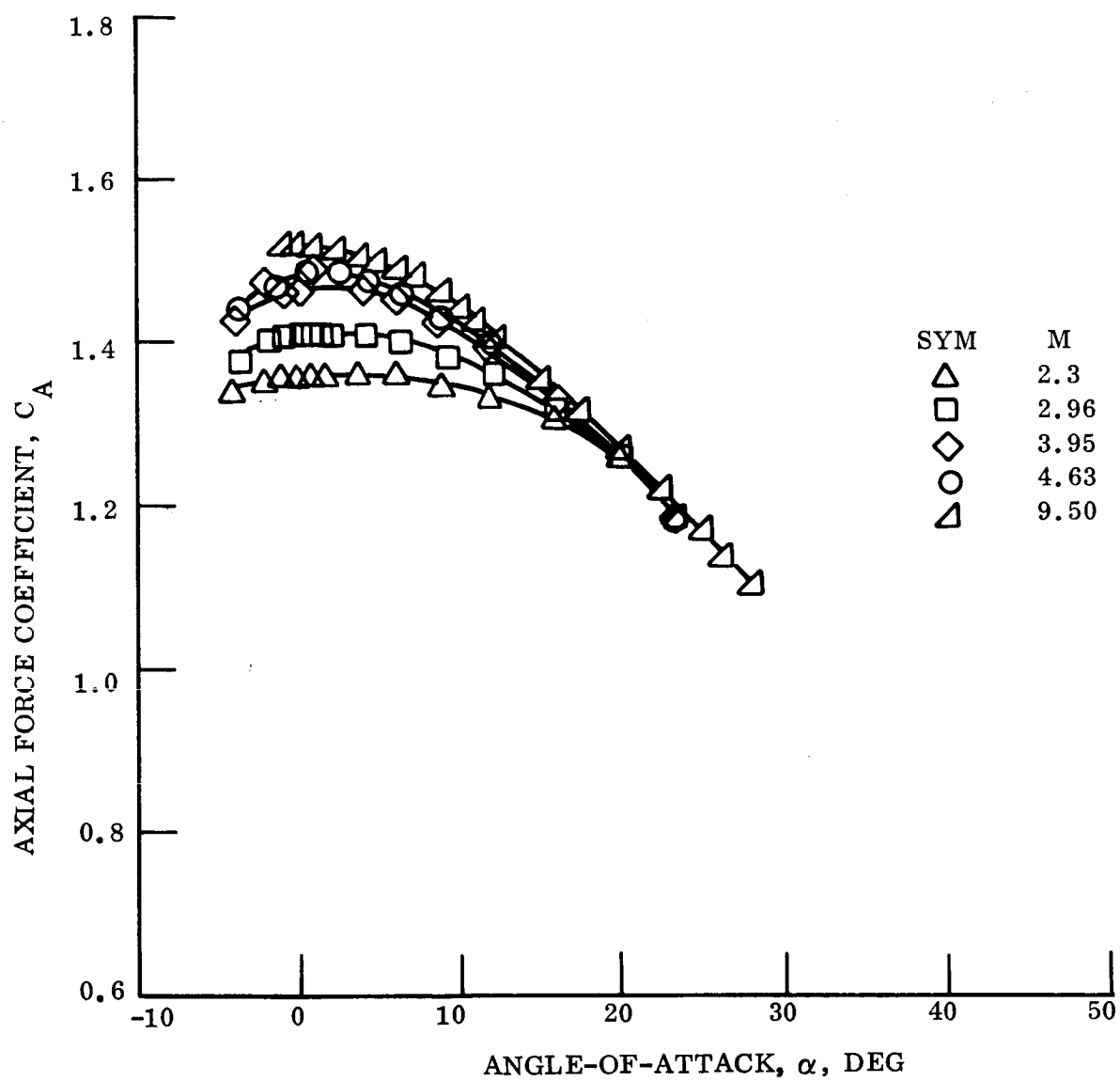


Figure 3.1.2-36. Axial Force Coefficient vs Angle-of-Attack
 $\theta = 60^\circ$ and $R_N/R_B = 0$

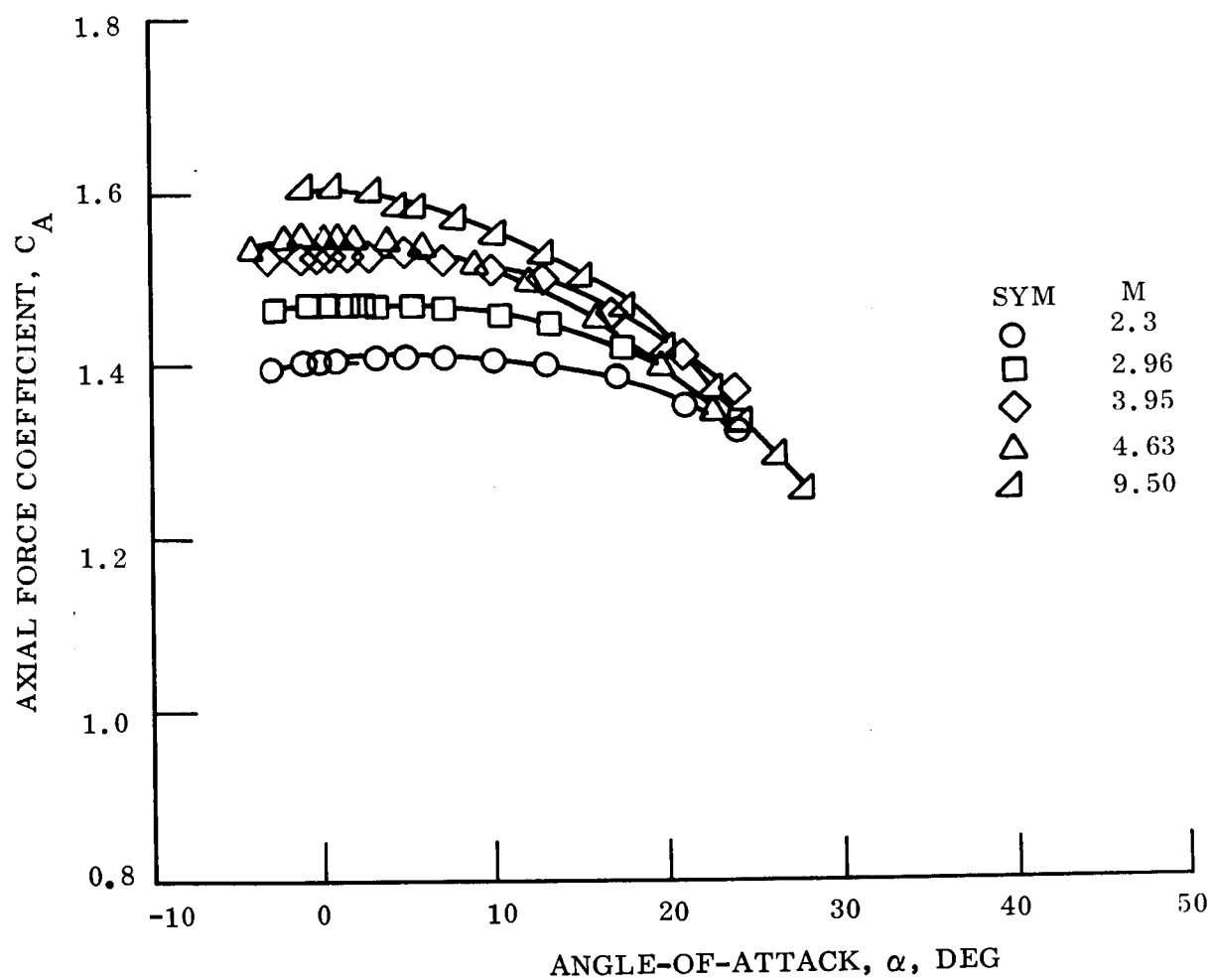


Figure 3.1.2-37. Axial Force Coefficient vs Angle-of-Attack
 $\theta = 70^\circ$ and $R_N/R_B = 0$

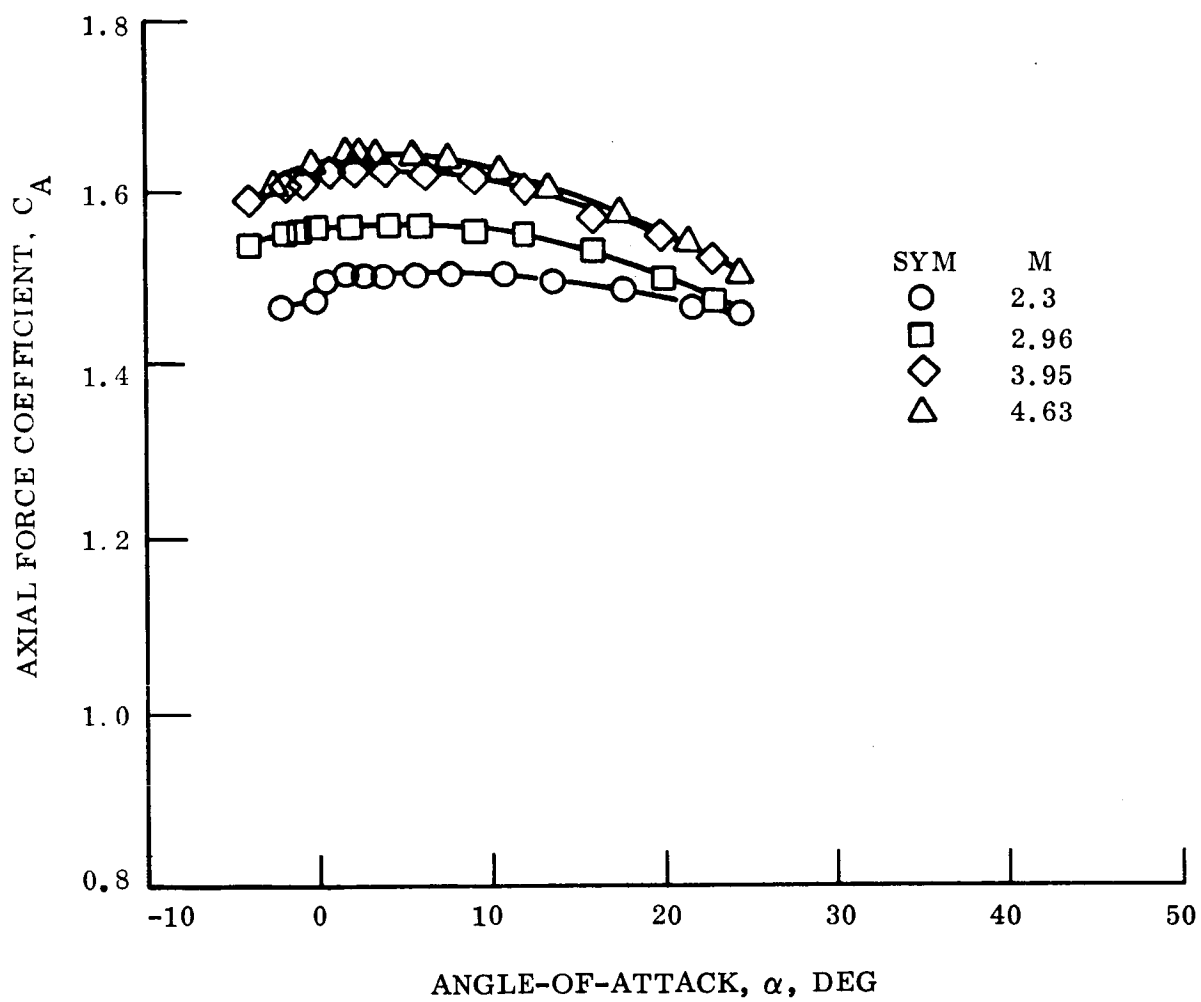


Figure 3.1.2-38. Axial Force Coefficient vs Angle-of-Attack
 $\theta = 80^\circ$ and $R_N/R_B = 0$

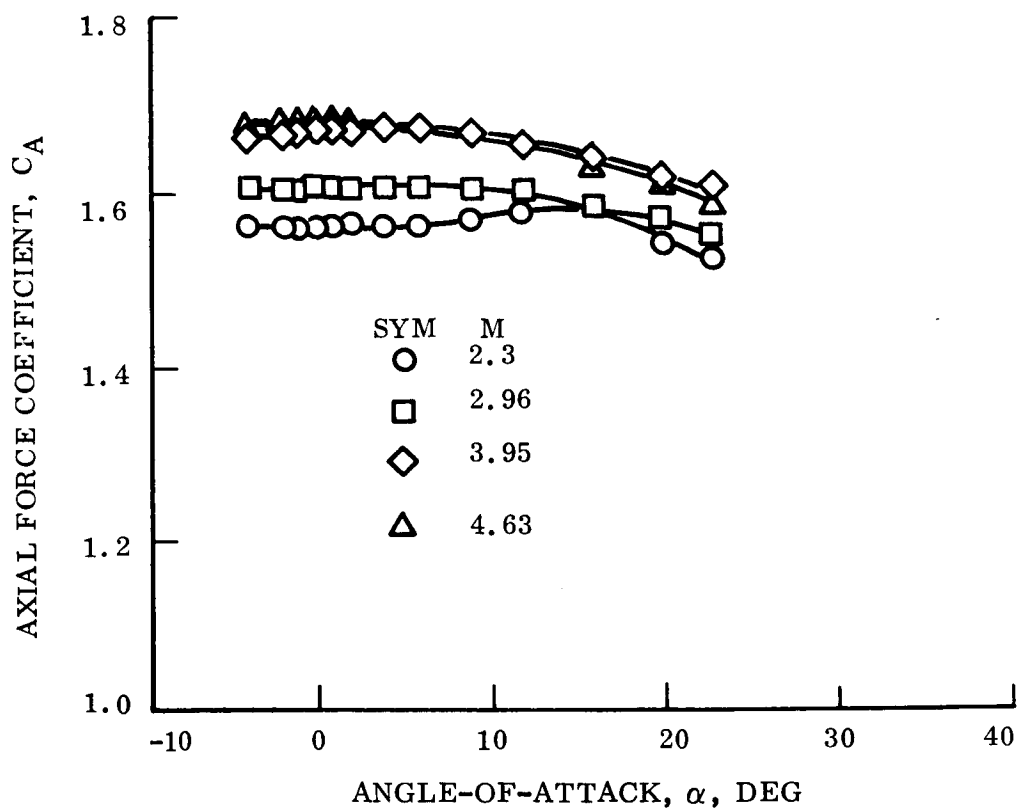


Figure 3.1.2-39. Axial Force Coefficient vs Angle-of-Attack
 $\theta = 90^\circ$ and $R_N/R_B = 0$

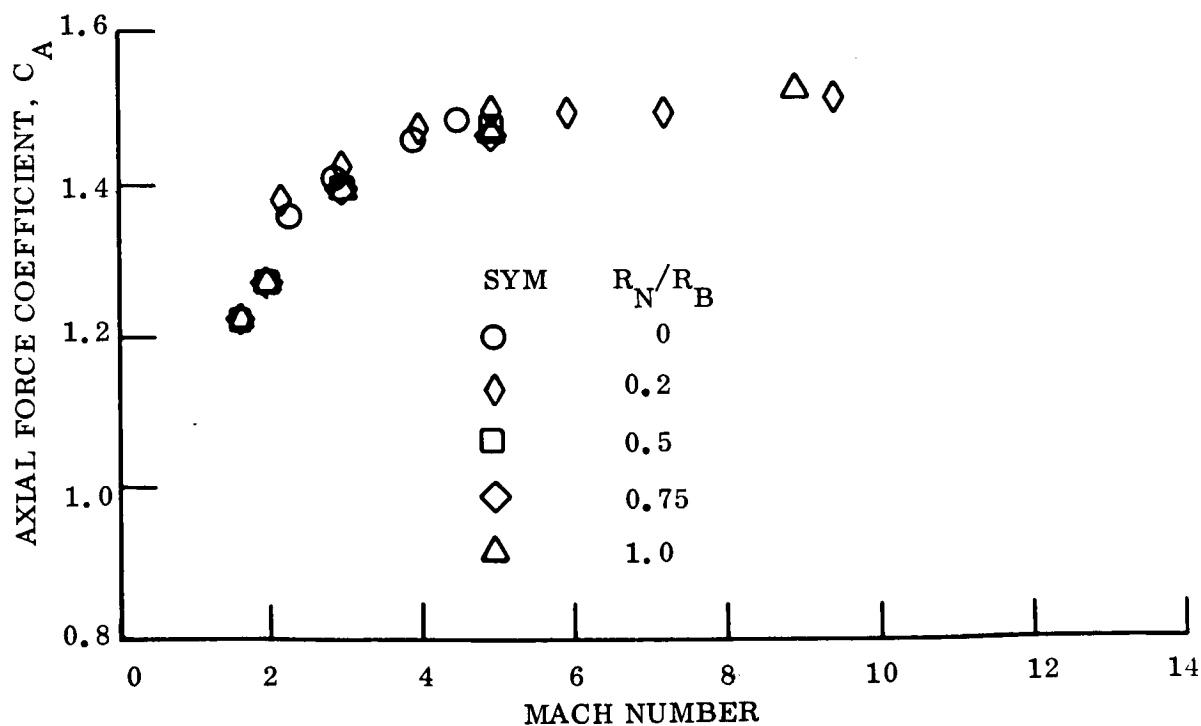


Figure 3.1.2-40. Effect of Bluntness on 60° Cone Axial Force Coefficient Variation with Mach Number

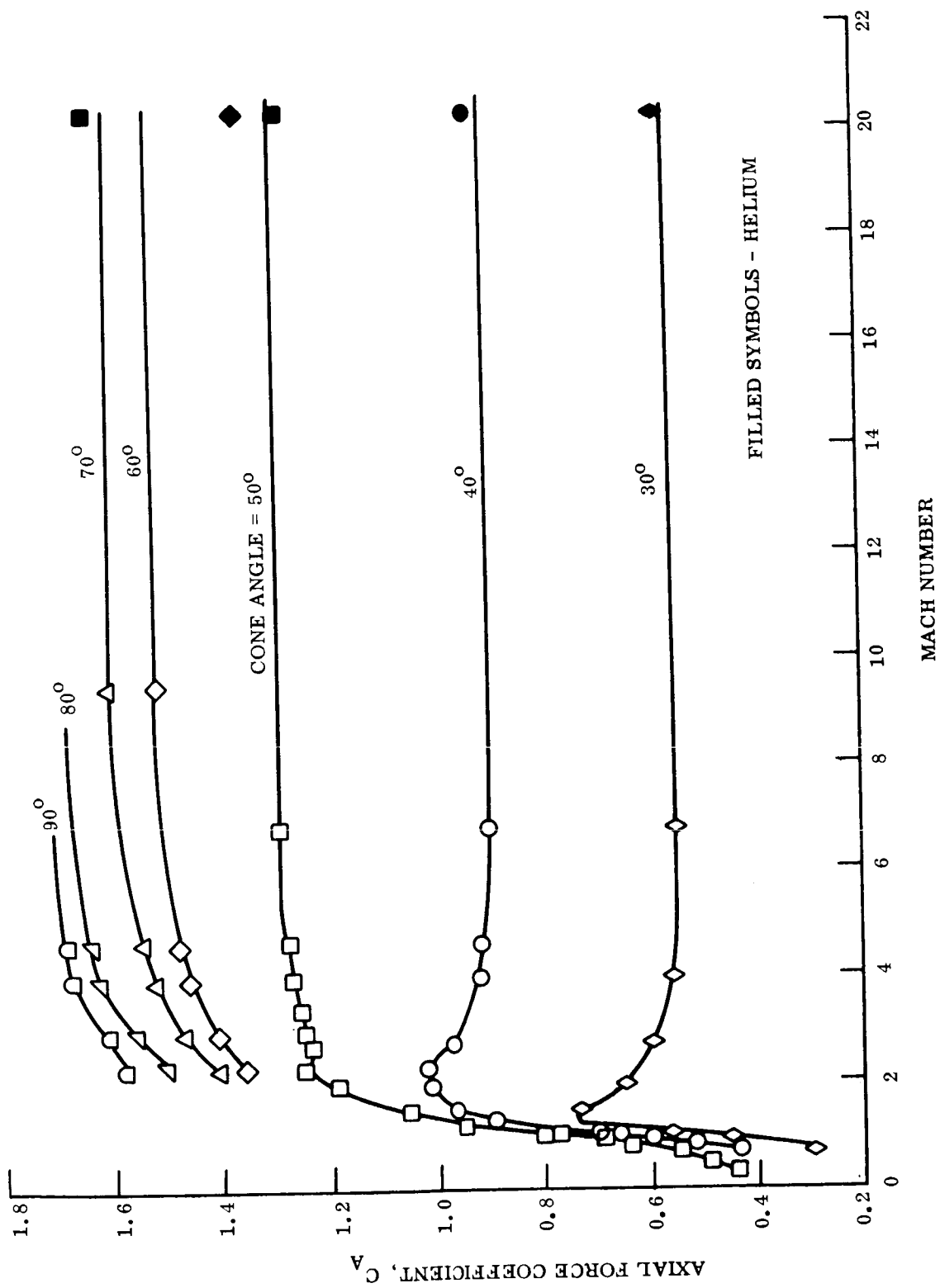


Figure 3.1.2-41. Effect of Cone Angle on Zero Angle-of-Attack on Axial Force Coefficient Variation with Mach Number - $R_N/R_B = 0$

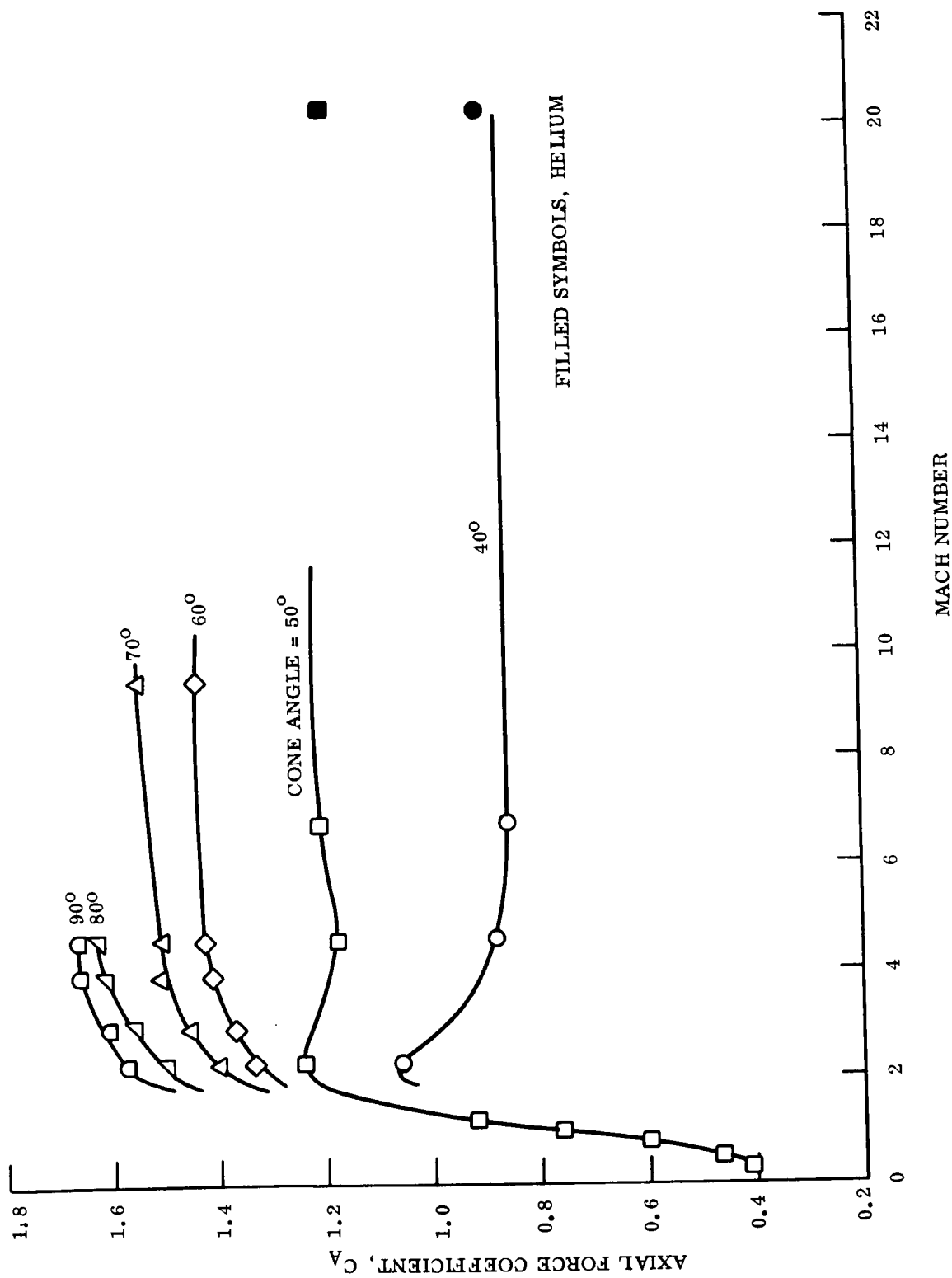


Figure 3.1.2-42. Effect of Cone Angle at 10° Angle-of-Attack on Axial Force Coefficient Variation with Mach Number - $R_N/R_B = 0$

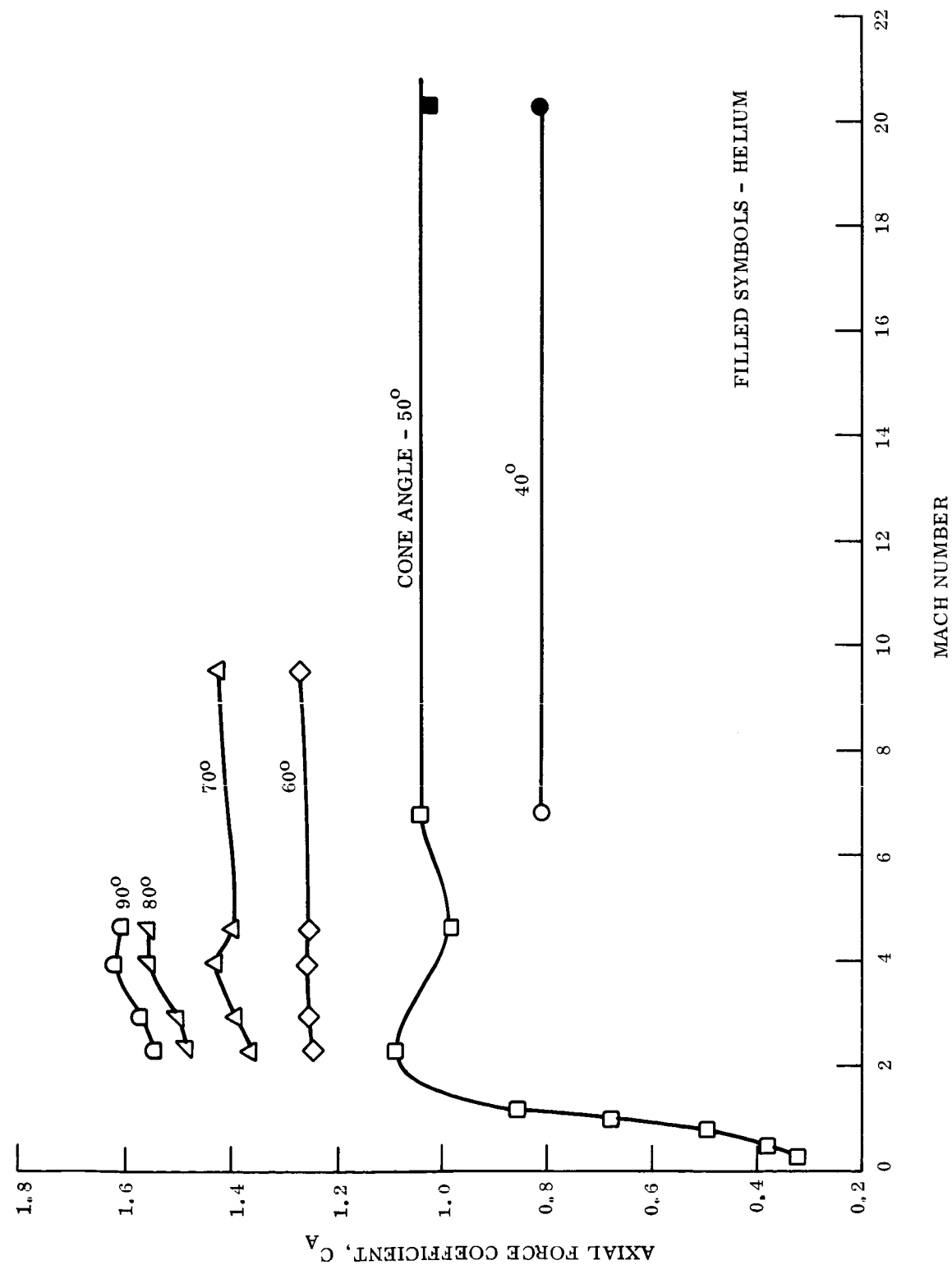


Figure 3.1.2-43. Effect of Cone Angle at 20° Angle-of-Attack on Axial Force Coefficient Variation with Mach Number - $R_N/R_B = 0$

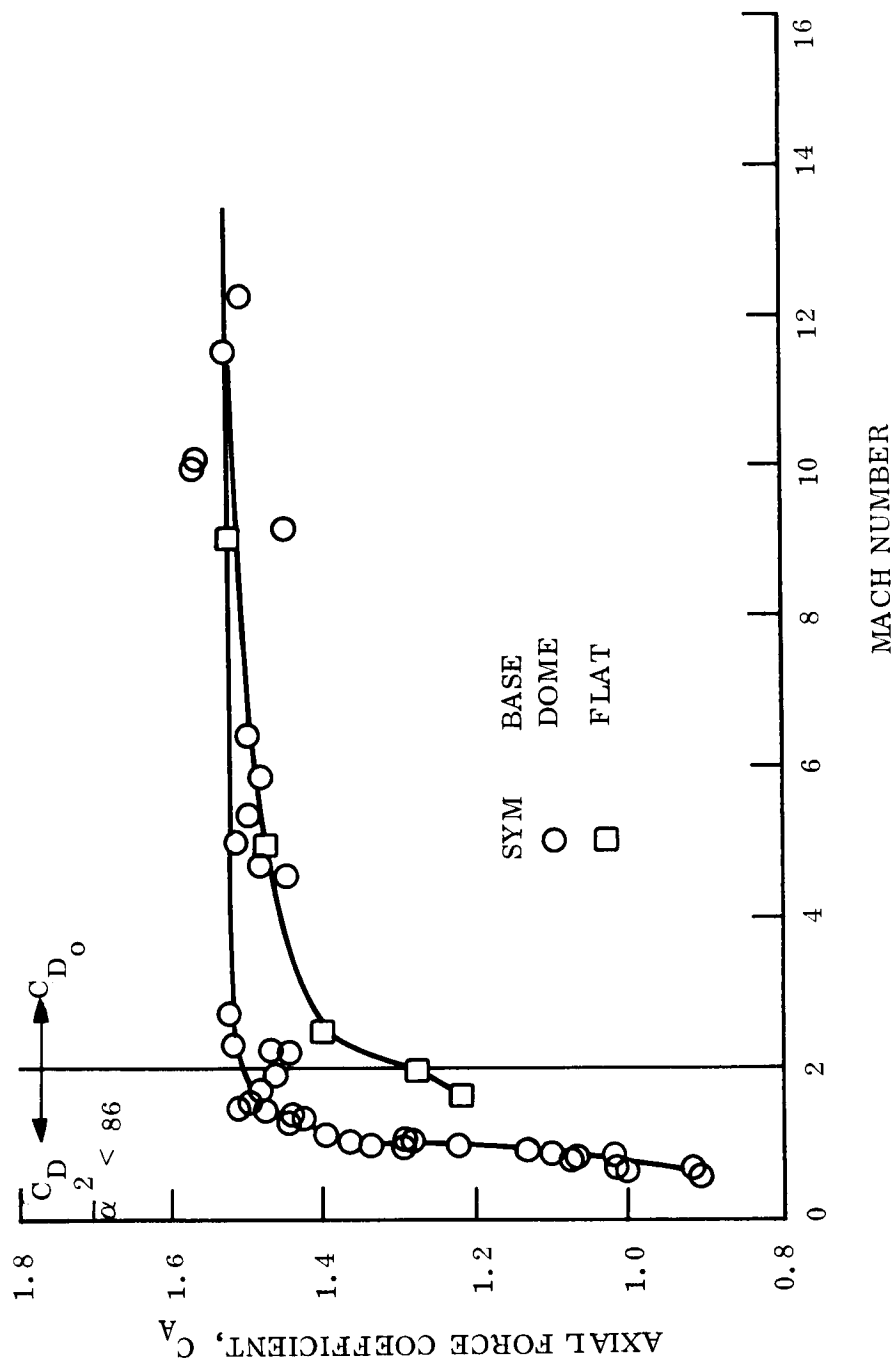


Figure 3.1.2-44. Effect of Afterbody Shape on Drag Coefficient Variation with Mach Number for a Blunt 60° Sphere - Cone

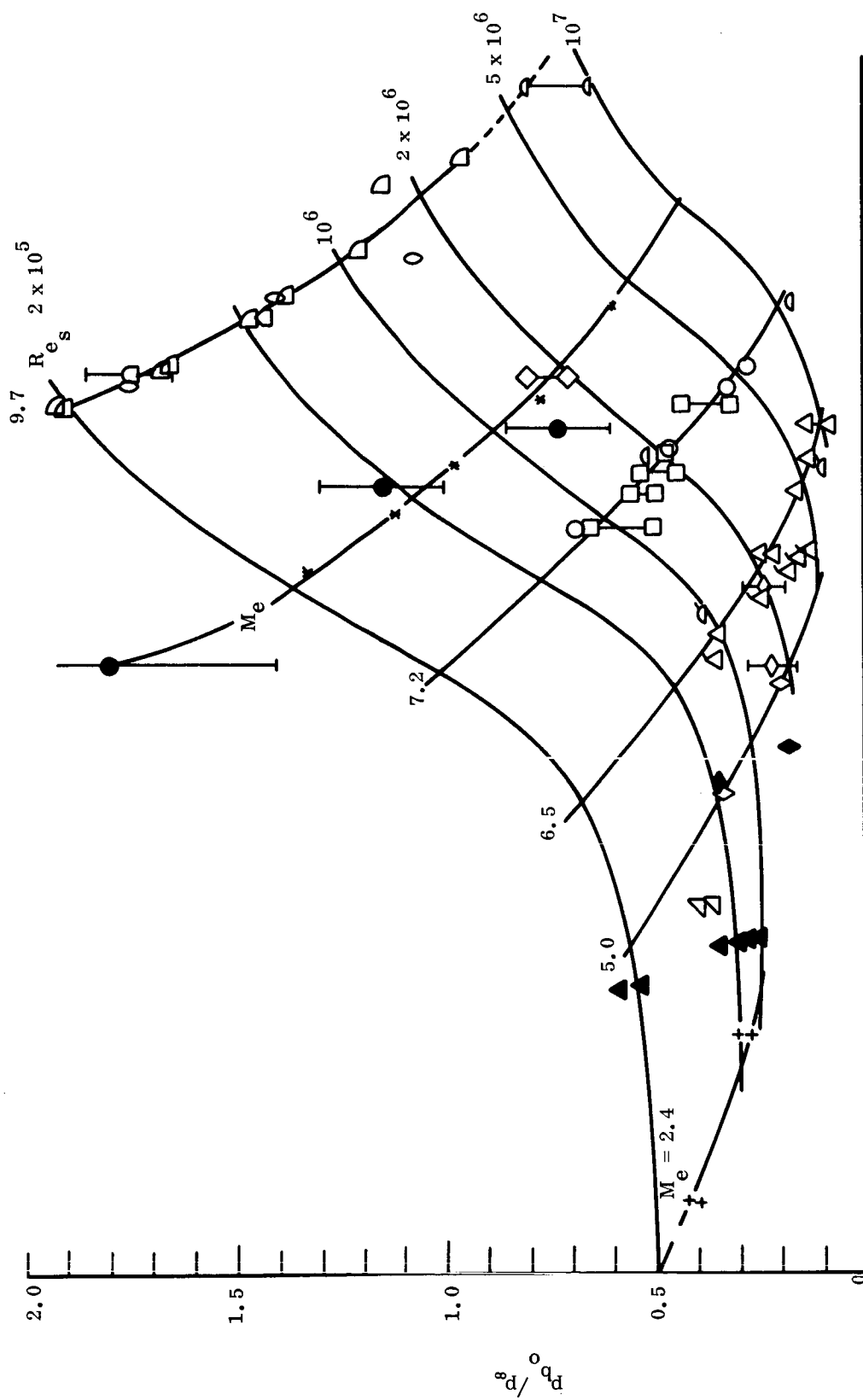


Figure 3.1.2-45. Laminar Centerline Base Pressure Correlation for Flat Base Cones

McMullen and Watson (ref. 3-15) presented aerodynamic data for four configurations with cone semi-vertex angles of 35°, 45°, 50°, and 60°. For purposes of completeness, and for comparison between the analytical techniques used and the experimental data, plots of the aerodynamic data for the 60° configuration are also presented. Fig. 3.1.2-46 shows that comparison of analytical and experimental data. While the experimental data values are consistently lower than the analytical values, the differences are well within specified tolerances. Also, as mentioned previously, correction of the experimental data for the base drag contribution would raise those values and lessen the differences.

B. Normal Force Coefficient - Normal force coefficients are presented in figs. 3.1.2-47 through -52 as a function of angle-of-attack and Mach number for the various cone angles. The C_N decreases significantly at a given angle-of-attack, with increasing cone angle, especially in the region at which the sonic point moves to the shoulder. This can be seen more readily in the normal force coefficient slope plots (figs. 3.1.2-53 through -56). As with axial force coefficient, bluntness does not appear to have a significant effect on normal force as shown in fig. 3.1.2-55 for the 60° cone body over a wide range of Mach number.

Once again, a comparison is made with McMullen's data of ref. 3-15, in fig. 3.1.2-57, which shows that the experimental data are somewhat high in the $M = 3$ region and slightly lower in the high Mach number region.

C. Aerodynamic Static Stability - Static stability coefficient data are presented in the form of center of pressure, $\frac{X_{CP}}{D}$, measured with respect to the base (maximum diameter) of the vehicle. This correlation format has been found to provide a reasonable stability matrix. Together with center of gravity and normal force coefficient, the pitching moment coefficient about any center of gravity can be determined:

$$C_{M_{CG}} = C_N \frac{(X_{CG} - X_{CP})}{D}$$

These center of pressure data for flat based vehicles are presented in figs. 3.1.2-58 through -62. As expected, the center of pressure moves further behind the base (maximum diameter) as the cone angle increases. The effects of bluntness are shown in fig. 3.1.2-60 for the 60° cone. With the exception of the largest bluntness ($R_N/R_B = 1.0$), there does not appear to be much effect for this large cone angle.

The effects on static stability of rounding the shoulder are presented in fig. 3.1.2-61 ($\theta_c = 60^\circ$). As expected, the center of pressure moves forward with increasing shoulder-to-base-radius ratio. The comparison of the analytical technique of ref. 3-15 with the experimental data is presented in fig. 3.1.2-63. The agreement is within specified tolerances.

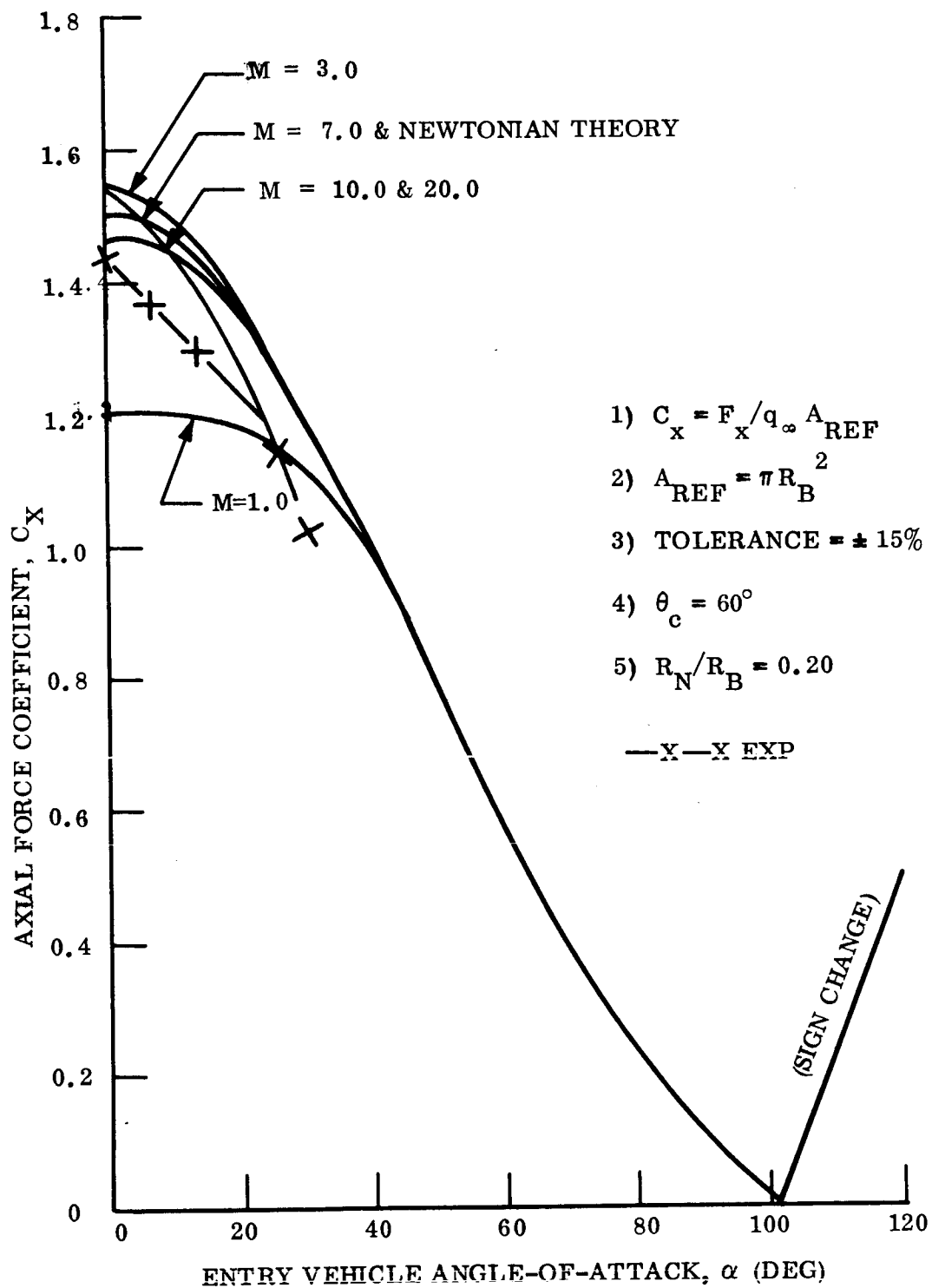


Figure 3.1.2-46. Variation of Axial Force Coefficient with Angle-of-Attack and Mach Number for a 0.2 Blunted 60° Cone

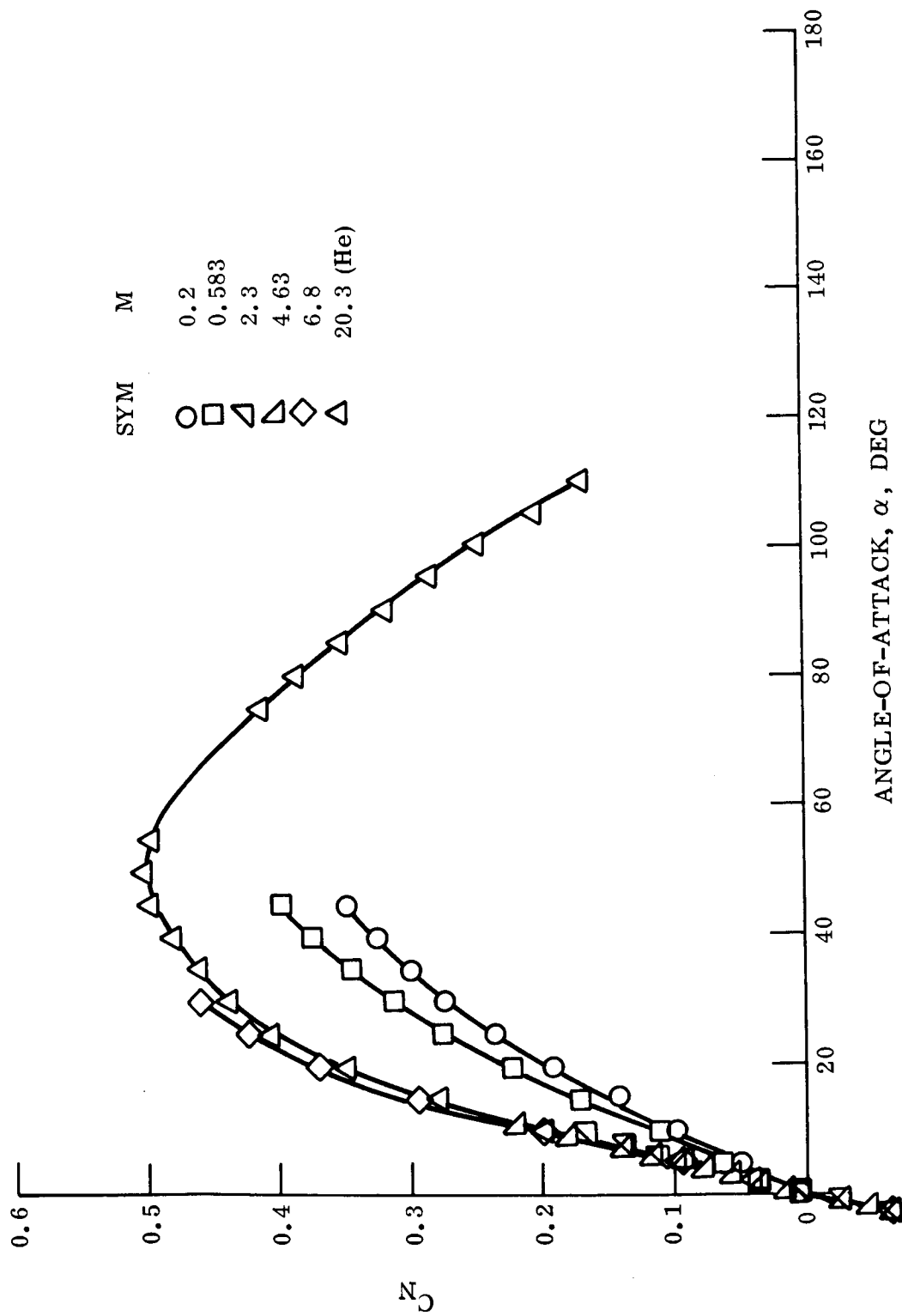


Figure 3.1.2-47. Normal Force Coefficient vs Angle-of-Attack, Cone Angle
 $(\theta) = 40^\circ$ and $R_N/R_B = 0$

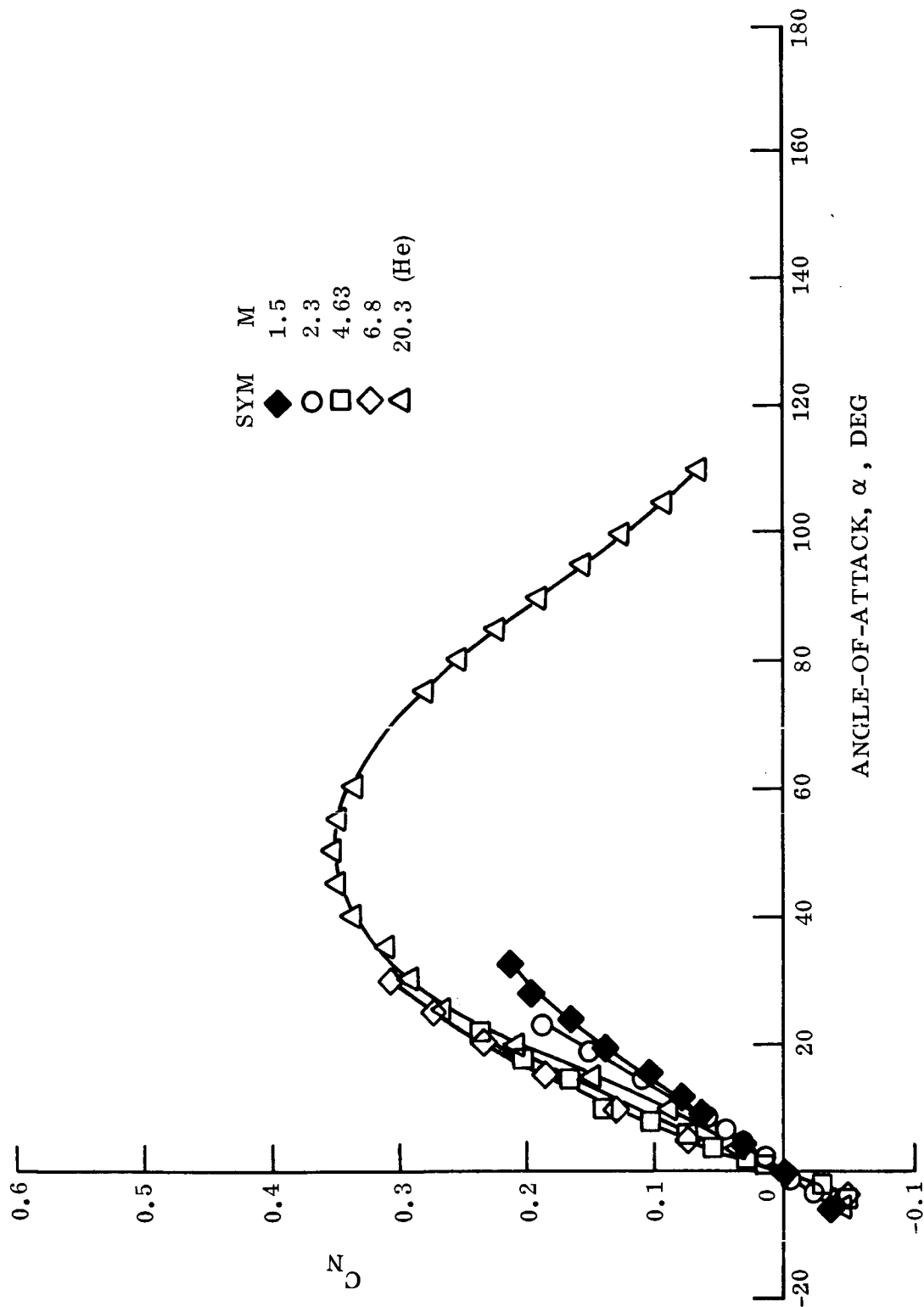


Figure 3.1.2-48. Normal Force Coefficient vs Angle-of-Attack, Cone Angle (θ) = 50° and $R_N/R_B = 0$

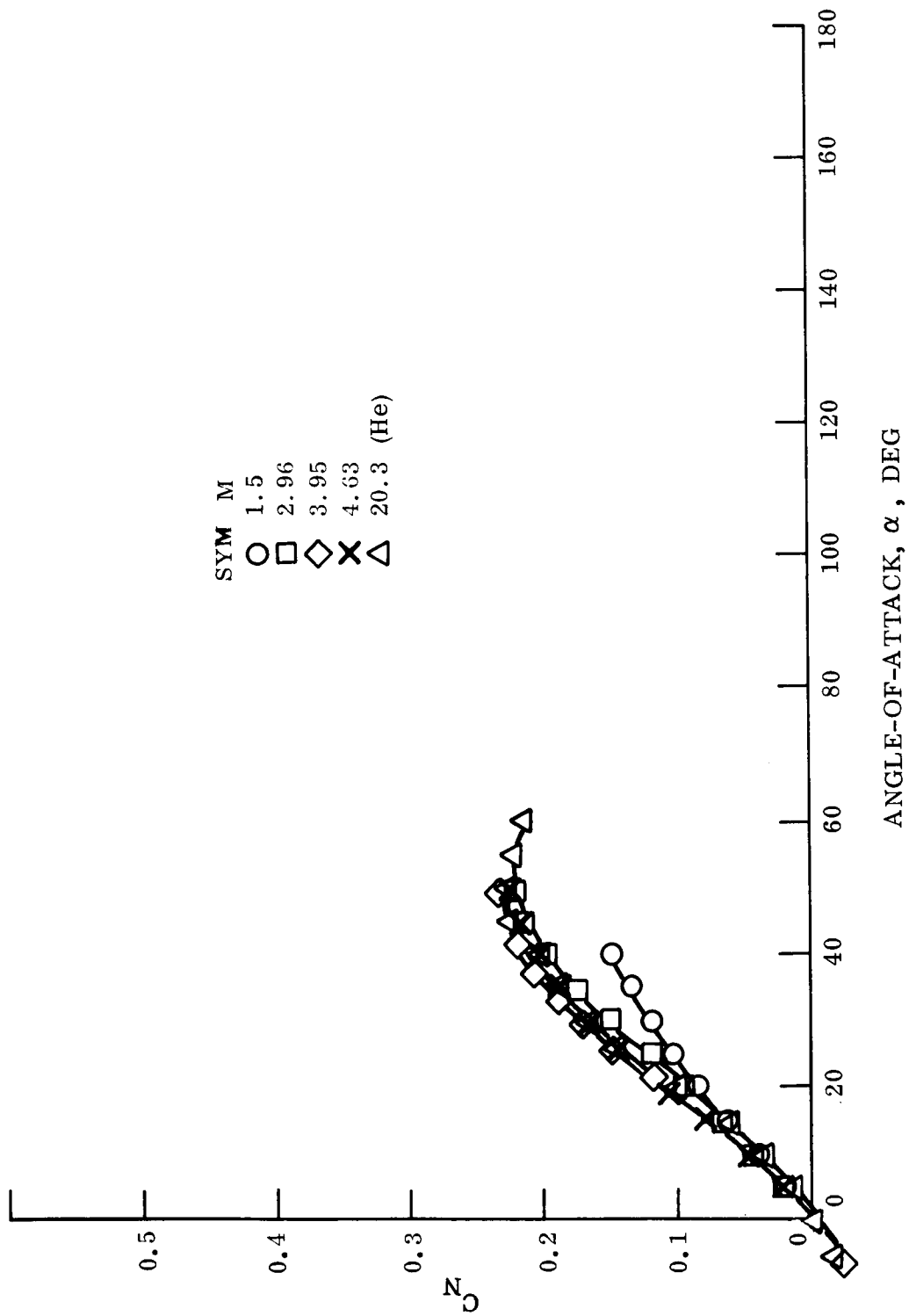


Figure 3.1.2-49. Normal Coefficient vs Angle-of-Attack, Cone Angle $(\theta) = 60^\circ$ and $R_N/R_B = 0$

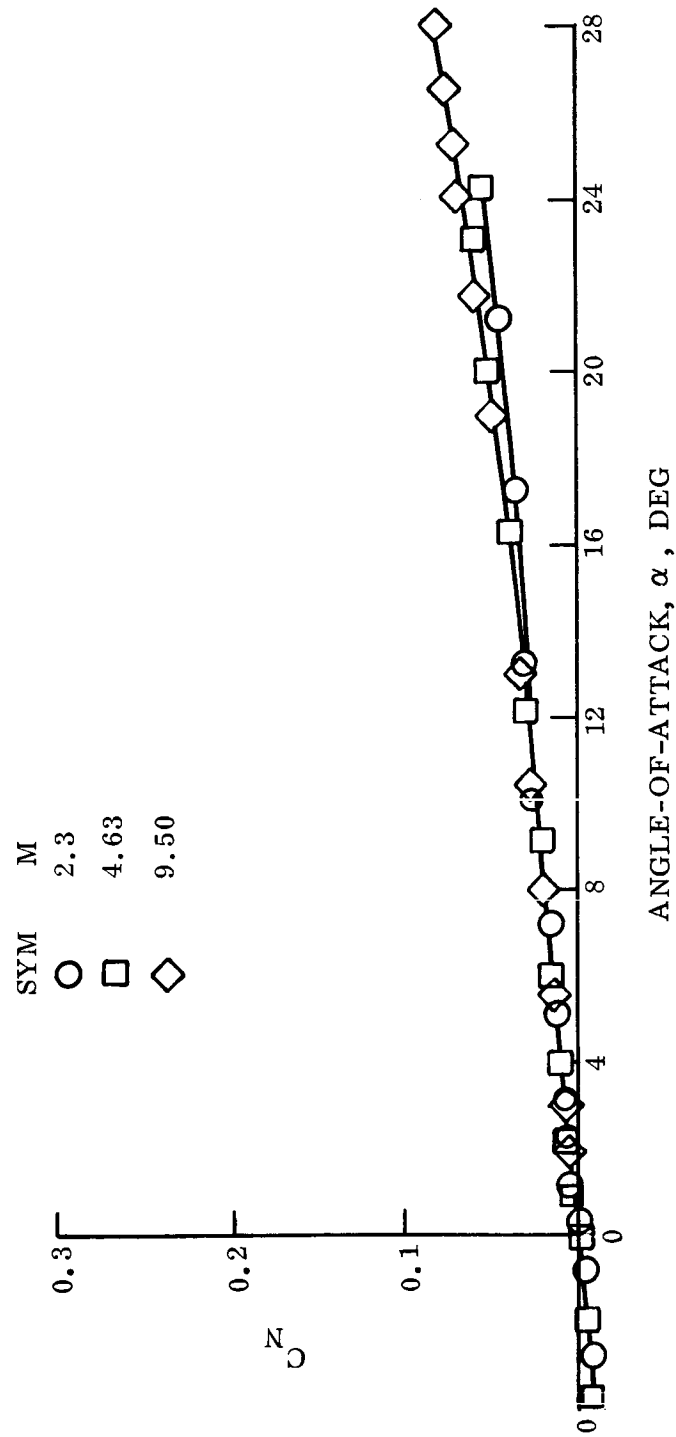


Figure 3.1.2-50. Normal Force Coefficient vs Angle-of-Attack, Cone Angle (θ) = 70° and $R_N/R_B = 0$

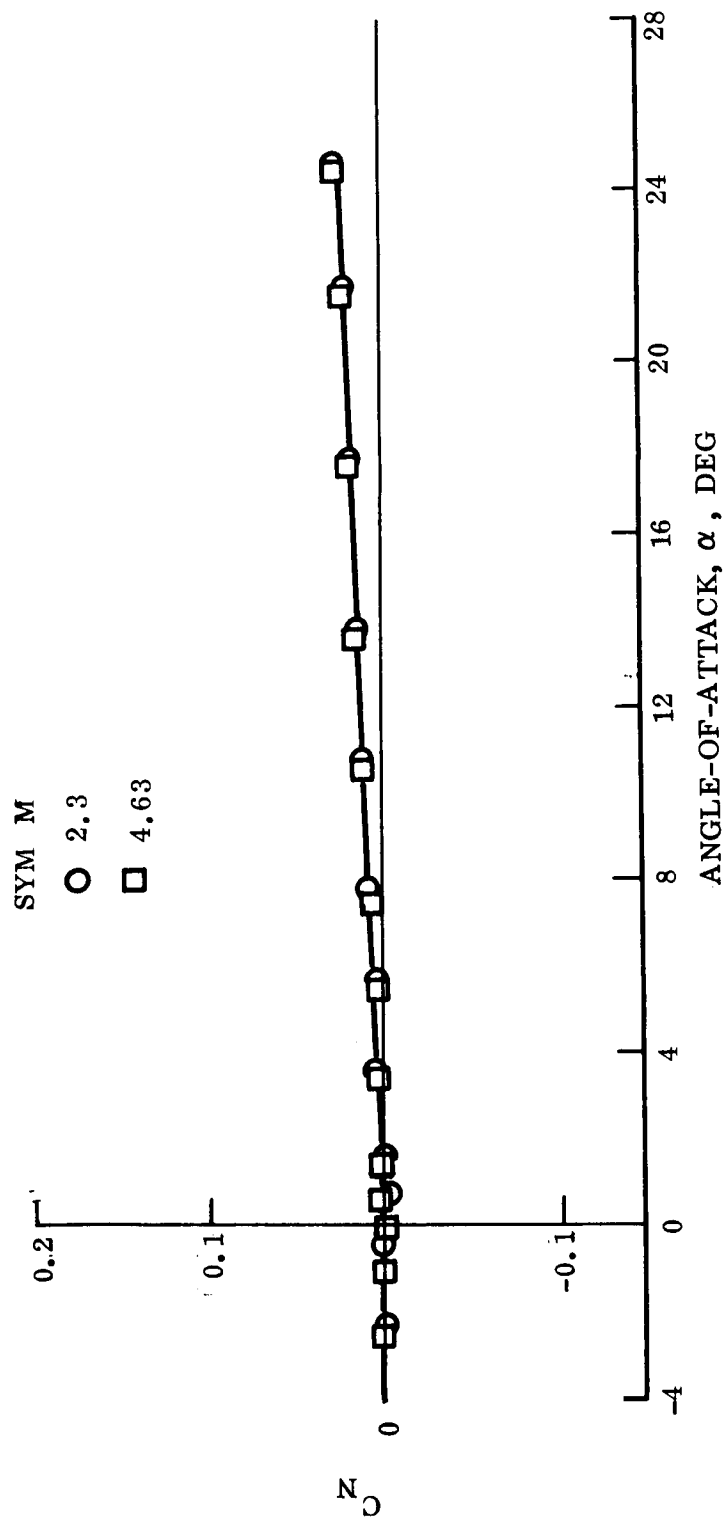


Figure 3.1.2-51. Normal Force Coefficient vs Angle-of-Attack, Cone Angle (θ) = 80° and $R_N/R_B = 0$

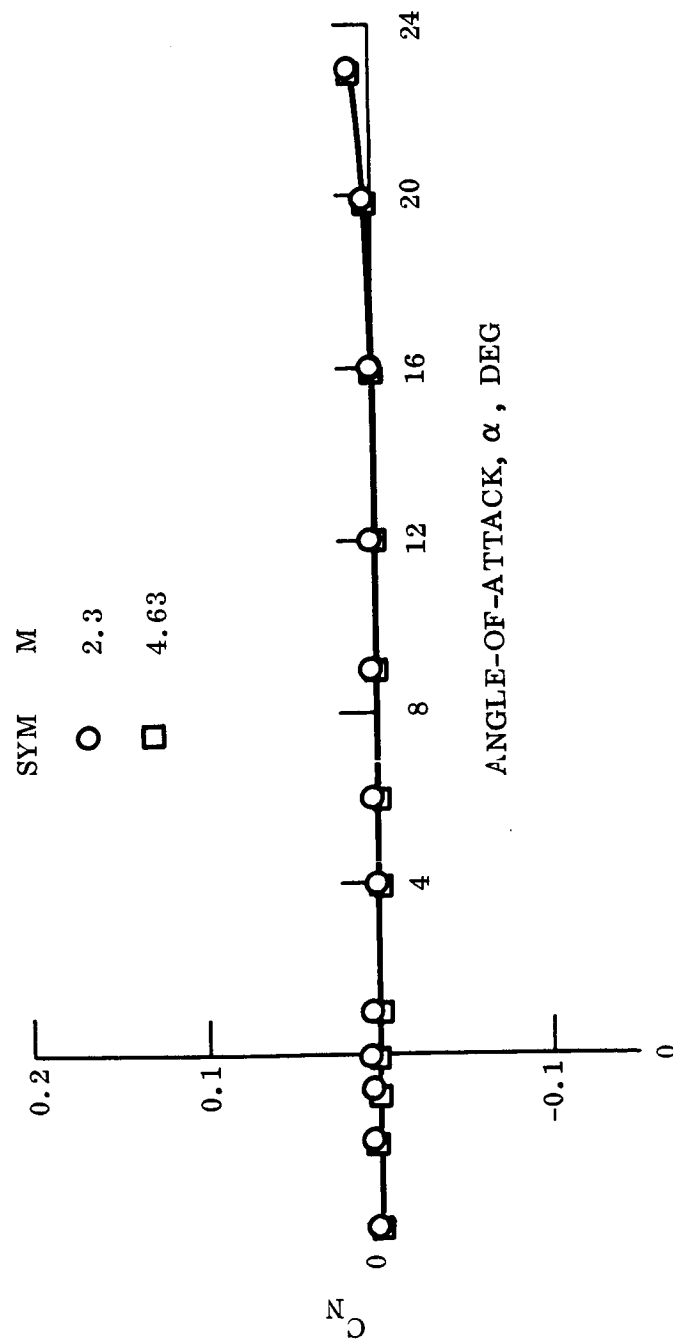


Figure 3.1.2-52. Normal Force Coefficient vs Angle-of-Attack, Cone Angle (θ) = 90°
and $R_N/R_B = 0$

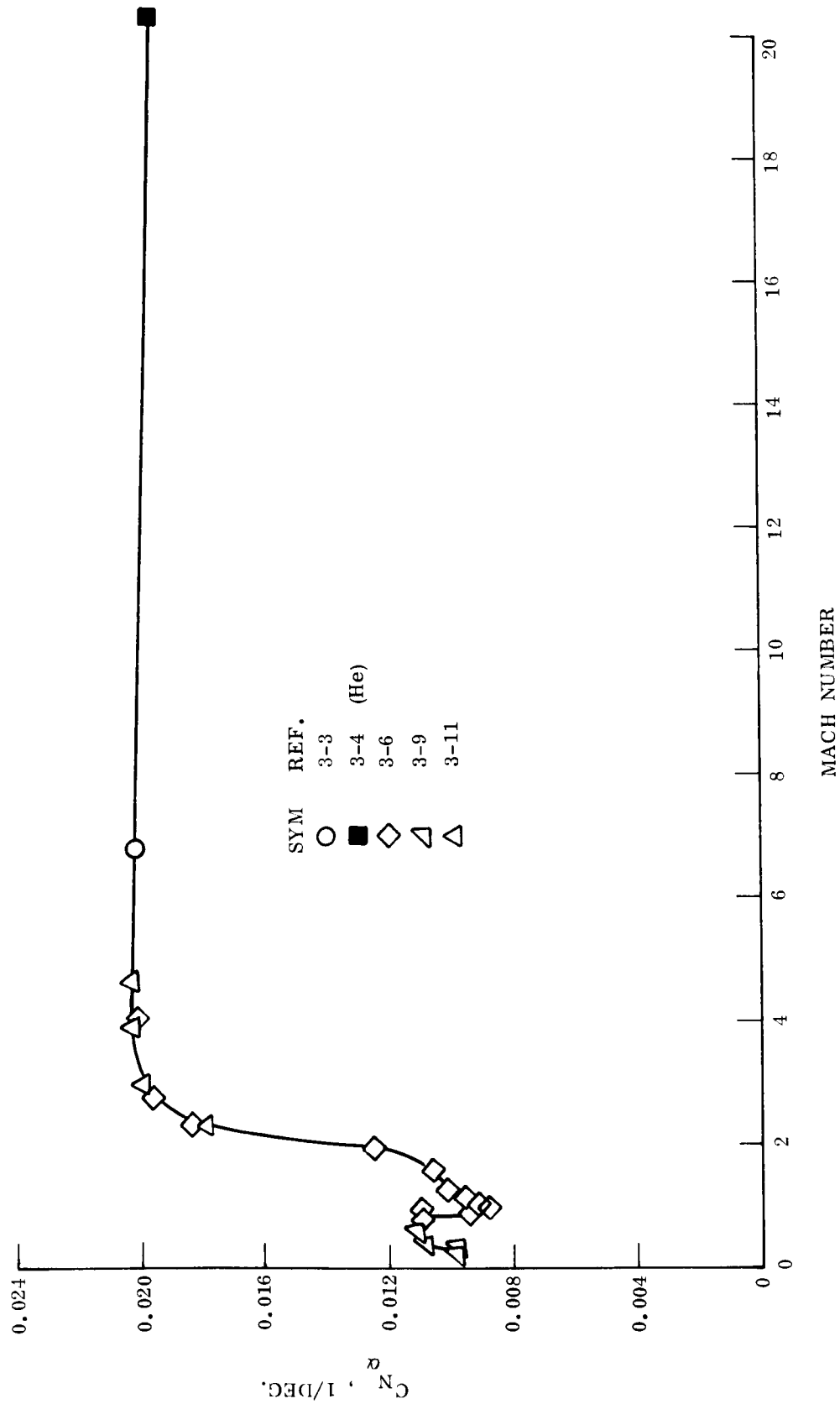


Figure 3.1.2-53. Normal Force Coefficient Slope vs Mach Number, Cone Angle $(\theta) = 40^\circ$ and $R_N/R_B = 0$

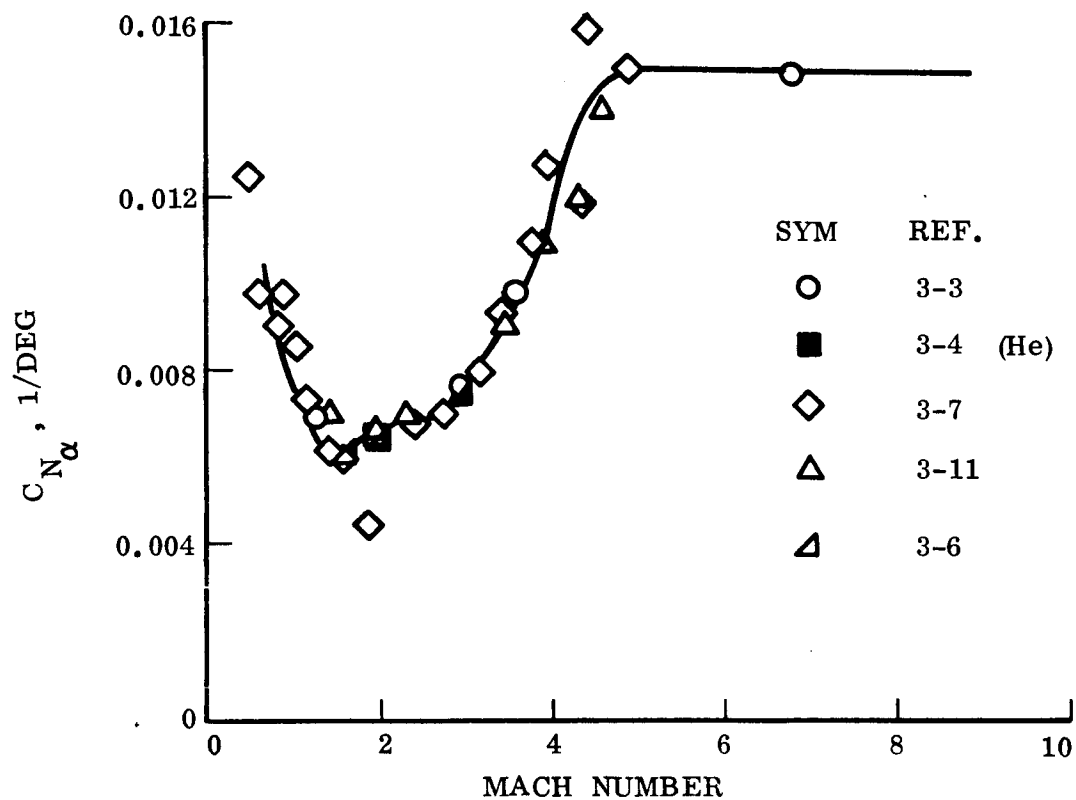


Figure 3.1.2-54. Normal Force Coefficient Slope vs Mach Number,
Cone Angle (θ) = 50° and $R_N/R_B = 0$

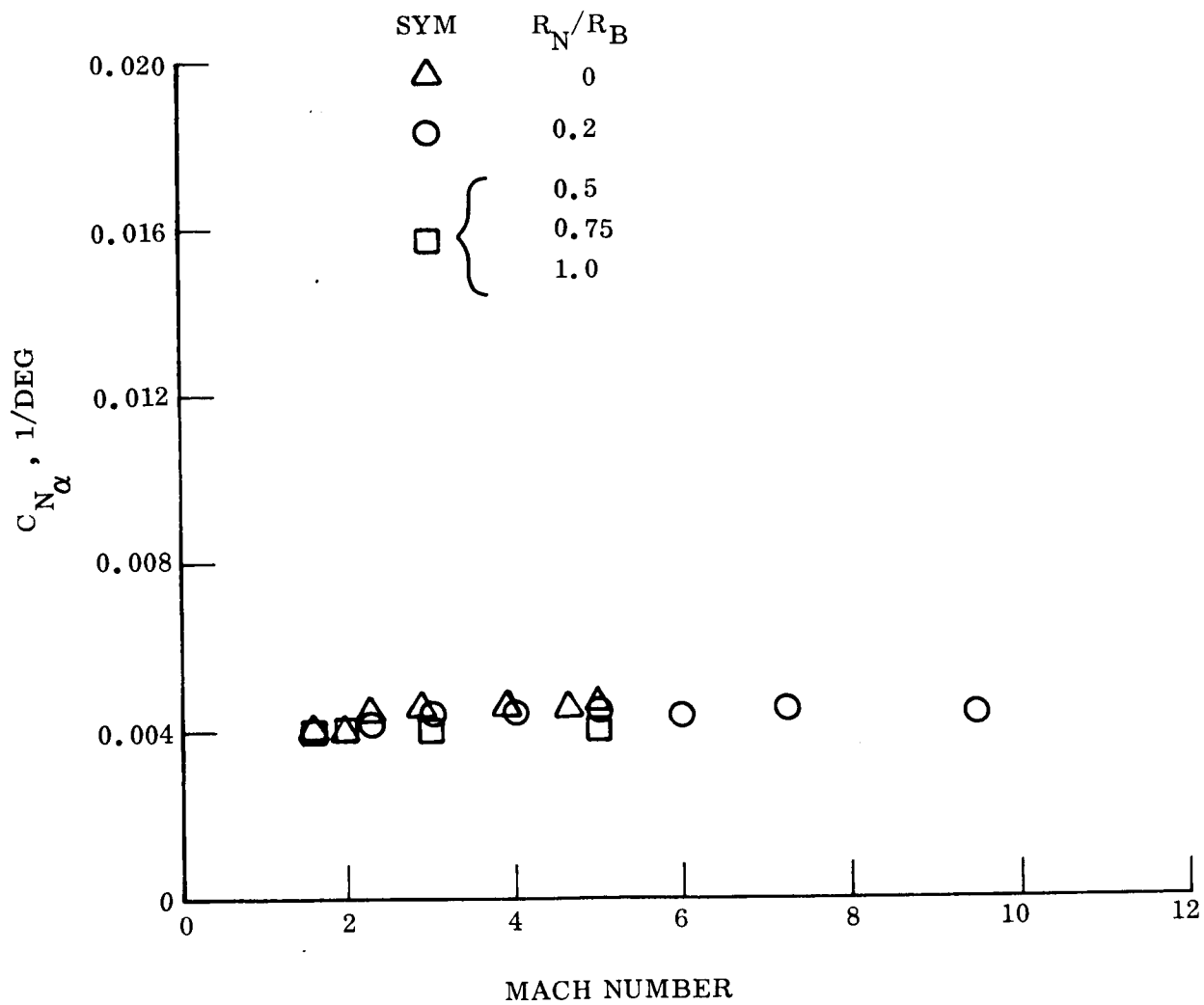


Figure 3.1.2-55. Normal Force Coefficient Slope vs Mach Number, Cone Angle (θ) = 60° and $R_N/R_B = 0$

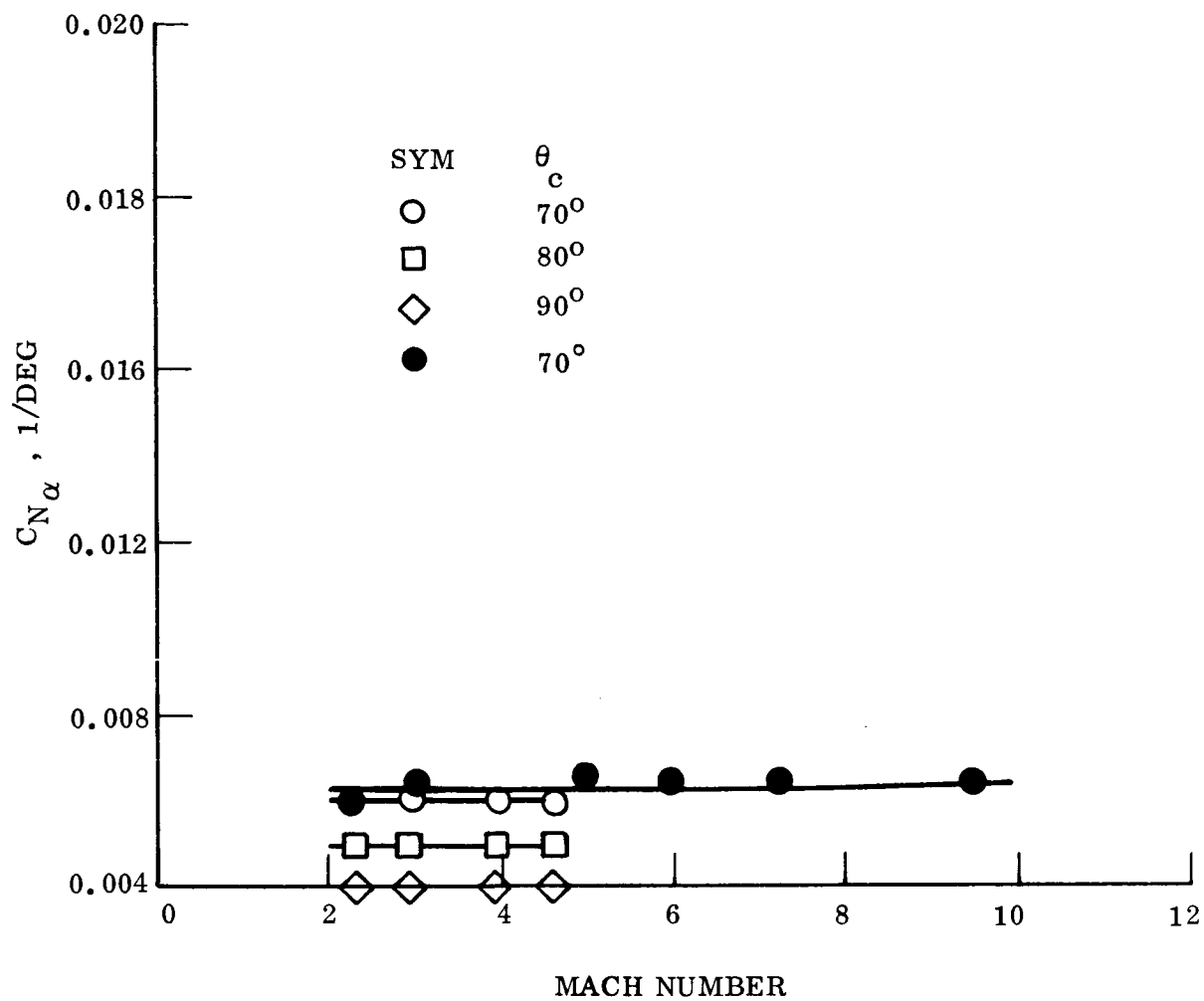


Figure 3.1.2-56. Normal Force Coefficient Slope vs Mach Number,
Cone Angle (θ) = 70° , 80° , and 90°
and $R_N/R_B = 0$

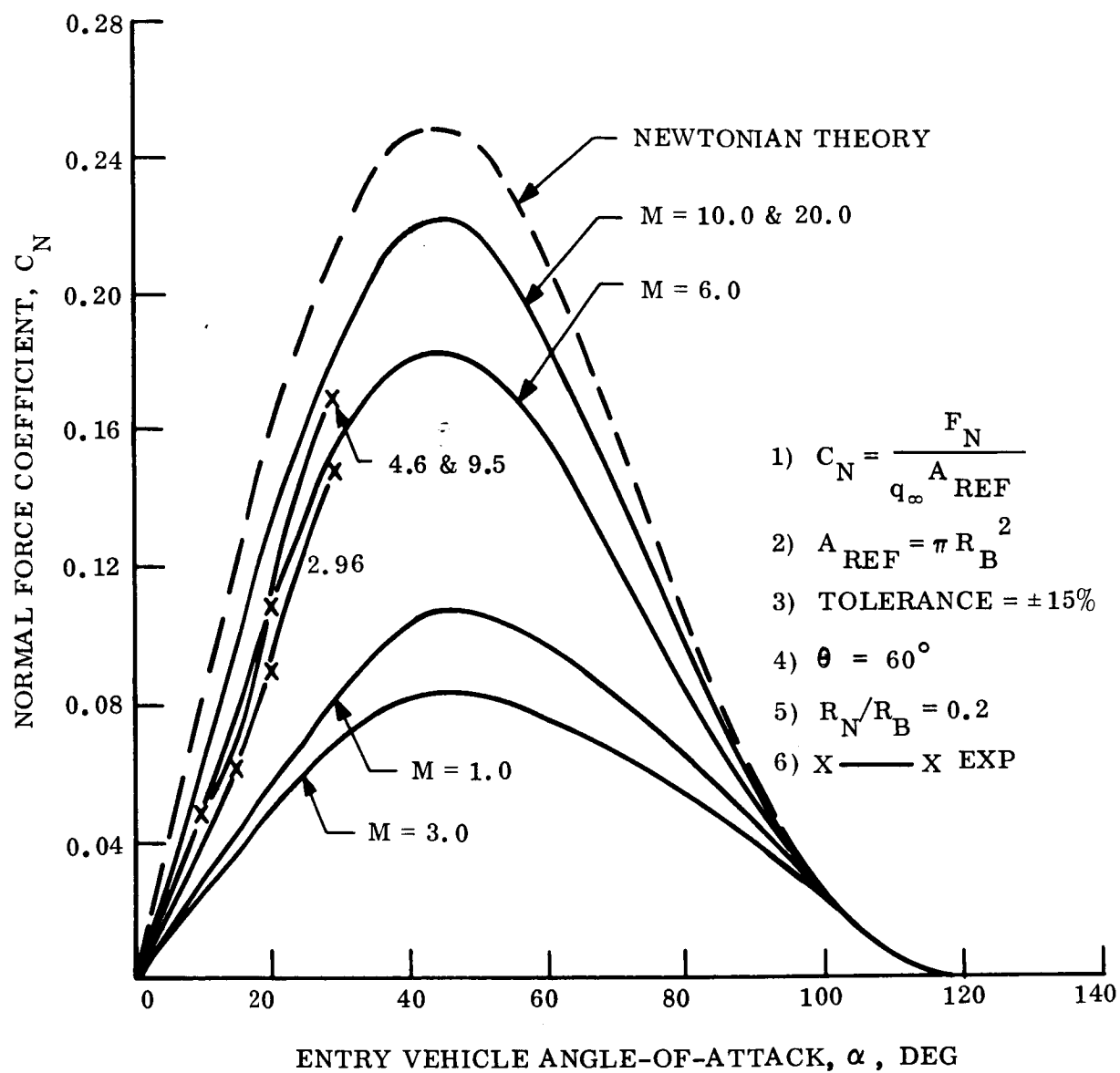


Figure 3.1.2-57. Variation of Normal Force Coefficient with Angle-of-Attack and Mach Number for a 0.2 Blunted 60° Cone

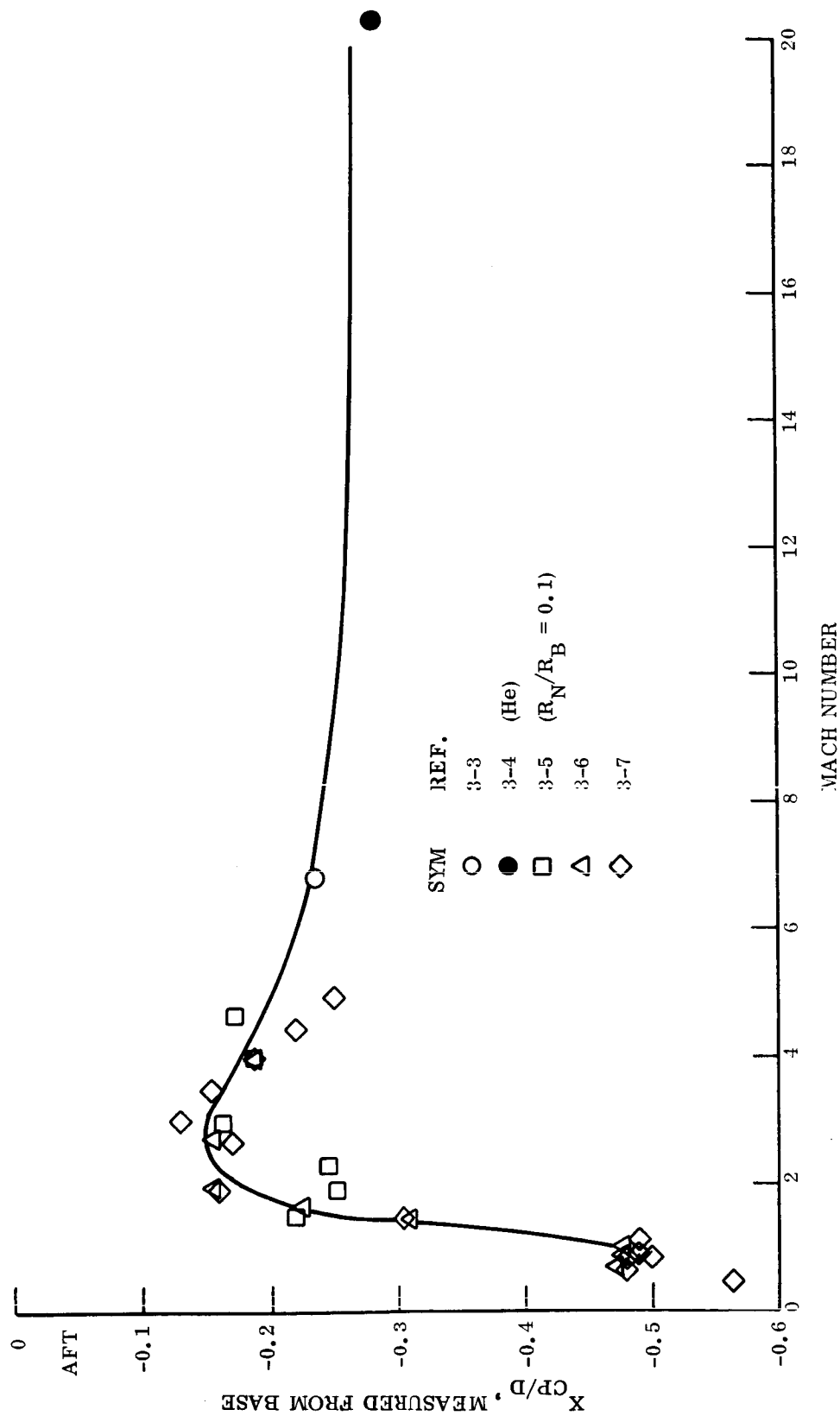


Figure 3.1.2-58. Center of Pressure vs Mach Number at Zero Angle-of-Attack,
Cone Angle (θ) = 40° and $R_N/R_B = 0$

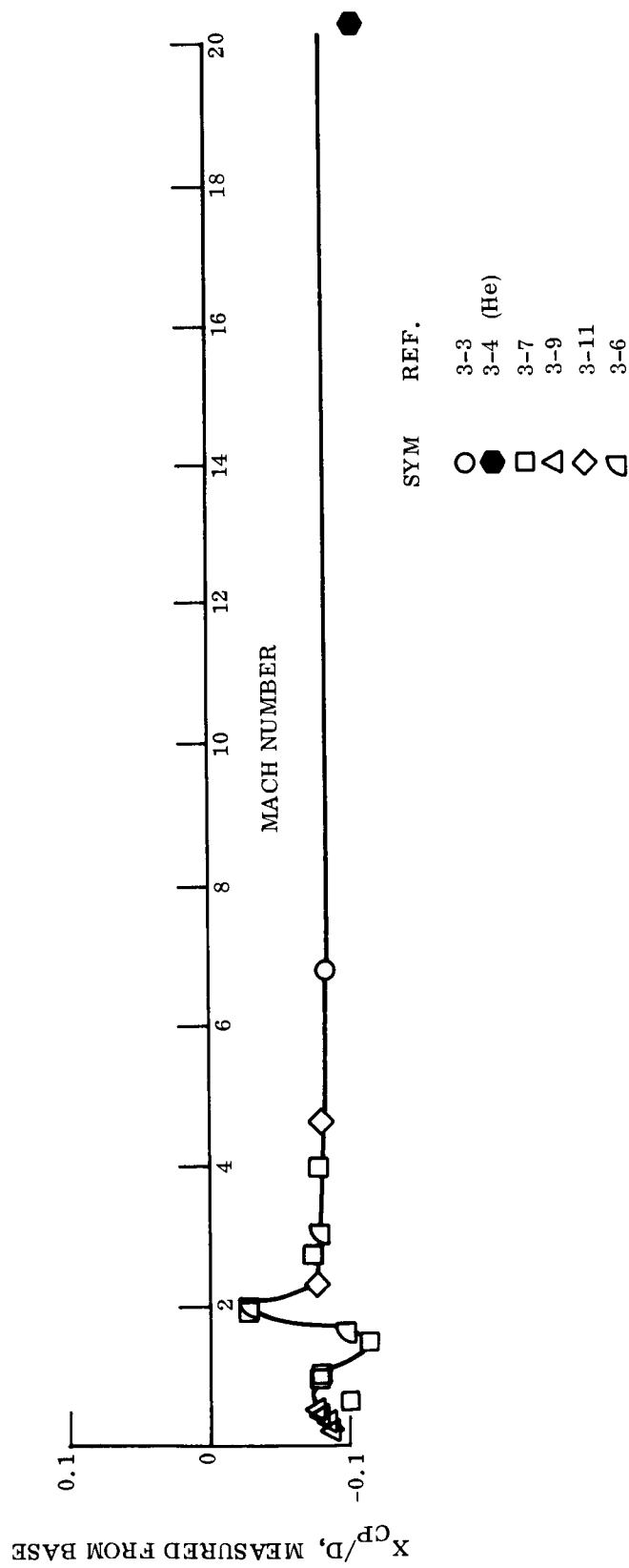


Figure 3.1.2-59. Center of Pressure vs Mach Number at Zero Angle-of-Attack, Cone Angle $(\theta) = 50^\circ$ and $R_N/R_B = 0$

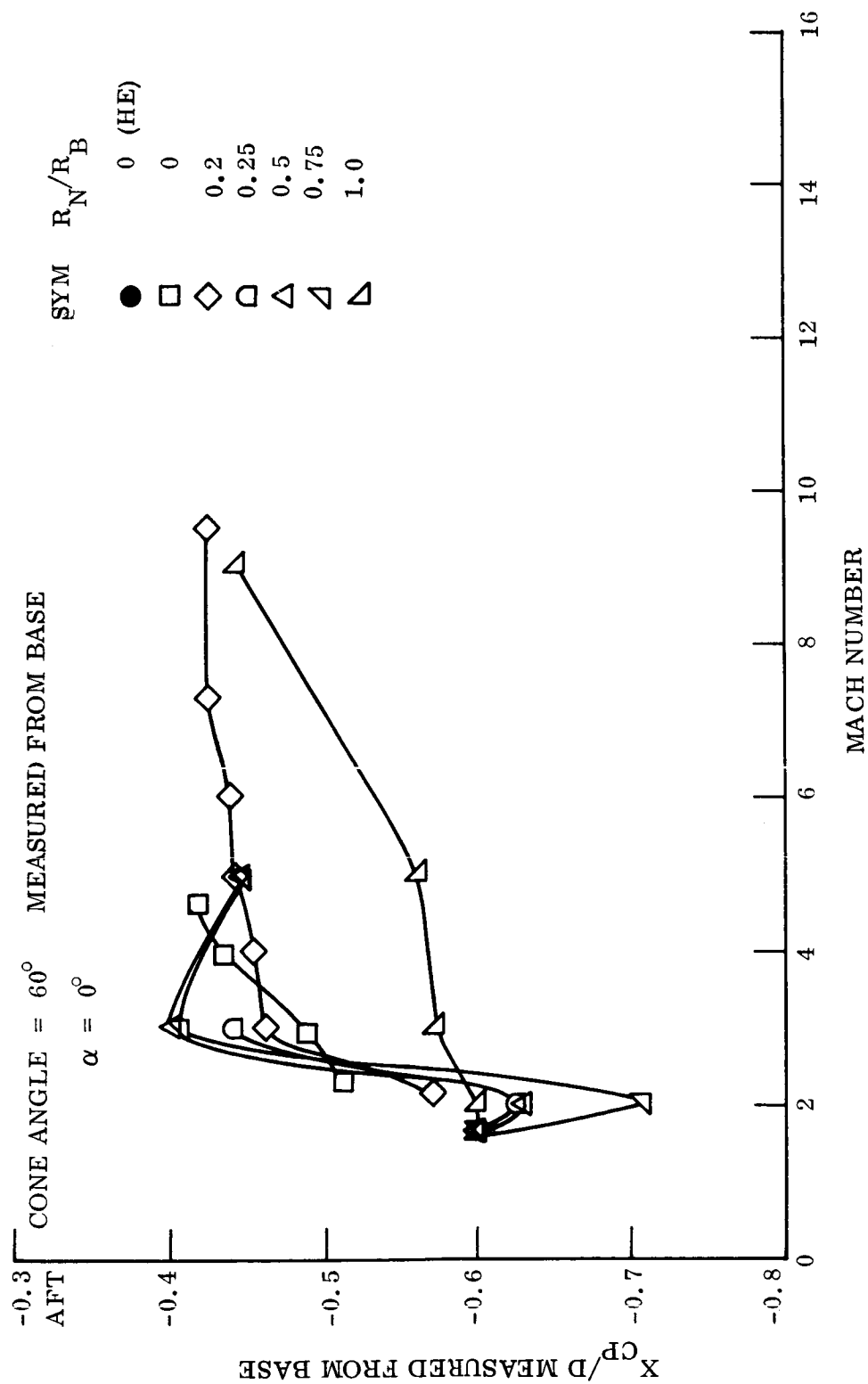


Figure 3.1.2-60. Effect of Bluntness on Center of Pressure Location with Mach Number for a 60° Cone

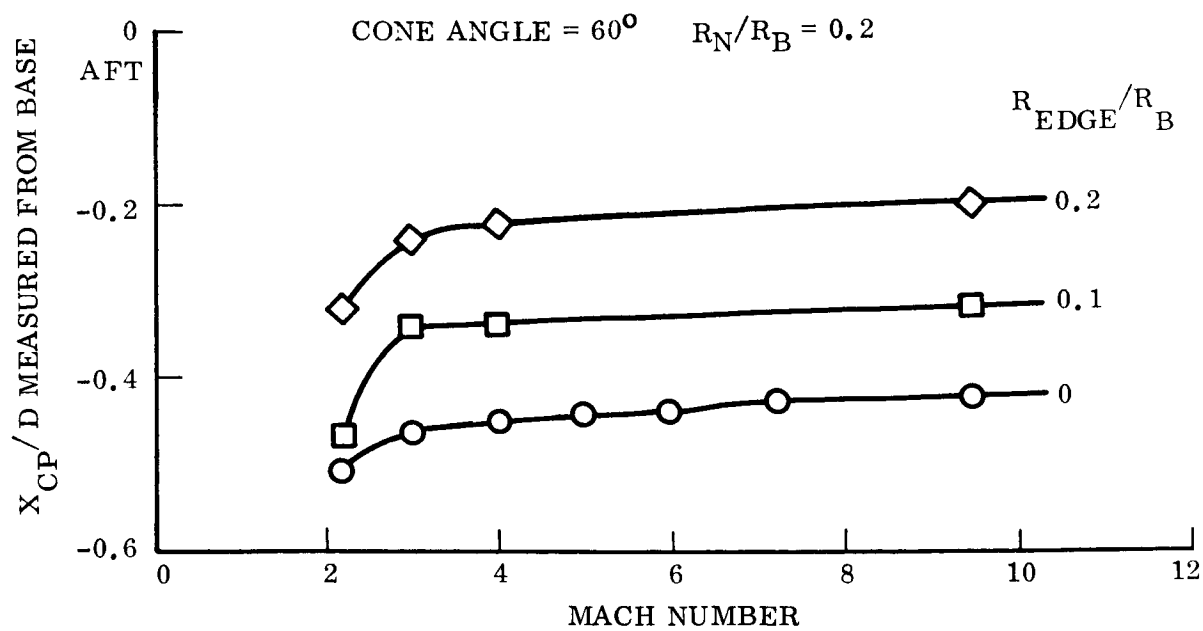


Figure 3.1.2-61. Effect of Rounded Edge on Center of Pressure Location with Mach Number for a 0.2 Blunted 60° Cone

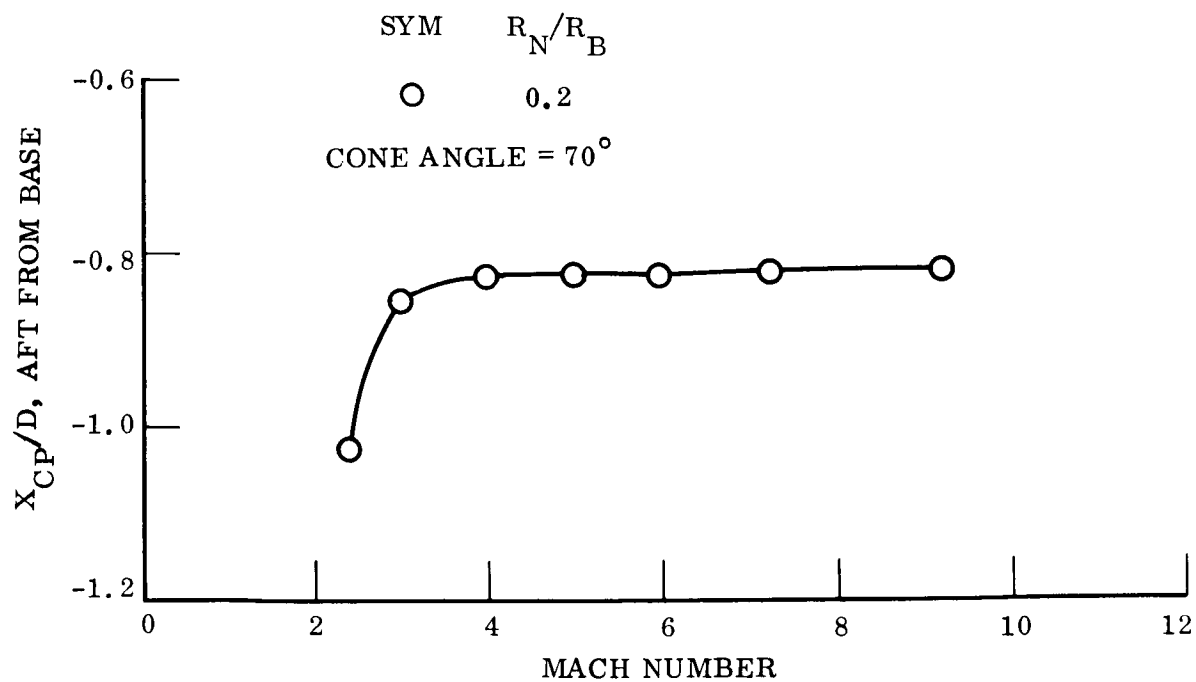


Figure 3.1.2-62. Center of Pressure vs Mach Number at Zero Angle-of-Attack for a 0.2 Blunted 70° Cone

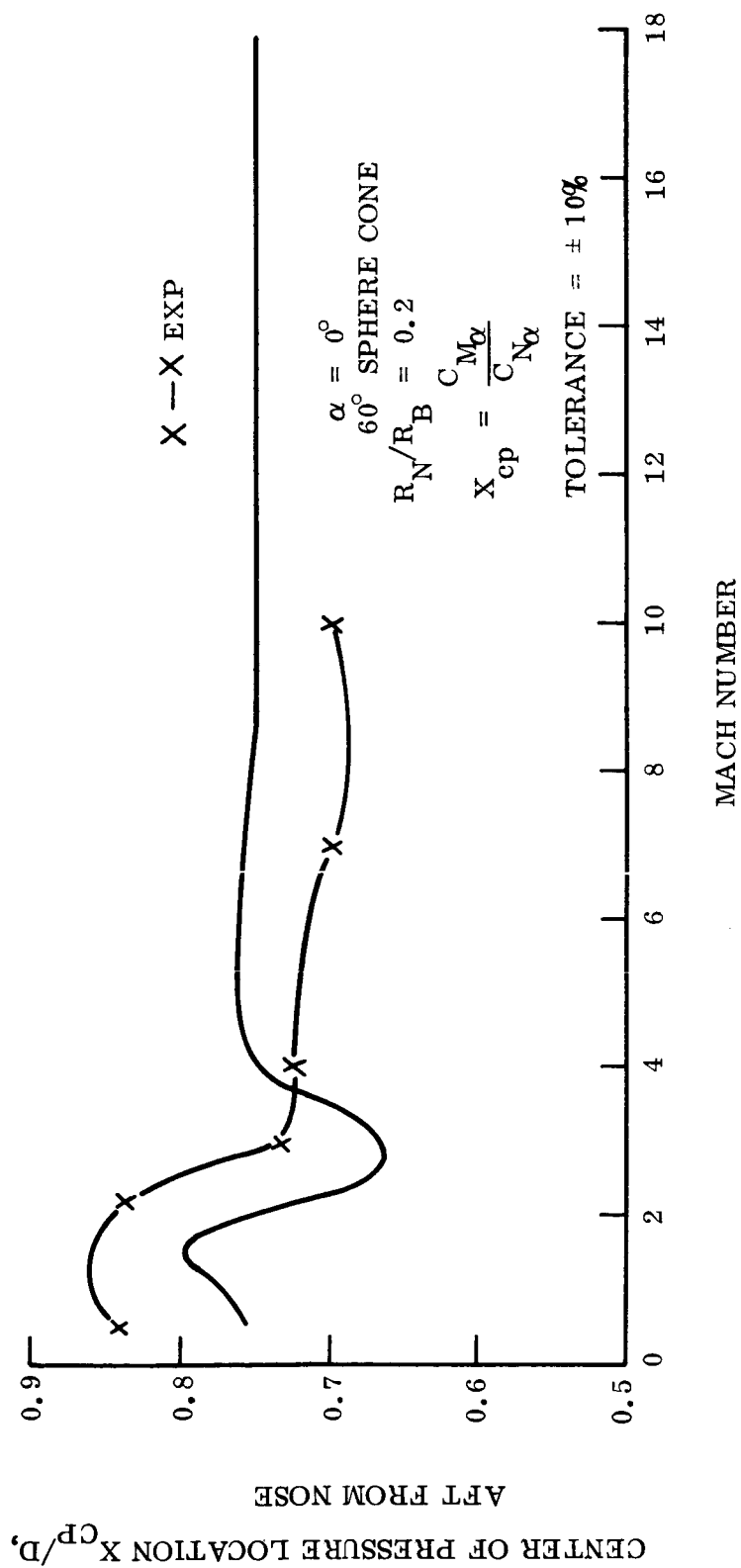


Figure 3.1.2-63. Variation of Center of Pressure with Mach Number for a 0.2 Blunted 60° Cone at Zero Angle-of-Attack

D. Dynamic Damping Coefficient - The dynamic damping coefficient is one of the most difficult aerodynamic characteristics to determine analytically or experimentally. The $\theta_c = 60^\circ$ estimates (fig. 3.1.2-64) are obtained from ref. 3-15 which made use of NASA AMES and NOL data. Fig. 3.1.2-65 presents dynamic data for various aft domes for a 60° blunted cone in air. Although the scatter is quite large, it appears that there is a definite decrease in stability for the large dome configuration at $\alpha = 0$. This same trend is also indicated from tests in freon (fig. 3.1.2-66) where the unstable region extends out to a Mach number of greater than 8. Fig. 3.1.2-67 presents a comparison of dynamic stability data for 60° and 70° cones in freon. While the centers of gravity in terms of base diameter are all different, the unflagged symbols have the same c.g. in terms of percent length. The 70° cone appears to have better damping characteristics. However, detailed and closely controlled tests are still required to obtain a better understanding of the dynamic damping characteristics of large cone angle configurations.

E. Aerodynamic Stability Criteria - The blunt configurations under consideration have good static margins and are generally well behaved in the early portions of atmospheric entry. The initial convergence of angle-of-attack is good even from very large displacement angles (up to 90° or more) though practical considerations such as communications or aft end heating will probably dictate an entry angle-of-attack limit nearer 40° or 50° . Past studies of blunt body dynamics have shown that the angle-of-attack convergence is only slightly affected by initial lateral rates, mass asymmetries, et cetera (ref. 3-17 for example).

After encountering peak dynamic pressure, when the density damping effect is removed, the blunt vehicles can display an unusual form of instability. This instability at zero angle-of-attack can be precipitated by spin and can exist even if the aerodynamic damping parameter C_{mq} is negative. Using linearized theory, a maximum spin rate for stability can be evaluated. The controlling equations are as follows (see ref. 3-17 for the details of the derivation):

$$\frac{m d^2}{8 I_y} C_{mq} - 1/4 C_{L\alpha} < 0 \quad (1)$$

$$P < \left(\frac{m d^2}{I_x} C_{mq} - 2 C_{L\alpha} \right) \left(\frac{I_y}{I_x} \right) \sqrt{\frac{-S q_\infty C_{m\alpha}}{2 m d C_{mq} C_{L\alpha}}} \quad (2)$$

where:

m	= vehicle mass	q_∞	= dynamic pressure
I_y	= pitch moment of inertia	S	= reference area $\frac{(\pi d^2)}{4}$
I_x	= roll moment of inertia	P	= spin rate in radians per second
d	= reference length (diameter)		

Aerodynamic derivatives are per radian

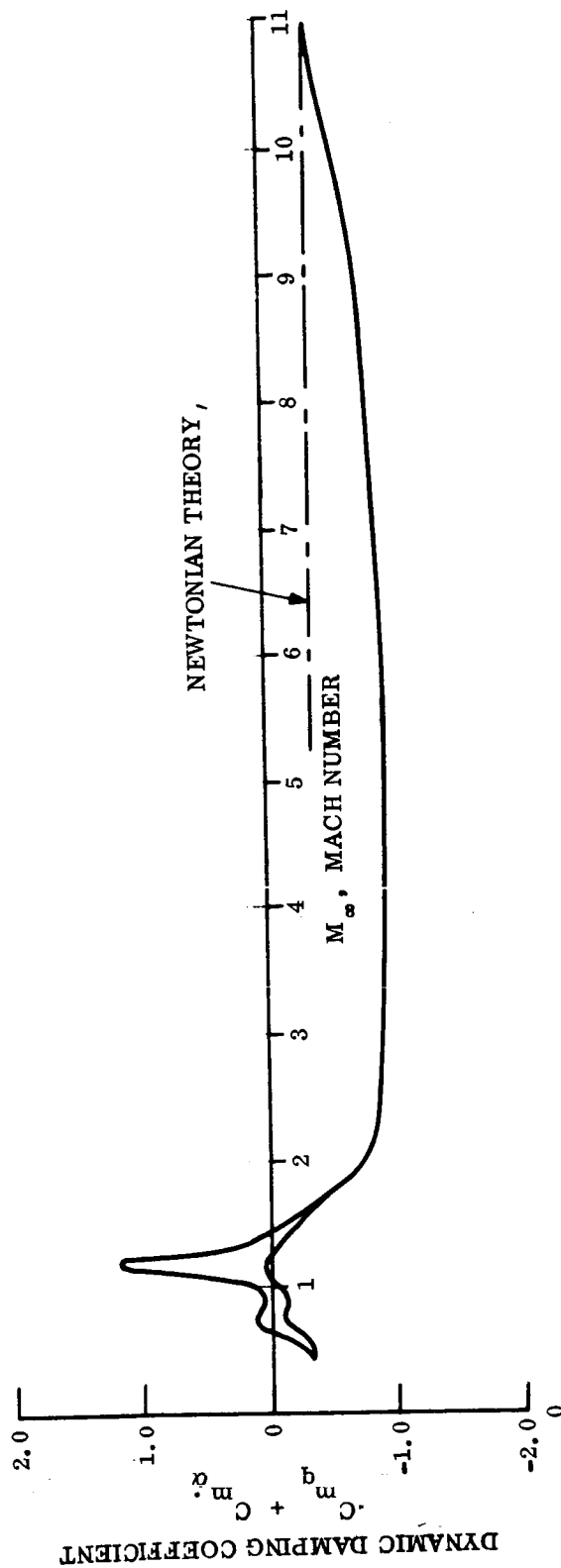


Figure 3.1.2-64. Variation of Dynamic Damping Coefficient with Mach Number and Angle-of-Attack

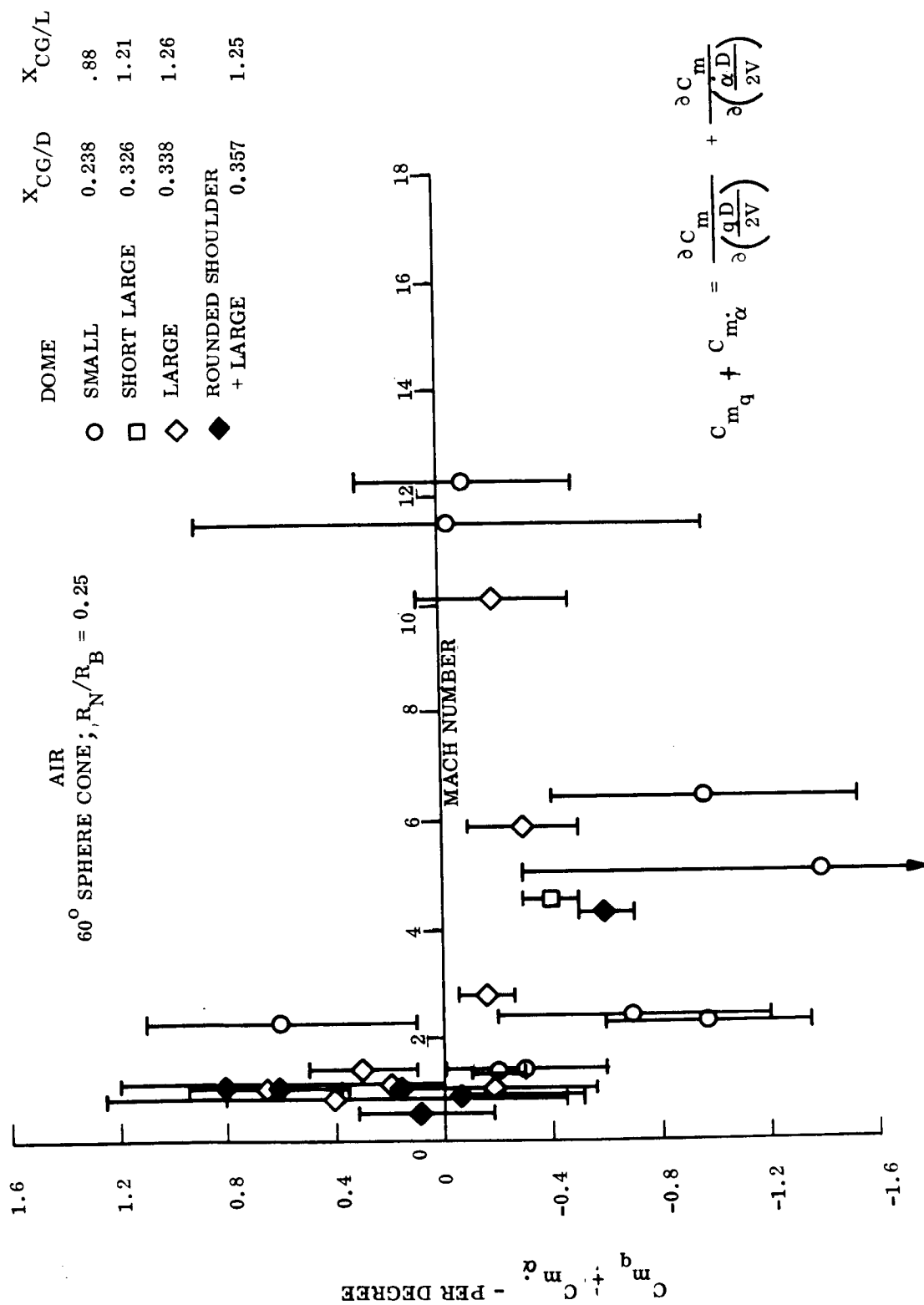


Figure 3.1.2-65. Dynamic Stability vs Mach Number for a 0.25 Blunted 60° Cone in Air

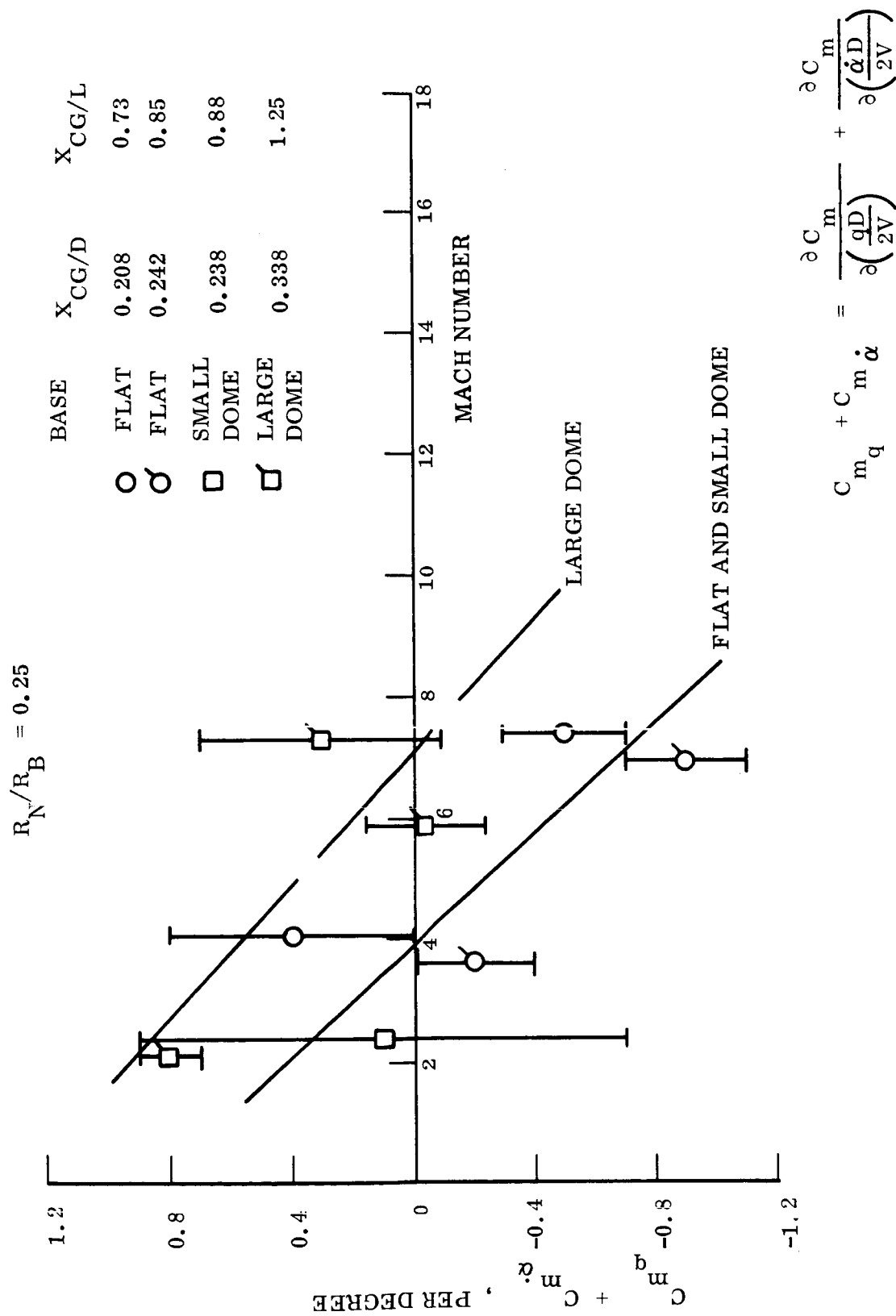


Figure 3.1.2-66. Effect of Afterbody on Dynamic Stability vs Mach Number for a 0.25 Blunted 60° Cone in Freon

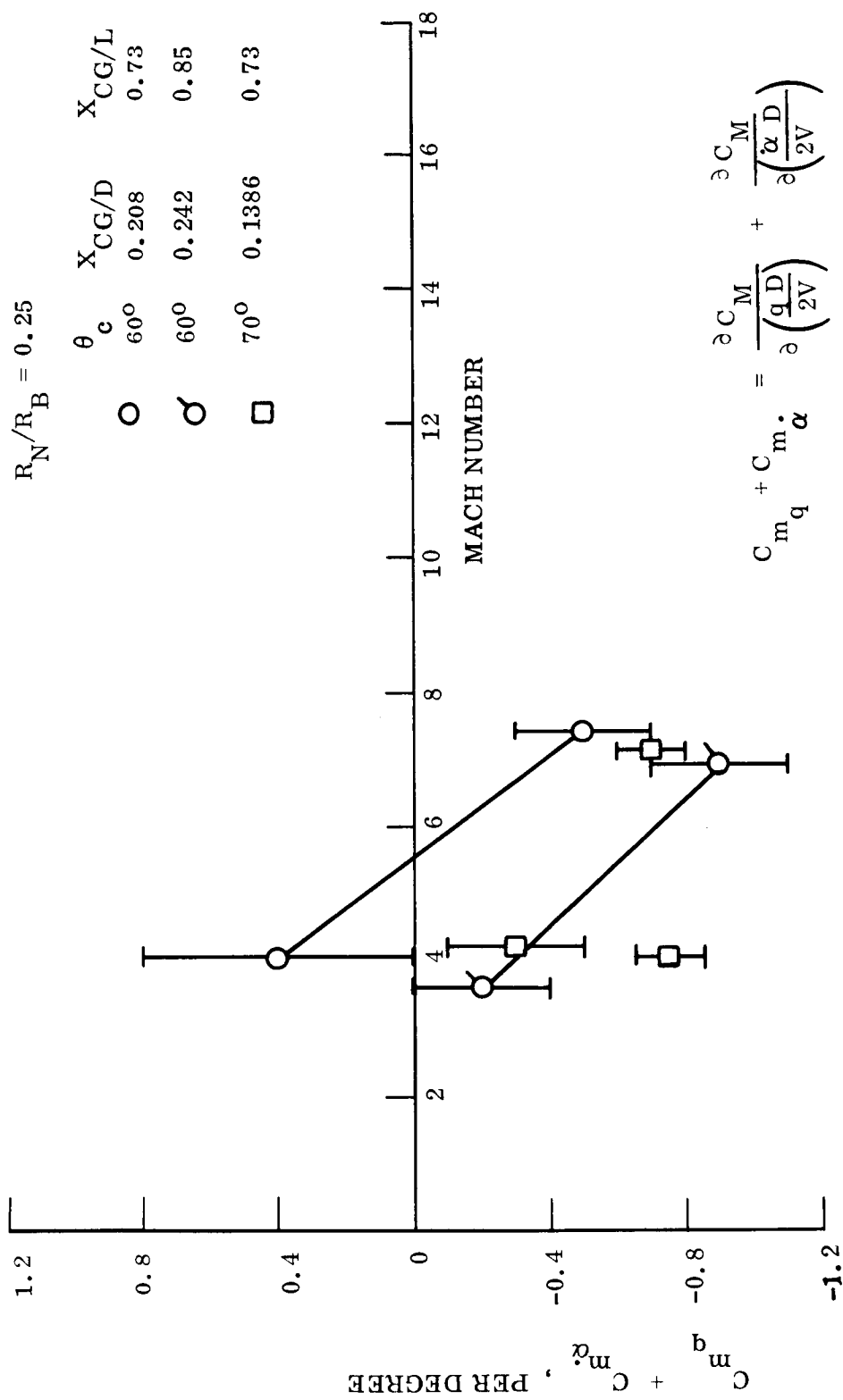


Figure 3.1.2-67. Dynamic Stability vs Mach Number for Flat Based 0.25 Blunted Cones in Freon

The effects of the ensuing divergence may or may not be significant, depending on the specific mission objectives and vehicle design. Particular items that would be affected are base and stagnation point pressure and temperature measurements, radar altimeter measurements, effective vehicle ballistic coefficient (which increases with angle-of-attack), and communication viewing angles.

The type of divergence to be expected if the limit roll rate is exceeded is shown in fig. 3.1.2-68. These curves show the initial convergence that would result as dynamic pressure increases with the final convergence or divergence depending on configuration, etc. Note the strong dependence on the aerodynamic damping parameter (C_{m_q}) which is always an elusive value to pin down. The degree and rapidity of divergence are dependent on the amount by which the limit value of roll rate is exceeded.

The equations just cited have been programmed for the IBM 1130 computer to enable the rapid generation of spin limits on a parametric basis. The curves described next have been reduced to the significant parameter combinations. The data are presented as a function of vehicle half cone angle, ignoring the very slight effect of configuration bluntness ratio (i.e., the effect of bluntness ratio on the aerodynamic parameters C_{L_α} , C_{m_q} , etc).

Figs. 3.1.2-69 through -74 are to be used to determine the limit spin rate for blunt body sphere cones in terminal descent. Each figure is for a specific diameter (from 7 to 14.5 ft) and presents the limit spin rate as a function of half cone angle and the parameter $\frac{m}{I_y}$ (mass/pitch inertia). To determine the limit spin rate for a specific case, the first step is to determine m/I_y (m in slugs, I_y in slug-ft²). From the proper figure (based on diameter), the limit spin rate can be found for half cone angles from 45° to 60°. Figs. 3.1.2-69 through -74 are based on $I_y/I_x = 1$. For I_y/I_x not equal to 1, the resulting spin rate must be scaled. This is accomplished by multiplying the spin rate by the actual value of I_y/I_x . This procedure will be illustrated by an example at the end of this section.

To obtain a limit spin rate at Mach 2, first obtain the limit spin rate in terminal descent as just described. Then, from fig. 3.1.2-75 obtain the proper scaling factor based on path angle. The limit spin rate at Mach 2 is the product of the limit spin rate in terminal descent and the scaling factor. The scaling factor N , is the acceleration in local g's determined at Mach 2. This procedure is illustrated as follows:

Example

Point Design 1 is used to illustrate the outlined procedures.

$$d = 8.5 \text{ ft}$$

$$m = 40 \text{ slugs}$$

$$I_y = 190 \text{ slug-ft}^2$$

$$I_x = 250 \text{ slug-ft}^2$$

$$\text{Half cone angle} = 60^\circ$$

$$\text{Path angle} = 16^\circ$$

I. Terminal Descent

$$\frac{m}{I_y} = \frac{40}{190} \approx 0.2$$

$$\frac{I_y}{I_x} = \frac{190}{250} \approx 0.8$$

Referring to fig. 3.1.2-70 for $d = 8.5$ ft, it can be seen that the limit spin rate (for $I_y/I_x = 1$) is 8 rpm. However, it is necessary to scale this spin rate for $I_y/I_x = 0.8$. Therefore, the limit spin rate is $8 \times 0.8 = 6.4$ rpm.

II. Mach 2

From I, the limit spin rate for terminal descent is 6.4 rpm. From fig. 3.1.2-75, the scaling factor is 1.6. Therefore, the limit spin rate at Mach 2 is 6.4×1.6 rpm = 10.2 rpm.

If this $M < 2$ instability must be avoided, at least a spin rate control system is required to ensure that the anticipated (not desired) roll torques developed during entry do not spin the vehicle above the limit rate. Further discussion of the roll torque problem is included in the following paragraphs.

F. Aerodynamic Induced Control Requirements - The control requirements associated with aerodynamic effects during entry are those of instability at zero angle-of-attack in the regime of decreasing dynamic pressures. A control system is necessary to hold the spin rate down for those designs which have a possibility of spinning up above the limit rate.

There are two prime sources of roll torques during entry. The first is the product of normal force and an effective center of mass (c.m.) offset (distance between the c.m. and the true aerodynamic centerline). This effect averages out in the ideal case; at one point in the motion cycle accelerating the vehicle in roll, at another decelerating it. However, in the actual case a slight surface asymmetry can result in a trim normal force fixed at a particular body direction, and if this force is perpendicular (or has a component perpendicular) to a body c.m. offset, a persistent trim torque is developed. Because this torque is associated with a trim angle-of-attack, it is commonly referred to as the $C_2 \alpha_T$ torque. The second source is the pure roll torque coefficient (C_l) which results primarily from heat shield surface irregularities developed during entry. Many persons are currently studying the phenomenology associated with the development of the roll torques, but no reliable technique is yet available to predict their magnitude.

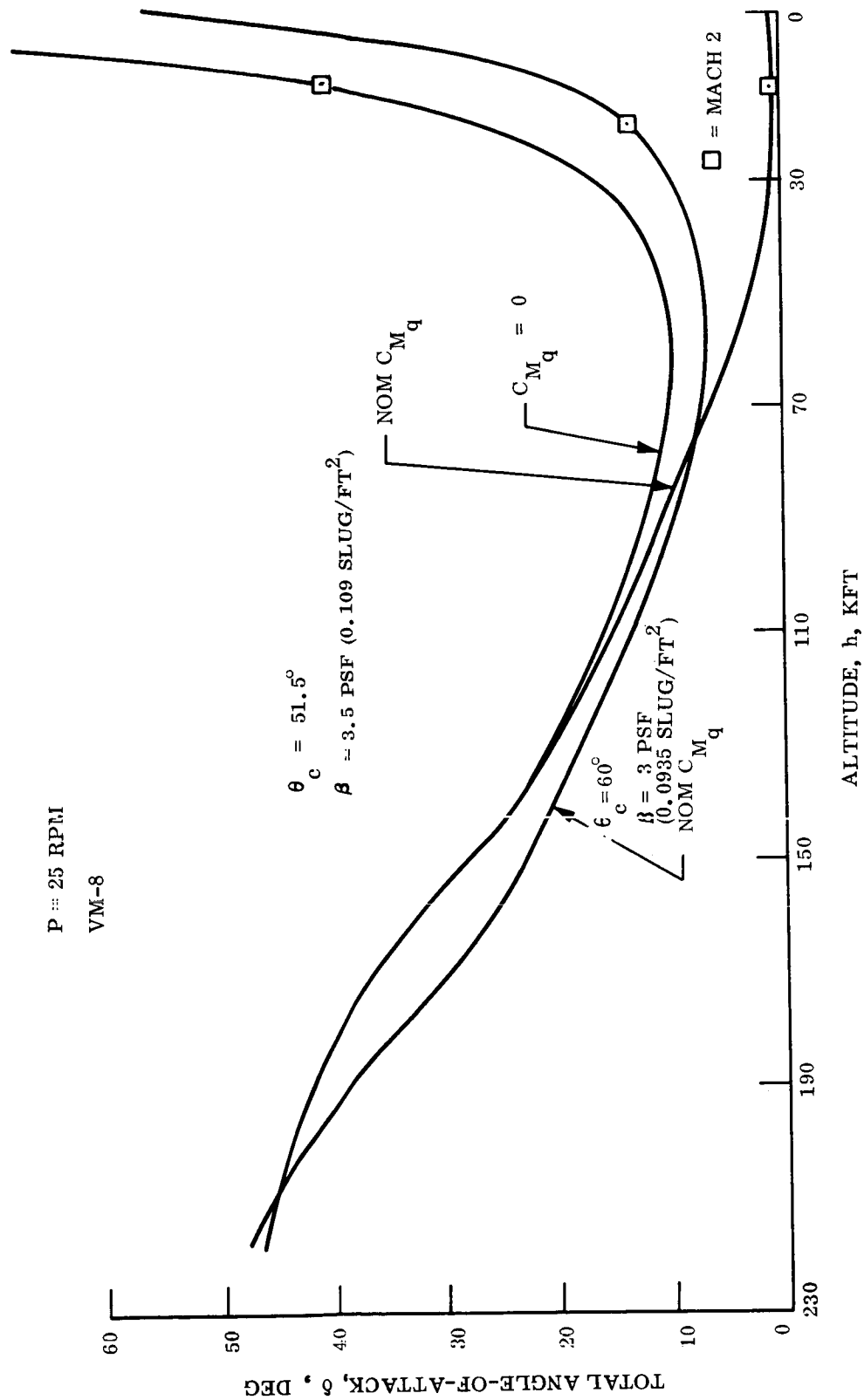


Figure 3.1.2-68. Typical Angle-of-Attack Time Histories

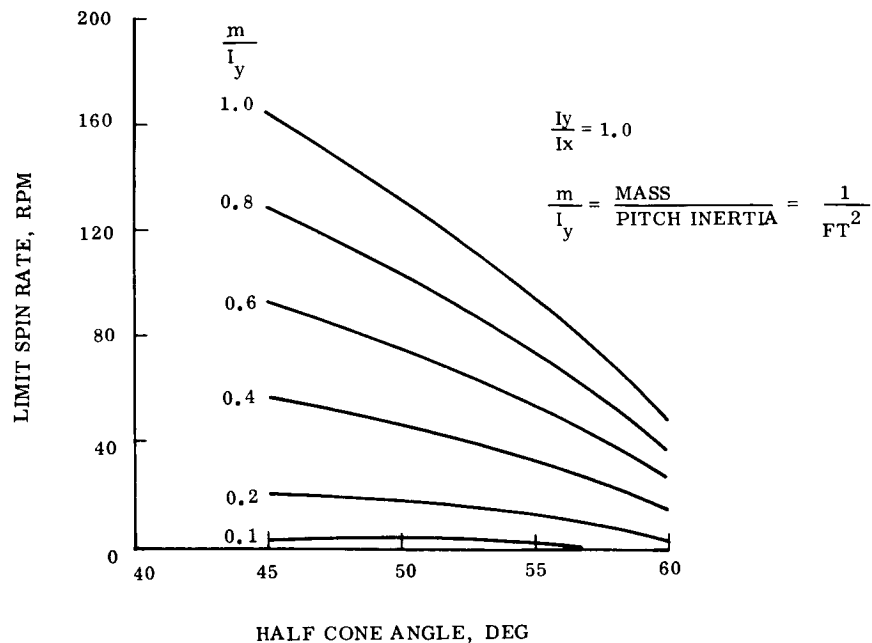


Figure 3.1.2-69. Limit Spin Rate for 7 ft Diameter Blunt Body Sphere Cones in Terminal Descent

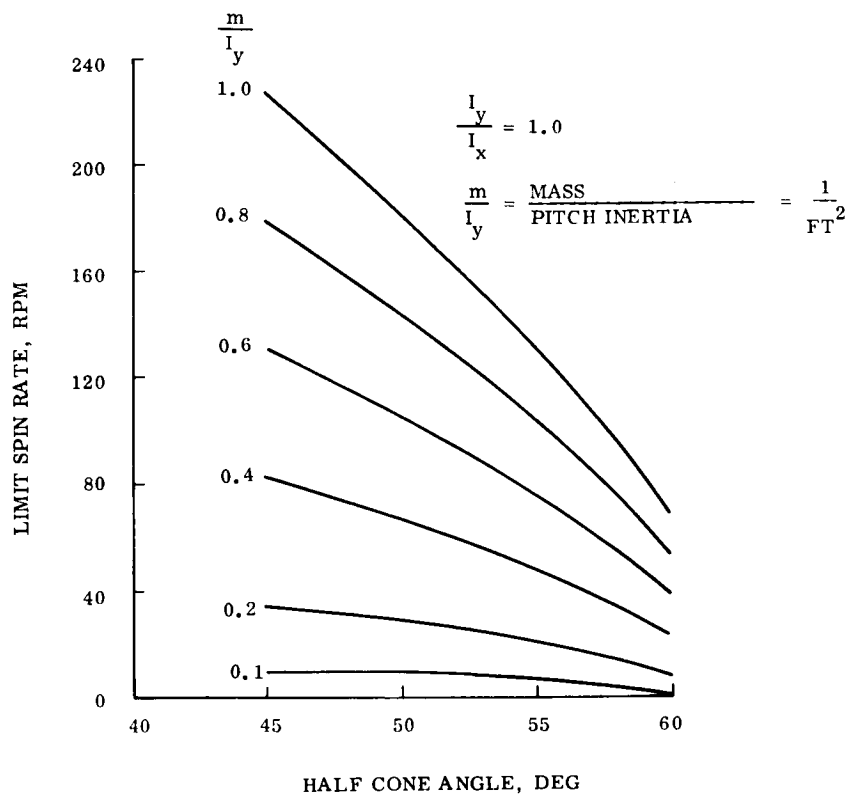


Figure 3.1.2-70. Limit Spin Rate for 8.5 ft Diameter Blunt Body Sphere Cones in Terminal Descent

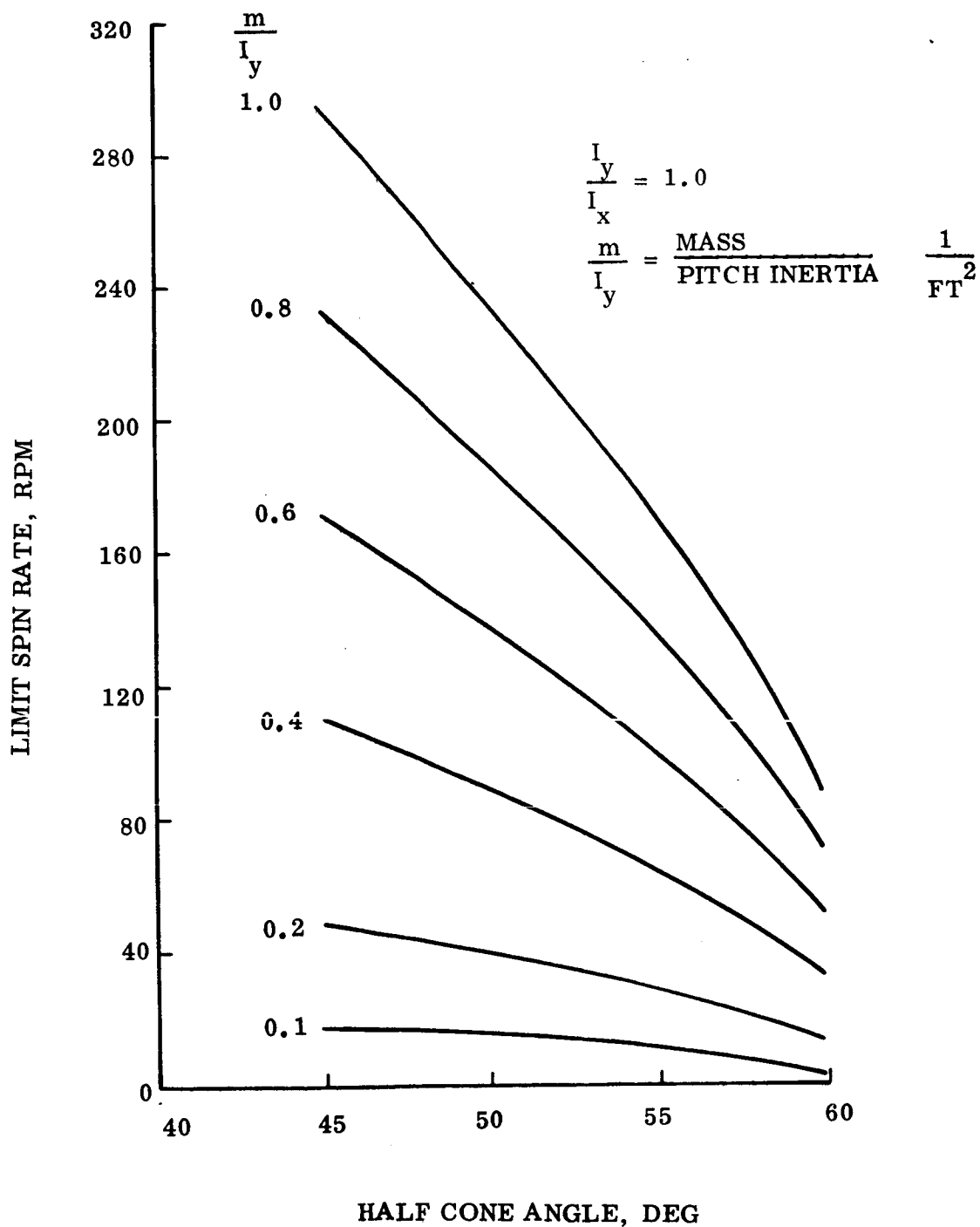


Figure 3.1.2-71. Limit Spin Rate for 10 ft Diameter Blunt Body Sphere Cones in Terminal Descent

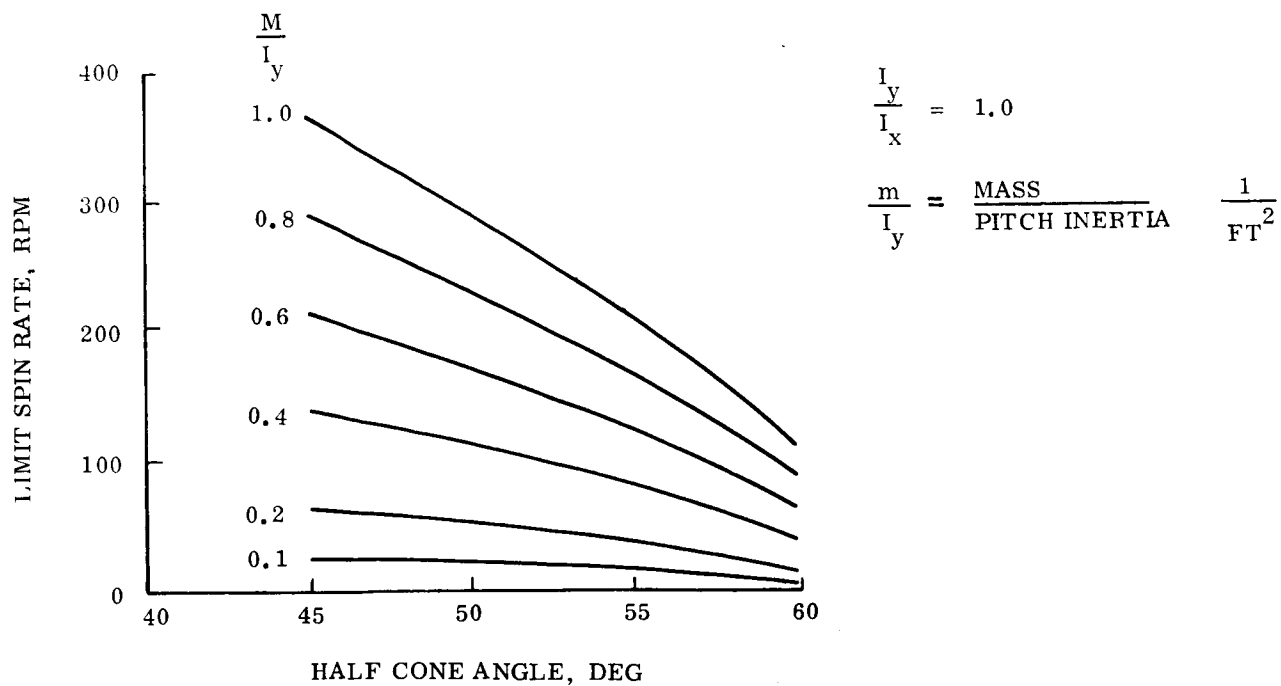


Figure 3.1.2-72. Limit Spin Rate for 11.5 ft Diameter Blunt Body Sphere Cones in Terminal Descent

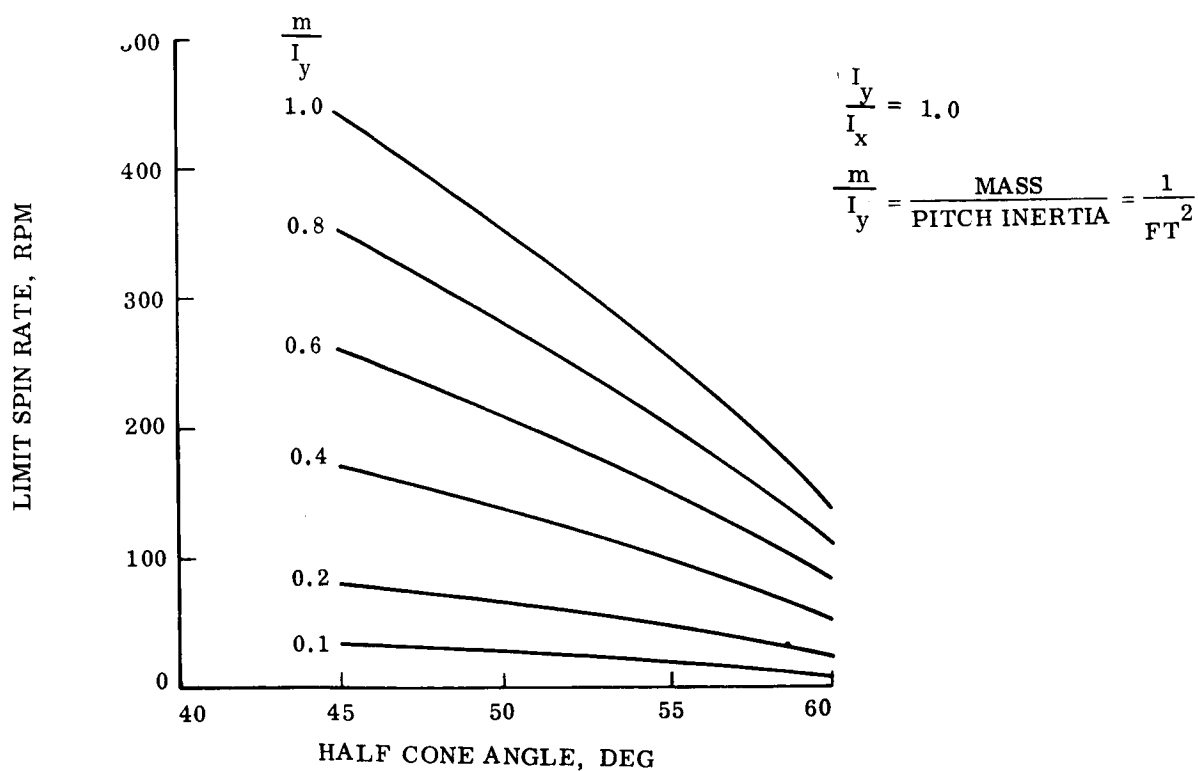
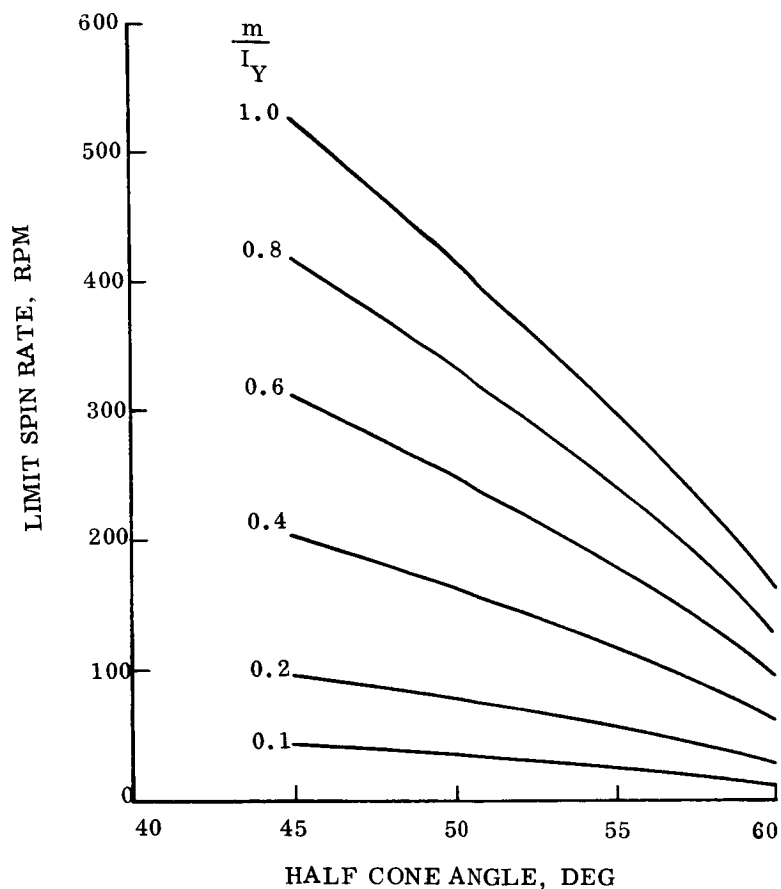


Figure 3.1.2-73. Limit Spin Rate for 13.0 ft Diameter Blunt Body Sphere Cones in Terminal Descent



$$\frac{I_y}{I_x} = 1.0$$

$$\frac{m}{I_y} = \frac{\text{MASS}}{\text{PITCH INERTIA}} = \frac{1}{\text{FT}^2}$$

Figure 3.1.2-74. Limit Spin Rate for 14.5 ft Diameter Blunt Body Sphere Cones in Terminal Descent

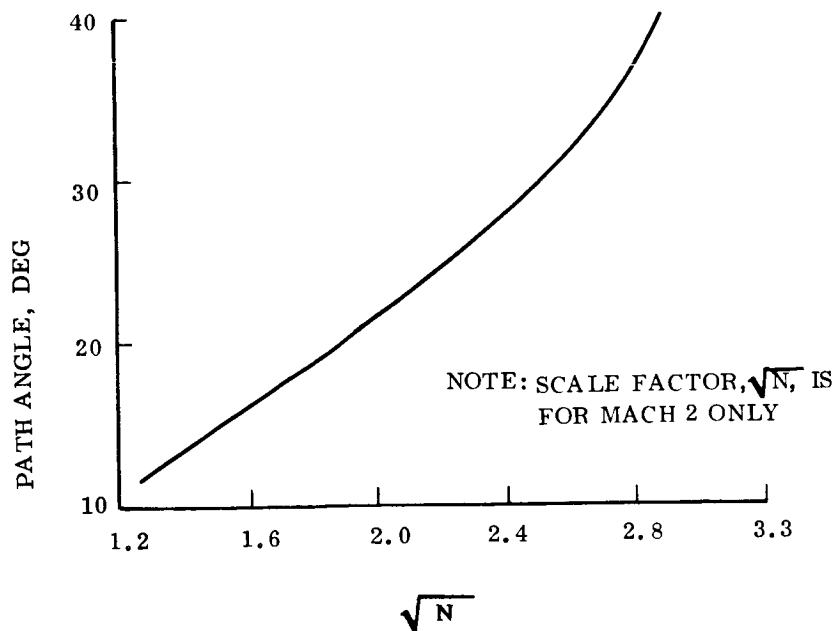


Figure 3.1.2-75. Scaling Factors for Limit Spin Rate at Mach 2

What are felt to be conservative estimates of the torque coefficient levels have been made for the cited sources. The pure roll torque coefficient (C_l) is estimated at 1.0×10^{-4} , which is about three times the value normally used for sharp slender cone vehicles (where flight data is available). The $C_2 \alpha_T$ value has been estimated at 0.25 inch-degrees.

Using these estimates, the torques are worked out as follows:

$$T_1 = C_{N_\alpha} \alpha_T C_2 q_\infty S \text{ (ft-lb)}$$

$$T_2 = C_l q_\infty S d \text{ (ft-lb)}$$

where:

C_{N_α} = normal force coefficient slope (0.007 per deg for 60° S-C)

q_∞ = dynamic pressure (PSF)

S = vehicle base area $\frac{\pi d^2}{4}$ (ft^2)

d = vehicle diameter, ft

The total torque is then the sum of T_1 and T_2 .

Putting in the values for $C_2 \alpha_T$, C_{N_α} , etc, the roll torque is $T = T_1 + T_2 = 0.0001 q d^2 + 0.00008 q d^3$. The torque coefficients are then a function of the vehicle diameter as follows:

<u>Diameter</u> (ft)	<u>K</u>
7	0.035
10	0.095
13	0.200
15	0.295

Then the torques in foot-pounds are:

$$T = Kq \text{ (ft-lb)}$$

and the time integral of torque is:

$$T \int dt = K \int q dt \text{ (ft-lb-sec)}$$

Parametric data on the peak dynamic pressure (q) and integrated pressure ($\int q dt$) is presented in the earlier "Trajectory Performance Trade-offs" portion of this section. Particular data for the torque levels and integrated torque values (needed to determine possible spin up and/or control system gas consumption) are shown in figs. 3.1.2-76 through -79 for the purpose of orienting the reader to the actual torque levels.

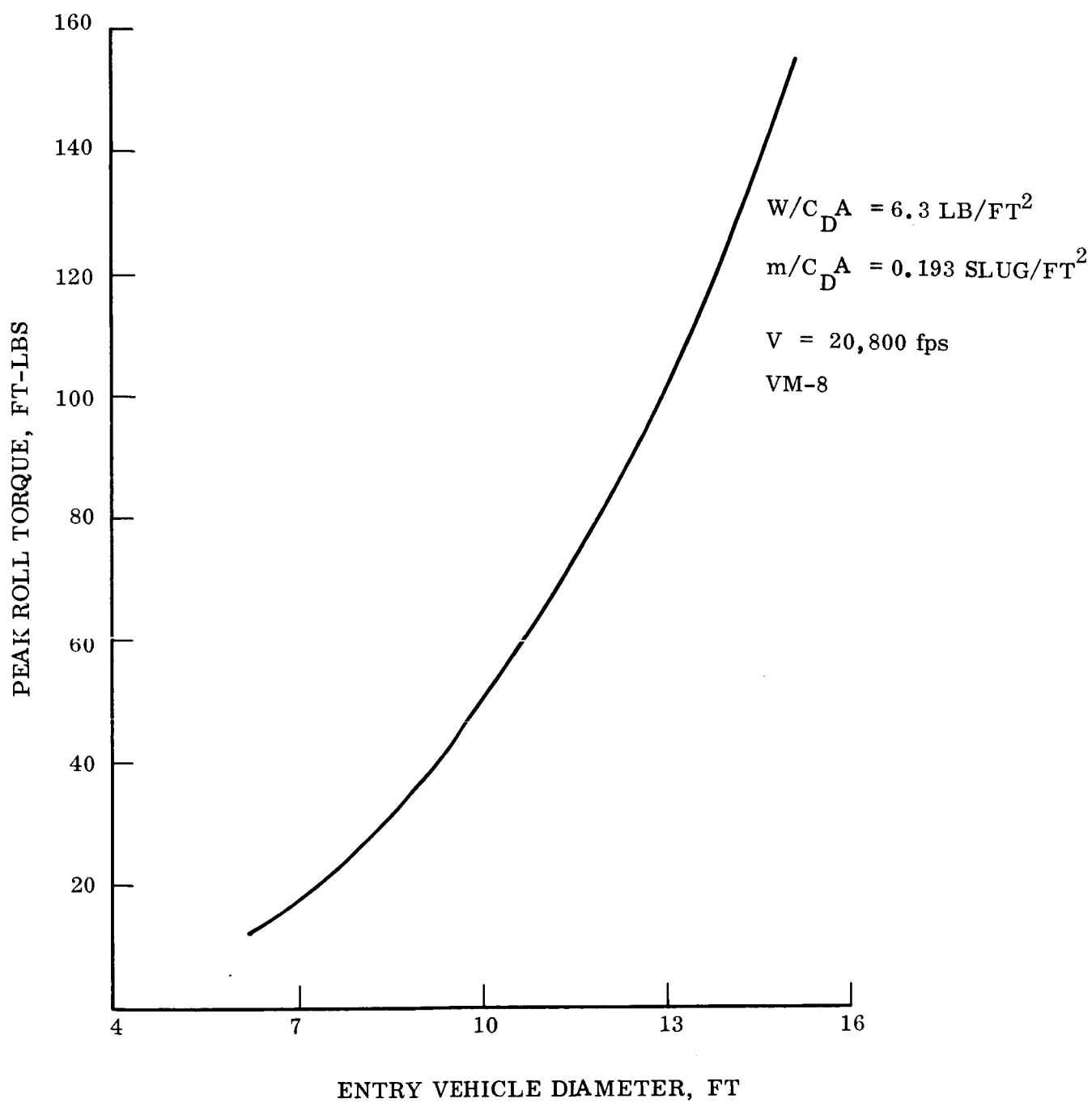


Figure 3.1.2-76. Peak Roll Torques for Direct Entry $\gamma_E = 40^\circ$

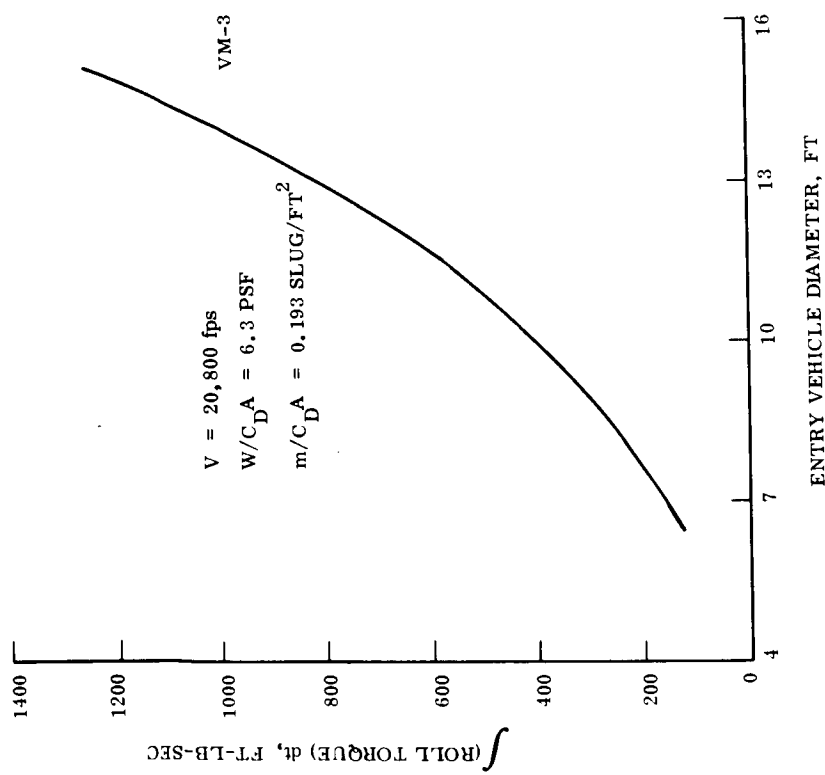


Figure 3.1.2-77. Integrated Roll Torque for Direct Entry $\gamma = 40^\circ$ and $\beta = 0$.

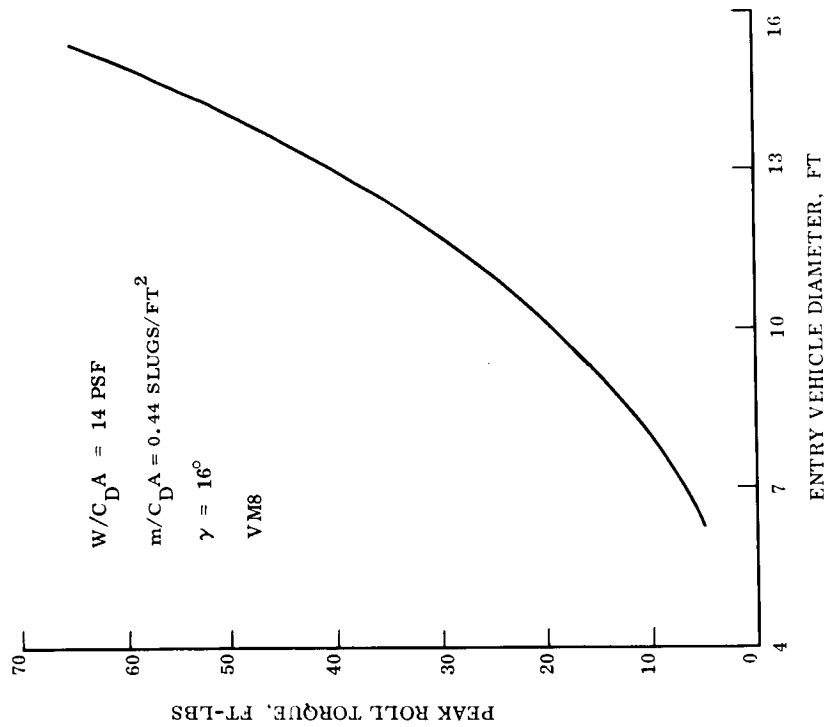


Figure 3.1.2-78. Peak Roll Torques for Out-of-Orbit Entry, $\beta = 0.44 \text{ Slug/ft}^2$

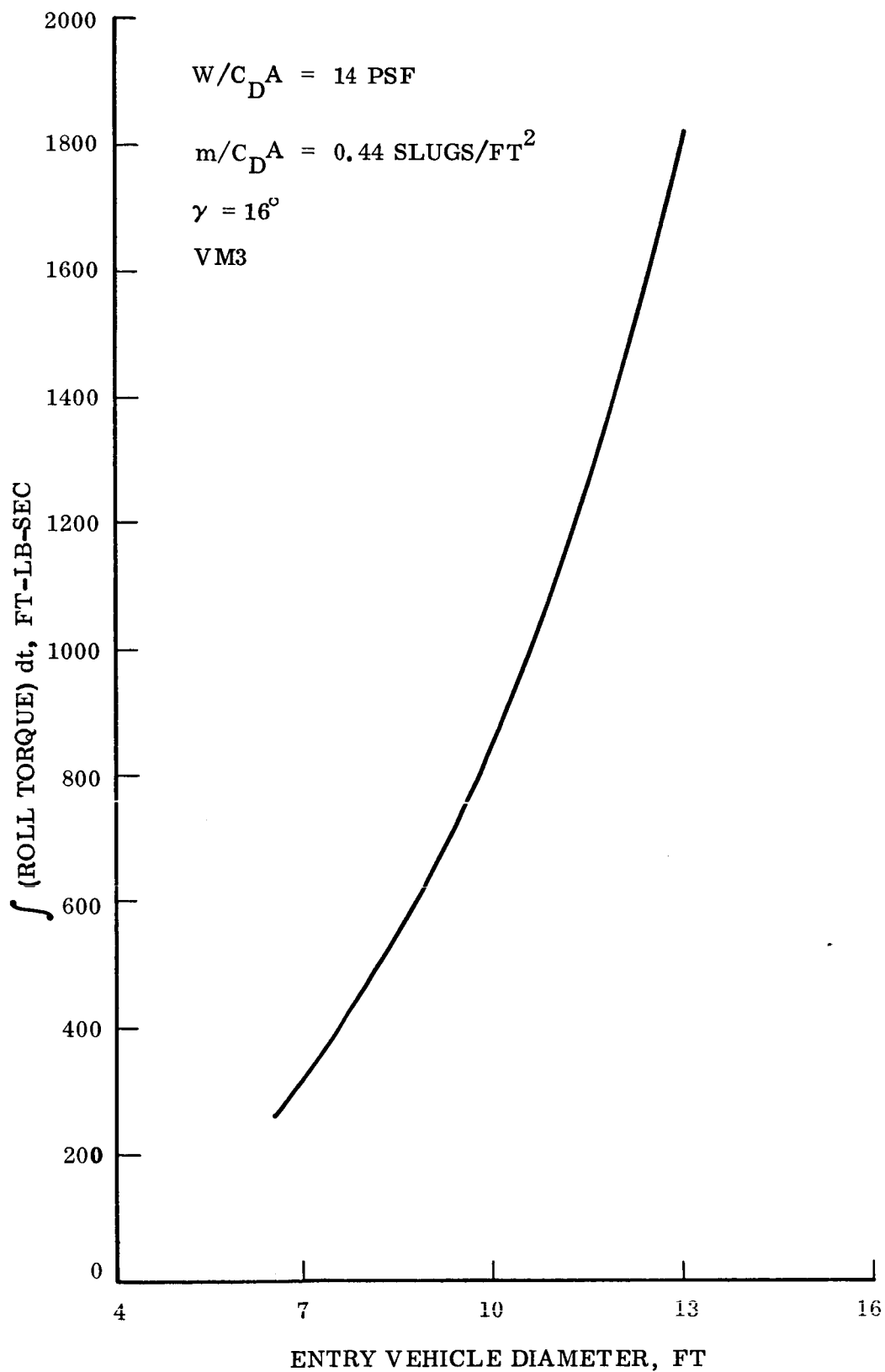


Figure 3.1.2-79. Integrated Roll Torques for Out-of-Orbit Entry, $\beta = 0.44 \text{ Slug/ft}^2$

3.1.2.1.3 Heat Shield Performance

A. Heating Environment

Effect of Atmosphere Model on Convective Heating - The amount of heat protection material required to maintain the structure at an acceptable temperature level is determined by the amount of convective and radiative heat transfer received by the vehicle during the Mars entry and by the thermal performance of the shield material exposed to the heating environment. The equilibrium aerodynamic heating to the capsule is calculated using the Planetary Aerodynamic Heating Program which is described in table A-1 of Appendix A and discussed in this section. The equilibrium radiative heating is calculated using the Hot Gas Radiation Program (HGR). The heat shield material response is calculated using the REaction Kinetics Ablation Program (REKAP). These programs are also summarized in table A-1 of Appendix A. For preliminary design trade-offs employing CONSEP, the comprehensive programs summarized in that table have been employed to generate simplified parametric results of aerodynamic and radiative heating and material response.

The two atmospheric characteristics that normally control the magnitude of the hypersonic entry heating environment are gaseous composition and effective density scale height, H . The gaseous composition and density scale heights for the various model Martian atmospheres are summarized in table 3.1.2-4. These include the minimum, mean, and maximum Mars engineering model promoted by NASA/LRC for mission and design studies.

TABLE 3.1.2-4. MODEL MARTIAN ATMOSPHERES

Model Atmosphere	Density Scale Height for Hypersonic Heating, (Ft)	Composition (% by mass)			Constant K (Ref 3-19)	K (H) ^{1/2}	K/H ^{1/2}
		CO ₂	N ₂	Argon			
VM-3	49900	28.2	71.8	0	2.987X10 ⁻⁹	6.67X10 ⁻⁷	134X10 ⁻¹¹
VM-8	16600	100	0	0	3.825X10 ⁻⁹	4.93X10 ⁻⁷	2.97X10 ⁻¹¹
LRC Minimum	23000	100	0	0	3.838X10 ⁻⁹	5.82X10 ⁻⁷	2.53X10 ⁻¹¹
LRC Mean	34000	74.4	12.8	12.8	3.603X10 ⁻⁹	6.65X10 ⁻⁷	1.95X10 ⁻¹¹
LRC Maximum	39000	25	50	25	3.144X10 ⁻⁹	6.22X10 ⁻⁷	1.59X10 ⁻¹¹

A study has been made (ref. 3-18) to ascertain the accuracy of the existing stagnation point heat transfer theories when applied to planetary atmospheres. Figs. 3.1.2-80 through -82 show comparisons between experimental data obtained from ground tests and several existing theories. The figures present $\dot{q}_s \sqrt{R_B/P_O}$ as a function of the enthalpy difference (h_o-h_w) for cold wall conditions. The method of

computing the stagnation point heat transfer in the Planetary Aerodynamic Heating Program (PAHP) gives essentially the same result as the theory of Scala and Gilbert (ref. 3-19) in air, using the constant $K_1 = 1.1$ in the heat flux equation, table A-1 of Appendix A. Fig. 3.1.2-80 shows data for the mixture 9 percent CO_2 - 90 percent N_2 - 1 percent A at enthalpies of between 10,000 and 15,000 Btu/lb and indicates that ref. 3-19 underestimates the stagnation point convective heat transfer. Also shown on the figure are results obtained using the PAHP with $K_1 = 1.1$ for a typical entry into the VM-3 atmosphere, which has a higher CO_2 content than the gas mixture assumed in the theories. Consequently, in this figure the PAHP results are high compared with theory as would be expected. Fig. 3.1.2-81 shows ground test data for 100 percent CO_2 which simulates the VM-8 atmosphere. The theoretical predictions for this atmosphere appear to give reasonable results over the enthalpy range shown. Thus, using the factor $K_1 = 1.1$ in the PAHP would give accurate estimates of the heat transfer in this atmosphere. The data shown in fig. 3.1.2-82 indicates that for an atmosphere containing large amounts of the monatomic gas argon, the existing theories underestimate the stagnation point heat transfer by (in some cases) a large percentage, particularly for the pure argon data. Thus, it is apparent that the factor K_1 in the PAHP should be increased from the assumed value of 1.1 to account for entry into argon rich atmospheres.

The procedure used for the calculation of the laminar convective heating distribution over a body is Lee's relation (ref. 3-32) modified by Eckert's reference enthalpy techniques. For turbulent boundary layers, a relationship is derived in ref. 3-33, which satisfies both the momentum and internal energy equations and includes the effect of a finite pressure gradient. The procedures for the calculation of the convective heat transfer in air have been verified by flight test data and by extensive hypersonic wind tunnel tests.

Results presented in ref. 3-34 indicate that air theories adequately predict the turbulent heat transfer results for a CO_2 atmosphere.

Having specified the trajectory, the vehicle's aerodynamic configuration and its pressure distribution, the convective heating to the vehicle is determined as a function of station and time. The program calculates all of the boundary layer edge conditions; such as, velocity, temperature, viscosity, boundary layer thickness, displacement thickness, local Mach number and Reynolds number. The normal procedure is to calculate both laminar and turbulent boundary layer conditions and then, depending on the Reynolds number, select which condition as a function of time or altitude is the actual condition of the boundary layer. Transition from laminar to turbulent flow has been found to be a function of many variables including Reynolds number, local Mach number, angle of attack, mass addition and wall temperature. The transition criteria are discussed fully in the following paragraphs.

Parametric Variation of Convective Heating and CONSEP Approach - An analysis has been performed, and is discussed in this section, showing that the approximate entry analysis of Allen and Eggers, (ref. 3-35) can be employed to estimate the stagnation point cold wall heating if the constants as listed in table 3.1.2-5 are used in the isothermal atmosphere model (where $\rho = \rho_{\text{REF}} e^{-h/H}$). Use of these isothermal atmosphere models greatly simplifies the preliminary trade-off studies and this approach has been included in the CONSEP program for parametric studies.

A series of PAHP results have been correlated by utilizing the approximate entry analysis approach of Allen and Eggers, ref. 3-35. The maximum stagnation point heat flux and time integrated heat flux are illustrated in figs. 3.1.2-83 through -86 for entry into the VM-3 and VM-8 atmospheres. In this approximate analysis, the maximum value of the stagnation point convective heat flux is given as

$$\left[\dot{q}_{\text{stag}} \sqrt{R_n} \right]_{\text{MAX}} = K \left[\frac{\left(\frac{W}{C_D A} \right) \sin \gamma_E}{3 e H} \right]^{1/2} V_E^3$$

TABLE 3.1.2-5. THERMOPHYSICAL PROPERTIES EMPLOYED IN REKAP

	ESM 1004 X	ESM 1004 AP
Virgin Density, ρ_v lb/ft ³	16.6	35.0
Char Density, ρ_c lb/ft ³	6.7	14.0
Pyrolysis Gas Specific Heat, Cpg Btu/lb° R	0.4	0.384
Molecular Weight of Injected series, Mg	24.5	24.5
Order of Reaction, n	2	2
Pre-exponential Factor, A sec ⁻¹	15000	30000
Activation Energy, E Btu/lb MOLE	44700	47500
Heat of Decomposition, Hgf 1335° R Btu/lb gas generated 1460 1710 1960	0	50 450 1000 2610
Specific Heat, Cp Btu/lb° R 600° R 710 1210 2075	0.31 0.33 0.44 0.44	0.305 0.360 0.44 0.44
Conductivity, k Virgin 610° R Btu/ft-sec° R 800 1335 1710 Char 1335 1710 2210	0.0000115 0.0000170 0.0000220 0.0000260 0.0000740 0.0000850 0.0001000	0.0000237 0.0000220 0.0000210 0.0000231 0.0000777 0.0000855 0.0001040
For the Decomposition Reaction Described by the Following Arrhenius Type Equation:		
$\frac{1}{W_o} \frac{dw}{dt} = \left(\frac{W-W_f}{W_o} \right)^n A(e)^{-\Delta E/RT}$		

where the cold wall value of stagnation point heat transfer is approximated by $\dot{q}_{stag} \sqrt{R_n} = K \rho_\infty^{1/2} V_\infty^2$. The strong effect on maximum convective heating of increases in either the product $(W/C_D A) (\sin \gamma_E)$ or the entry velocity V_E is illustrated in figs. 3.1.2-83 and -85. The time integrated value of the stagnation point convective heat flux is given as

$$\int \dot{q}_{stag} \sqrt{R_n} dt = 1.25 K \left[2 \frac{W}{C_D A} \right]^{1/2} \left[\frac{\sin \gamma_E}{H} \right]^{-1/2} V_E^2$$

A typical generalized expression for the stagnation point convective heat transfer is shown in fig. 3.1.2-87 which is essentially a more generalized method of presenting the data of fig. 3.1.2-84. The hemisphere and frustum distribution of the convective heating employed in CONSEP were generalized forms of the relationships presented in ref. 3-32 and 3-33.

The turbulent heat transfer to a flat plate can be closely approximated in air by

$$\left(\dot{q}_T S^{0.2} \right)_{\text{max on vehicle}} = 4.26 \times 10^{-8} \rho_\infty^{0.8} V_\infty^3$$

where the local turbulent heat transfer rate, $\dot{q}_T$, is determined by

$$\dot{q}_T = \frac{\Gamma}{S^{0.2}} \left(\dot{q}_T S^{0.2} \right)_{\text{max on vehicle}}$$

where

$$\Gamma = f \left(\frac{P_e}{P_s} \right)$$

The turbulent heat transfer distribution parameter Γ is pressure ratio dependent. It can be shown that the point on a hemisphere where $\dot{q}_T$ is a maximum, is 33° off the stagnation point, assuming a modified Newtonian pressure distribution. Reference 3-33 has shown that this approximate flat plate equation overestimates the turbulent heat transfer obtained at 33° off the stagnation point using Walker's turbulent integral equation by about 24 percent. Hence, reducing the flat plate constant by 24 percent and introducing the wetted length, S , to 33° off the stagnation point, the maximum turbulent heat transfer on the hemispherical nose cap can be written as:

$$\dot{q}_T R_n^{0.2} = 3.62 \times 10^{-8} \rho_\infty^{0.8} V_\infty^3$$

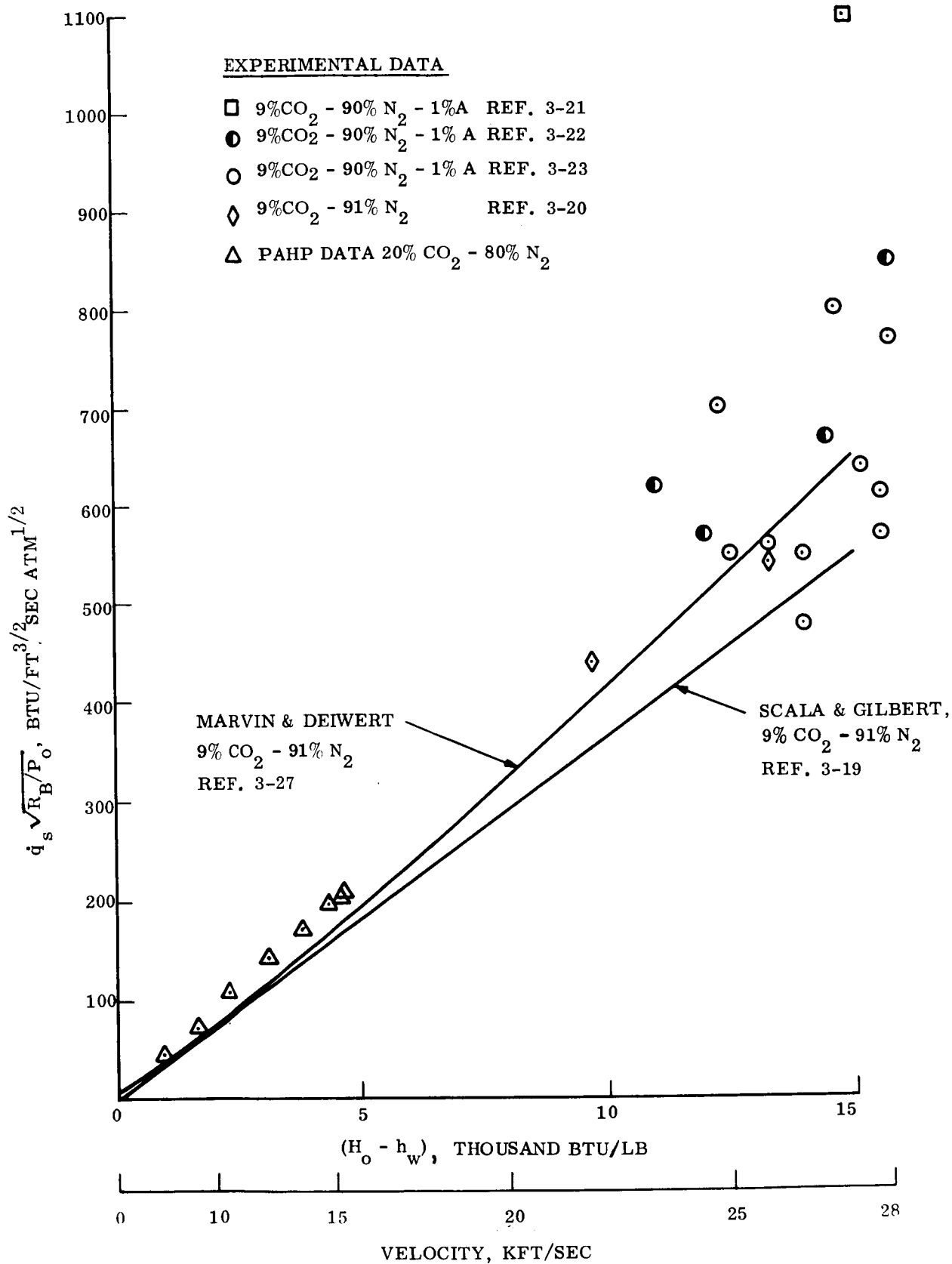


Figure 3.1.2-80. Comparison of Available Theories
with Experimental Data for CO₂-N₂ Mixes

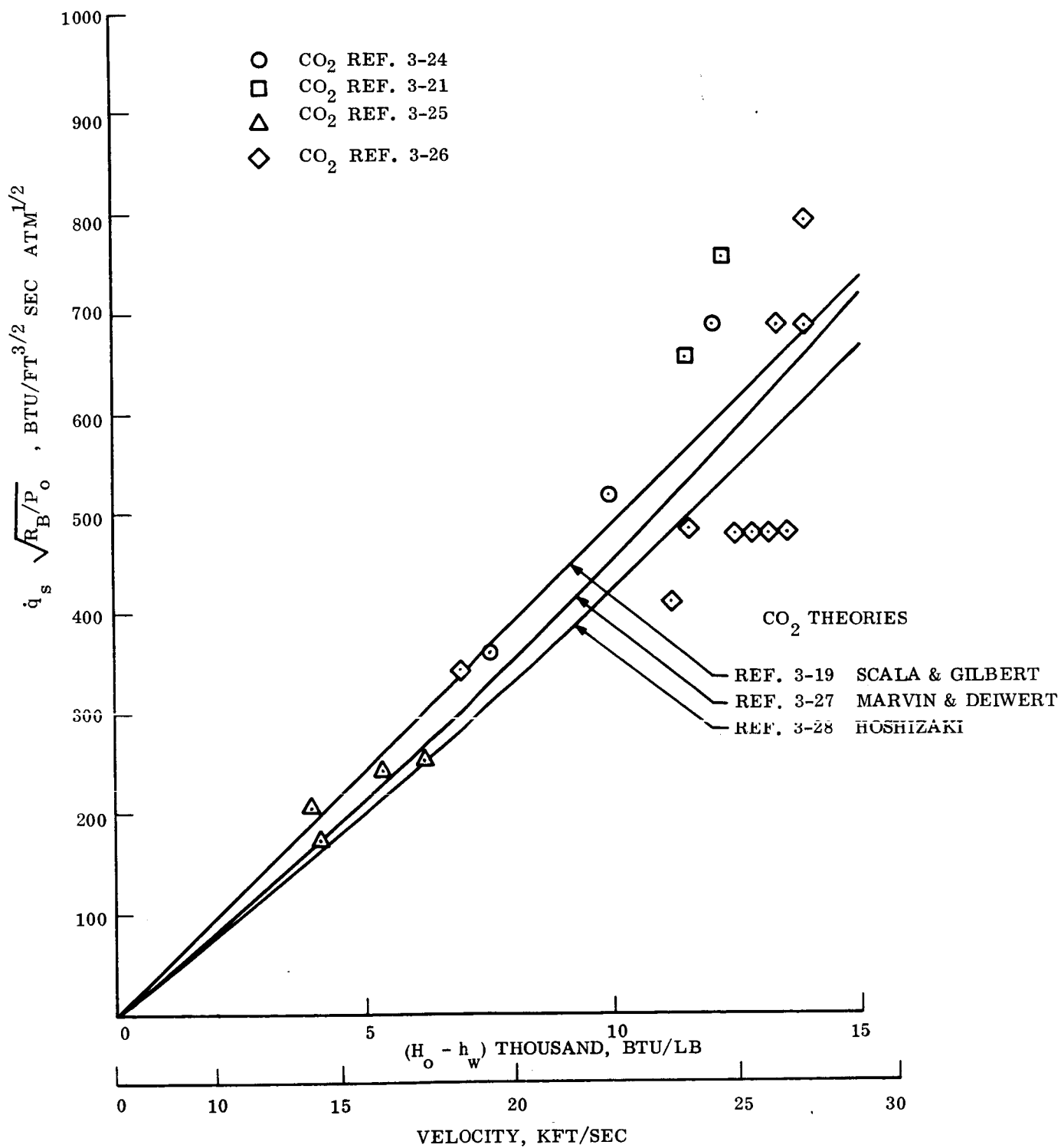


Figure 3.1.2-81. Comparison of Available Theories with Experimental Data for CO₂ Environment

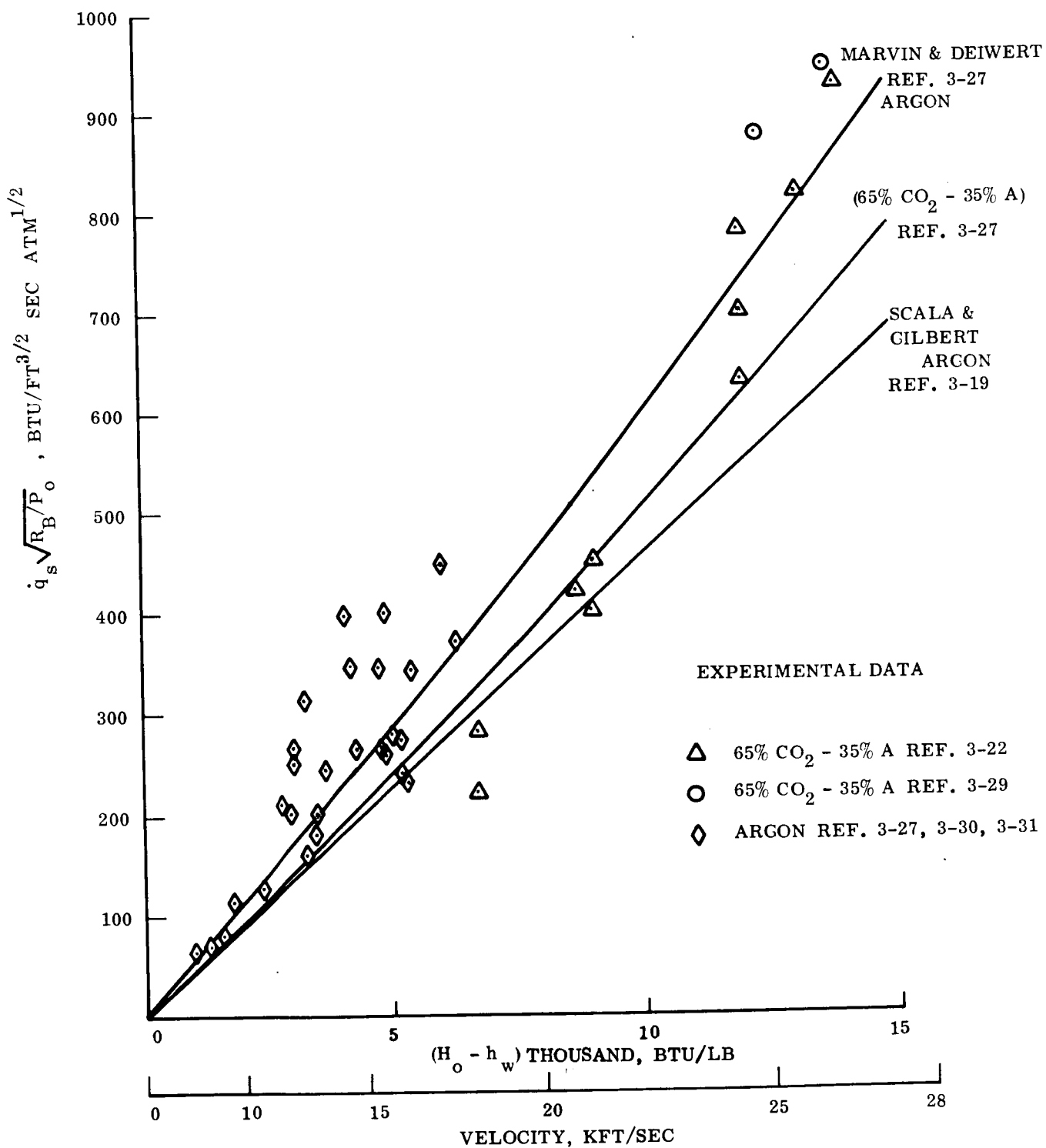


Figure 3.1.2-82. Comparison of Available Theories with Experimental Data for Pure Argon and an Argon Rich Mixture

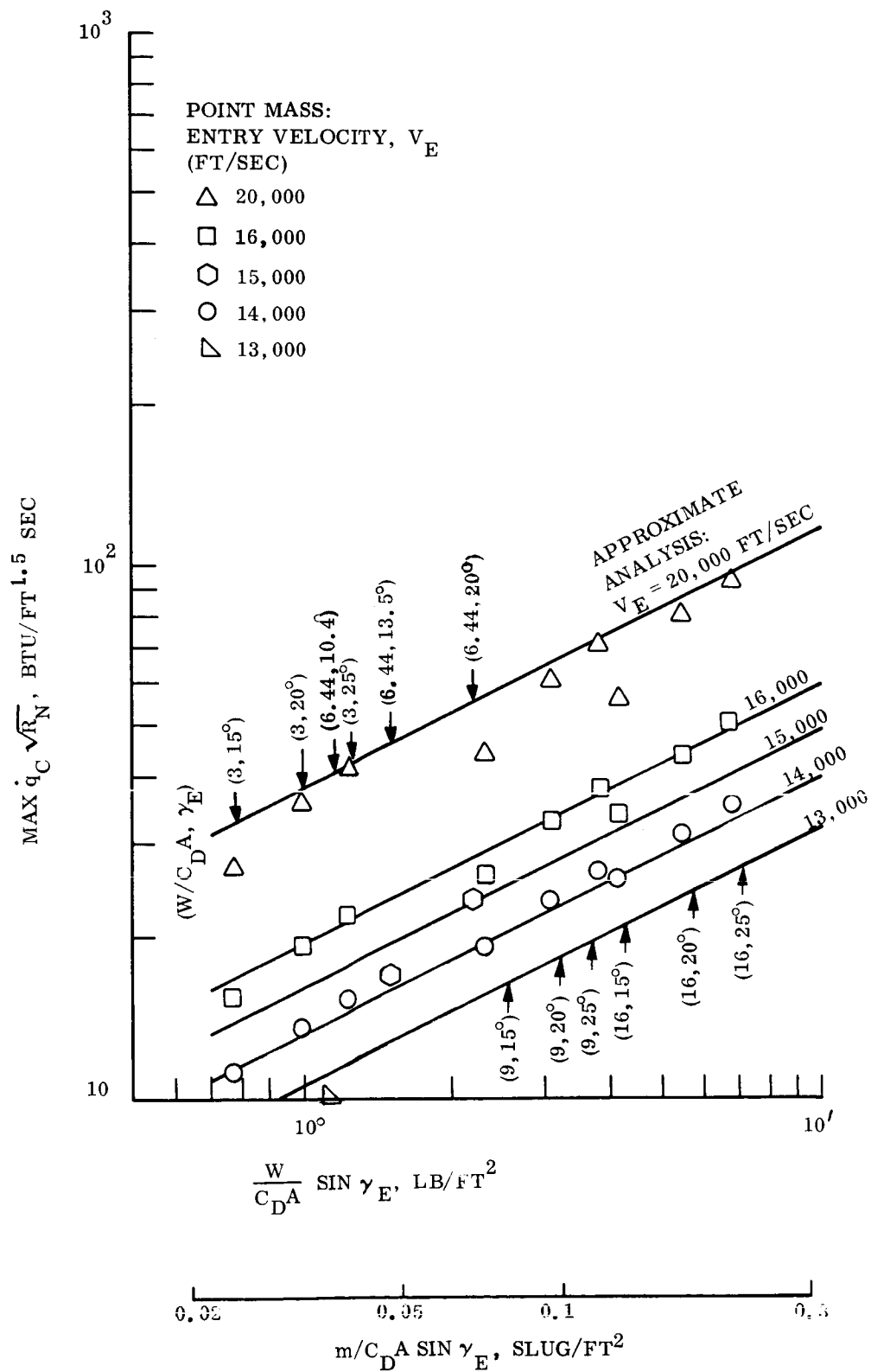


Figure 3.1.2-83. Comparison of Approximate Entry Analysis with Point - Mass Trajectory Heating Results - Time Integrated Cold - Wall Convective Heat Flux in VM-3

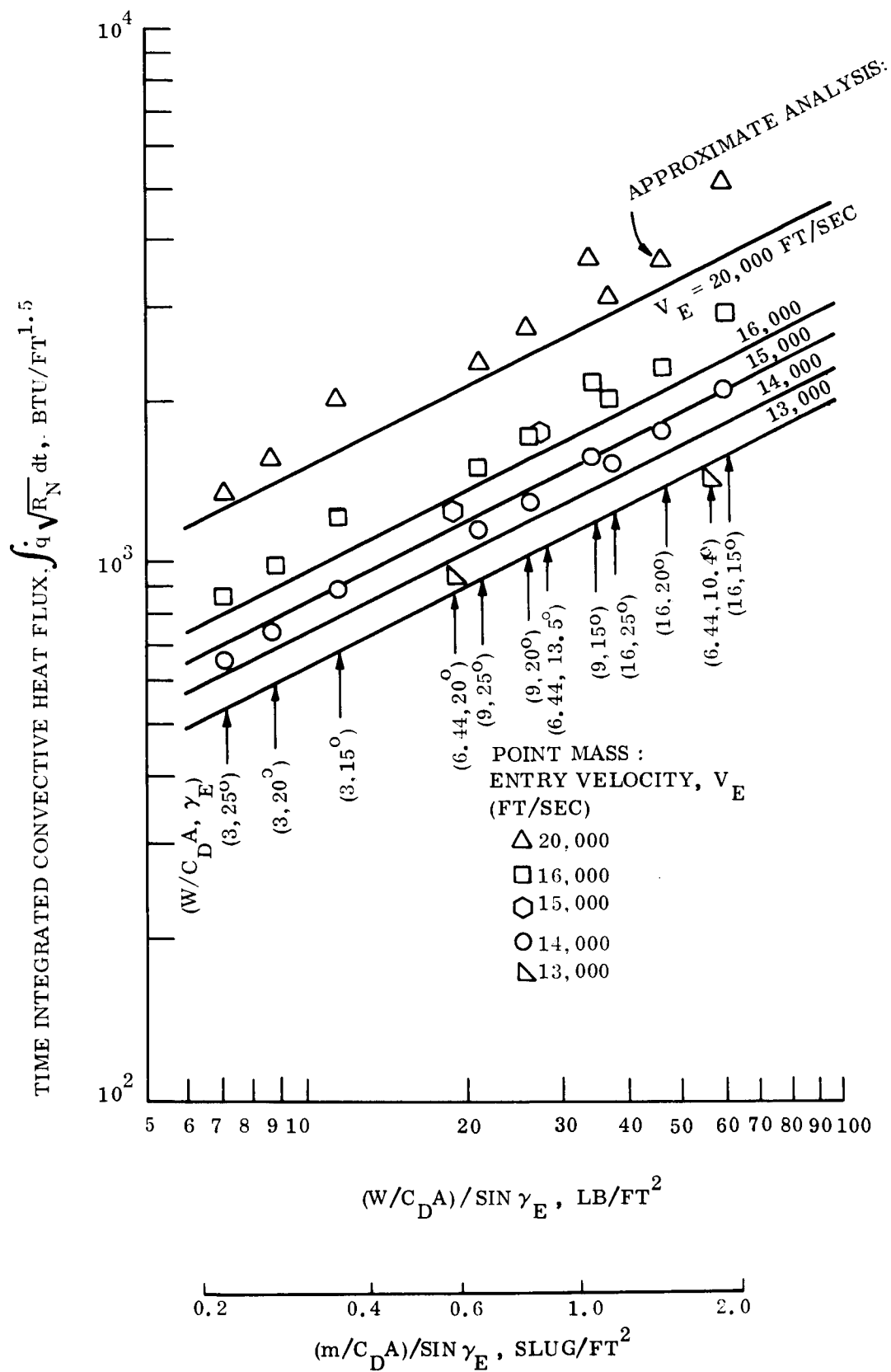


Figure 3.1.2-84. Comparison of Approximate Entry Analysis with Point-Mass Trajectory Heating Results-Time Integrated Cold-Wall Convective Heat Flux in VM-3

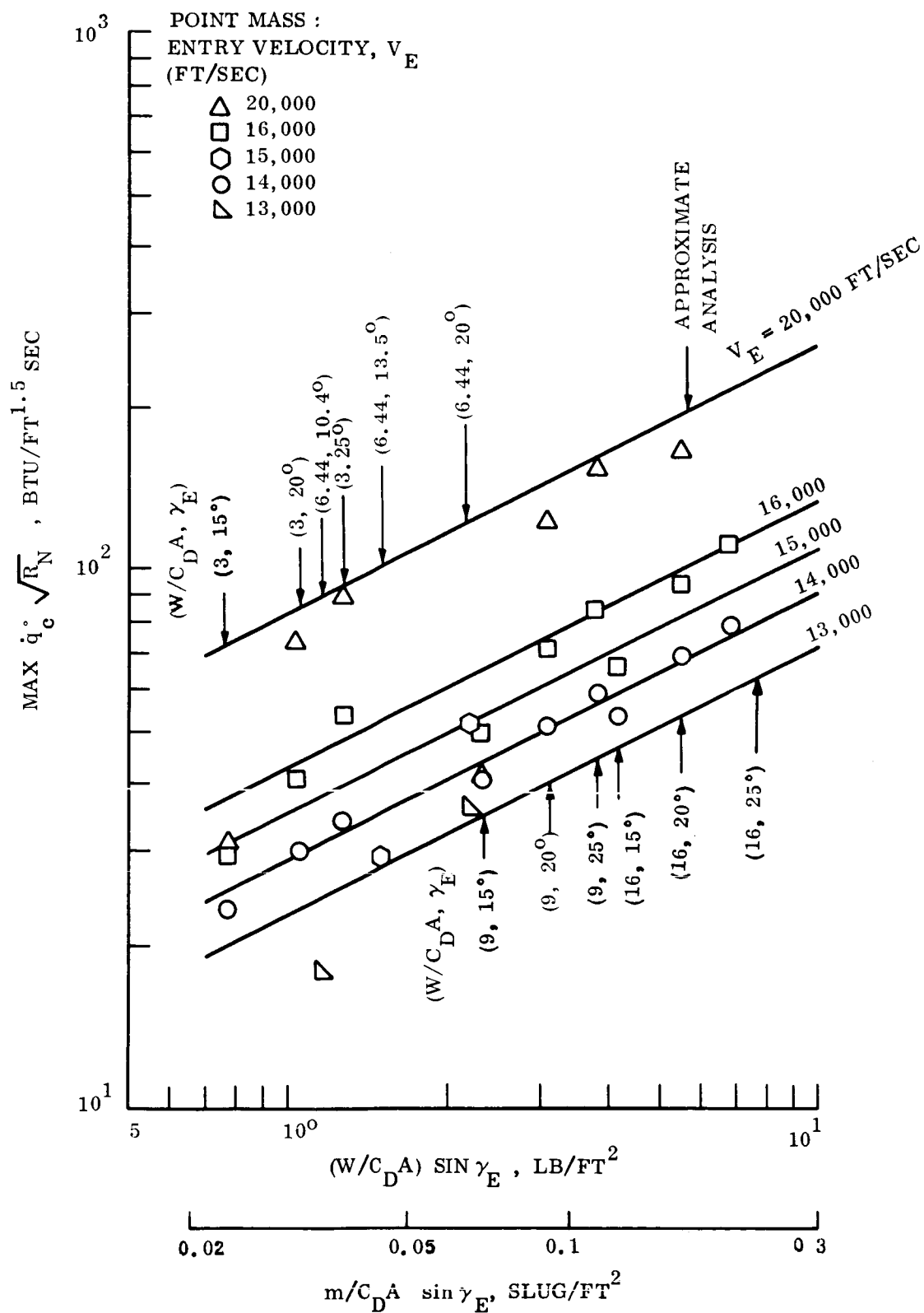


Figure 3.1.2-85. Comparison of Approximate Entry Analysis with Point-Mass Trajectory Heating Results-Maximum Cold-Wall Convective Heat Flux in VM-8

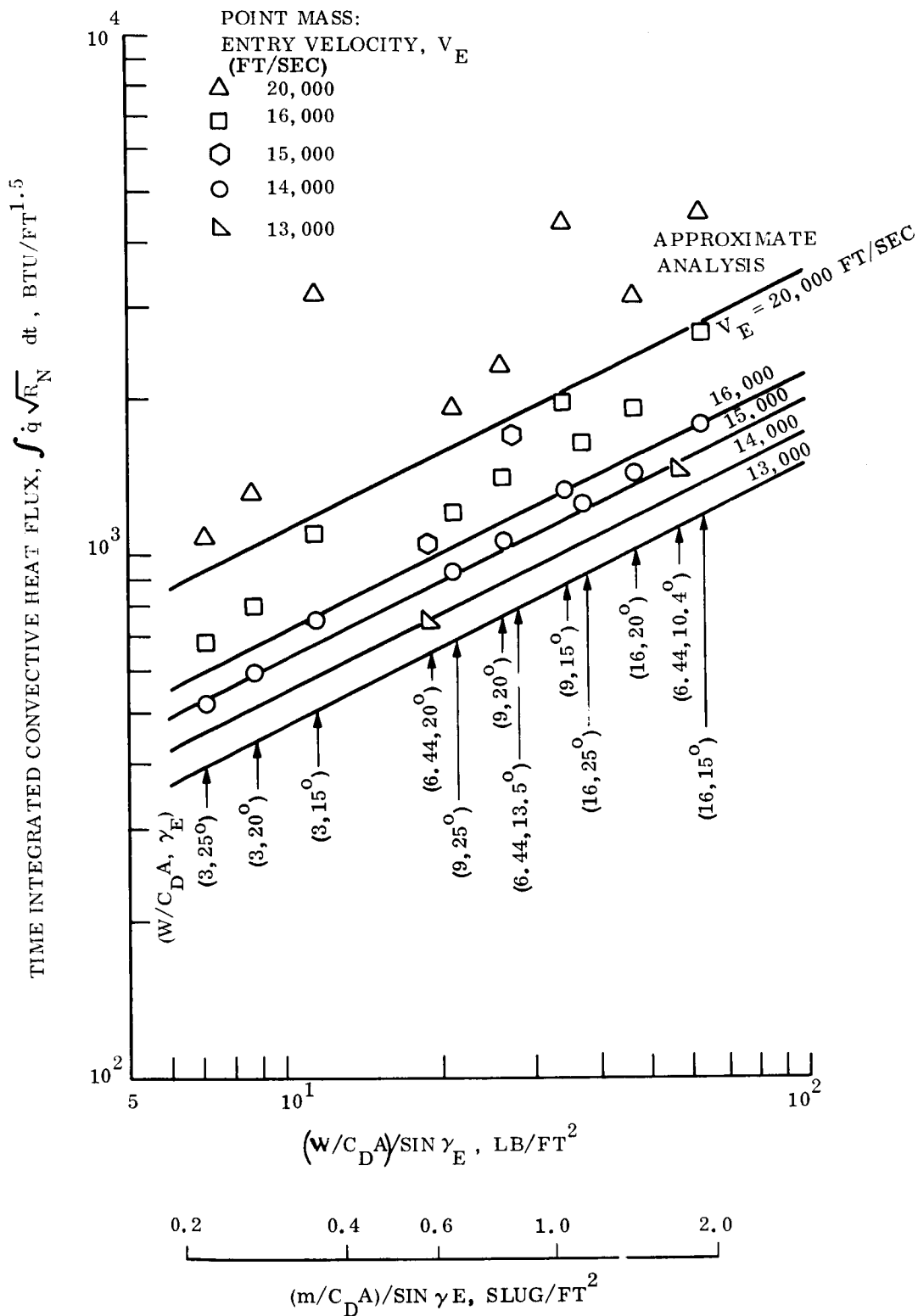


Figure 3.1.2-86. Comparison of Approximate Entry Analysis with Point-Mass Trajectory Heating Results - Time Integrated Cold-Wall Convective Heat Flux in VM-8

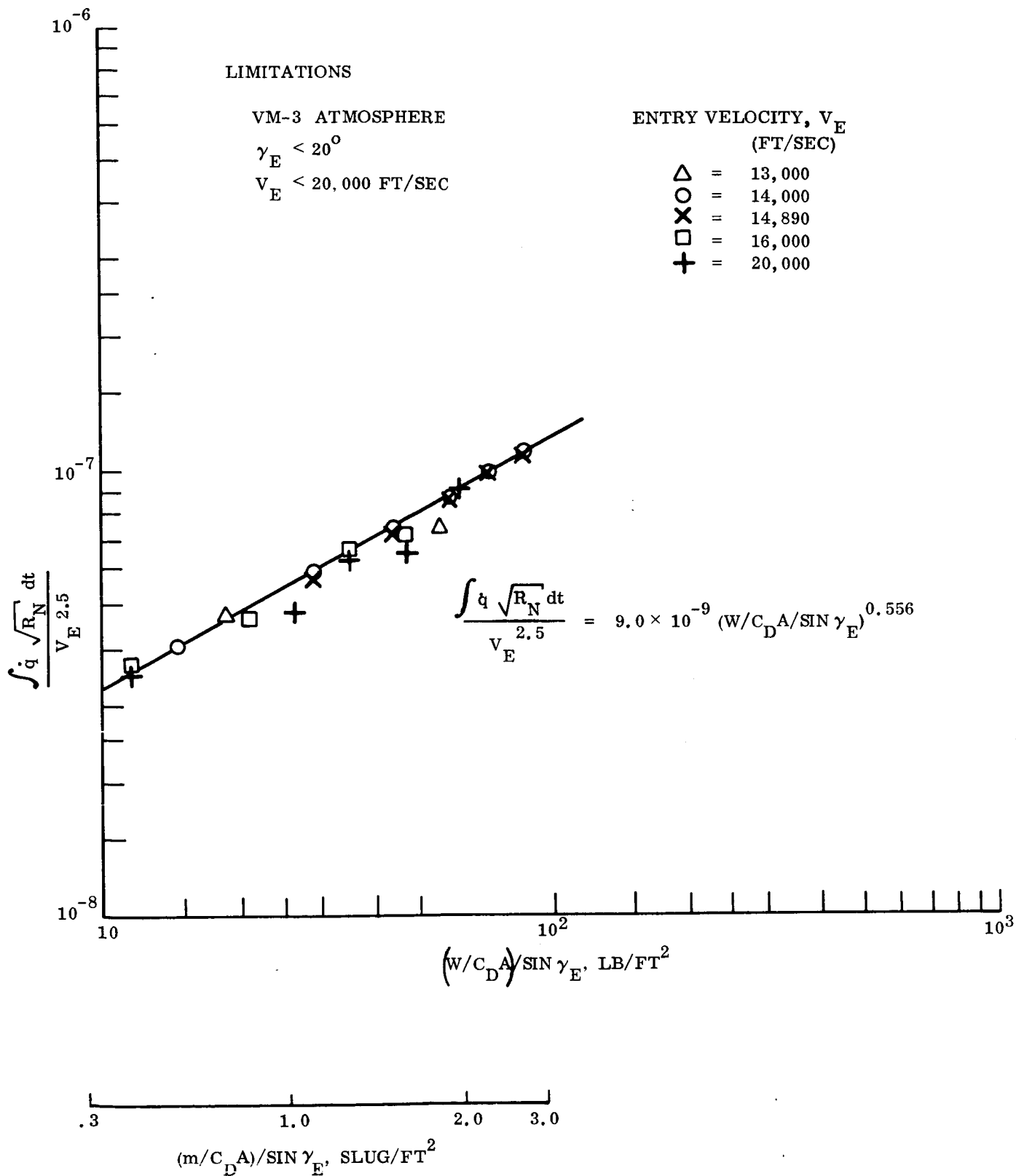


Figure 3.1.2-87. Parametric Variation of Stagnation Point Convective Heat Transfer

Using again the approximate entry analysis of ref. 3-35, the maximum turbulent heat transfer rate experienced during entry is given by:

$$\dot{q}_T R_n^{.2} = 3.62 \times 10^{-8} \left[2 \frac{W}{C_D A} \frac{\sin \gamma_E (.766)}{H} \right]^{0.8} (.766 V_E)^3$$

The heat shield is normally sized by the entry condition that provides the longer soak time, maximum time integrated heat flux. If $\dot{q}_{stag} \sqrt{Rn} = K \rho_\infty^{1/2} V_\infty^3$, (where K is constant given in ref. 3-19, then the time integrated heat flux is proportional to $K (H)^{1/2}$ and the maximum time integrated heat flux will occur for entry into the VM-3 atmosphere as shown in table 3.1.2-4. Since the maximum heat flux is proportional to $K / (H)^{1/2}$, the maximum heat flux will occur for entry into the VM-8 atmosphere, ignoring the effects of large quantities of the monatomic gas argon in the atmosphere. However, considering the high argon content the maximum heat flux will occur for entry into the VM-4 or the NASA/LRC maximum atmosphere.

Boundary Layer Transition - As the ballistic entry parameter $(W/C_D A \sin \gamma_E)$ increases, the altitude where hypersonic heating is experienced decreases. With this decrease in altitude goes an increase in the free stream density and hence, the free stream Reynolds number. A previous design study with a $\theta_c = 60^\circ$, $R_n/R_B = 0.5$, $R_B = 8.5$ -ft design for out-of-orbit entry into VM-3 and VM-8 $(W/C_D A) \sin \gamma_E = 1.5$ lbs/ft $[0.0466 \text{ slugs/ft}^2]$ ref 3-36, indicates that a vehicle sized for laminar heating into VM-3 was adequate for entry into VM-8 with local boundary layer transition not occurring until a Reynolds number of 150,000 is reached (based on wetted length and edge of boundary layer properties). As the path angle is steepened towards 90° and the ballistic coefficient increased above 6.4 lbs/ft² $[0.199 \text{ slugs/ft}^2]$ the effect of boundary layer transition becomes more pronounced and must not be ignored in making the heat shield material selection.

Fig. 3.1.2-88 indicates the boundary layer transition criteria employed in this study. The upper and lower curves show the transition Reynolds number (based on wetted length and local edge of boundary layer properties) for low and high mass addition materials, respectively, as a function of local Mach number. The local Mach numbers for the sphere-cones of this study, are generally subsonic for all body locations and thus the range of transition Reynolds number is between 1.5 and 5×10^5 . Due to the lack of transition data on ESM materials in this environment and the resultant uncertainty of ESM mass addition on boundary layer transition, a Reynolds number of 3×10^5 was chosen for instantaneous boundary layer transition in this study, although it is realized that a transitional region exists between fully laminar and fully turbulent flows.

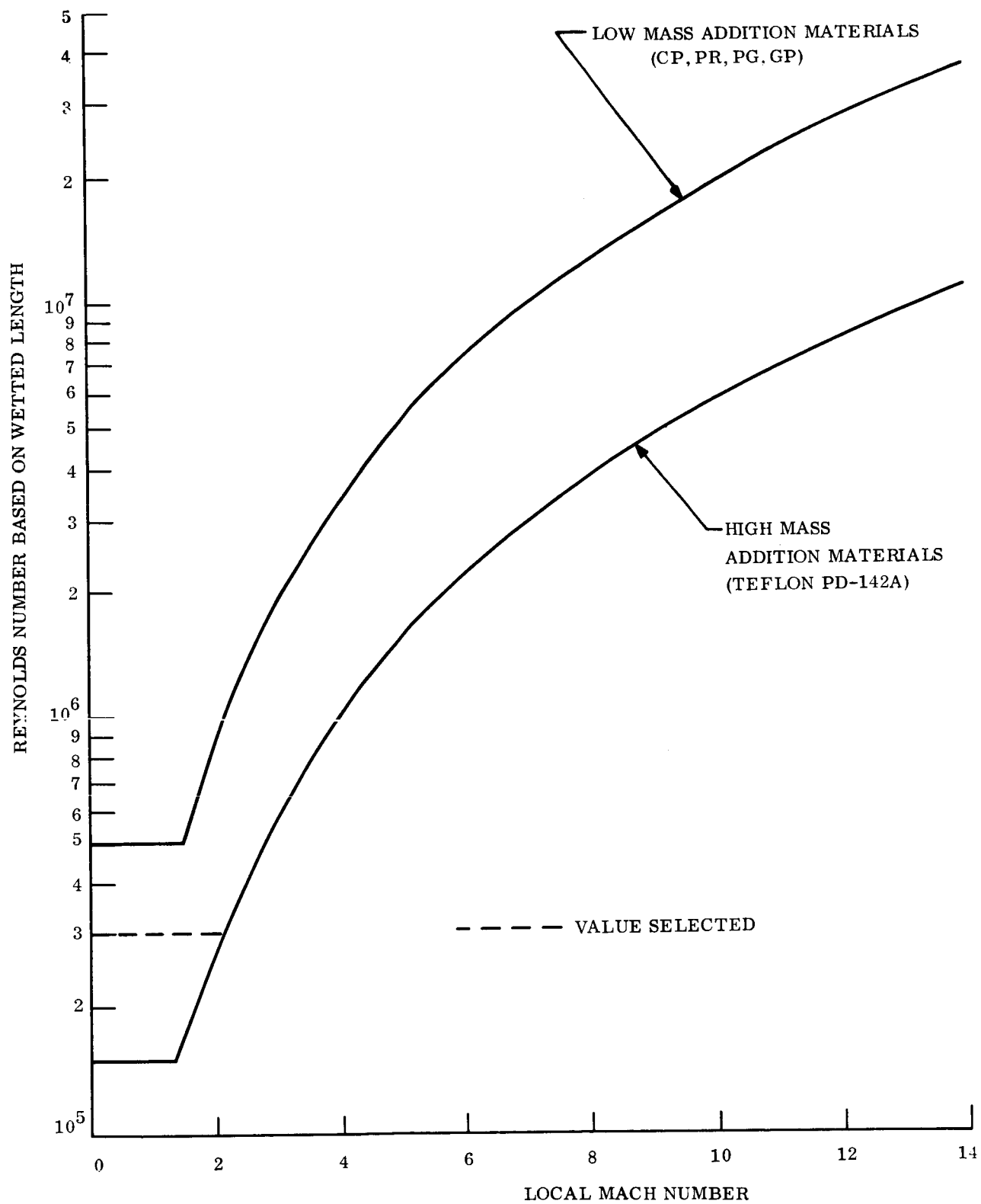


Figure 3.1.2-88. Transition Reynolds Number Based on Wetted Length

Hot Gas Radiative Heating - During planetary entry the hot gas cap surrounding the entry vehicle can emit an appreciable amount of radiant energy toward the vehicle surface. In order to ascertain the magnitude of the radiative heating as seen by the vehicle surface, it is necessary to estimate both the equilibrium and non-equilibrium radiant heat transfer and to consider possible radiant heating reductions due to self-absorption and radiative coupling of the flow field.

Equilibrium Radiation - For the Mars entry velocity range of 14,000 to 24,000 ft/sec, the molecular band radiation is the principal contributor to the radiant heating. The continuum and atomic line radiation result from molecular dissociation at higher speeds and are small for the Mars entry velocity range considered.

The molecular band equilibrium radiation to the stagnation point of the vehicle can be reasonably obtained with the use of 30 percent CO₂ - 70 percent N₂ radiation data of ref. 3-37, which is in the chart form of radiation heat flux versus local density with local temperature and gas layer thickness as parameters.

These parametric charts were generated by use of the HGR program with the assumption that the temperature and density are constant across the shock layer. The charts include the principal radiation source, CO (4+), and the effects of self-absorption. The state variable of temperature and density from these charts can be related to free stream flight conditions and thus, the charts yield equilibrium radiant heating for flight velocities up to about 21,000 feet per second. For the higher velocities, Wolf and Spiegel, ref. 3-38, have summarized comparisons of available experimental data and the theoretical predictions of both JPL and NASA/ARC.

An attempt was made by ref. 3-38 to obtain a correlation for radiant intensity with local density and flight velocity as parameters for a 30-percent CO₂, 70 percent N₂ atmosphere. The form of the correlation is given here where the coefficients A, B, and C differ accordingly with the approximate mean or maximum curve drawn through the data.

$$\frac{I}{(10^5 \rho_\infty)^{1.4}} = A \left(\frac{V}{10^4} \right)^2 + B \left(\frac{V}{10^4} \right) + C \quad (3)$$

The values of the constants for a 30 percent CO₂, 70 percent N₂ gas mixture are given below:

<u>Constants</u>	<u>30 percent CO₂ - 70 percent N₂</u>	
	<u>Mean Curve</u>	<u>Max Curve</u>
A	+ 700	+ 1700
B	-2180	- 6040
C	+1690	+ 5440

The stagnation equilibrium radiant heating is approximate as the product of one half of the shock layer thickness and the correlated radiant intensity. These data and theory do not include the effects of self-absorption or the contributions of the strong CO (4+) radiating species. Only the black body limit of the CO (4+) contribution to the radiation from the shock layer has been treated in the work of ref. 3-38. It was treated as a separate contribution that could be added to the other radiation sources, when the thickness of the shock layer was equal to or greater than the CO (4+) penetration depth for the system.

On the conical portion of a sphere cone configuration, the equilibrium radiant heating increases with increasing distance from the stagnation point. This increase occurs since the shock continually thickens while the temperature and density remain essentially constant along the cone. The level of the radiant heating at the end of the cone was obtained from empirical relations using the data from ref. 3-38 and -39.

The thickness of the shock layer used in the calculation is obtained from the shock stand off correlation of ref. 3-38 and -39. For the end of the cone, the correlation of ref. 3-38 (repeated here in fig. 3.1.2-89) is used. The stand-off distance at the stagnation point is obtained from the correlation of ref. 3-40 (repeated here in fig. 3.1.2-90). These correlations are in reasonable agreement with results of some recent JPL flow field gas radiation studies (ref. 3-41).

Generalized expressions for estimating the equilibrium radiant heating were required for use in the CONSEP program parametric studies. The charts of radiation heat flux from ref. 3-37, discussed previously, were related to free stream flight conditions and were used to estimate the equilibrium radiant heating. A straight line trajectory was assumed and generalized radiant heating parameters derived. It was found that the maximum radiative heat flux could be expressed as

$$F_1(V_E) = \frac{\dot{q}_{r \max}}{\left(\frac{\delta}{0.05}\right)^{0.88} (\beta \sin \gamma_E)^{1.14}} \quad (3)$$

and the time integrated radiant heat flux expressed as

$$F_2(V_E) = \frac{\sin \gamma_E \int \dot{q}_r dt}{\left(\frac{\delta}{0.05}\right)^{0.88} (\beta \sin \gamma_E)^{1.14}} \quad (4)$$

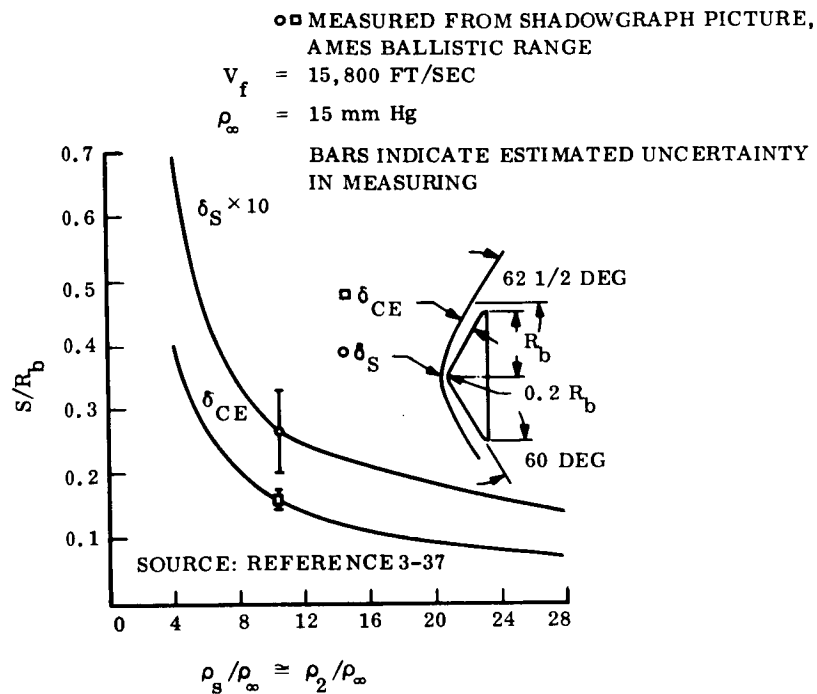


Figure 3.1.2-89. Shock Standoff Distances Used for Present Calculations

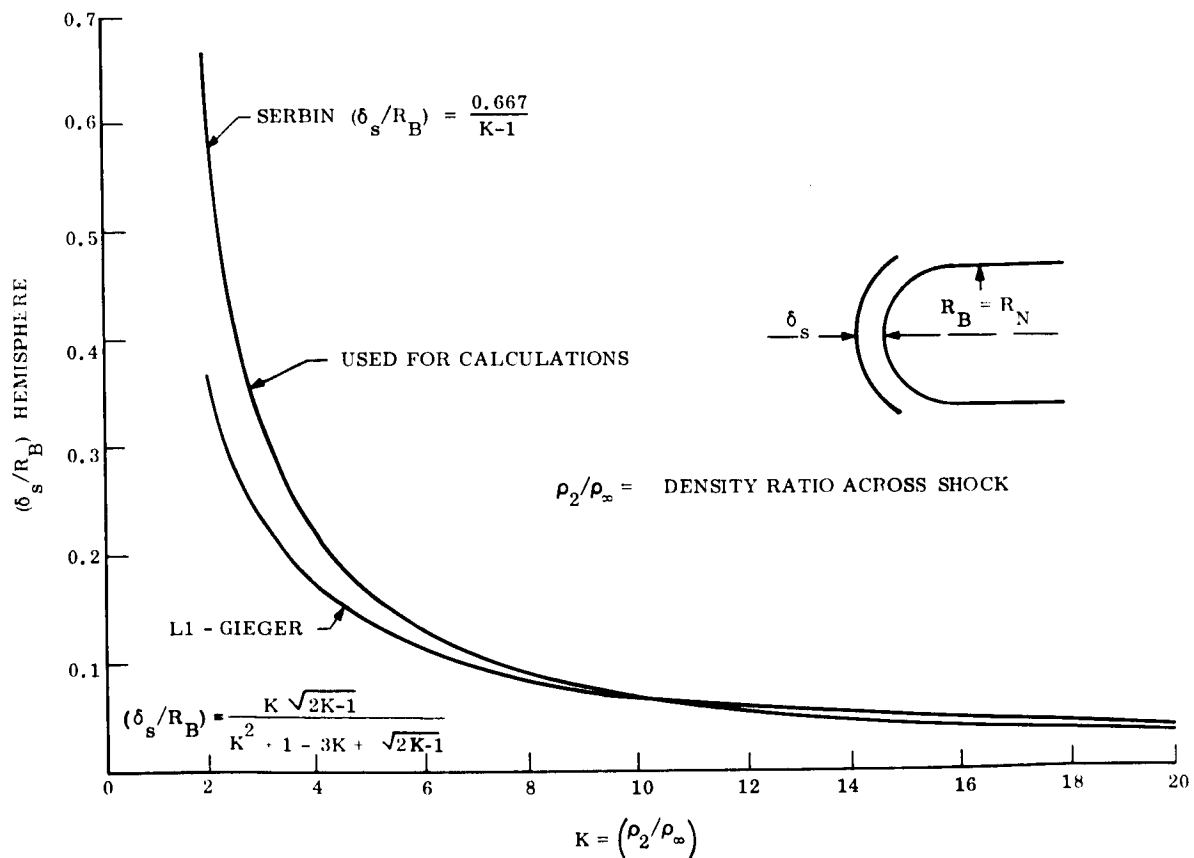


Figure 3.1.2-90. Theoretical Shock Standoff Distance for Hemispheres

These results are shown in figs. 3.1.2-91 and -92 for typical Mars entry velocities.

For out-of-orbit Mars entry velocities of 14,000 to 15,000 fps, the equilibrium radiative heating can be considered negligible compared to the convective heating; however, as the entry velocity is increased, the equilibrium radiant heating can be significant compared to the convective heating.

For estimates at the stagnation point, the shock stand-off distance may be approximated as 5 percent of the nose radius for Mach numbers greater than 10. The expression $(\delta/.05)$, eqs. 4 and 5 and in figs. 3.1.2-91 and -92 may, therefore, be replaced by the nose radius R_N .

The distribution of the radiant heating along the cone is difficult to estimate in generalized terms since the properties of the shock layer and the shock thickness vary with flight conditions. Therefore, following the trends indicated in refs. 3-38 and 3-39, the stagnation point equilibrium radiant heating is doubled to approximate the level at the end of 60° half-angle cone. To obtain the equilibrium radiant heating along the body surface a linear interpolation between the stagnation point and end of the cone was used in the CONSEP program. This ignores the drop shown in ref. 3-39 in radiant heating around the sphere portion of the vehicle to the tangency point, but it is considered acceptable since it results in an over-estimate of radiant heating at the tangency point.

It is noted here that the radiative coupling effect has been neglected in this analysis. When the radiation from the shock layer is relatively large in comparison to the energy influx to the shock layer, there occurs a reduction in shock layer temperature and a consequent reduction in radiation level. Since the radiation level of Martian entry is small to begin with, a negligible reduction from the adiabatic level occurs and may be neglected. Since the coupling effect reduces net heat flux to the vehicle, neglecting it will yield slightly conservative heat shield requirements.

Non-Equilibrium Radiation - Non-equilibrium effects present in the shock layer give rise to a radiative heat flux that may be of the same order or greater than the equilibrium portion determined by most calculations. This results from the high non-equilibrium temperature overshoot experienced by the gas as it passes through the hypersonic shock front. Because of these high temperatures, which tend to gradually decay to their equilibrium value, a strong radiation overshoot occurs, resulting in an increase in radiation over the equilibrium value. Determination of the non-equilibrium portion of radiation has been made by use of test data. A comparison of the data for total non-equilibrium radiance as given in ref. 3-38 is shown here in fig. 3.1.2-93. This figure shows the nonequilibrium contribution to be independent of density (i.e., a function of only flight velocity and composition). The reason for this is that as the density is increased, the non-equilibrium temperature behind the shock front is also

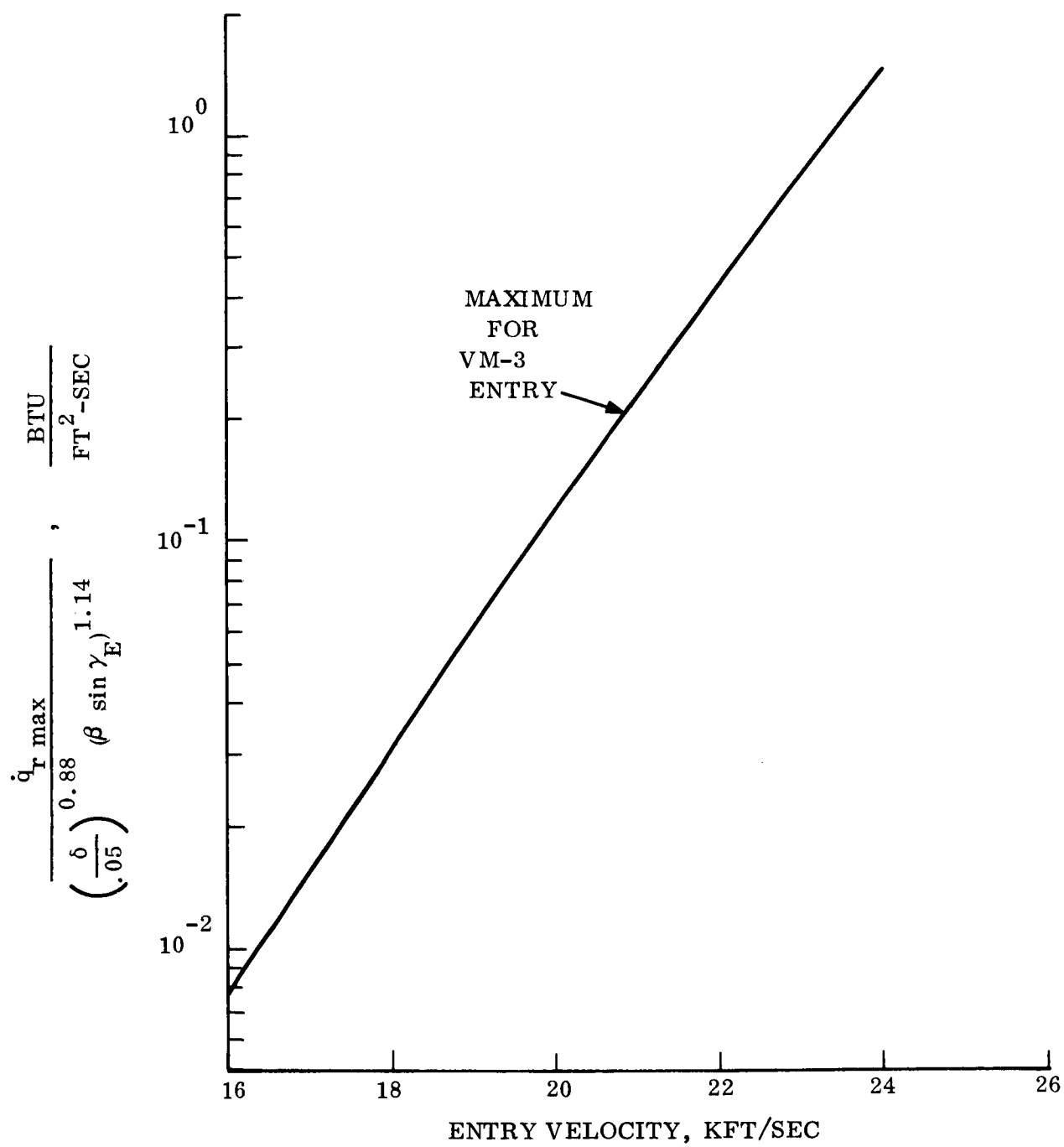


Figure 3.1.2-91. Variation of Maximum Radiant Heating Parameter

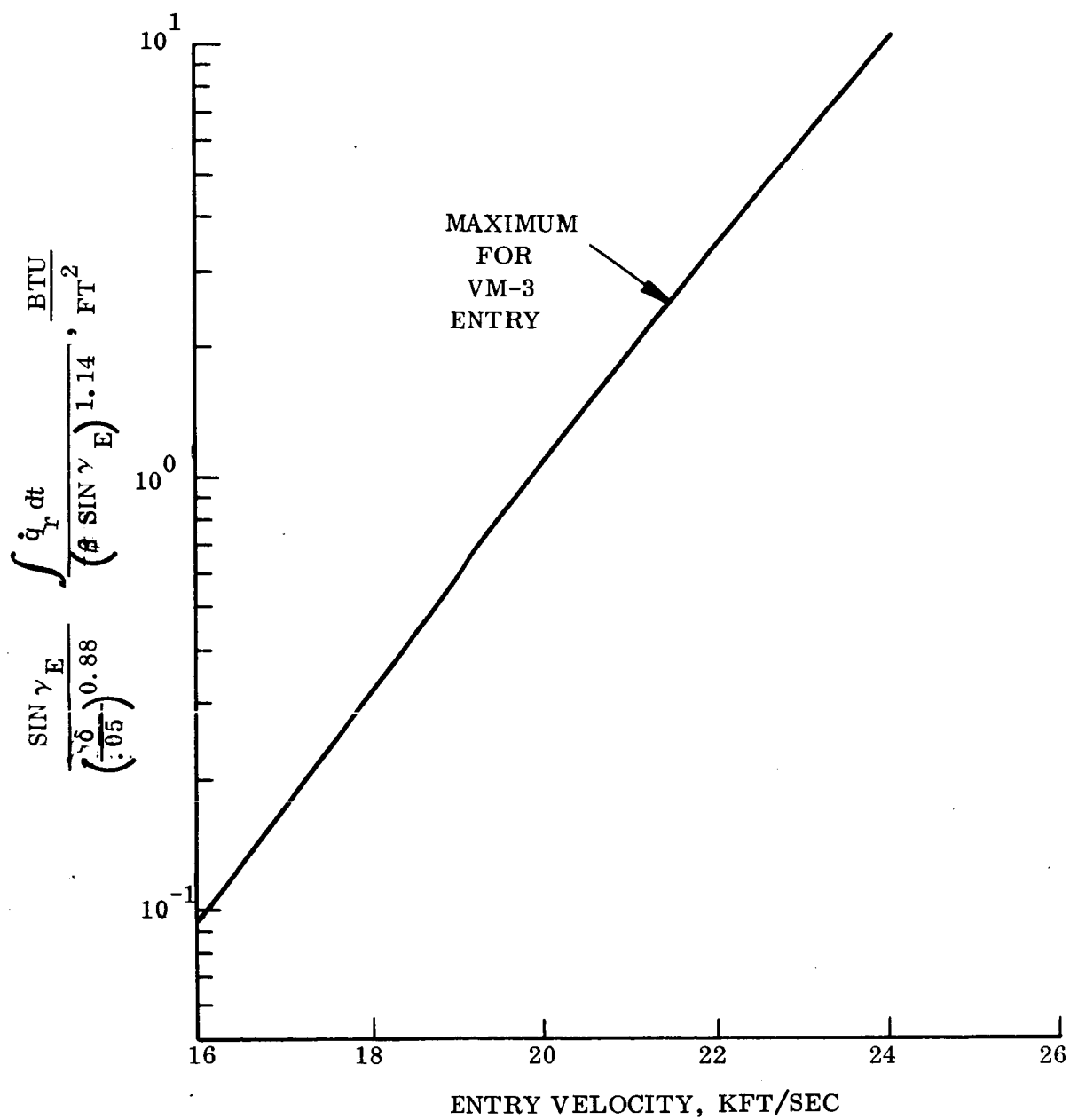


Figure 3.1.2-92. Variation of Total Integrated Radiant Heating

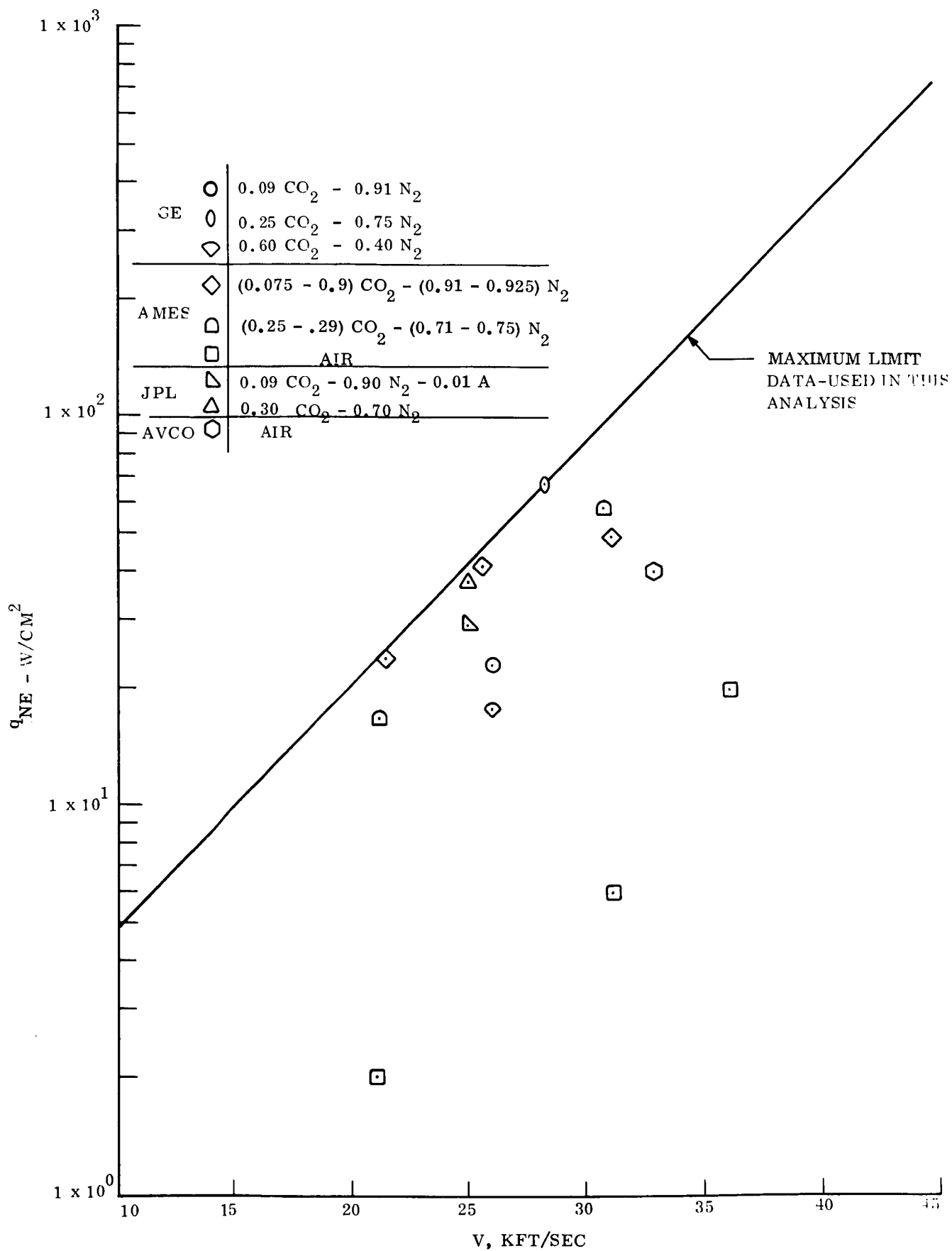


Figure 3.1.2-93. Comparison of Total Non-equilibrium Heat Transfer Rate for Several Gas Mixtures

increased; however, the relaxation time (time for the temperature to "relax" to an equilibrium value) is shorter due to the higher density (i.e., more interactions between particles can occur). The net result of these two effects is to produce a value of non-equilibrium radiation which is essentially independent of density.

As more data are collected concerning non-equilibrium radiation for various atmosphere compositions, better definition can be given to fig. 3.1.2-93. For the present, the straight line through the maximum of the data extrapolated down to the velocities of interest in this study is assumed to be representative of the non-equilibrium contribution. This contribution represents the maximum radiation that can be contained within the relaxation distance behind the shock. When relaxation distances are longer than the shock standoff distances, the amount of non-equilibrium radiation is reduced. The technique of ref. 3-38 for calculating this reduction contains the assumption that the non-equilibrium radiation intensity can be represented by the area of a triangle whose base length and peak location are given by:

$$\delta_{NE} = \frac{2.3 \times 10^{-6}}{\rho_{\infty} \left(\frac{V_{\infty}}{10^4} \right)^{4.3}} \quad (6)$$

and

$$\delta_{peak} = \frac{0.23 \times 10^{-6}}{\rho_{\infty} \left(\frac{V_{\infty}}{10^4} \right)^{3.3}} \quad (7)$$

respectively. When the shock stand-off distance is less than the length of the triangle, the triangle is appropriately truncated, which results in a reduction of the radiant heat flux, see fig. 3.1.2-94. The maximum radiant flux, fig. 3.1.2-93, is assumed valid at both the stagnation point and at the cone edge where the normal component of velocity is used to define $\delta_{NE} + \delta_p$. The shock stand-off distances used in the equilibrium radiation calculations are also used here to determine the amount of truncation. To develop a generalized expression for non-equilibrium radiative heating in the CONSEP program, the non-equilibrium heating rate histories for a typical out of orbit entry and for several values of ballistic coefficient were evaluated. The time integrated non-equilibrium heating at the stagnation point is shown in fig. 3.1.2-95 and was used as an estimate for all out-of-orbit entry cases since the entry parameters are close to the

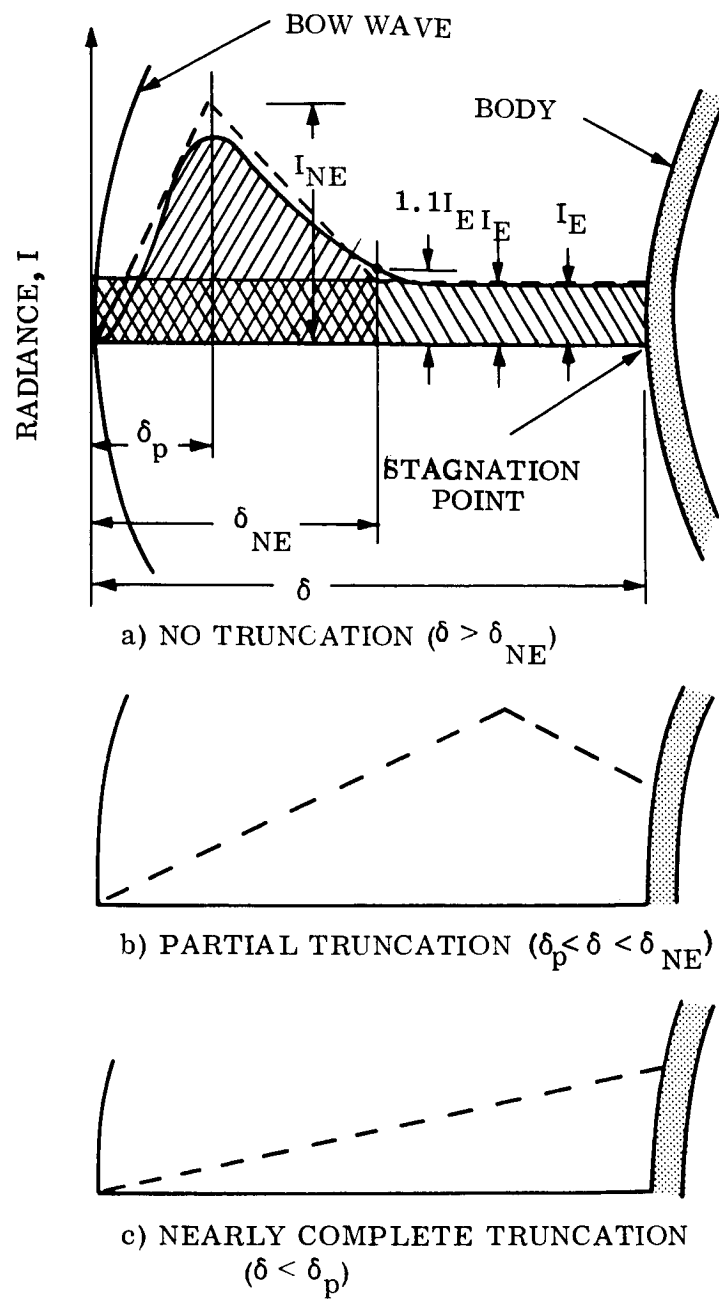


Figure 3.1.2-94. Stagnation Point Radiation Model

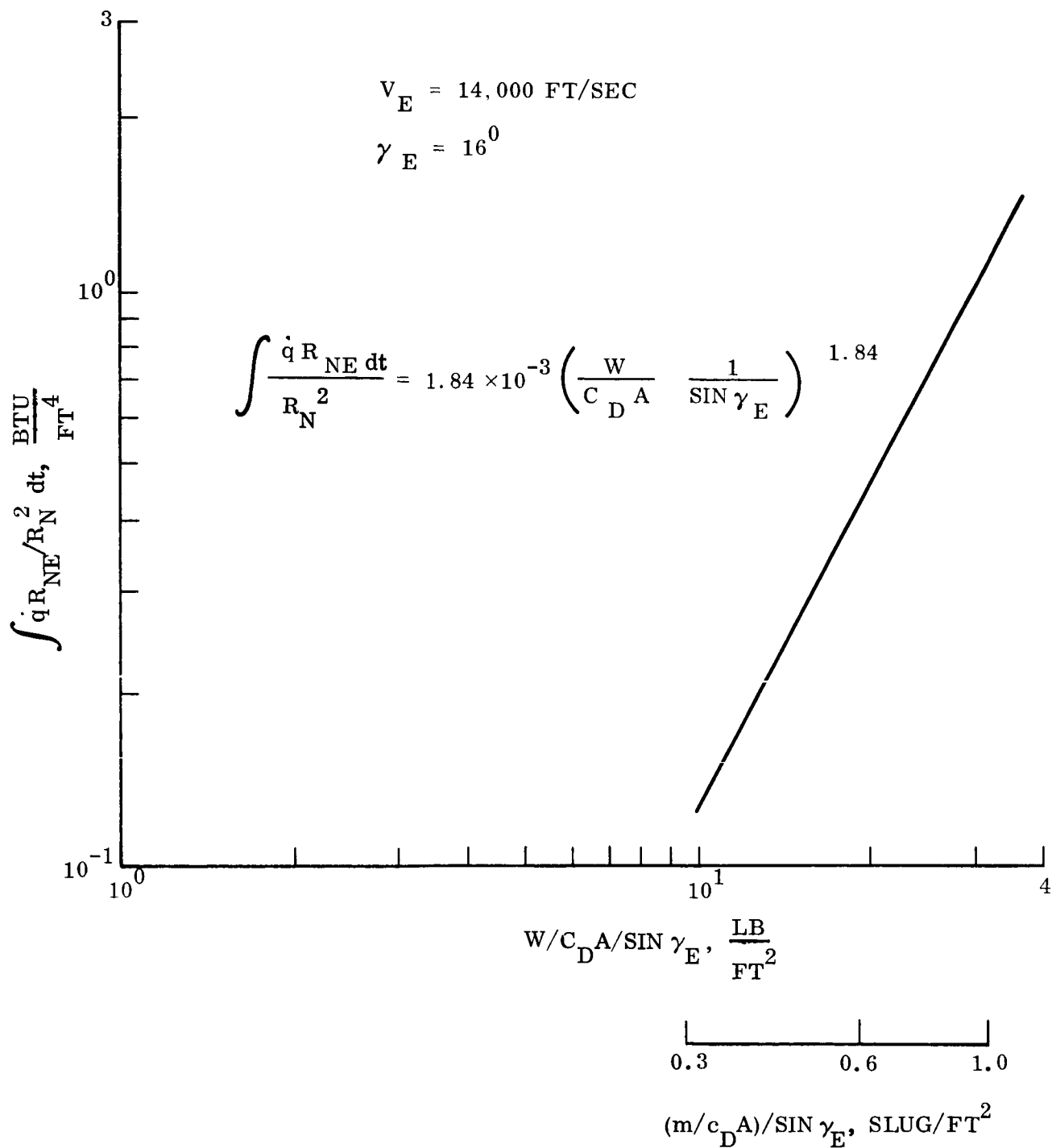


Figure 3.1.2-95. Integrated Non-equilibrium Radiant Heating for the Stagnation Point

case evaluated. An expression for the time integrated stagnation point non-equilibrium radiant heating can be written in the form

$$\frac{1}{R_N^2} \int \dot{q}_{r_{NE}} dt = 1.84 \times 10^{-3} \left(\frac{\beta}{\sin \gamma_E} \right)^{1.84} \quad (8)$$

Estimations of non-equilibrium radiant heating at the stagnation point for direct Mars entry were obtained by the trends shown in ref. 3-38. The entry velocity Wolf and Spiegel used was a typical direct Mars entry value of 22,350 feet per second. The ratio of the total integrated equilibrium radiative heating to the integrated non-equilibrium heating varying with ballistic coefficient and entry angle is shown in fig. 3.1.2-96. The equilibrium value is calculated as previously discussed and thus defines the non-equilibrium value at a specific ballistic coefficient and entry angle.

The values for non-equilibrium radiative heating for out-of-orbit and direct Mars entry at the end of the skirt of the vehicle have been estimated by the same procedures as at the stagnation point. For the out-of-orbit case, calculations were made which resulted in radiant heating values dependent on the base radius, R_B . An expression for the time integrated non-equilibrium radiant heating to the conical portion of the vehicle can be written in the form:

$$\frac{1}{R_B^{1.56}} \int \dot{q}_{r_{NE}} dt = 0.0216 \left(\frac{\beta}{\sin \gamma_E} \right)^{1.48} \quad (9)$$

and is presented in fig. 3.1.2-97. An approximate approach for estimating the effects of varying cone angle resulted in the following variation with cone angle of the constant in this equation:

<u>Cone Angle</u>	<u>Constant</u>
50°	0.00972
60°	0.0216
70°	0.0345

For the direct entry case, the data of Wolf and Spiegel was used again. The ratio of the integrated equilibrium radiative heating to the integrated non-equilibrium heating varying with ballistic coefficient and entry angle is shown in fig. 3.1.2-98. Again, the equilibrium value on the cone is calculated as discussed previously. In addition, some variation with entry velocity is shown for the end of the cone in this figure.

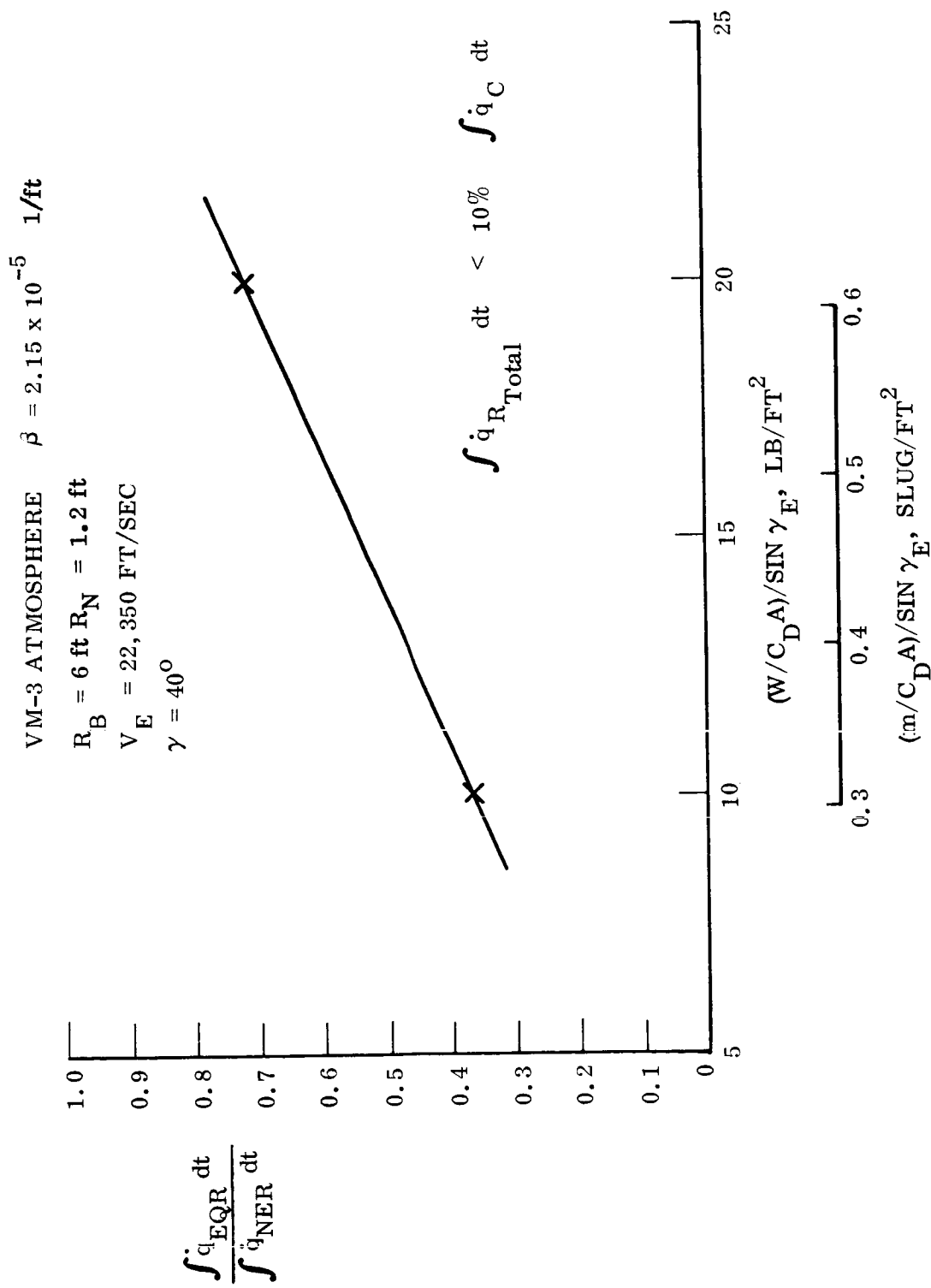


Figure 3.1.2-96. Ratio of Equilibrium to Non-equilibrium Time Integrated Heating for the Stagnation Point

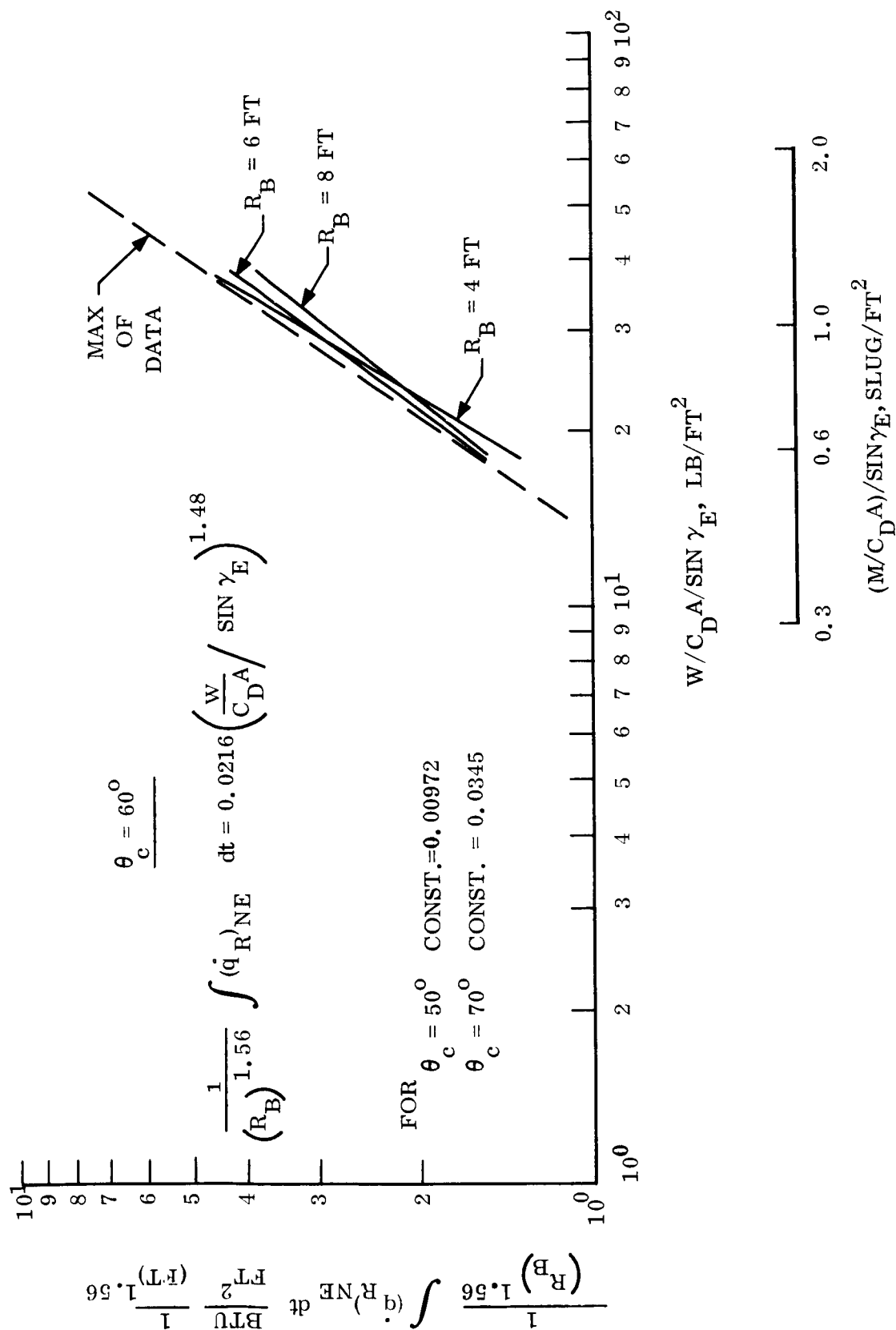


Figure 3.1.2-97. Integrated Non-equilibrium Radiant Heating for the Conical Frustum

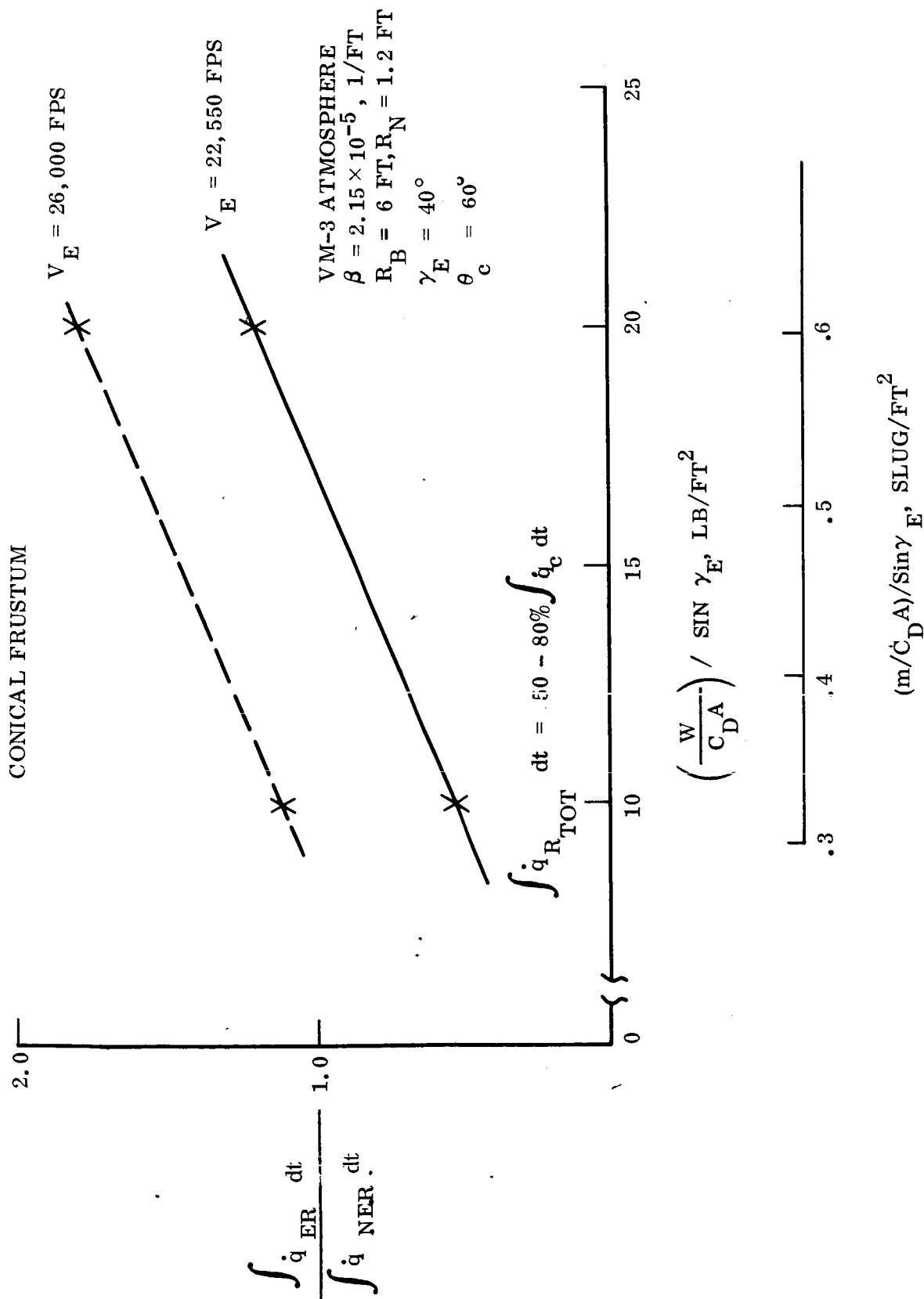


Figure 3.1.2-98. Ratio of Equilibrium to Non-equilibrium Time Integrated Radiative Heating for the Conical Frustum

These approximations were input to the CONSEP program for estimating the non-equilibrium radiative heating for the parametric design studies.

The Effect of Argon Presence in a CO_2N_2 Mixture on Equilibrium Radiant Heating - A significant quantity of argon may be present in the Martian atmosphere which, as discussed in refs. 3-42, -43 and -44 has three main effects on the level of equilibrium radiant heating: (1) The temperatures in the shock layer containing argon will be higher due to the monatomicity of argon. This higher temperature will mean higher radiative emission; (2) It was observed in ref. 3-42 that the CN concentration was lower for the argon case and also the peak CN concentration occurs at a lower temperature (or flight velocity). Since CN is one of the strongest contributors among the molecular band radiators, the equilibrium gas radiation will be reduced in the temperature range where this type of radiator is important; (3) The emissivity of argon increases faster with temperature than that of nitrogen. At lower speeds argon's emissivity is insignificant. At 10,000°K nitrogen and argon have approximately the same radiance (ref. 3-42). At higher temperatures and velocities the radiance of argon is greater than that of nitrogen.

In the lower velocity range from 14,000 to 21,000 fps, the argon emissivity is low. For the gas mixture with argon in it, the temperature is higher but there is less CN present (the CN molecular band radiation is important in this velocity range). These last two effects tend to cancel each other out and the equilibrium heat transfer is not changed much (ref. 3-42). In the velocity range from 21,000 to 28,000 fps, the radiation level for the mixture including argon is lower due to the lower concentration of the very strong molecular band radiator CN. Above 28,000 fps, the radiance of the mixture containing argon is an order of magnitude higher than the $\text{N}_2 - \text{CO}_2$ mixture due to the high emissivity of argon in this temperature range.

Therefore, it appears that predictions using the $\text{N}_2 - \text{CO}_2$ gas mixture will be good up to 28,000 fps and will be conservative in the range of 21,000 to 28,000 fps with respect to heat shield design.

B. Material Response

Preferred Approach - Several generic types of materials can be considered as potential candidates for a Mars entry vehicle thermal protection system. These include re-radiative, heat sink, low temperature sublimers, and high or low density charring ablators. The relative advantages and disadvantages of these various systems have been evaluated in detail for the out-of-orbit Voyager Lander studies (ref. 3-36). From a minimum weight basis, a hot beryllium heat sink can be made competitive with the class of low density charring ablators designed to a maximum bondline temperature of 800° F. However, the heat sink has the disadvantage of being "less forgiving" than the charring ablator to an off-nominal entry mission.

At the heating rates expected for the out of orbit and direct entry cases, it was shown in ref. 3-45 that, in general, the most efficient material on a weight basis, is a member of the low density ablator class, e.g. a silicone elastomer.

During the past several years, General Electric has been developing a class of low density ablators. This ESM 1000 series consists of fiber-filled foamed silicone and silicone-epoxy elastomers available over a density range of 14 to 60 lbs/ft³. In cases of relatively high entry heat flux and aerodynamic shear conditions, the elastomer can be further supported by incorporation into a discontinuous, specifically oriented, phenolic glass matrix. The methyl-phenyl silicone base elastomer has a glass transition temperature of -180° F and has excellent high temperature stability and environmental resistance. The base elastomer is also used as the bonding agent and permits the efficient use of high temperature structural materials. The ESM system has been fully characterized by extensive ground testing and data are available for analysis by each of the GE thermodynamic computer programs. Thus, because of the relatively mild Martian entry environment where total surface recession is small in most cases and thermal insulation is of prime importance, the low density charring ablators, ESM 1004 X and ESM 1004 AP, were selected as a design solution for the Mars Hard Lander heat protection system on the basis of

- Minimum system weight
- Maximum flexibility to accommodate non-design conditions
- Fewer anticipated fabrication and development problems than with the other approaches
- Sensitivity to the sterilization/decontamination and low temperature/hard vacuum environments can be circumvented by proper material formulation.

For the relatively mild thermal environment of the out of orbit entries the low density ESM 1004 X ($\rho_v = 16 \text{ lbs/ft}^3$) is the preferred shield material, since no surface recession is anticipated even for the steep VM-8 entries and on a unit weight basis this results in a lighter weight shield.

For the direct entry missions where steep entry into VM-8 results in ESM surface melting, ESM 1004 AP ($\rho v = 36 \text{ lb/ft}^3$) is the preferred approach due to its good performance in relatively mild melting regimes.

Mathematical Simulation of ESM - After defining the range of expected aerodynamic heating, the in-depth temperature response and subsequent mass loss can be calculated by use of the well calibrated charring ablator mathematical model, REKAP, given a heat flux history and selected material thicknesses.

The primary parameters affecting the thermal response of a silicone elastomer charring ablator are as follows:

- 1) Maximum instantaneous heat flux: for $q_c \text{ LAM} < 100 \text{ Btu/ft}^2 \text{ sec}$ (surface temperature $< 3400^\circ \text{R}$) the SiO_2 and SiC char surface behaves as an efficient re-radiator, for a surface temperature $> 3400^\circ \text{R}$ the char surface melts or vaporizes resulting in surface recession. Data exists for defining the char mass loss rates as a function of surface temperature.
- 2) Total time integrated heat flux: as the total heating load is increased, a greater thickness of heat shield is required to hold the maximum allowable bondline temperature.
- 3) Length of time during which heating is experienced: for a given total heat load, as the heating time is increased, the heat shield requirement is again increased.

A general description of the REKAP mathematical model of the ESM materials is given in Appendix A. Table 3.1.2-5 shows the thermophysical properties employed in REKAP for ESM 1004 X and ESM 1004 AP.

Generation of Parametric Results - From approximate entry analysis techniques, it is known that the maximum instantaneous heat flux is a function of vehicle configuration, entry velocity, V_E , atmospheric model (gaseous composition and density scale height) and the product $W/C_D A \sin \gamma_E$. Hence, the silicone elastomer melt regime can be parametrically defined. The total time integrated heating load is a function of the same parameters as the maximum instantaneous heat flux. Hence, given a range of possible vehicle configurations, the total time integrated heating loads (both convective and radiative) can be parametrically defined across the entry corridor.

The heating time is dependent only on the density scale height of the model atmosphere and the product $V_E \sin \gamma_E$. Hence, for a given atmosphere, the range of heating times can be established. The range of entry path angles, γ_E , provides the greatest effect on the heating time.

The parametric heat shield requirements provided for use in CONSEP are illustrated in figs. 3.1.2-99 and -100 for both ESM 1004 AP and ESM 1004 X.

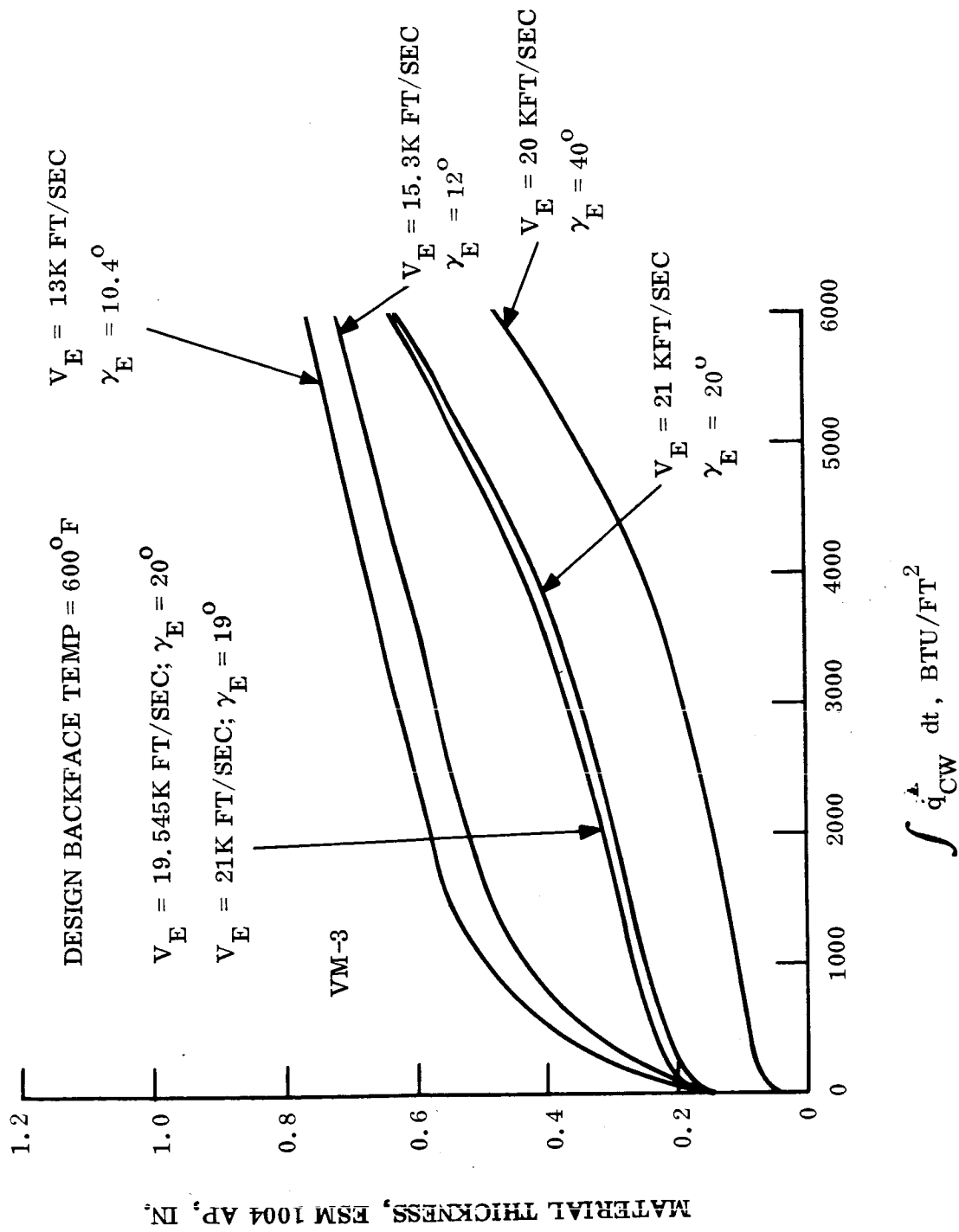


Figure 3.1.2-99. Shield Thickness Requirements of ESM 1004AP for
Several Entry Conditions

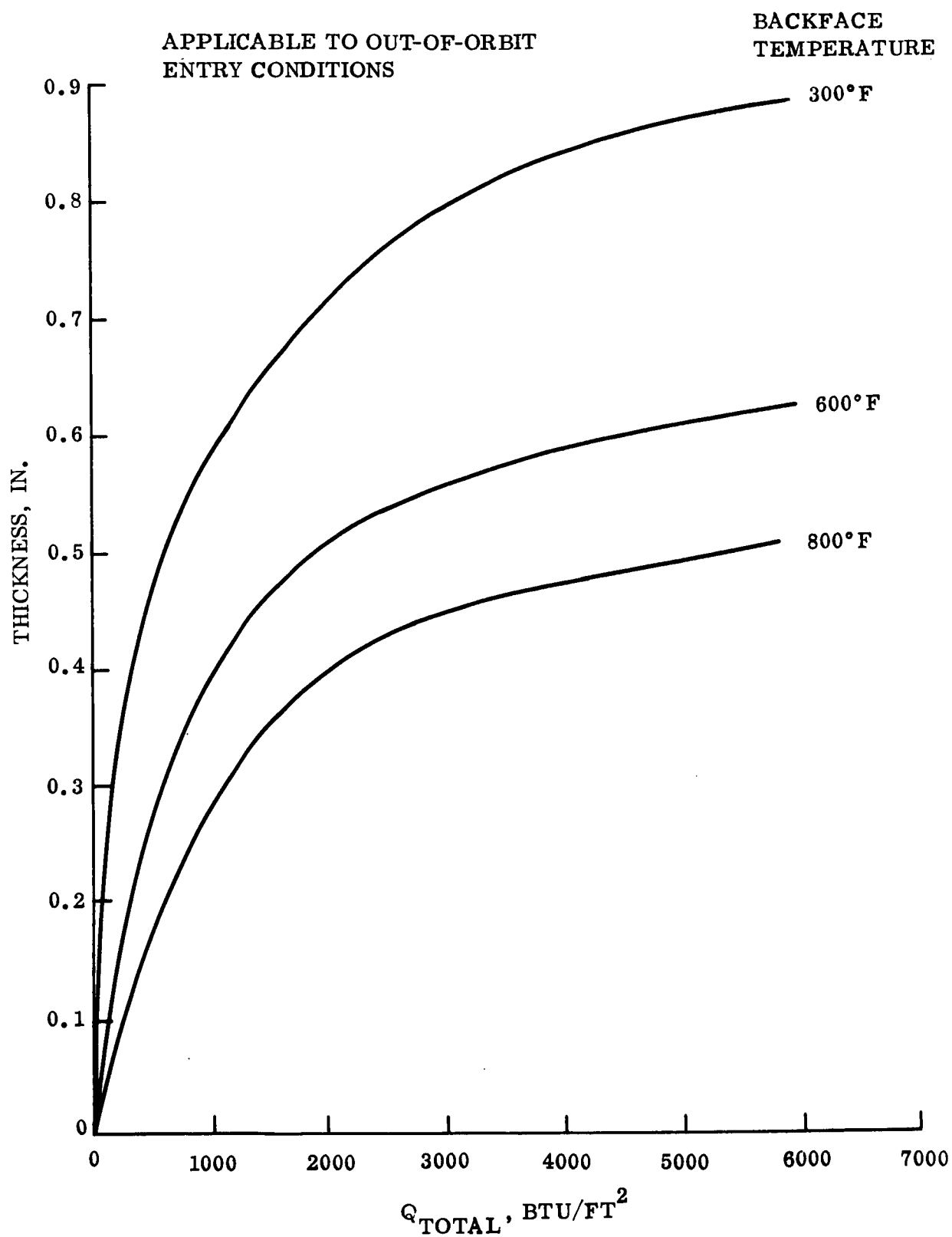


Figure 3.1.2-100. Shield Thickness Requirements for ESM 1004 X
with Adiabatic Backface

Heat shield requirement curves were generated for several heating times, i.e., combinations of entry path angles, and entry velocities. The flow chart shown in fig. 3.1.2-101 illustrates the development of the final shield thickness and unit weight in the CONSEP program.

C. Material Characteristics

The two materials selected as the preferred approach to the thermal protection of the out-of-orbit and direct entry Mars Hard Lander vehicles are ESM 1004 X and ESM 1004 AP, respectively, both developed at General Electric. The ESM 1000 series are char forming materials consisting of chemically blown, phenyl methyl silicone elastomeric foams. The ESM 1004 materials contain from 6 to 20 percent of fibrous aluminum silicate and are available over a wide range of densities from 15 to 60 lb/ft³ depending on the concentration of blowing agent and curing conditions. The specific formulation ESM 1004 AP uses the base elastomer RTV 560, contains 12 percent aluminum silicate fibers and has a density of 35 lb/ft³. ESM 1004 X is an unsupported foamed, methyl phenyl silicone reinforced with iron oxide and aluminum silicate fibers. It has been scaled up to full size sheets and its manufacturing cycle is similar to the more familiar ESM 1004 AP, including a postcure at 300° F.

The ESM can be fabricated with a molded nose cap to eliminate seams in the stagnation area. The remainder of the vehicle should be covered with cut sheet stock joined on a diagonal to the direction of the aerodynamic flow.

ESM has good electrical properties, and has been demonstrated in test to produce a non-carbonaceous, non-conductive char which is transparent to RF signals at ambient temperatures. (See ref. 3-36).

The ESM 1004 X formulation was selected for the out of orbit entry model because of its low density (14-16 lb/ft³) and improved thermal conductivity over more familiar ESM 1004 formulations. It is soft bonded to the structure after fabrication, with RTV 560 adhesive.

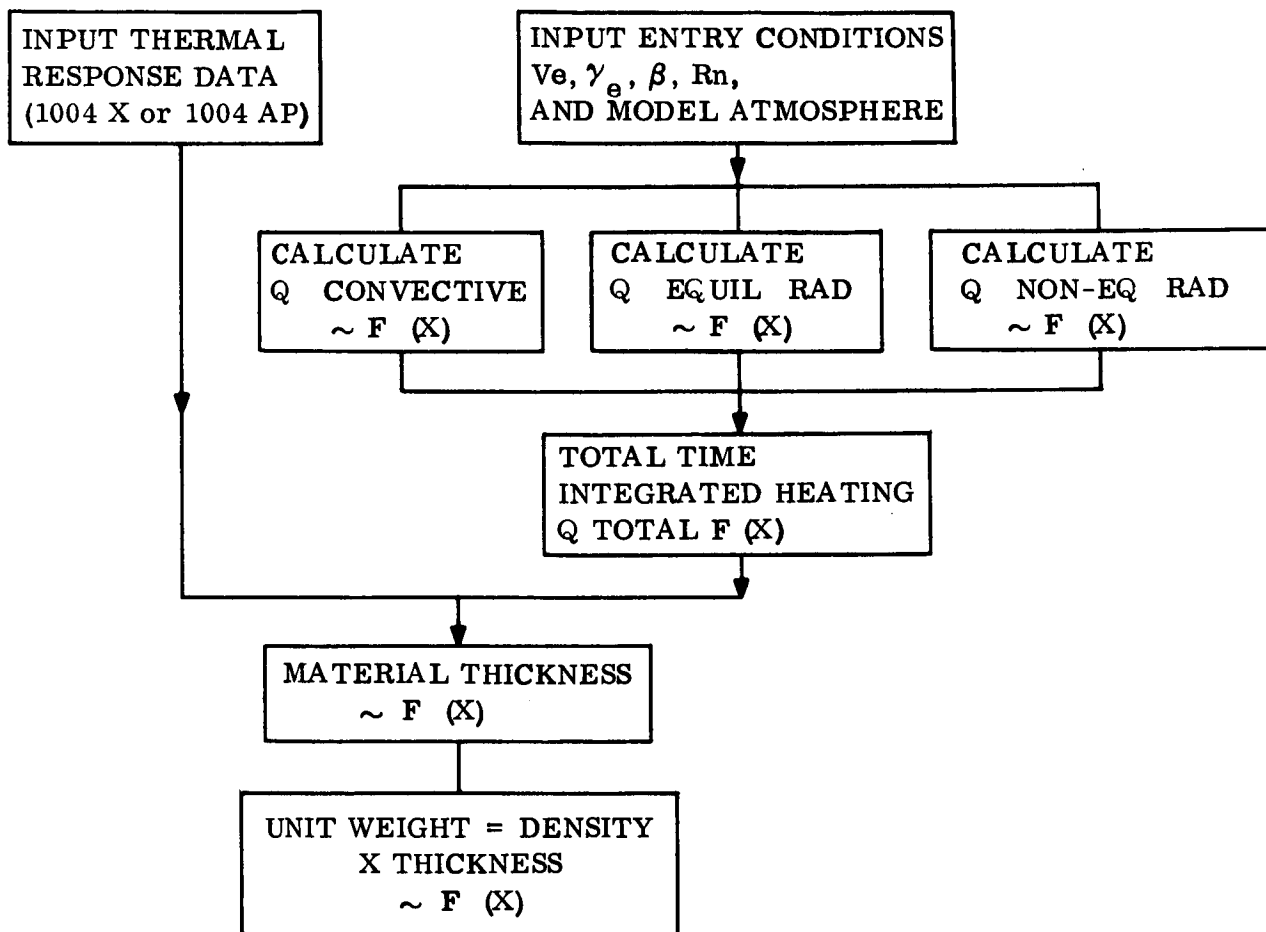


Figure 3.1.2-101. Flow Chart of Development of Shield Thickness Requirements in CONSEP Program

3.1.2.1.4 Structural Performance

The structural weight of the entry aeroshell is a major element of the overall entry vehicle system weight and, as such, is critical in determining vehicle performance. The aeroshell is composed of two elements: the heat shield, for entry heat dissipation and protection of Lander, and the substructure which supports the heat shield and the Lander under aerodynamic pressure and inertia loading during entry deceleration. The primary criteria for aeroshell design is the temperature at the heat shield and substructure interface/bondline. Trade-offs are required to minimize the total aeroshell weight for this parameter so as to derive an optimum structural configuration of suitable construction and material.

For entry into Martian atmosphere, with characteristic low density and pressure, a blunt body sphere cone configuration is required for the aeroshell. Parametric studies of sphere cones of half apex angles between 50° and 70° have been studied by using CONSEP (ref. 3-46) program. Based on these studies, a 60° half cone angle has been chosen as the baseline for structural optimization studies.

The structural performance requirements are directly related to the entry mode, the trajectory parameters, the aeroshell geometry and the entry vehicle weight. Both the direct and out-of-orbit entry modes are considered for deriving critical performance requirements. Of the trajectory parameters, the variation of Martian standard atmosphere model influences the structural requirements. The aerodynamic deceleration is critical for the VM-8 atmosphere and dictates the maximum pressure and inertia loading. On the other hand, a VM-3 atmosphere yields the maximum thermal and temperature environment to which heat shield and structure are subjected. An important consideration in evaluating structural performance is the time gap between the occurrence of maximum loading and maximum temperature. Although maximum heating rate occurs before maximum loading, temperatures at maximum loading are lower than maximum temperatures experienced during flight. Maximum temperatures occur towards the end of the trajectory where loading as well as structural strength at elevated temperatures, is lowered, although not necessarily by the same ratio.

In addition to strength requirements, the aeroshell must show adequate aerothermoelastic stiffness. The aerothermoelastic effects have been studied in detail in Ref. 3-47 for the sphere-cone configuration, but the type of study conducted does not lend readily to this parametric study. Ref. 3-47 shows that with the aeroshell structure designed for strength and stability under entry loadings, the aerothermoelastic stiffness requirements are satisfied.

A. Structural Requirements and Criteria

For ballistic entry vehicle aeroshells, the critical structural loadings are aerodynamic pressure and axial inertia loads during entry deceleration, combined with the heating of the structure. The loading is a function of the entry mode and entry parameters such as the ballistic coefficient, entry velocity, and entry path angle.

Trajectory analyses show that, for the aeroshell configurations considered, the maximum axial deceleration occurs simultaneously with peak dynamic pressure. For simplicity of preliminary analysis, the entry vehicle weight is assumed to be uniformly distributed over the length of the aeroshell. The conical aeroshell structure is designed to withstand the combined pressure and inertia loading with operating temperature considered as a parameter. The loading is increased by an ultimate factor of 1.25, while the temperature is taken to be the unfactored nominal value, with allowance being made in thermodynamic analysis to obtain a conservative estimate of the heat shield/substructure interface temperature.

B. Parametric Data Requirements

For the present study, the parametric analysis for the aeroshell must take into account the following variables and their ranges as shown below:

Trajectory Parameters:

Entry Mode	Direct and out-of-orbit
Entry Velocity	15,000 to 21,000 ft/sec
Entry Path Angle	16 to 40 deg
Ballistic Coefficient	3 to 20 lb/ft ² (0.01 to 0.625 slug/ft ²)

Configuration Parameters:

Base Diameter	7 to 15 ft
Cone Angle	60 ±10 deg
Bluntness Ratio	0.5 ±0.2
Shape	Sphere-cone

The critical parameter combinations of loading conditions and vehicle geometry considered are as shown with the interface temperature varied between 200 to 1000°F.

C. Approach

The structural analysis of the aeroshell is based on the theory for strength and stability estimation of blunt body sphere cones of large diameter and with large R/t (radius to thickness ratio of the shell) parameter values, subjected to external aerodynamic pressure and meridional inertia loading. The parametric analysis was carried out in three successive stages, with increasing refinement of analytical treatment being introduced to involve more significant detail considerations at each stage.

Stage 1: Initially, a gross analysis for overall system trade-offs with a large parametric matrix was carried out using GE's CONSEP IV program. This included variation of trajectory, aerodynamic, flight dynamic, aerothermodynamic and design geometry parameters.

Stage 2: Critical trajectory and geometry parameter combinations were then selected for refinement of structural analysis and optimization using GE-MSD's Structural Internal Load and Optimization program (ref. 3-48). (SILC-SILO.)

Stage 3: Finally, the effects of structural weight were determined for analytical treatment of significant detail considerations. These included local shell bending due to Lander (decelerator load), inertia, and aerodynamic loading; and thermal gradients between honeycomb sandwich faces, both of which increase structural weights. On the other hand, recognition of the fact that maximum loading occurs before maximum temperature, and that loading at maximum temperature is appreciably lower, offers some weight saving. Such extended detail analysis was based on a point design with structure shield interface temperature as a parameter. The results of this phase, using GE's MULTISHELL program, are used as a final calibration of the trade-offs.

D. Analysis

Stage 1: Details of the approach for structural analysis of spherecone shells used for the CONSEP studies are described in Appendix A.2.

Stage 2: Structural Configuration Selection - Construction and Material - Various structural configurations were studied using different materials and constructions:

<u>Type of Construction</u>	<u>Materials</u>
Monocoque	Aluminum Alloys 2024 - T4
Honeycomb Sandwich	Aluminum Alloys 7075 - T6
Ring Stiffened	Magnesium Alloy HK31A-H24
Waffles	Titanium Alloy, 6AL-4V
Corrugation Stiffened	Beryllium Alloy
Corrugation Sandwich	Stainless Steel 15 Ph. 7 Mo.
	Fiberglass, Phenolic

Types of construction are schematically shown in fig. 3.1.2-102. Mechanical strength properties used are summarized in table 3.1.2-6. Minimum gages and corresponding structural weight densities used for various configurations are shown in tables 3.1.2-7 and -8.

The critical combinations of geometric and trajectory conditions investigated are shown in table 3.1.2-9, with aeroshell structural geometry as in fig. 3.1.2-103.

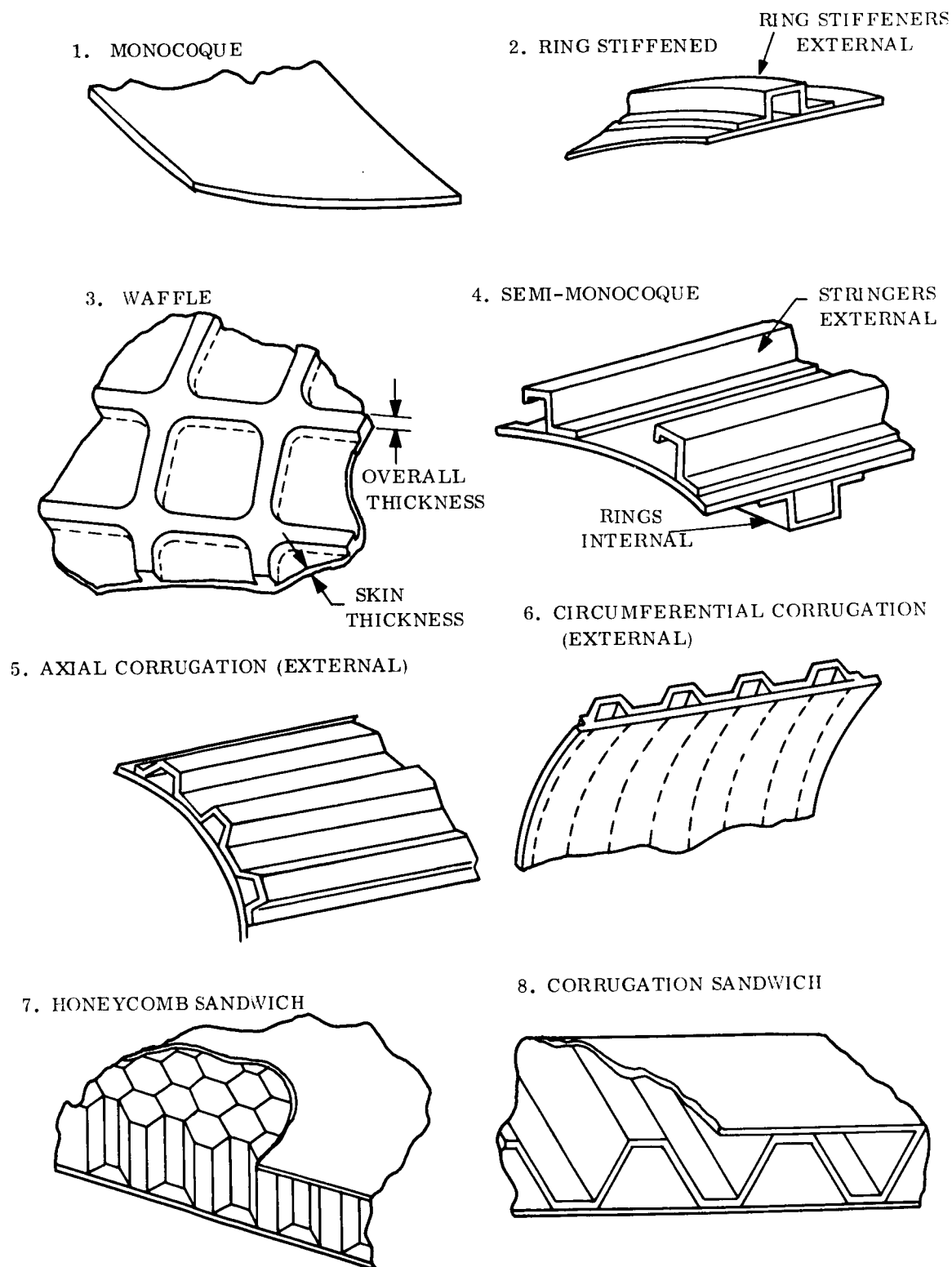


Figure 3.1. 2-102. Types of Structural Construction

TABLE 3.1.2-6. MECHANICAL STRENGTH PROPERTIES OF MATERIALS CONSIDERED

Material	Temperature °F	Ult Tensile Strength F _{tu} , 10 ³ psi	Tens Yield Strength F _{ty} , 10 ³ psi	Comp Yield Strength F _{cy} , 10 ³ psi	Young's Modulus E, 10 ⁶ psi	Poisson's Ratio ν	Density ρ #/in ³
Aluminum (2024-T4)	-300 RT 250	70 ¹ 62 54.6	52 ¹ 40 37	52 ¹ 40 37	11.7 10.5 10.19	0.3 ¹ 0.33 0.33	0.10
Aluminum (7075-T6)	-300 RT 250	89.7 76 51.4	76.1 65 51.4	83 ¹ 67 61.6	11.3 10.3 9.48	0.3 ¹ 0.33 0.33	0.101
Titanium (6AL-4V)	-300 RT 250	201 134 116.5	196.6 126 105.8	182 ¹ 132 114.8	17.1 16 15	0.3 ¹ 0.29 0.29	0.160
Beryllium (X-rolled sht)	-300 RT ³ 250	70	50 43.4 ²	50 43.4 ²	42 41.6 ²	0.05	0.067
Fibreglass (Phenolic)	-300 RT ³ 250	40	20	20 37 ²	2.5 2.82 ²	0.22	0.066
Magnesium (HK31A-H24)	-300 RT 250	50 ¹ 34	29 ¹ 26 21.6	29 ¹ 20 19.2	7.0 ¹ 6.5 6.37	0.3 ¹ 0.35 0.35	0.065
Stainless Steel (PH15-7M0)	-300 RT 250	255 ¹ 223 210	240 ¹ 200 188	240 ¹ 210 200	31 ¹ 29 28.1	0.3 ¹ 0.3 ¹ 0.3 ¹	0.277

NOTES: 1. From NASA Report "Structural Systems and Program Decisions". Vol 1 (Ref. 16) (Apollo Program).

2. From GE Technical Info Series, "Ballistic Re-Entry Vehicle Thermo-Structural Optimization" TIS No. 62SD199 by T. E. Hess, J. Bailey and H. Most (ref. 3-50).

3. From Material Producers: Beryllco & Brush Beryllium Corp.

4. Unmarked values taken from Military Handbook 5 are S-basis or A-basis (whichever one applies) rather than B-basis (ref. 3-51).

TABLE 3.1.2-7. COMPARISON OF MATERIAL CHARACTERISTICS: MINIMUM GAUGES AND STRENGTH AND STIFFNESS TO WEIGHT RATIOS

Material Specification	Aluminum		Titanium 6Al 4V	Beryllium Cross-Rolled Sheet	Magnesium HK31A-H24	Phenolic Fiberglass	Stainless Steel Ph 15-7 Mo	Lock Alloy
Material Density (lb/in. ³)	0.100	0.101	0.160	0.067	0.065	0.066	0.277	0.076
Minimum Skin Thickness (in.)	0.012	0.012	0.008	0.020	0.016	0.030	0.008	0.020
Minimum Face Thickness (Sandwich Construction) (in.)	0.010	0.010	0.005	0.010	0.010	0.010	0.005	0.010
Ultimate Tensile Strength (lb/in. ² x 10 ³)	62	76	134	70	34	40	223	60
Elastic Modulus (lb/in. ² x 10 ⁶)	10.5	10.3	16	42	6.5	2.82	2.9	18.5
U.T.S./density (in. x 10 ³)	620	753	838	1045	523	606	805	790
Modulus/density (in. x 10 ⁶)	105	102	100	627	100	38	105	375

TABLE 3.1.2-8. WEIGHT/DENSITY IN LB/FT² FOR MINIMUM GAUGE DESIGNS USING VARIOUS MATERIALS AND CONSTRUCTIONS

Material/Alloy Construction	Fb, Fabrication Factor Included	Al 2024	Al 7075	Ti	Be	Mg	PFG	SS	LA
Monocoque	1.05	0.1814	0.1830	0.1936	0.2026	0.1572	0.2993	0.3342	0.2408
Ring Stiffened	1.20	0.2311	0.2334	0.2465	0.2581	0.2003	0.3813	0.4075	0.2928
Semi-Monocoque	1.20	0.2916	0.2945	0.3110	0.3256	0.2527	0.4811	0.5142	0.3694
Waffle	1.4	0.3087	0.3117	0.4938	0.2068	0.2006	0.2037	0.8550	0.2346
Honeycomb Sandwich	1.25	0.4041	0.4077	0.3321	0.2853	0.2781	0.2648	0.5427	0.3177
Axial or Circumferential									
Single-Face Corrugated	1.20	0.5344	0.5398	0.5701	0.5968	0.4632	0.8818	0.9424	0.6769
Double-Face Corrugated Sandwich	1.20	0.5433	0.5488	0.4347	0.3640	0.3532	0.3586	0.7525	0.4129

Notes: 1. For waffle construction, minimum over-all thickness is 0.125 and minimum skin thickness is 0.0125 for all metallic materials

2. Minimum thickness for stiffeners (rings, stringers, and corrugations) are taken to be same as skin thickness.

TABLE 3.1.2-9. VEHICLES ANALYZED

Vehicle No. Loading Condition	<u>1</u>	<u>2</u>	<u>3</u>	<u>4</u>	<u>5</u>	<u>6</u>	<u>7</u>	<u>8</u>	<u>9</u>
Diameter (ft)	7	7	7	7	9	15	15	15	15
γ E Path Angle (deg)	40	40	25	25	40	40	40	25	25
β (lb/ft ²)	3	5	3	8	10	3	5	3	8
V_E (kft/sec)	20	20	19	19	20	20	20	19	19
G_x (earth g's)	86	80	44	42	68	86	80	44	42
Alt. (kft)	64	61	74	62	--	64	61	74	62
Weight (lb)	168	280	158	448	923	772	1287	772	2060
Length (ft)	1.75	1.75	1.75	1.75	2.25	3.75	3.75	3.75	3.75
Stagnation Pressure (lb/in. ²)	3.38	5.22	1.69	4.32	8.55	3.28	5.22	1.69	4.32
Skirt Pressure (lb/in. ²)	2.59	4.03	1.33	3.41	6.74	2.59	4.03	1.35	3.41

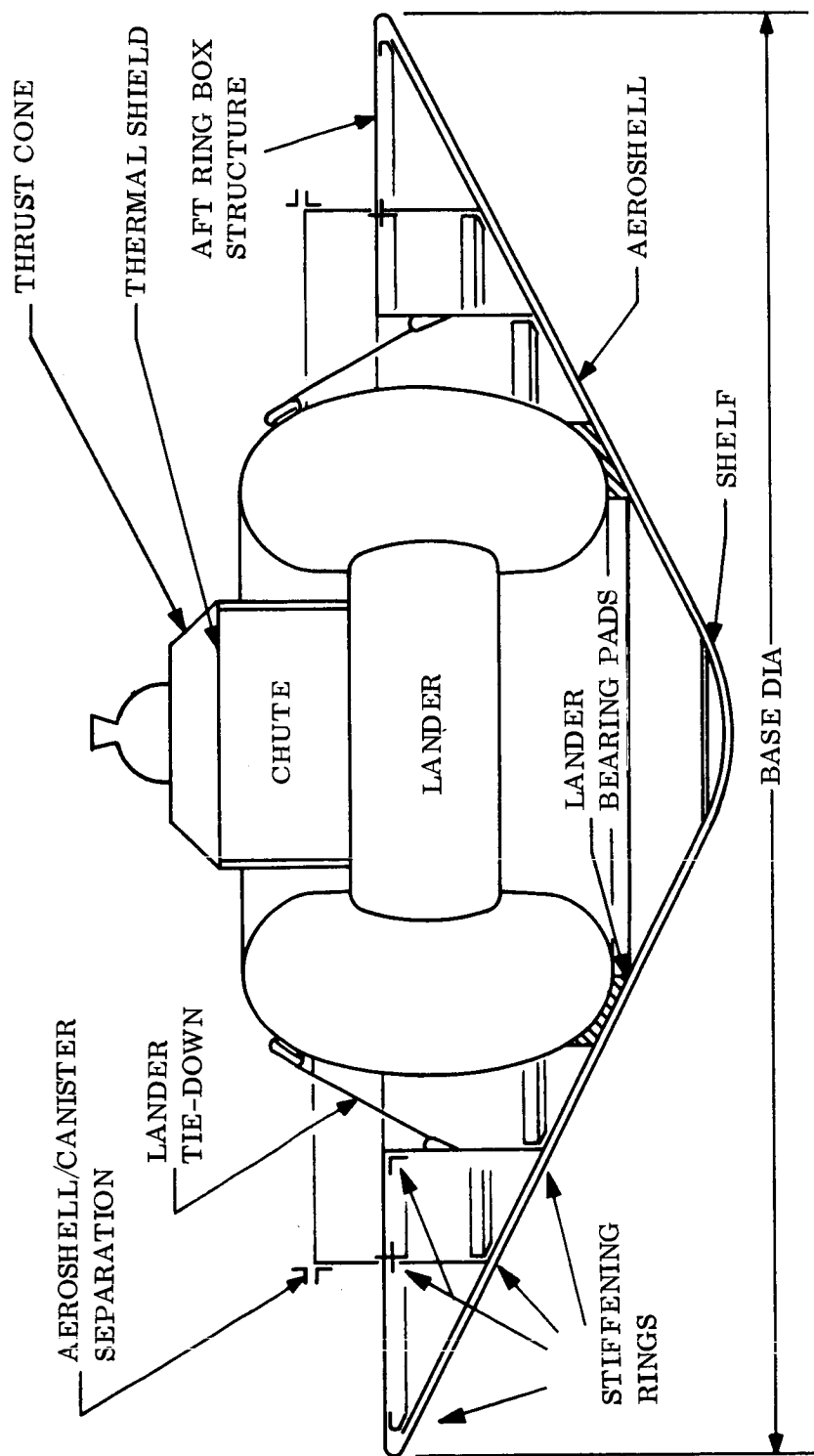


Figure 3.1.2-103. Aeroshell Structural Geometry

The SILC-SILO program (ref. 3-48) determines the internal loads (meridional and hoop loading) in each geometric section of the shell subjected to the critical pressure and inertial loading. Optimum structural sizes required are then calculated for a range of temperatures for each type of structural configuration, taking into account both strength (based on a shell membrane theory) and buckling (thin shell theory) considerations (ref. 3-48). The interactions between the heat shield and structure and resulting thermal stresses are accounted for in the analysis. More suitable configurations were chosen to give minimum weight design and ease of fabrication. The structural weight densities are summarized for the critical parametric loading conditions analyzed (tables 3.1.2-10 and figs. 3.1.2-104 and 105. For the nose spherical cap, monocoque and honeycomb sandwich constructions were optimum; while for the conical frustum skirt, honeycomb sandwich and ring stiffened constructions give comparable structural weight densities. Corrugation based stiffened or sandwich constructions give inherently heavier structural weight densities, while waffle is not suitable for large structure, as envisaged here, from fabricability considerations. For the honeycomb shell, each geometric section is of constant thickness. In the ring stiffened shell, a ring is placed at the end of each section and the optimum spacing obtained. Fabrication factors are used on the computer calculated structural weight densities to make allowance for structural items such as doublers, fasteners, bonding, etc., which are not readily accountable.

From the materials listed, Aluminum, Titanium, Steel, and Beryllium alloys were chosen for comparison as each shows certain merits: Aluminum alloys being more easily fabricated, titanium and steel both have better high temperature capabilities, while beryllium offers the lightest structure even at high temperatures, but presents difficulties in fabrication. Magnesium and phenolic glass have very limited temperature and stiffness capabilities, respectively, and are therefore not included.

The loading conditions for the out-of-orbit entry mode are less severe than the direct entry mode. During direct entry, for minimum aeroshell weight (i.e., combined heat shield and structure), titanium honeycomb sandwich construction using minimum face and core thicknesses is optimum with stainless steel honeycomb as the second choice, again with minimum gage construction, followed by aluminum.

Taking into account the rapid decrease of the aerodynamic and inertial loading, and the short duration of the loading pulse, it is considered feasible to use aluminum honeycomb sandwich construction as a possible candidate with interface temperature reduced to the 200 to 300° F range. This approach offers a less costly design using conventional materials at a small weight penalty as a fraction of total entry vehicle weight.

TABLE 3.1.2-10. STRUCTURE WEIGHT DENSITY (PSF)

Temp.		Mat'l	Entry Angle (deg)			Base Diameter (in)			Temp.		Mat'l	Entry Angle (deg)			Base Diameter (in)												
			β (lb/ft ²)	β (lb/ft ²)	β (lb/ft ²)	7	8	25				40	9	10	40	15	3	40									
			*6.0 in. ring spacing			*6.0 in. ring spacing						*6.0 in. ring spacing			*6.0 in. ring spacing												
Al	SS	Al	0.37	0.70	0.61	0.61	0.59	1.08	0.43	0.82	0.60	0.70	0.67	0.90	0.67	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.40	0.70	0.64	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.44	0.70	0.68	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.48	0.70	0.72	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.52	0.70	0.76	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.56	0.70	0.80	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.60	0.70	0.84	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.64	0.70	0.88	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.68	0.70	0.92	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.72	0.70	0.96	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.76	0.70	1.00	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.80	0.70	1.04	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.84	0.70	1.08	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.88	0.70	1.12	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.92	0.70	1.16	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	0.96	0.70	1.20	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.00	0.70	1.24	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.04	0.70	1.28	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.08	0.70	1.32	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.12	0.70	1.36	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.16	0.70	1.40	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.20	0.70	1.44	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.24	0.70	1.48	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.28	0.70	1.52	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.32	0.70	1.56	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.36	0.70	1.60	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.40	0.70	1.64	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.44	0.70	1.68	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.48	0.70	1.72	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.52	0.70	1.76	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.56	0.70	1.80	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.60	0.70	1.84	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.64	0.70	1.88	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.68	0.70	1.92	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.72	0.70	1.96	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.76	0.70	2.00	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.80	0.70	2.04	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.84	0.70	2.08	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.88	0.70	2.12	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.92	0.70	2.16	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	1.96	0.70	2.20	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.00	0.70	2.24	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.04	0.70	2.28	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.08	0.70	2.32	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.12	0.70	2.36	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.16	0.70	2.40	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.20	0.70	2.44	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.24	0.70	2.48	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72	1.30	0.38
Al	SS	Al	2.28	0.70	2.52	0.61	0.44	0.69	0.44	0.67	0.54	0.70	0.68	0.90	0.65	0.70	0.86	2.62	1.99	0.61	0.32	0.61	0.70	0.66	1.72</		

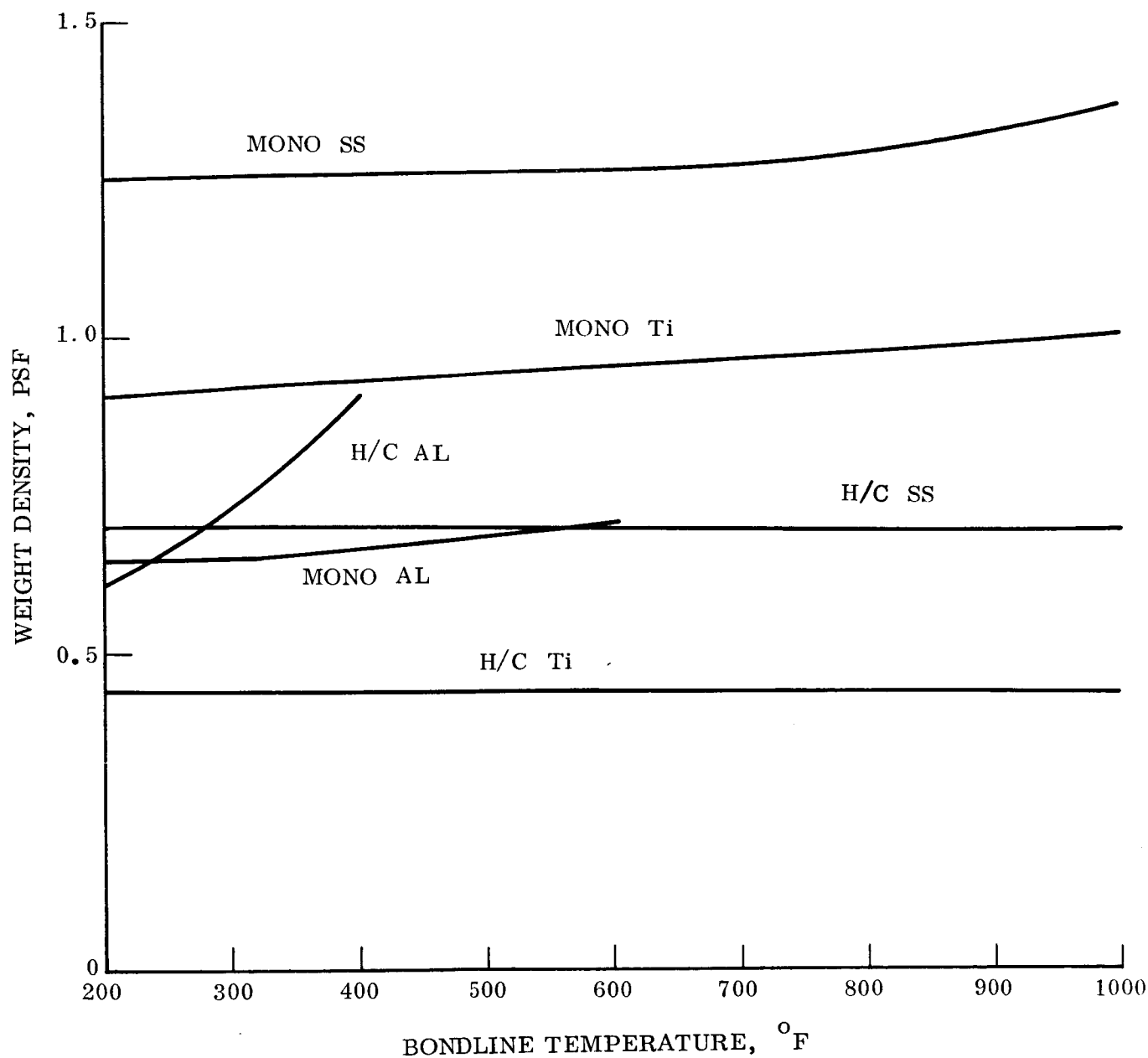


Figure 3.1.2-104. Weight Density of Nose Structure for Direct Entry:
 $\gamma = 40^\circ$, $\beta = 10$ psf, $D = 9$ ft

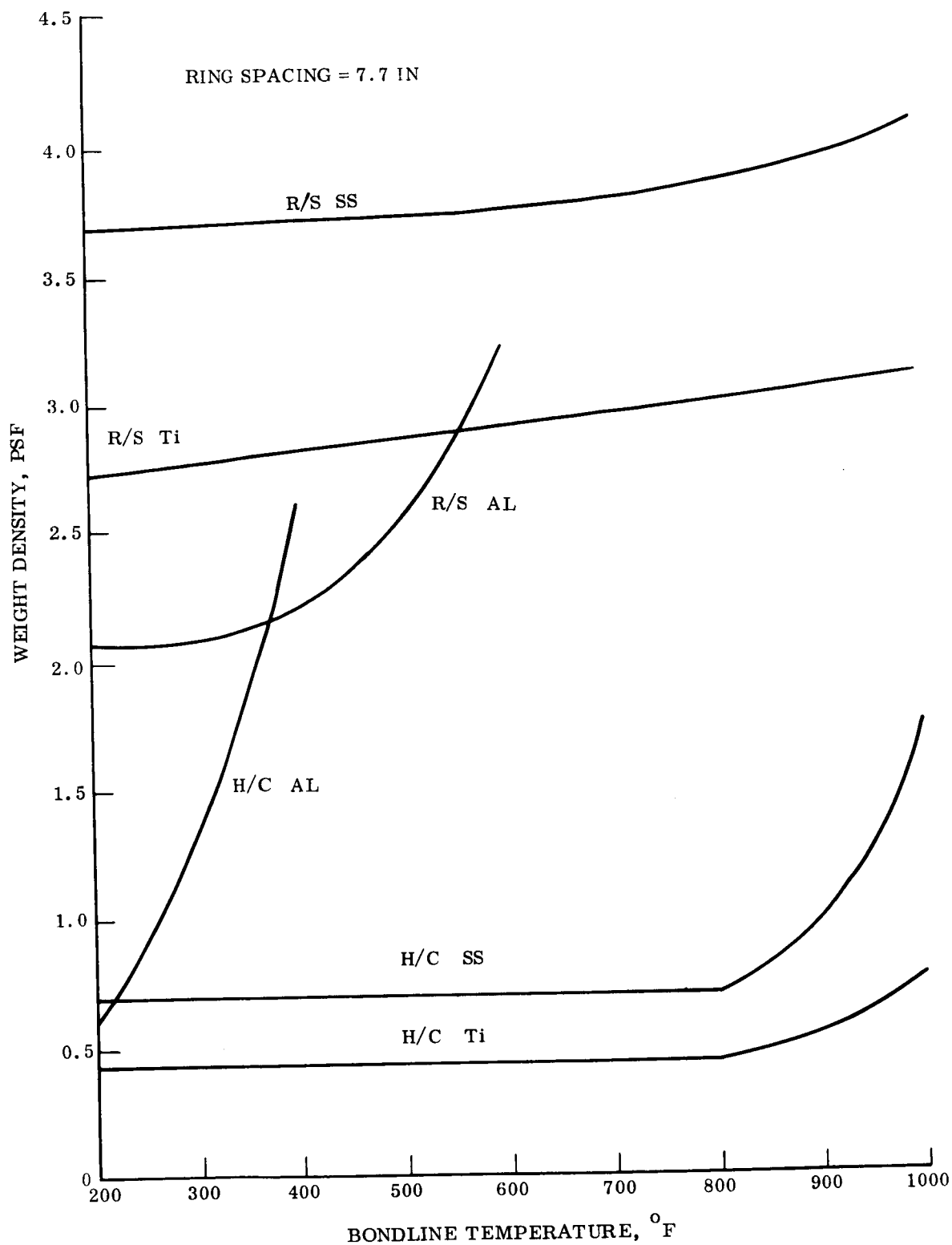


Figure e 3.1. 2-105. Weight Density of Skirt Structure for Direct Entry:
 $\gamma = 40^\circ$, $\beta = 10$ psf, $D = 9$ ft

D. Discussion

Stage 1 CONSEP Analysis. The overall parametric analysis was carried out using the CONSEP Program for charting the entire ranges of the parametric matrix. Results are detailed in para 3.1.2.2.

Stage 2 Structural Concept Optimization Analysis (SILC-SILO). Nine combinations of vehicle sizes, ballistic parameters, and entry angles were considered (table 3.1.2-9). The vehicles are all of a nominal 60° half-cone angle, 0.5 bluntness ratio, and have a zero angle-of-attack during entry. The structure was sized for entry into VM-8 atmosphere, whereas the shield was sized for entry into VM-3 atmosphere. A limit factor of 1 was used for shield and structure with yield criteria. The weight of each entry vehicle is assumed distributed along the structure. The distributed weight is assumed proportional to the vehicle radius or station.

The structure weight density is listed in tables 3.1.2-10 and is given for the spherical nose and conical skirt separately. For isolated cases (e.g., - low ballistic parameter or large vehicle diameter) the minimum weight structure may be a monocoque structure in the nose or a beryllium ring stiffened structure in the skirt. However, usually honeycomb structure is the optimum. Ignoring Beryllium because of fabricability considerations titanium honeycomb is the lightest, followed by stainless steel and aluminum alloy honeycomb. For both, the titanium and steel face sheets are 0.005 in. thick and the core thickness is 0.125 in. Large structures, such as shrouds, have been manufactured by Solar Corp. in titanium honeycomb and prove feasibility of the construction. The weight densities of tables 3.1.2-10 do not include the weights of rings at the tangency point and at the base of the vehicle. In order to include the ring weights, the structure weight densities should be increased by about 0.1 lb/ft^2 .

The structural weight densities shown include fabrication factors of:

<u>Type of construction</u>	<u>Monocoque</u>	<u>Honeycomb sandwich</u>	<u>Ring stiffened</u>
Fabrication factor	1.05	1.25	1.20

Figs. 3.1.2-104 and 105 show the variation of structure weight density with operating temperature for the 9 feet diameter structure. Figs. 3.1.2-106 and -107 show the variation of total weight density of structure and shield. The shield is ESM 1004 AP of 35 lbs per cu ft density. The shield weight variation with temperature used is shown in fig. 3.1.2-108.

The weight density of stainless steel honeycomb (0.8 lb per sq ft) is the same as that of titanium corrugation structure used by McDonnell-Douglas for the Mars Voyager Phase B Study (ref. 3-49). However, that structure was designed for out-of-orbit trajectory loading and is incapable of sustaining the loads of direct entry considered here for the 9 ft vehicle.

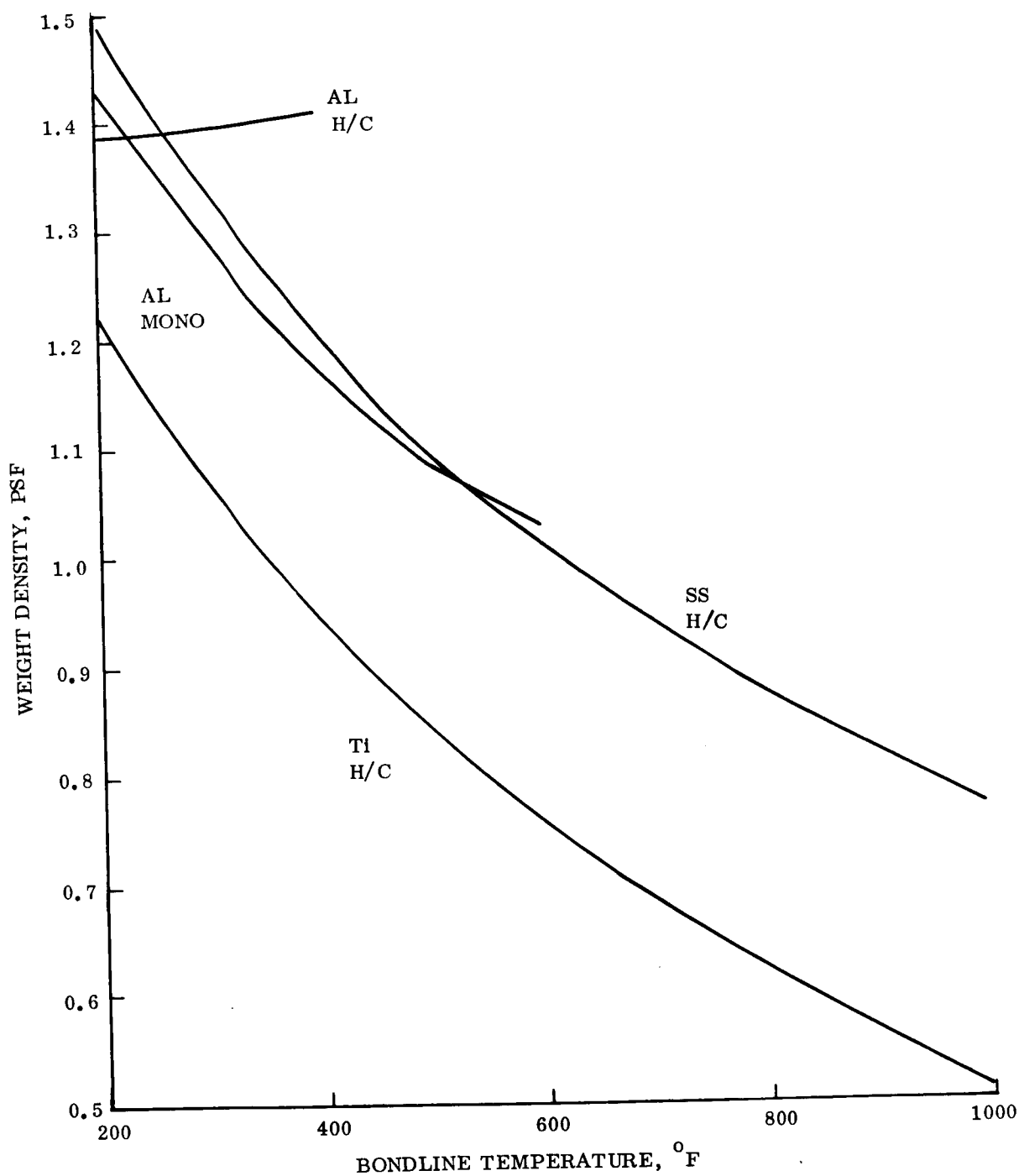


Figure 3.1. 2-106. Weight Density of Nose Structure and Shield for Direct Entry:
 $\gamma = 40^\circ$, $\beta = 10$ psf, $D = 9$ ft

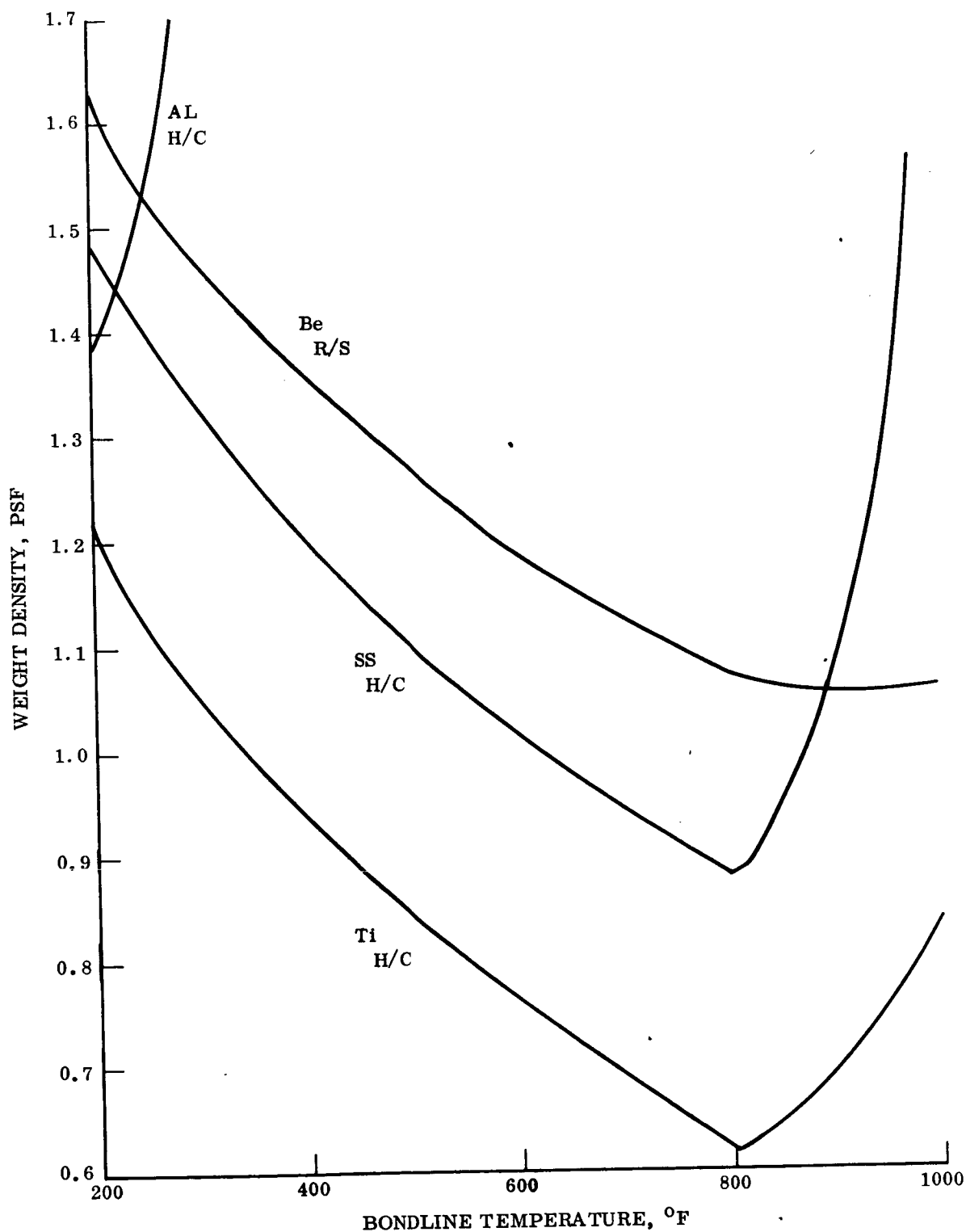


Figure 3.1.2-107. Weight Density of Skirt Structure and Shield for Direct Entry:
 $\gamma=40^\circ$, $\beta=10$ psf, $D=9$ ft

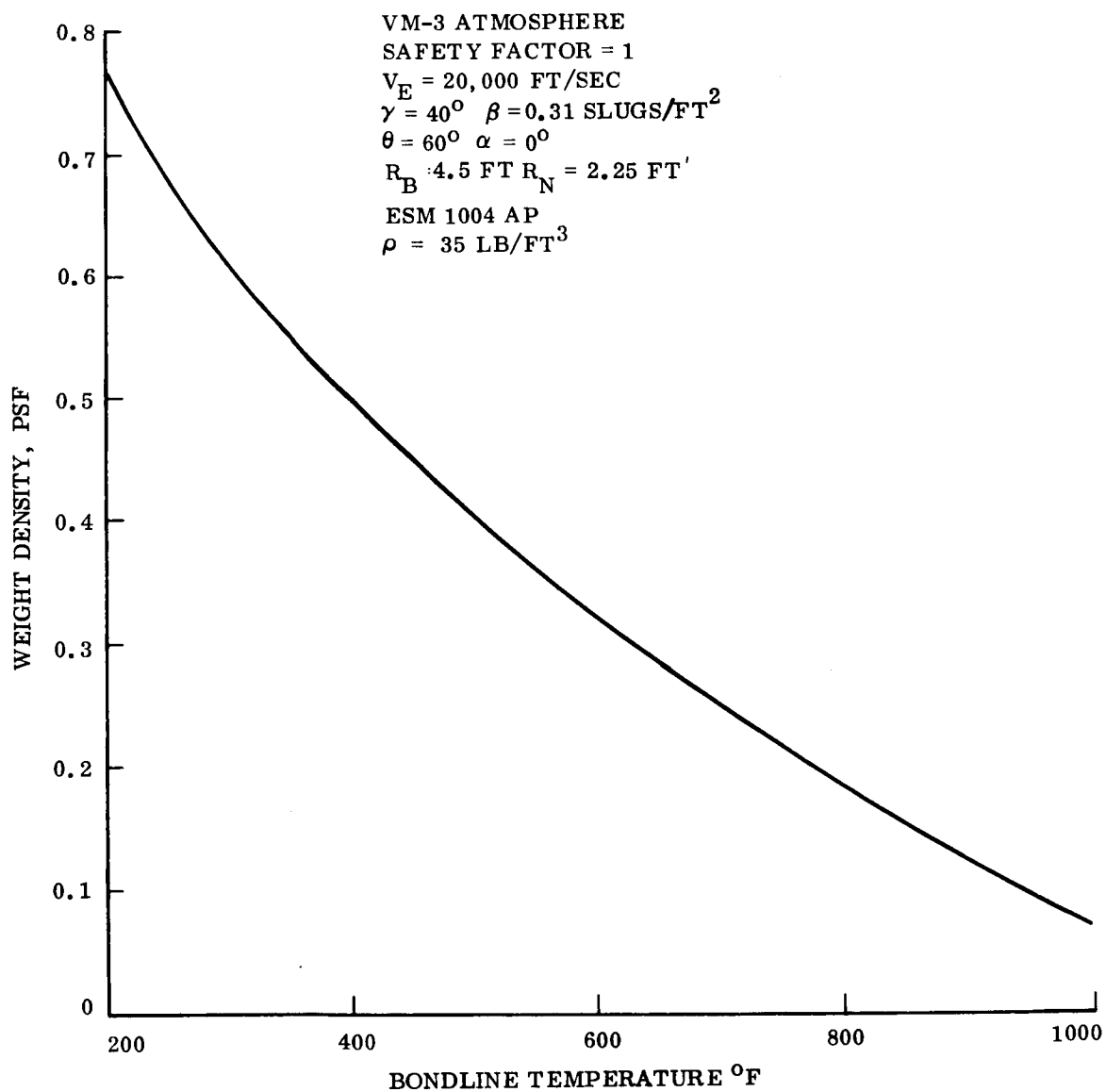


Figure 3.1.2-108. Shield Weight Density of Nose and Skirt for Direct
 Entry: $\gamma = 40^\circ$, $\beta = 10 \text{ psf}$, $D = 9 \text{ ft}$

Stage 3 - Extended Analysis of Honeycomb Sandwich Shell Concept. In this stage of analysis those structural concepts which appeared to potentially offer the least weight were analyzed in further detail. The concepts were all honeycomb sandwich in four materials: 7075-T6 aluminum, 2024-T4 aluminum 6A1-4V titanium, and 15 Ph-7 Mo stainless steel. The actual detailed analysis was done for the point designs and these results were used to calibrate, that is, to refine the parametric trends.

A major result of this stage of analysis was that shell bending in the area of the lander attachment required increased shell sizing over that calculated from membrane analysis. A detailed evaluation was performed for Point Design 2A (see Volume IV), at the time of maximum entry acceleration of 48 g's. The actual heat shield thickness on the aeroshell is such that at the time of peak loading the skirt shield-outer face structure temperature is only 100°F. The first comparison made was to determine the relative honeycomb weight required, excluding thermal stresses with the outer facing and inner facing at the source temperature and at the time of peak loading (48g's). The critical condition was shell bending in the meridional direction. This bending is due to the interaction between payload inertia loading and the aerodynamic pressure loads. The meridional bending stresses override the hoop membrane stress levels and the core thickness required precludes any buckling of the aeroshell. Fig. 3.1.2-109 shows the comparative structural weight requirements for different structural temperatures. This trade-off indicates very little advantage of one material over the other if the max. loading occurs below 200°F. Fig. 3.1.2-110 shows the comparative weights when the temperature gradients between the outer and inner facing of the honeycomb are included. The weights are approximately the same at lower temperatures, but the weight requirements for aluminum increase rapidly if the structure is allowed to operate at higher temperature at the time of max. loading. For all of the designs, the maximum differential temperature between the outer and inner facing was held to 200°F. In fig. 3.1.2-110, the temperature is that of the outer facings and at temperatures above 300°F, the inner facing is 200°F lower. When the thickness of the heat shield is designed so that the maximum interface temperature is 600°F during entry, the temperature for this direct entry aeroshell is between 100°F and 200°F at the time of max. loading. The actual Point Design 2A has 0.02 in. 7075-T6 aluminum facings with a 1.50 in. core thickness.

3.1.2.2 Ballistic Configuration Selection Trade-off Studies

A considerable portion of the entry vehicle study has been devoted to establishing basic entry vehicle design trends and associated design trade-off influences. The design data was generated by a high speed computer program (CONfiguration SElection Program, or CONSEP), expressly constructed for Mars/Venus entry analyses. A description of this program is presented in Appendix A. The CONSEP Program provides a complete description of each vehicle design in terms of external geometry, mass, and inertia properties; $W/C_D A$ and $m/C_D A$; stability margin, heat shield and

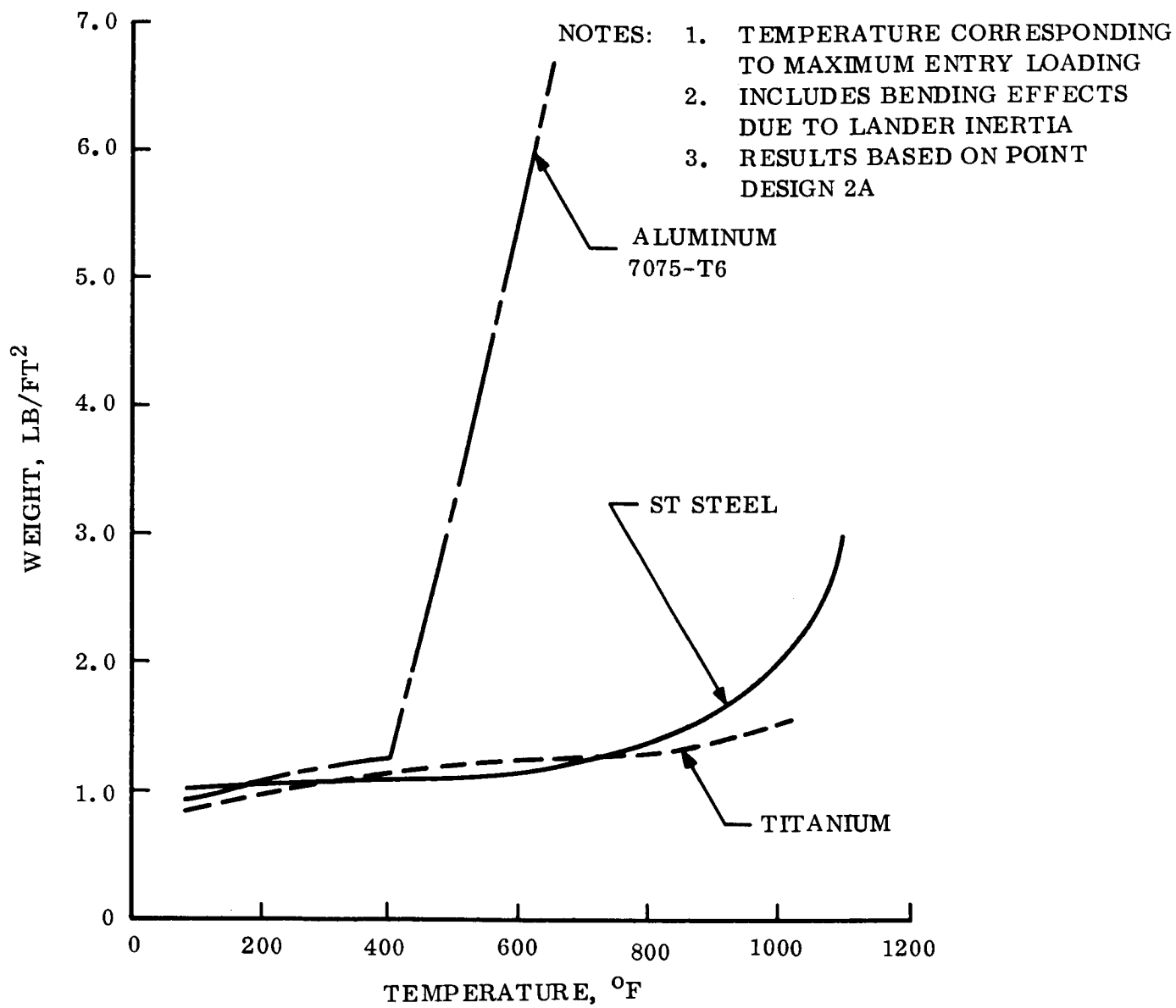


Figure 3.1.2-109. Skirt Structural Weight Density vs Temperature

- NOTES: 1. TEMPERATURE CORRESPONDING TO MAXIMUM ENTRY LOADING
 2. INCLUDES BENDING EFFECTS DUE TO LANDER INERTIA
 3. RESULTS BASED ON POINT DESIGN 2A
 4. EFFECT OF THERMAL STRESSES DUE TO 200° DIFFERENTIAL BETWEEN HONEYCOMB SANDWICH FACES

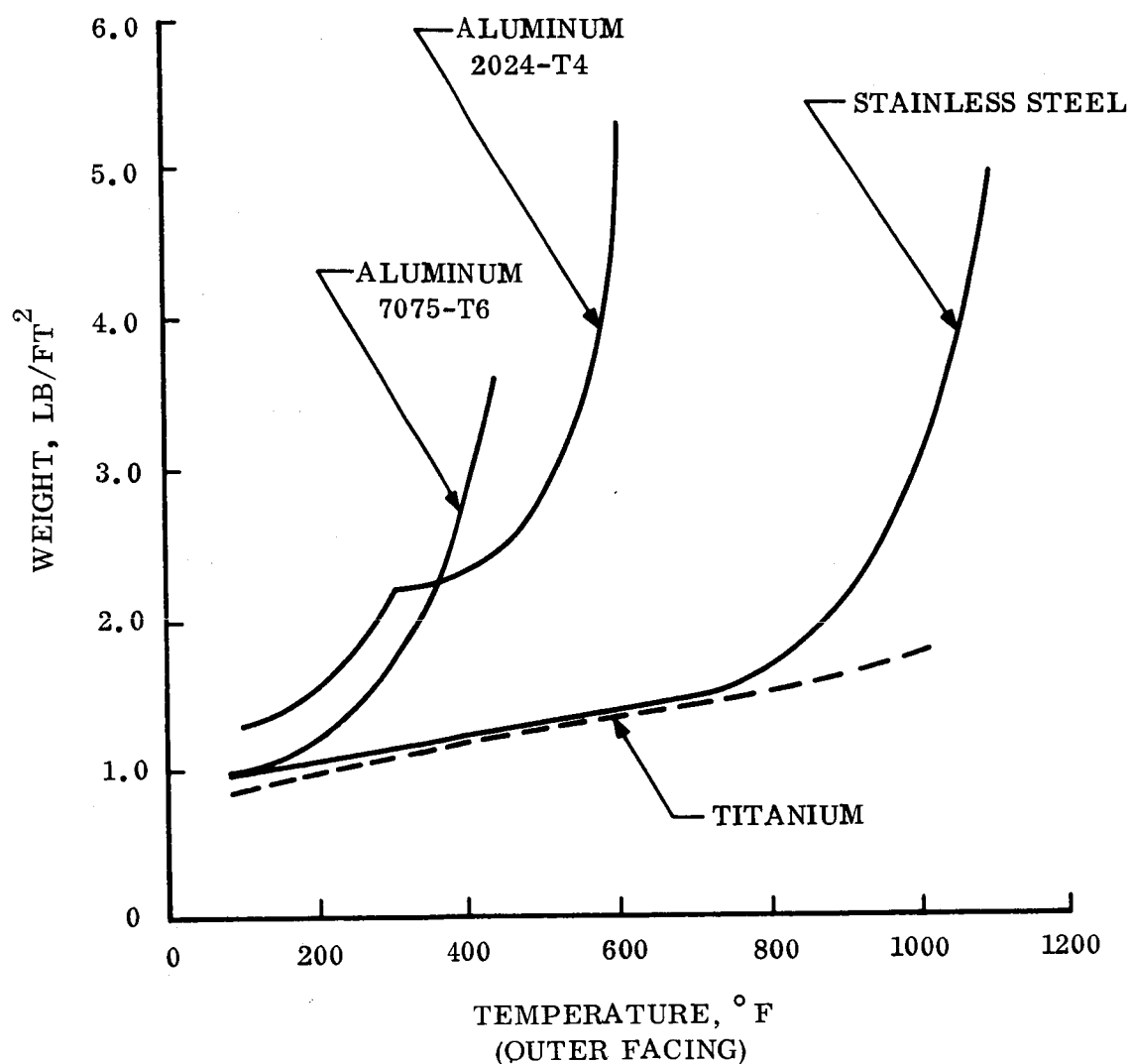


Figure 3.1.2-110. Skirt Structural Weight Density vs Temperature

substructural thickness distributions; weight break-down, etc. Such information is the numerical equivalent of a detailed design layout but is made at considerably less cost and in much less time. The CONSEP Program provides design data at the approximate rate of 100-200 designs per minute. The following basic parametric studies resulted in the evaluation of 1890 candidate vehicles, required less than one-half hour of computation time, and significantly reduced the number of design layouts required.

The resulting design data of this section are presented in three basic subsections. First, entry vehicle design trade-offs are described for Lander design packaging concept, size and weight, and aeroshell geometry for typical out-of-orbit and direct entry flight trajectories. Second, the aeroshell heat shield weight trade-offs with design backface temperature are described, along with variations in structural design concept and materials. Third, the heat shield design trade-offs are shown for varying entry velocity and path angle, using design information and the results of materials analysis from the first two subsections.

The primary conclusions derived from these design trend studies are:

1. The principal factor affecting entry vehicle and aeroshell weight, for any selected decelerator load, is that of physical size (aeroshell diameter). Bluntness ratio has virtually no effect, while cone semi-apex angle has only a secondary influence (three to five percent of entry Capsule weight variation).
2. The body cone angle influences the resultant vehicle $m/C_D A$ due to its effect on Capsule drag. The large cone angles considered for the Capsule design display little variation of drag coefficient with bluntness ratio.
3. For a given decelerator load, wide variations in Lander design concept produce a negligible variation in Capsule weight and $m/C_D A$. Fixed weight science packages result in omni-directional decelerator-aeroshell combinations whose weight is 350 to 400 lbs heavier than the corresponding multi-directional design. This weight increase, for a 10 ft vehicle, would result in an approximate 33 percent increase in vehicle $m/C_D A$.
4. Items 1 through 3 indicate that design vehicle selection is basically insensitive to geometric influences, but primarily dependent on systems constraints such as maximum allowable vehicle weight (not significant during this study), shroud diameter, and decelerator performance limits.

Heat shield melt limits were established for the low density ablator selected for application to the out-of-orbit designs, but lie outside the $m/C_D A$ limit values imposed by recovery system performance considerations.

5. All design concepts exhibited positive static stability. Minimum levels (for 50° , $R_N/R_B = 0.2$, low diameter designs), approached 15 percent of base diameter. More representative capsule concepts, in terms of mission preference, exhibited stability levels between 30 and 60 percent. Static stability increased with increasing cone angle and/or bluntness ratio.
6. The vehicle inertia properties I_x/I_y and m/I_y , while comparatively invariant with cone angle and bluntness ratio, varied considerably with vehicle diameter and Lander design. This trend is due to the attenuator (impact limiter) contributions to decelerator load, inertia and weight, which control Capsule characteristics.

The inertia parameters are of particular significance during exospheric and atmospheric portions of the trajectory, because of their influence on Capsule dynamics. Reasonable entry Capsule designs result in I_x/I_y values from 1.0 to 1.5 and m/I_y values from 0.2 to 0.35 (these establish allowable roll rate limits for avoidance of low-altitude attitude instabilities).

7. Design backface temperature variations (300°F and 800°F , versus the nominal 600°F value) indicate that heat shield weight savings amount to only 10 or 20 lbs through use of the maximum 800°F design value. These savings are most likely offset by the associated increase in structural weight. A 300°F design requirement would increase heat shield weight by 10 to 50 lbs, for out-of-orbit designs, and by 10 to 90 lbs for direct entry designs over a 7 to 15 ft vehicle diameter range. These shield weight variations represent a design confidence factor of unity for the heat shield. Uncertainties in atmospheric composition, etc., would result in larger penalties.
8. Approximate structural weight variations with material and design concept, for both out-of-orbit and direct entry, show a clear advantage of honeycomb design over ring-stiffened shell structure. The use of titanium or aluminum honeycomb is preferred over stainless steel.
9. Studies of variations in entry conditions were made to validate the current design approach. The lowest entry path-angle resulted in the most conservative heat shield design. Further, increasing entry velocity increased shield requirements, particularly for out-of-orbit designs. However, the heat shield weight varied only 20 lbs from the nominal designs (about 1.2 percent of entry capsule weight).

3.1.2.2.1 Entry Capsule Design Trends

The following discussion presents a summary of the design data obtained via the CONSEP Program. Capsule performance is described as a function of Lander configuration, and corresponding aeroshell weight contributions (Heat shield and structure).

A. Decelerator Load Packaging Procedure and Assumptions

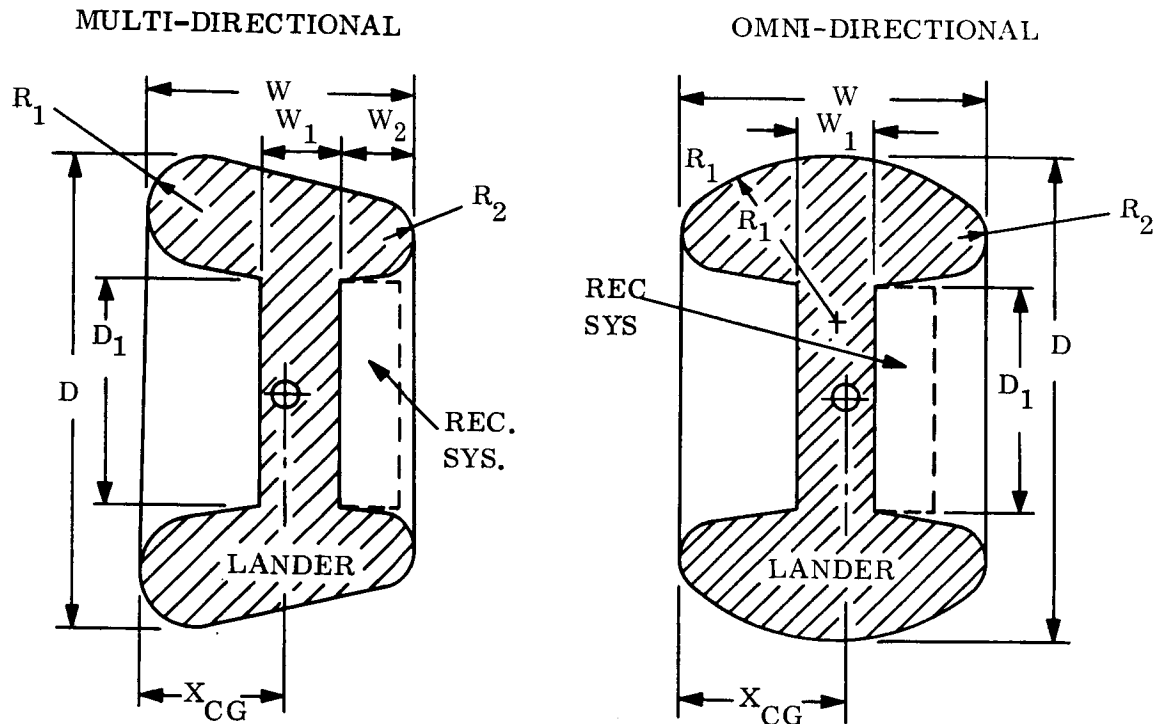
Aeroshell and entry vehicle characteristics were generated for two Lander design concepts (multi- and omni-directional lander). Three lander designs were supplied for each concept. Fig. 3.1.2-111 lists the basic geometric characteristics of these Lander concepts. Table 3.1.2-11 lists the weight and inertia values for each Lander, together with decelerator and spin/de-spin systems weight allowances. The latter items were obtained from scaling laws as a function of Lander/weight decelerator load. The recovery systems characteristics were combined with the Lander for final decelerator load definition. Each decelerator load is identified by a prefix (u/d or o/d), which signifies multi- or omni-direction Lander, and a number representing decelerator load.

Table 3.1.2-12 summarizes materials characteristics, design concepts, and trajectory characteristics.

Two densities of elastomeric shield material were selected on the basis of efficiency and performance: 1) ESM 1004 X, weighing 16 lbs/cu ft, for out-of-orbit designs, and 2) ESM 1004 AP, at 35 lbs/cu ft, for direct entry. The higher density provides increased ablation and shear resistance. A 600°F baseline backface temperature was selected.

Structural analyses indicated that a minimum gage stainless-steel honeycomb structure (with supporting rings) would provide the necessary structural integrity for both out-of-orbit and direct entry designs. An equivalent surface density of 0.8 lb/ft² (including ring weight allowances) was derived for this approach, along with a conservative thickness allowance of 0.5 in.

The capsule design concept allows the Lander to project through the aft cover. An aft cover annular disk was assumed to be constructed of minimum gage (0.04 in.) aluminum with thermal protection. Thermodynamic analysis indicated that a required thickness would approximate the value applied at the aft edge of the conical skirt. Modifications were made to the CONSEP Program to include the inertia effects of the cut-out (for Lander protrusion) and the heat shield thickness application.



<u>D/L</u> <u>NO.</u>	<u>W</u>	<u>D</u>	<u>R₁</u>	<u>R₂</u>	<u>W₁</u>	<u>W₂</u>	<u>D₁</u>	<u>X_{CG}</u> *

1. MULTI-DIRECTIONAL LANDERS:

U/D594	33.0	54.9	6.0	4.0	11.0	8.0	28.0	19.49
U/D955	35.7	66.3	7.5	5.0	14.6	8.0	34.0	21.42
U/D1255	37.28	73.2	8.3	5.5	16.0	8.0	38.0	22.28

2. OMNI-DIRECTIONAL LANDERS:

O/D868	40.8	67.0	22.6	3.8	11.0	-	28.0	22.73
O/D1292	44.0	75.0	24.9	5.0	14.6	-	34.0	24.70
O/D1540	44.0	80.0	25.8	5.5	16.0	-	38.0	24.67

*INCLUDES RECOVERY SYSTEM

NOTE:

ALL DIMENSIONS IN INCHES

Figure 3.1.2-111. Decelerator Load Basic Geometry

TABLE 3.1.2-11. LANDER AND DECELERATOR LOAD INERTIAL CHARACTERISTICS

Item	D/L Type No.	Multi-Directional			Omni-Directional		
		M/D594	M/D955	M/D1255	O/D868	O/D1292	O/D1540
Science Weight, (lb)		160	306	440	160	306	440
Container Weight, (lb)		120	160	190	120	160	190
Attenuation Weight, (lb)		195	298	374	414	567	602
Lander Weight, (lb)		475	764	1004	694	1033	1232
Recovery System Weight, (lb)		119	191	251	174	259	308
Deceleration Load Weight, (lb)		594	955	1255	868	1292	1540
Pitch Inertia, (slug-ft ²)		36.3	75.5	132.9	33.9	136.0	187.8
Roll Inertia, (slug-ft ²)		32.8	53.8	107.4	67.6	118.0	164.8
CG, (in.)		19.49	21.42	22.28	22.73	24.70	24.67
Spin/De-spin system, Weight, (lb)		22.4	31.6	54.9	37.6	59.5	79.8

TABLE 3.1.2-12. ENTRY CAPSULE DESIGN CONDITIONS

Item	Characteristics		
1. <u>Heat Shield</u>	<u>Out-of-Orbit</u>	<u>Direct Entry</u>	
Material	ESM 1004 X*	ESM 1004 AP	
Density (lb/ft ³)	16	35	
Backface Temp (°F)	600	600	
Confidence Factor	1.0	1.0	
*Limit Integrated Heat (BTU/FT ²)	1900	1900	
2. <u>Forebody Structure</u>			
Type:	Stainless Steel Honeycomb w/Rings		
Surface Density (lb/ft ²)	0.8 (w/rings)		
Thickness Allowance (in.)	0.53		
3. <u>Aft Cover</u>			
Type:	Flat Annular Disk		
Materials:			
Structure:	0.04 in. of aluminum		
Heat Shield:	ESM 1004 X or 1004 AP		
4. <u>Entry Conditions</u>			
<u>Entry Mode</u>	<u>Velocity</u>	<u>Angle</u>	<u>Tolerance</u>
Out-of-Orbit	15,300 fps	16°	±2°
Direct	20,800	25°	±8

Trajectory performance analyses specified an out-of-orbit entry velocity of 15,300 fps and an entry path angle of 16° , with a tolerance of $\pm 2^\circ$. The direct entry conditions were specified as 20,800 fps, and $25^\circ \pm 8^\circ$. Maximum heat shield requirements occur at minimum flight path angles due to increased soak time. The heat shield calculations therefore utilized reference entry angles of 14° and 17° , respectively, for out-of-orbit and direct entry.

A further heat shield limitation existed for the ESM 1004 X application (out-of-orbit). Ablation test data indicated where backface melting would occur if sufficient heating rates and surface shear were encountered. Analysis indicated that a limiting integrated heat load of 1900 Btu/ft² should be used in CONSEP studies. This limitation was not, however, programmed into CONSEP, but was applied as a limit line to CONSEP results. Recovery system performance requirements, as will be discussed in subsequent sections, eliminated consideration of vehicles whose $m/C_D A$ resulted in this limiting heat load.

The aerodynamic parameters utilized for CONSEP application (drag coefficient and center of pressure location) are based on Newtonian theory. The integrated heat load distribution was calculated for convective and equilibrium/non-equilibrium radiative heating. These calculations, and the associated materials performance are discussed in detail in para 3.1.2.1 of this document.

B. Entry Capsule Weight - Performance Design Variations

Examination of entry Capsule weight variations with aeroshell cone angle and bluntness ratio indicated virtually no sensitivity to bluntness ratio and very little to cone angle. This trend permitted selection of a nominal cone angle-bluntness ratio combination (60° , $R_n/R_B = 0.5$) for the data presentations of table 3.1.2-13 and figs. 3.1.2-112 through -115, which illustrate representative variations in entry vehicle weight and $m/C_D A$ with capsule diameter and decelerator load.

Out-of-Orbit Design Characteristics - Figs. 3.1.2-112 and -113 illustrate the entry capsule weights and $m/C_D A$ levels as a function of capsule diameter for the given range of multi-directional Lander packages. The basic expected trends are indicated. Reduction in capsule diameter for a given decelerator load results in reduced entry weight due to the associated reduction in capsule surface area. This, in turn, results in lower aeroshell weight, even with increasing capsule $m/C_D A$. A breakdown of aeroshell weight is given in the following subsection.

The entry Capsule weight varies directly with decelerator load, in an approximate one-to-one ratio; i.e., an increase in decelerator load weight of 100 lb results in a similar increase in entry Capsule weight. This correspondence is virtually invariant with diameter and decelerator load, through a considerable variation in Capsule $m/C_D A$. The decrease in allowable decelerator load level with increasing diameter, for a given entry Capsule weight, reflects the aeroshell weights associated with the surface area-diameter effect discussed in the preceding paragraphs. The basic

TABLE 3.1.2-13. ENTRY CAPSULE AND AEROSHELL WEIGHT VARIATIONS

$$\theta_c = 60^\circ \quad R_N/R_B = 0.50$$

Capsule Diam (ft) =		Out-of-Orbit			Direct		
Item		7	11	15	7	11	15
Decelerator (lbs)		←		594 to 1540			→
Heat Shield Wt (lbs)		26-30	48-64	64-96	59-63	117-132	289-228
Structural Wt (lbs)		40	124	246	46	130	251
Entry Capsule Wt (lbs)		688-1691	794-1808	932-1962	713-1724	863-1876	1072-2093
Payload Efficiency (D/L Wt Fraction)		0.86-0.91	0.75-0.85	0.64-0.78	0.88-0.89	0.69-0.82	0.55-0.74

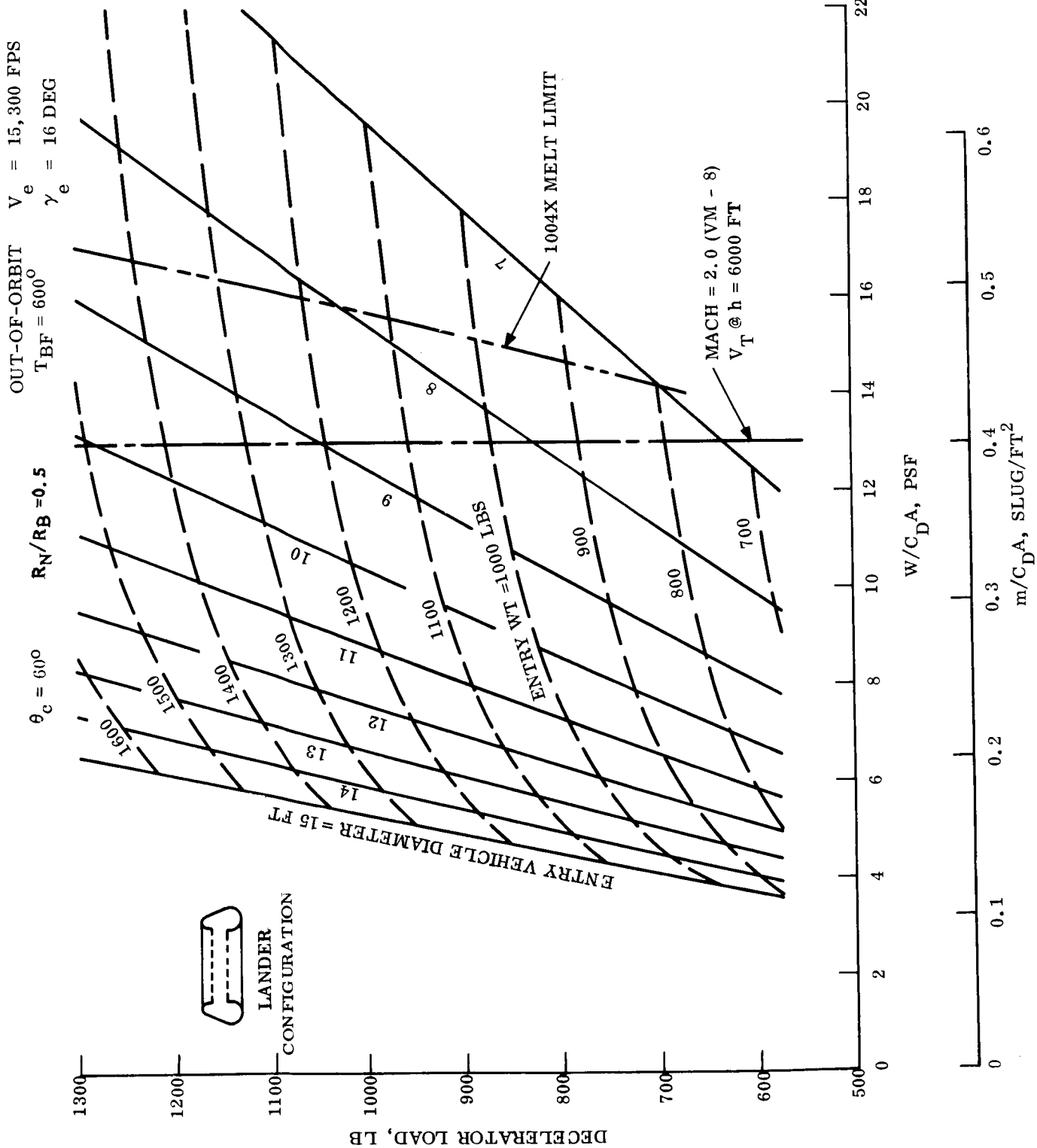


Figure 3.1.2-112. Decelerator Load vs Ballistic Coefficient-Multidirectional Landers-Out-of-Orbit

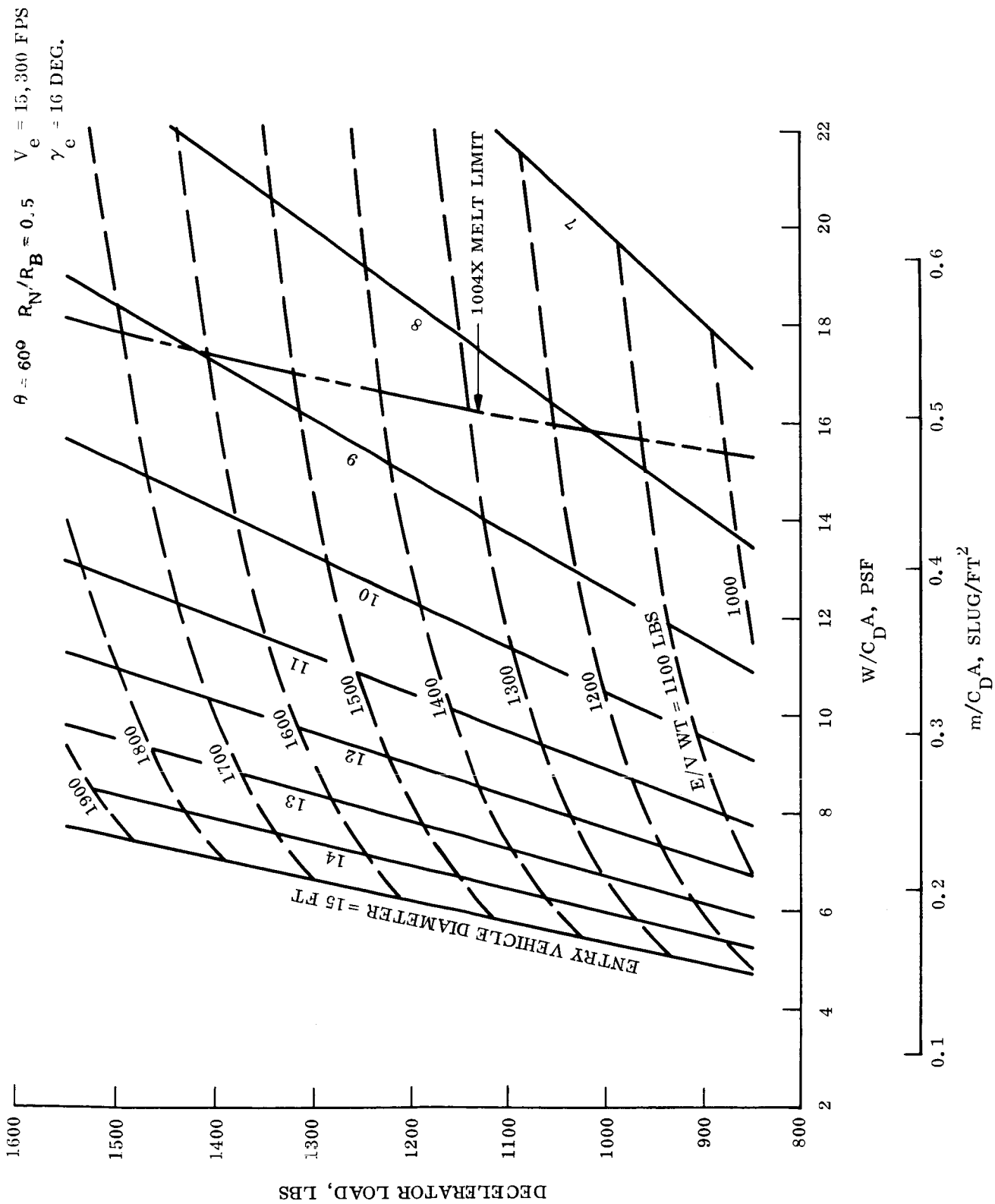


Figure 3.1.2-113. Decelerator Load vs Ballistic Coefficient-Multidirectional Lander, Out-of-Orbit

finding is that aeroshell weight is virtually independent of decelerator load and $m/C_D A$, being affected only by capsule size (diameter). This also indicated that variations in total entry Capsule weight for a given Capsule diameter almost entirely result from decelerator load variations.

The melting limit associated with the low density ESM 1004X heat shield material has been investigated. Analyses indicated that recovery system performance requirements will limit maximum allowable $m/C_D A$ levels to values lower than those indicated by the melt limit. This heat shield material is therefore recommended for point design studies.

The two weight performance curves of figs. 3.1.2-115 and -116 can be superposed, but are shown separately to point out the influence of Lander concept on decelerator load. Reference to table 3.1.2-11 shows that the two decelerator load scales encompass the same landed subsystem weight range. Landed subsystems (with container) weighing 466 lb result in a multi-directional decelerator load package of 955 lb, and an omni-directional package of 1292 lb. Insertion of these two decelerator loads into a 10 ft diameter vehicle results in Capsule weights of 1136 lb and 1505 lb, respectively, a difference of 369 lb. Most of this Capsule weight difference lies in the decelerator load and assumed spin/de-spin system weights; the aeroshell contribution has little influence; for this example contributing 150 lb.

Direct Entry Design Characteristics - Figs. 3.1.2-114 and -115 present data for direct entry in the same data format as the two preceding curves for out-of-orbit design. The same trends and comments apply, the basic difference occurs in the aeroshell weight levels due to the use of ESM 1004 AP (35 lb/ft³) instead of ESM 1004 X (16 lb/ft³).

The increased heat shield density results in a slight increase in heat shield weight sensitivity to $m/C_D A$ and therefore in deceleration load, over that exhibited by the out-of-orbit designs. This increase is so slight that the out-of-orbit conclusions still apply. Aeroshell weight is virtually independent of decelerator load and $W/C_D A$. The Capsule weight is only sensitive to decelerator load and vehicle diameter. Capsule weights of 1189 and 1569 lb result from placing 466 lb of landed subsystems into multi/omni-directional decelerator loads of 955 and 1292 lb respectively for a 10 ft vehicle. This represents an aeroshell weight of 210 lbs, indicating a 60 lb growth over the corresponding out-of-orbit aeroshell heat shield design.

Figs. 3.1.2-112 to -115 illustrate the influence of Capsule diameter on $m/C_D A$. Drag coefficient, C_D , for all of the selected capsules is constant for the specified values of cone semi-apex angle and bluntness ratio. $m/C_D A$, therefore, varies

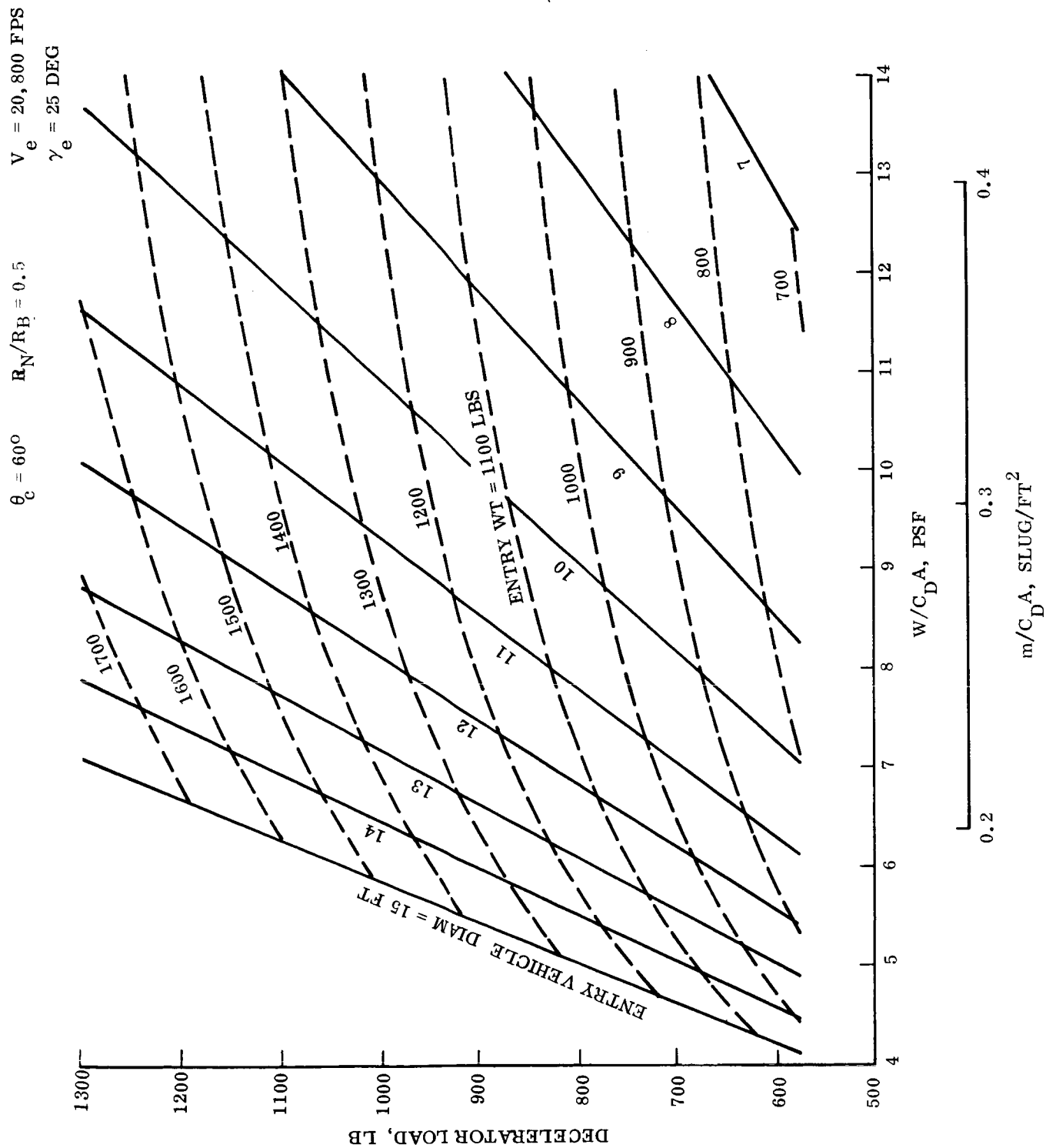
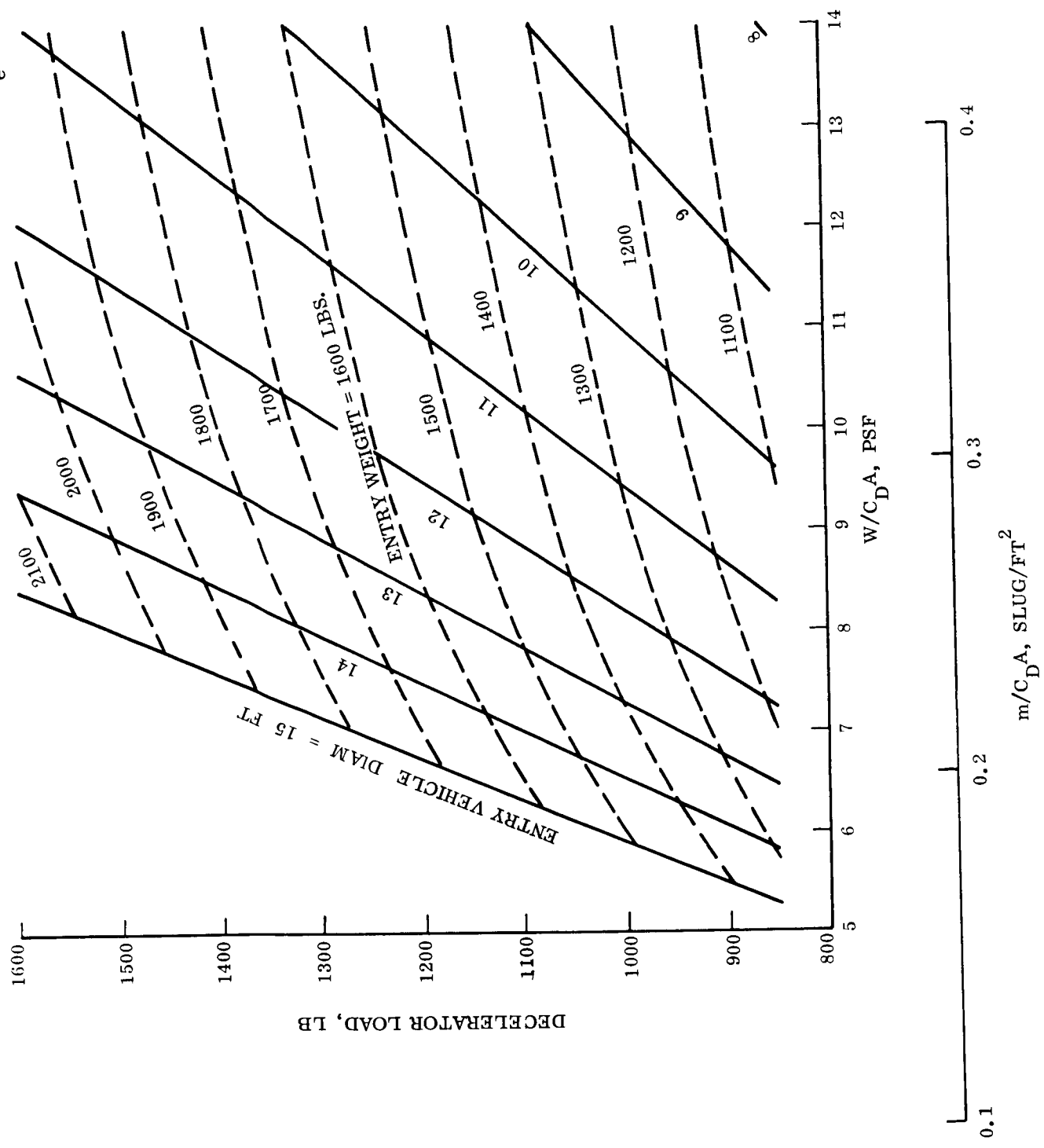


Figure 3.1.2-114. Decelerator Load vs Ballistic Coefficient-Multidirectional Landers-Direct Entry

$\theta_C = 60^\circ; R_N/R_B = 0.5$ $V_e = 20,800 \text{ FPS}$
 $\gamma_e = 25 \text{ DEG}$



III

Figure 3.1.2-115. Decelerator Load vs Ballistic Coefficient-Omnidirectional Landers-Direct Entry

only as a function of weight and base area. For a given Capsule weight, increased diameter results in an inverse-square variation in vehicle $m/C_D A$. There is a decreasing sensitivity of $m/C_D A$ to Capsule or decelerator load weight as diameter is increased.

The aeroshell weight sensitivity to base diameter, together with the small aeroshell weight contribution to total vehicle weight (5 to 20 percent), indicates that vehicle $m/C_D A$ should vary in a nearly linear manner with decelerator load. This is confirmed by the data with the exception of a slight non-linearity, due to the 5 to 10 percent weight contribution, and also to variations in spin/de-spin system weight allotments, as a function of decelerator load.

C. Basic Design Trend Variations with Capsule Geometry

Entry Capsule design characteristics were generated for sphere-cone concepts, with variations in semi-apex angle ranging from 50° to 70° . Bluntness ratios from 0.2 to 0.8 and base diameters from 7 to 15 ft were treated in the study. Only one complete set of entry Capsule design trend variations is presented for brevity. The example case is for a single decelerator load (M/D 1255), in the out-of-orbit entry mode.

Entry Capsule weight variations with semi-apex angle and bluntness ratio for various diameters are shown in figs. 3.1.2-116 and -117. Inspection indicates the nearly complete insensitivity of Capsule weight to bluntness ratio. This trend indicates that aeroshell structural and heat shield weight increments are also insensitive.

Weight variation with cone angle is noted, but is of secondary importance. The total variation for 7 ft diameter Capsules is one percent; for 15 ft diameter vehicles, 3.6 percent (due to the larger percentage of total weight represented by the aeroshell). Constant $m/C_D A$ curves are provided to indicate capsule performance.

The inverse square variation of $m/C_D A$, mentioned previously, is apparent in these plots, although minor variation occurs due to cone angle effect on drag. The bluntness ratio shows such a weak $m/C_D A$ sensitivity that each diameter curve can be represented by a single $m/C_D A$ value. This is done in subsequent plots.

Figs. 3.1.2-118 and -119 present heat shield, and structural weight variations with cone angle and diameter. Variations with R_N/R_B are not shown as they are virtually non-existent. The values shown at R_N/R_B of 0.5 are therefore representative of those at any other bluntness ratio, for each cone angle-diameter combination.

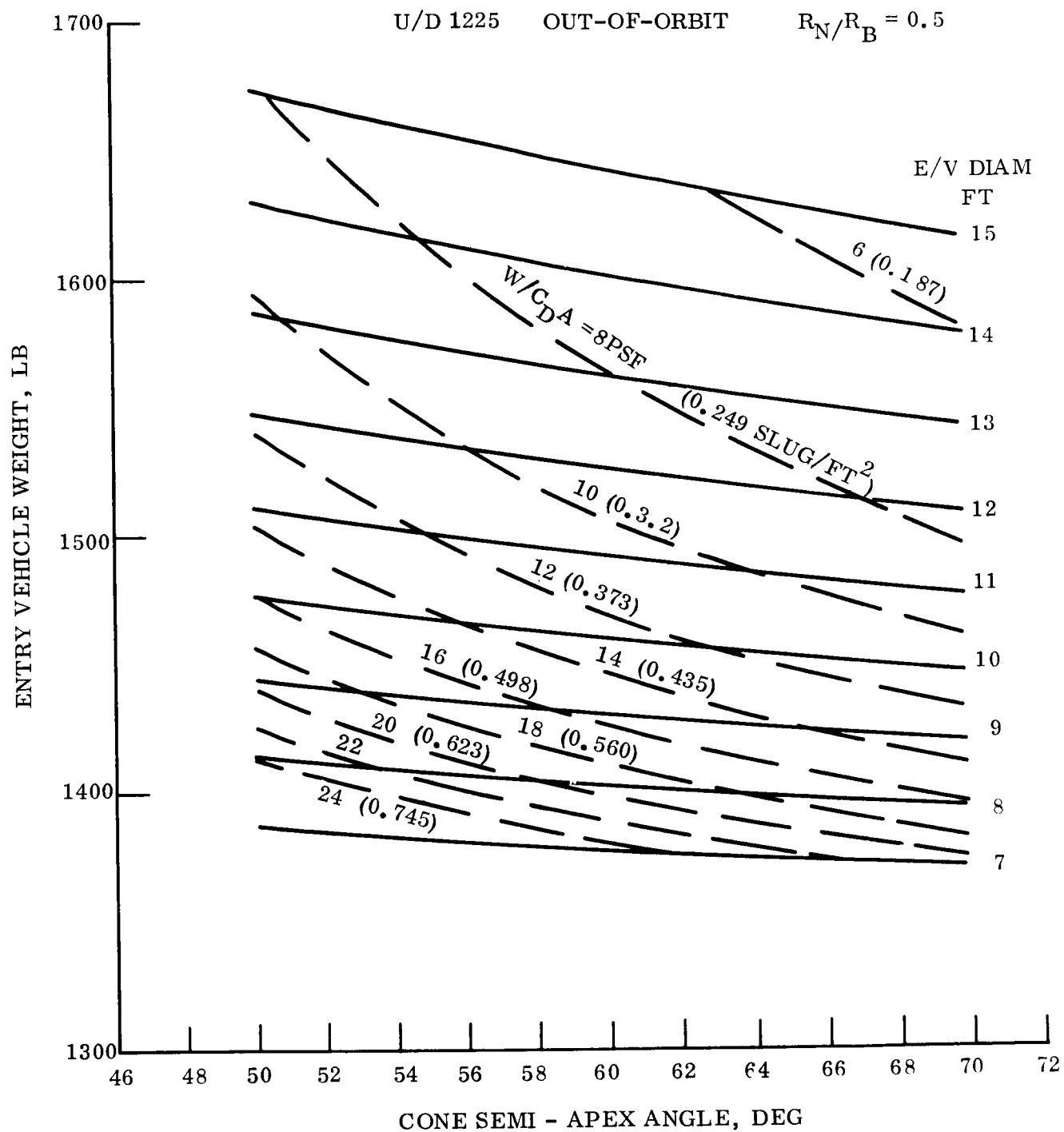


Figure 3.1.2-116. Entry Vehicle Weight vs Cone Angle

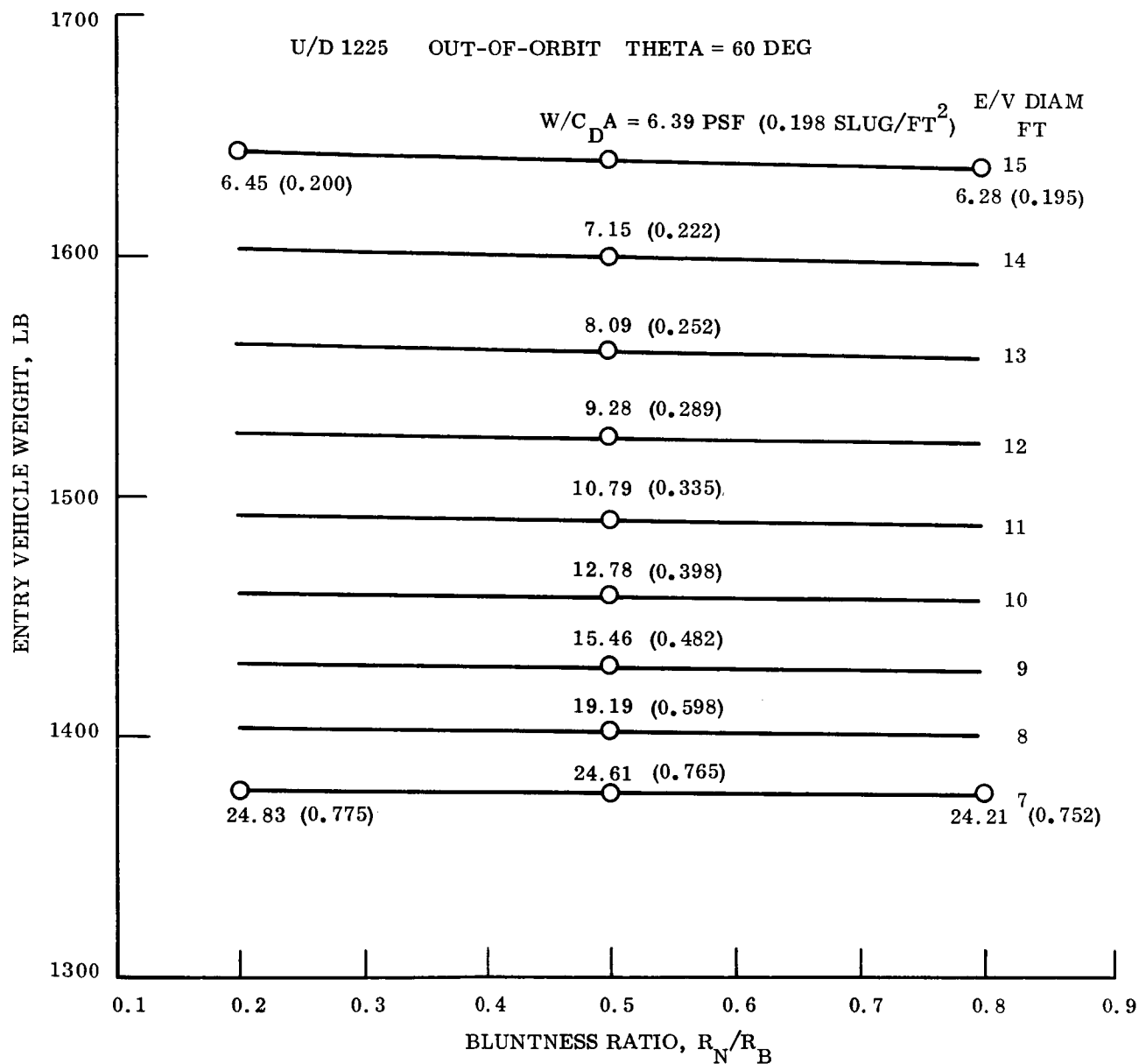


Figure 3.1.2-117. Entry Vehicle Weight vs Bluntness Ratio

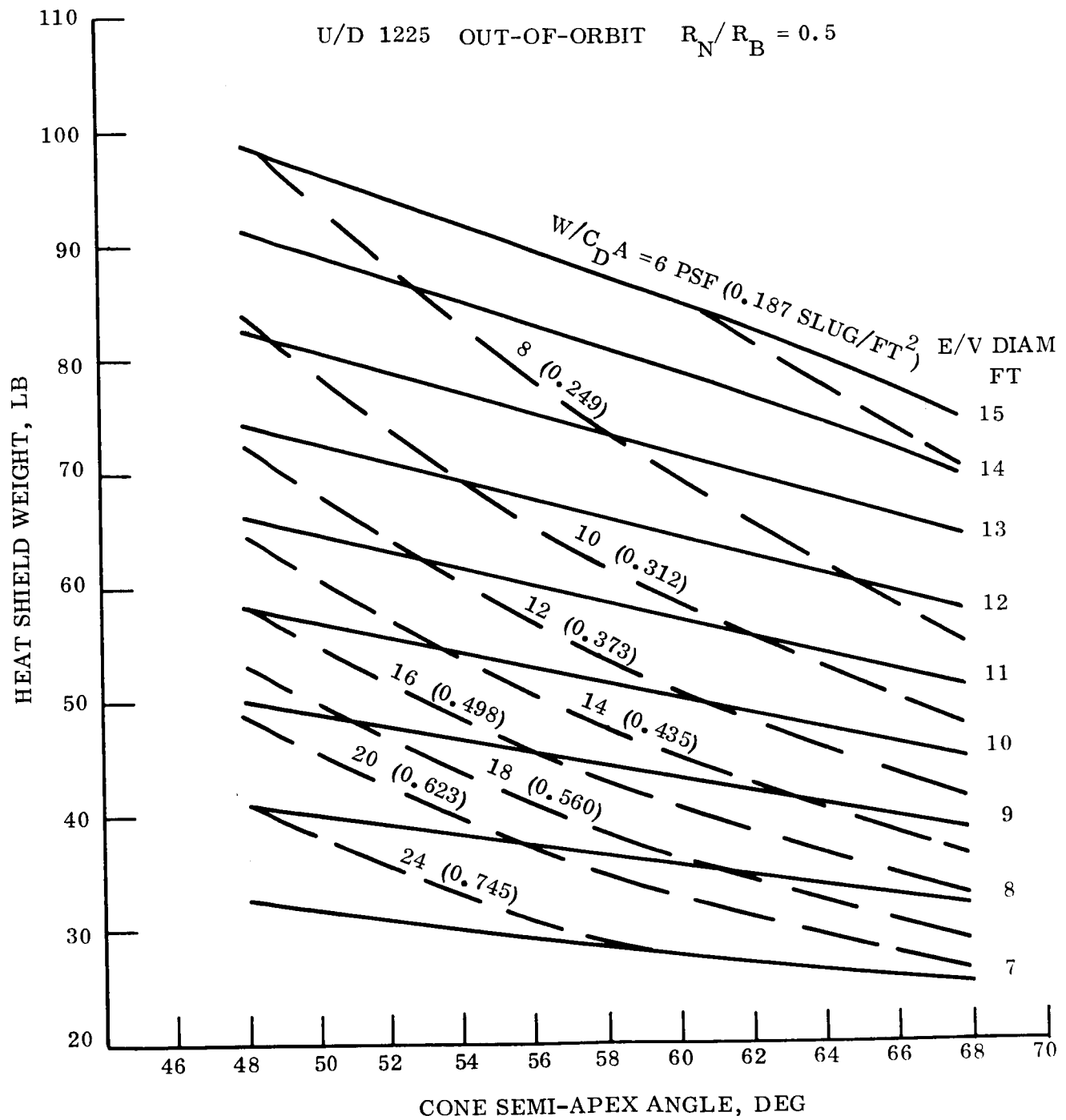


Figure 3.1.2-118. Heat Shield Weight vs Cone Angle

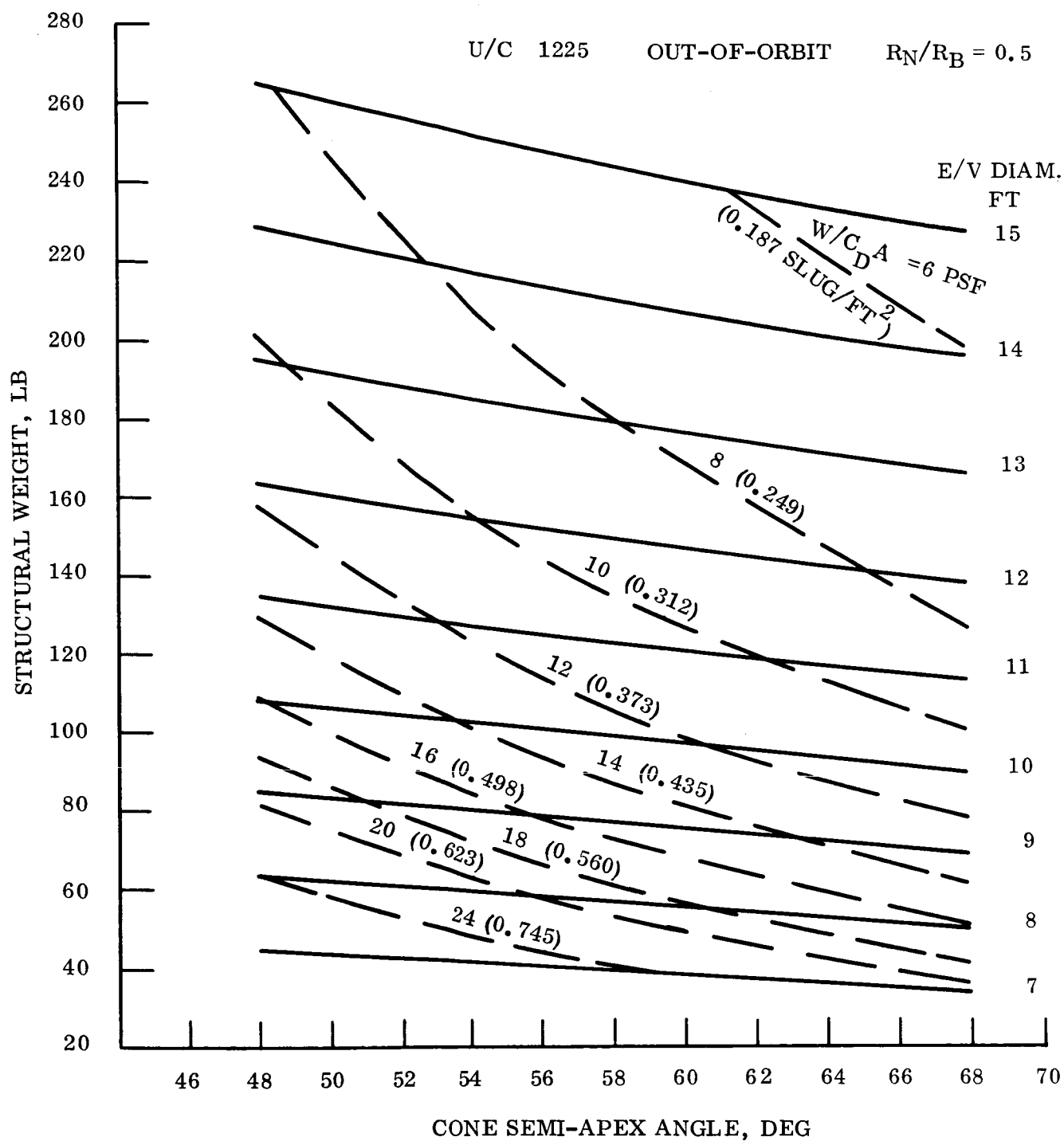


Figure 3.1.2-119. Structural Weight vs Cone Angle

The indicated reduction in weight contributions with increasing cone angle is basically due to a similar reduction in forebody surface area. While increased cone angles result in increased surface pressure coefficients, the attendant decrease in $m/C_D A$ results in lower total integrated heat load. Heat shield thickness requirements decrease slightly with increasing cone angle, thereby contributing to the heat shield weight trend.

Variations in heat shield weight with diameter indicate a less than one to one dependency upon surface area. This is again due to the reduction in $m/C_D A$ (and consequently in integrated heating) with increasing diameter. The structure is held at constant thickness and density is, therefore, insensitive to $m/C_D A$ influences. It does, however, exhibit a greater than one-to-one correspondence with forebody surface area. This is due to the aft cover contribution. Structural weight increases for aft covers between 7 and 15 ft diameter are, for example, about 60 lb.

Figs. 3.1.2-120 and -121 present data in terms of entry capsule weight percentages. It can be seen from fig. 3.1.2-120, that the heat protection weight fraction for out-of-orbit designs varies between two and five percent. The corresponding structural weight fraction range is from 3 to 16 percent (fig. 3.1.2-121).

Payload efficiency (recovery load weight fraction), is given in fig. 3.1.2-122. It can be seen that this parameter varies from 0.95 at 7 ft diameter to approximately 0.8 at 15 ft. The increasing gradient at any diameter with cone angle is due to surface area effects which result in less aeroshell weight contribution.

Figs. 3.1.2-123 and -124 present entry Capsule center of gravity location, in percent of aeroshell length, as a function of diameter, cone angle, and bluntness ratio. Variations with cone angle and bluntness ratio are principally due to normalizing to the Capsule length. If c.g. location had been expressed in distance from the aft cover, and had been normalized by base diameter, the θ and R_N/R_B variations would be virtually eliminated. The Capsule c.g. location is basically controlled by the decelerator load package. The Capsule c.g. is therefore near that of the decelerator load.

Small diameter Capsule concepts result in considerable Lander protrusion from the base of the aeroshell. This is aggravated by increasing cone angle. When the c.g. location is normalized by length, the trend is increased, ultimately resulting in X_{cg}/L values of 1.6 to 1.7 for a 7 ft vehicle diameter. Figs. 3.1.2-125 and -126 illustrates variations in stability margin, which have been normalized by base diameter. These curves illustrate the more powerful influence of cone angle upon vehicle center of pressure location, and consequently on static margin. The nominal 60° , 0.5 bluntness Capsule designs of the preceding section are indicated as having margins varying from 33 to 56 percent of diameter over the diameter variation of 7 to 15 ft. All Capsule concepts exhibited positive static stability even at the lower values of diameter and cone angle.

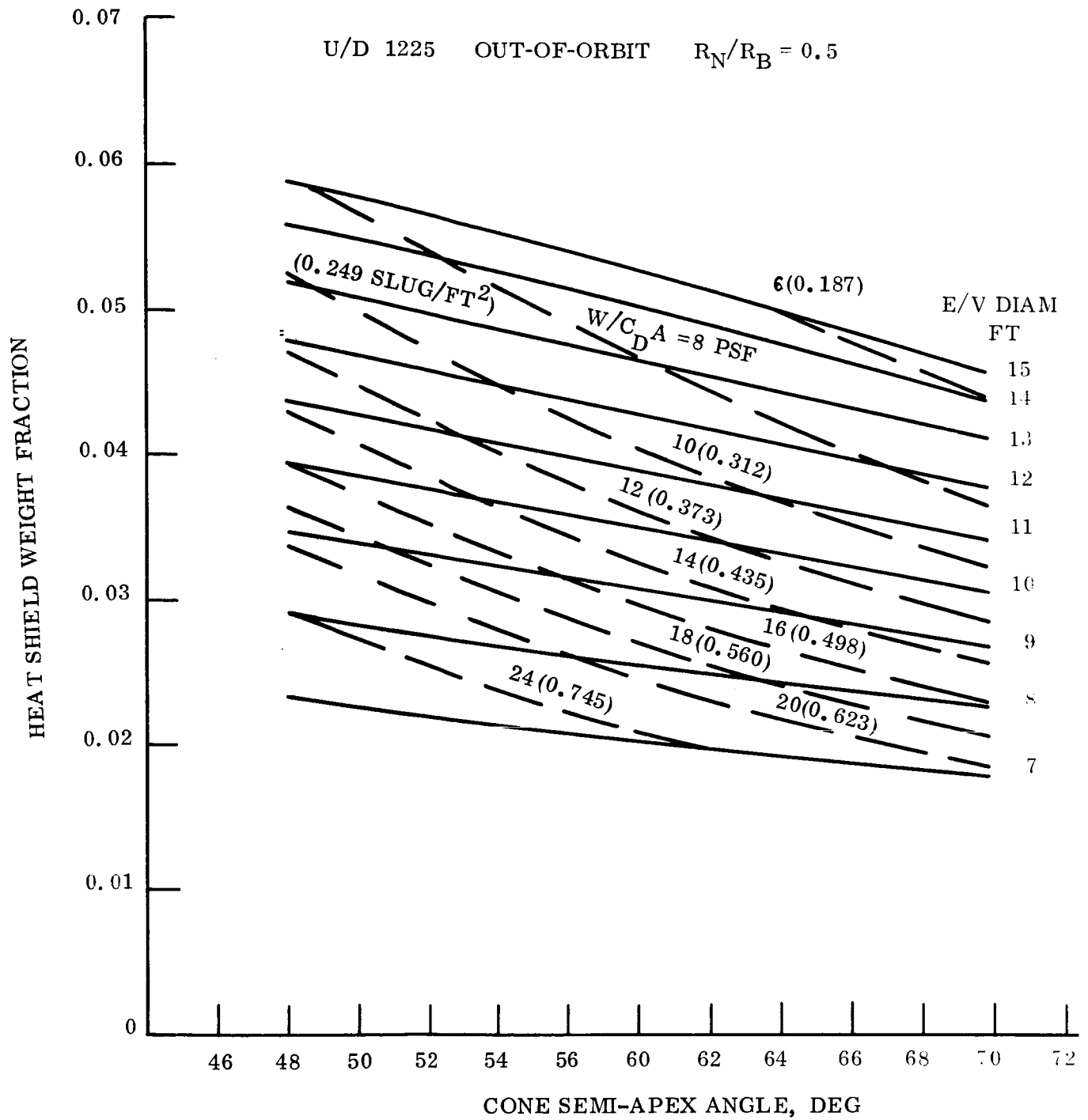


Figure 3.1.2-120. Heat Shield Weight Fraction vs Cone Angle

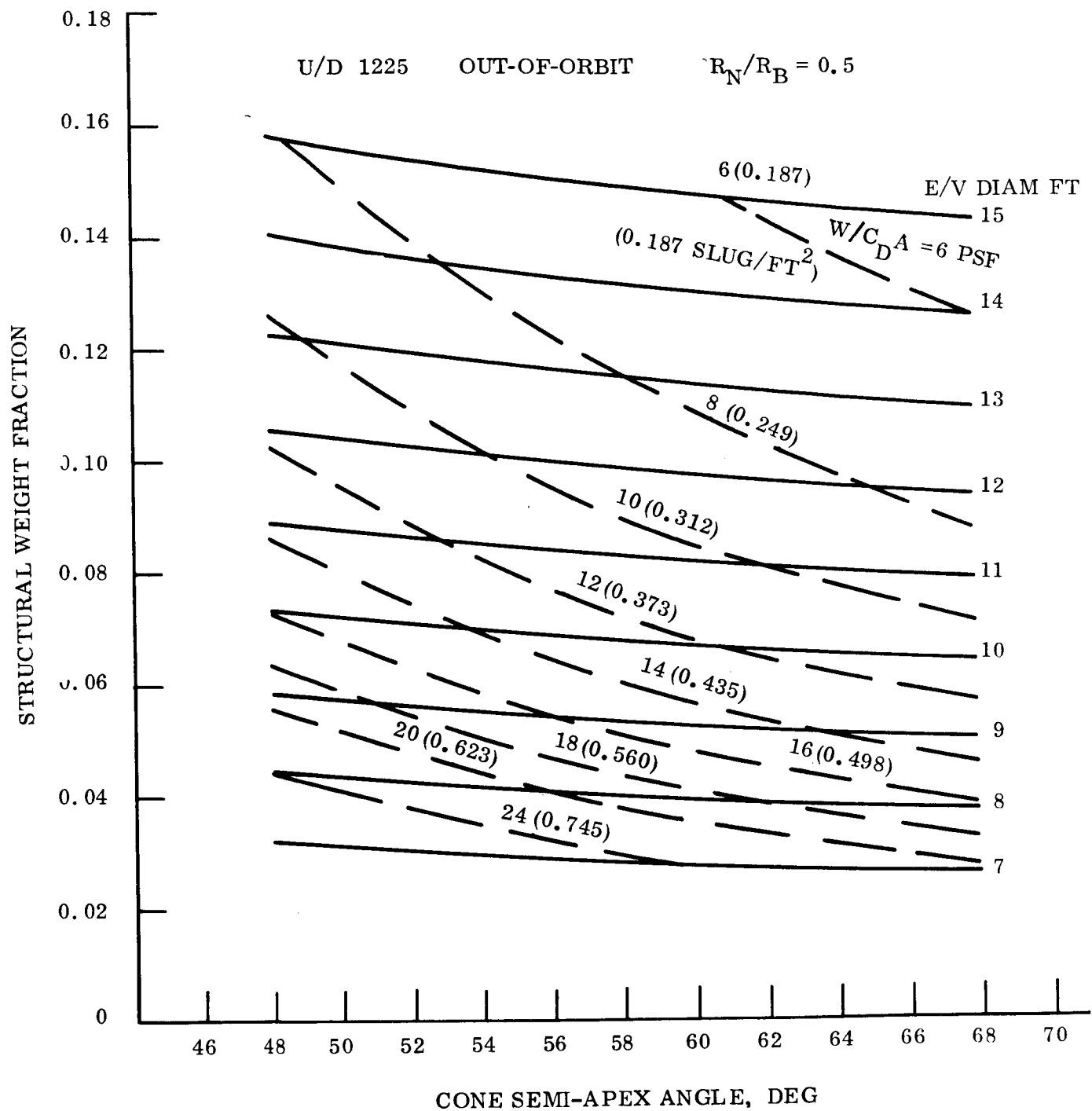


Figure 3.1.2-121. Structural Weight Fraction vs Cone Angle

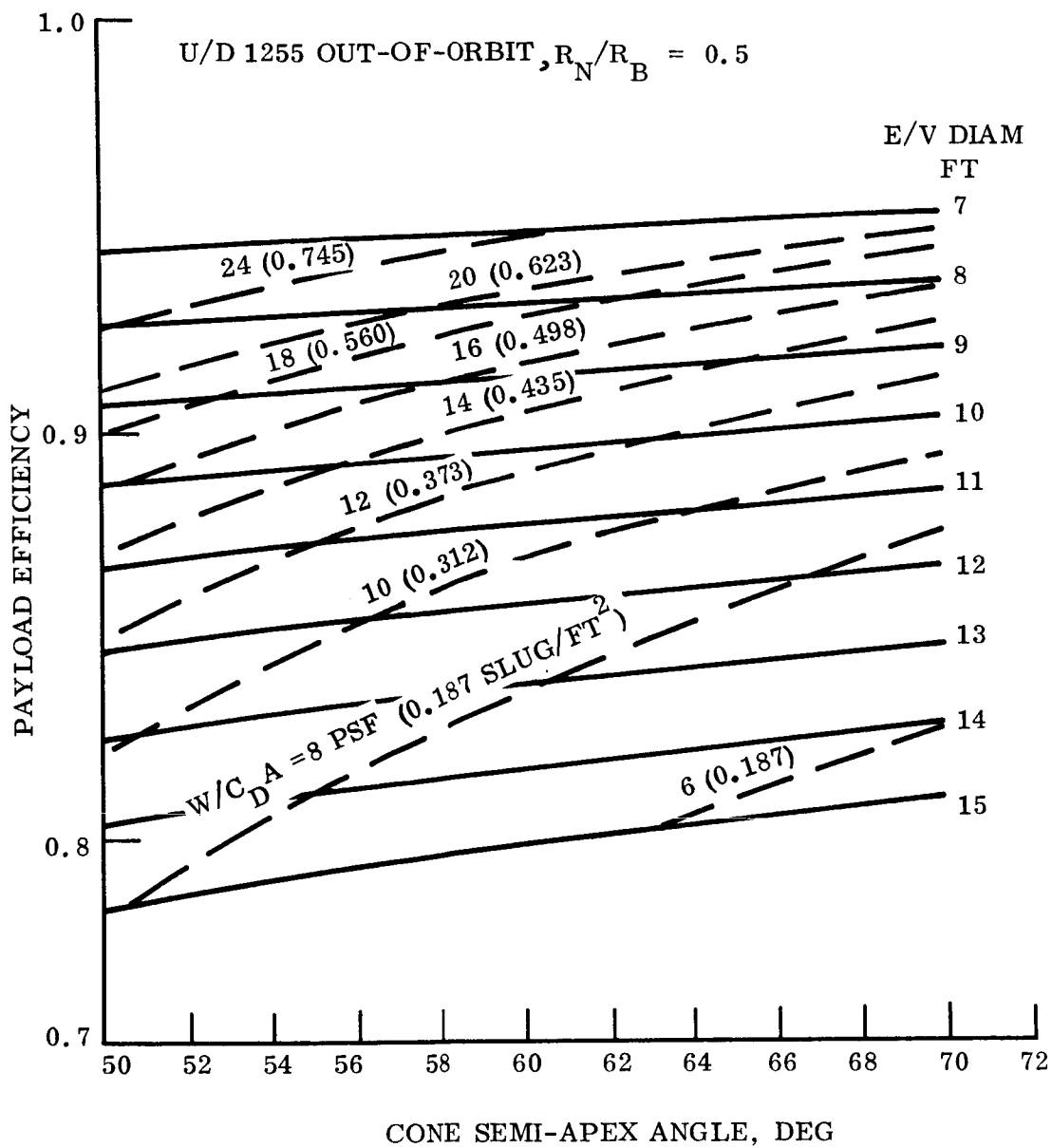


Figure 3.1.2-122. Payload Efficiency vs Cone Angle

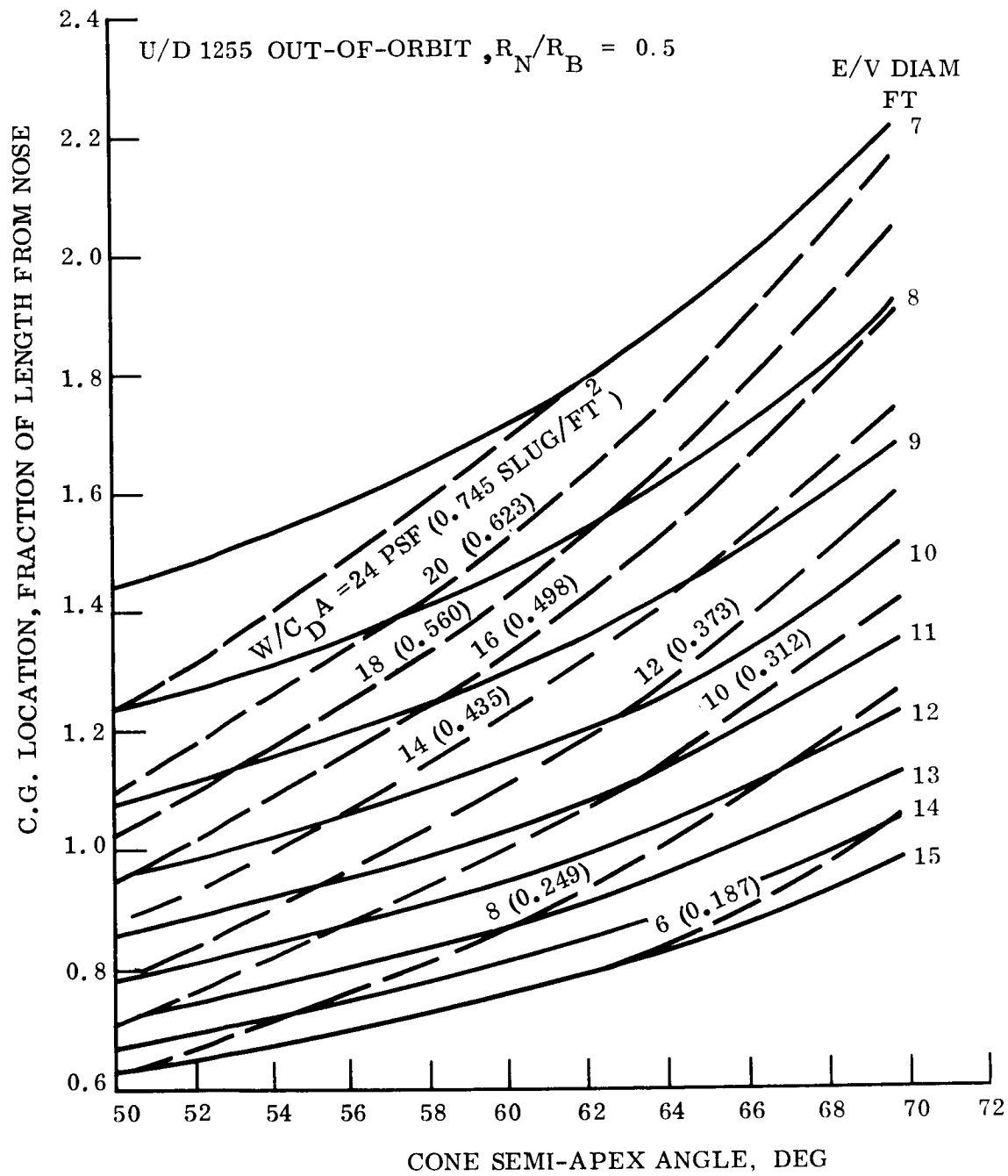


Figure 3.1.2-123. C.G. Location vs. Cone Angle

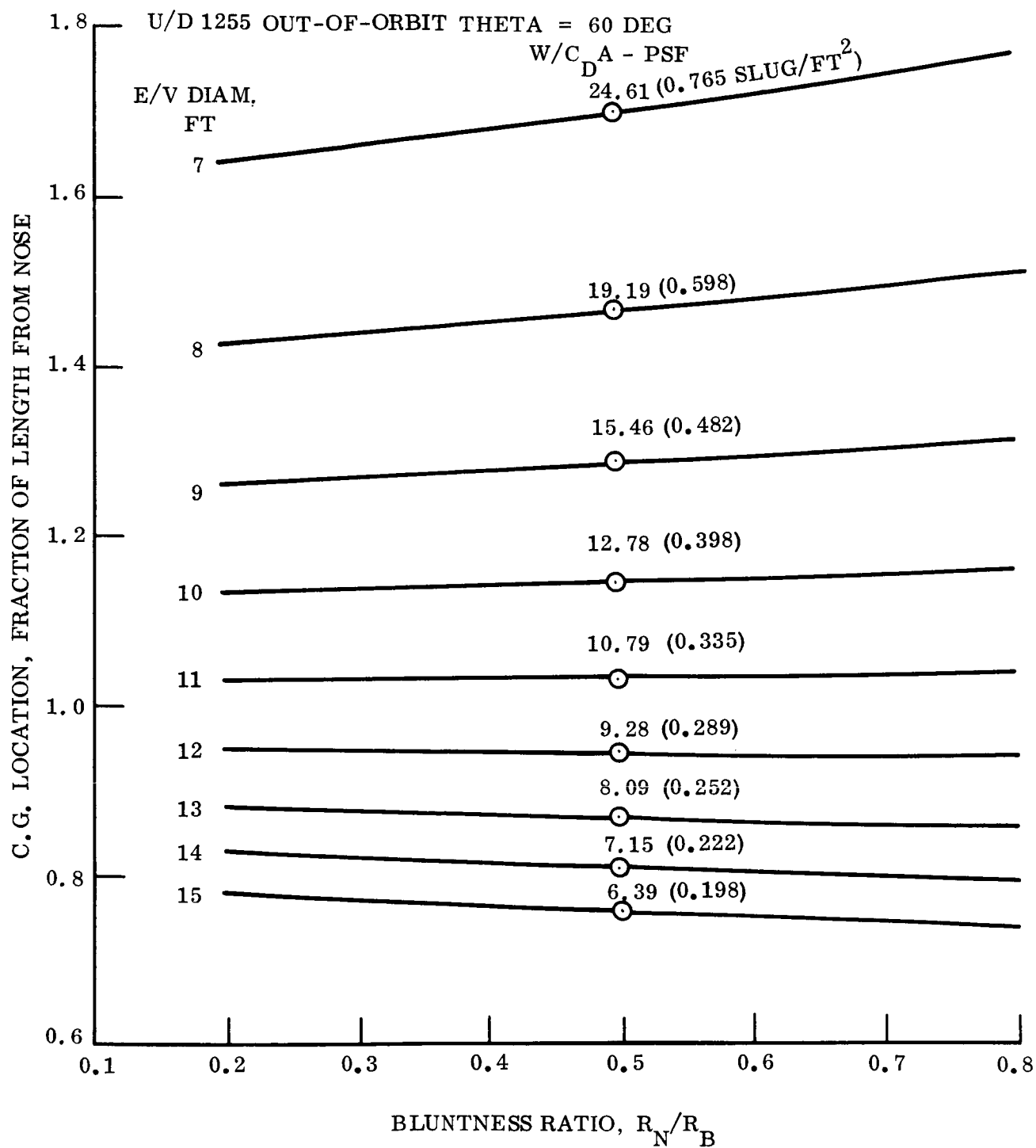


Figure 3.1.2-124. C. G. Location vs Bluntness Ratio

U/D 1225

OUT-OF-ORBIT

$R_N/R_B = 0.5$

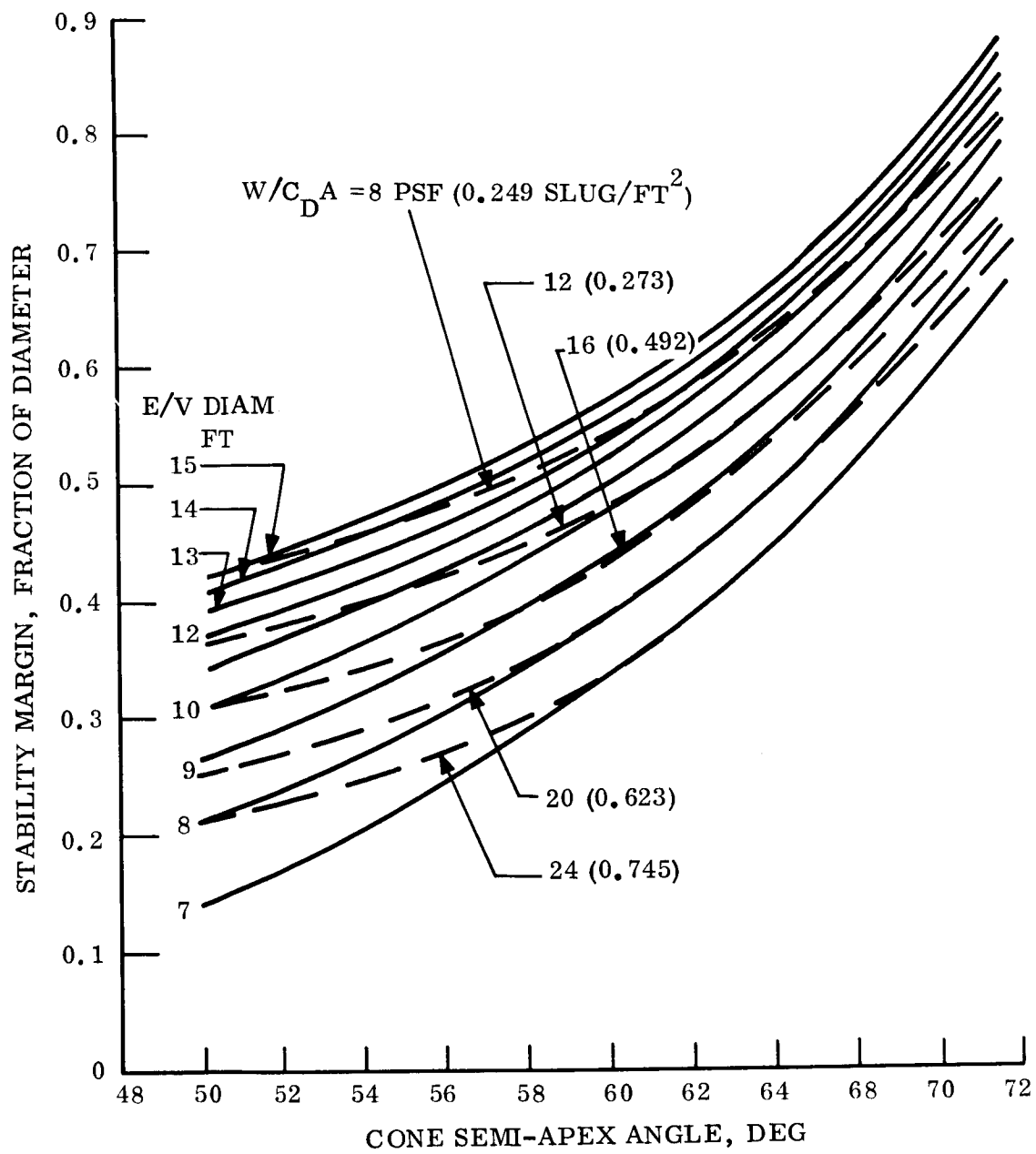


Figure 3.1.2-125. Stability Margin vs Cone Angle

U/D 1225

OUT-OF-ORBIT

THETA = 60 DEG.

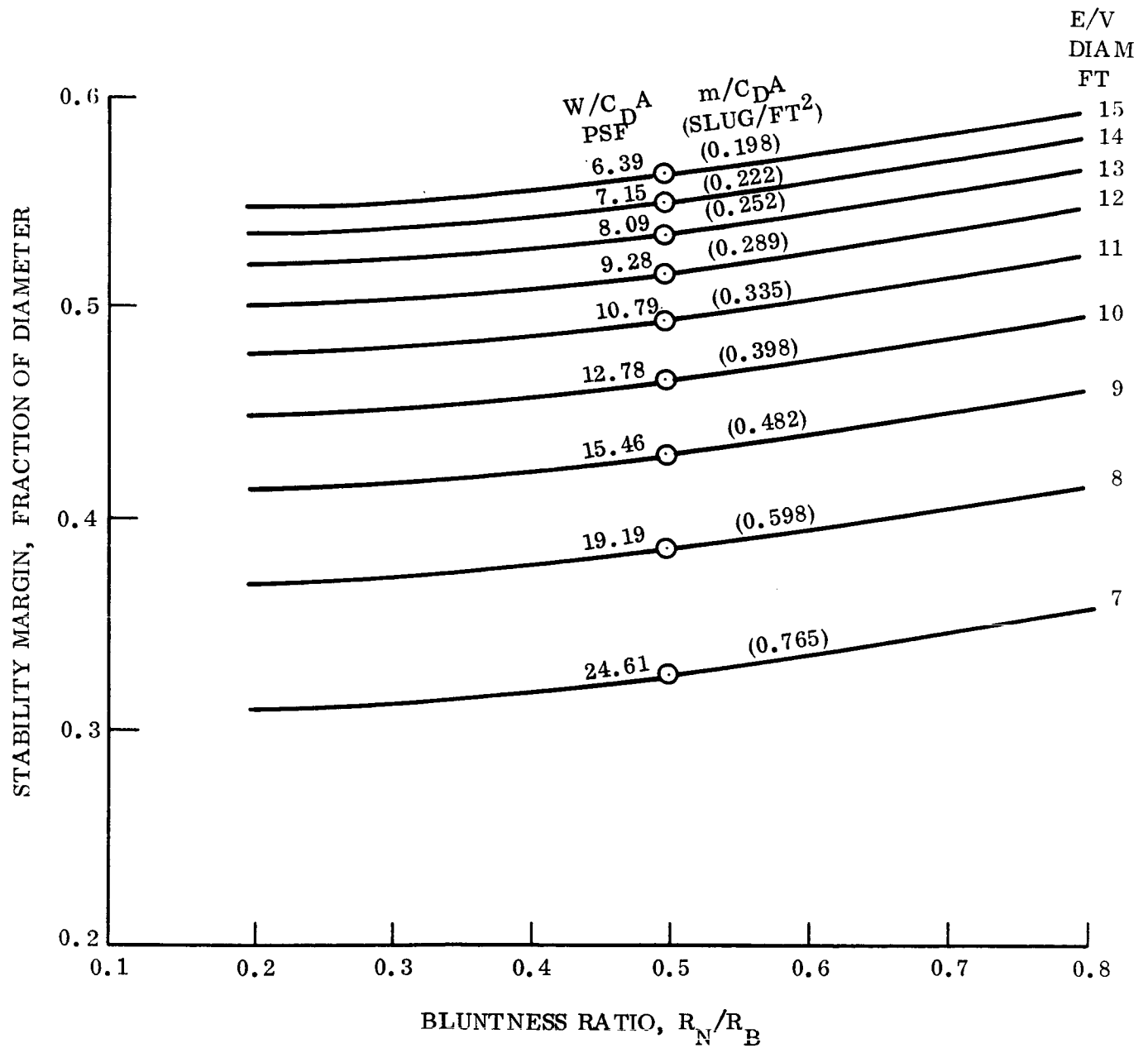


Figure 3.1.2-126. Stability Margin vs Bluntness Ratio

Figs. 3.1.2-127 and -128 present inertial property variations; roll-to-pitch moment of inertia ratios, and Capsule mass-to-pitch moment of inertia ratios, respectively. These are of particular significance for Capsule dynamic studies. The mass-to-pitch moment of inertia ratios vary from 0.17 to 0.29 per sq ft, indicating a low allowable spin rate for maintenance of dynamic stability during the final phases of ballistic flight. Cone angle and bluntness ratio have little influence upon this inertia parameter. The indicated variations are the result of a rather complex interaction between aeroshell and decelerator load properties and the c.g. location relative to the entry Capsule.

Entry Vehicle Design Characteristics Comparison - The following material is presented to permit comparison of Capsule design trends between the multi- and omni-directional Lander concepts as a function of decelerator load.

The data presented represents a Capsule geometry which utilizes a 60° cone angle and a bluntness ratio of 0.50. This selection was made to provide direct comparison with the weight and performance plots contained in the last section.

Total entry Capsule weight variations with these parameters are given in fig. 3.1.2-129 as a function of decelerator load. It is apparent that little variation in Capsule weight is encountered due to Lander concept for a given decelerator load. The entry Capsule weights are theoretically identical for a given decelerator load and Capsule diameter. Slight differences in spin/de-spin system weight allowances, aft cover weight, and heat shield weight perturbations account for this difference. Equivalent decelerator loads indicate a variation of 100 to 180 lbs in landed subsystems over the total decelerator load range between the omni- and multi-directional concepts. The entry Capsule weight variation with Lander concept is indicated as being between 10 and 15 lbs; approximating the probable CONSEP data accuracy (about a 0.1 percent variation in m/C_{DA} level). Examination of entry Capsule weight variations (at each diameter) with decelerator load indicates that almost all the weight variation is in the decelerator weight itself, reinforcing the low aeroshell weight fractions given for a single decelerator load in figs. 3.1.2-120 and -121. This results in almost perfect linearity between Capsule weight and decelerator load.

Comparison of the out-of-orbit and direct entry designs indicates the influence of the more severe, direct entry flight environment, coupled with the required use of a higher density heat shield material (35 versus 16 lb/ft³). Direct entry designs result in weight increases of 28, 68, and 137 lb over out-of-orbit designs for 7, 11, and 15 ft diameter Capsules, respectively.

Since the structural design concept is independent of m/C_{DA} and entry conditions, these weight increments are entirely due to heat shield weight variation.

The corresponding heat shield and structural weight variations are shown in figs. 3.1.2-130 and -131. The weight increments are nearly identical to those exhibited by the heat shield weight curve.

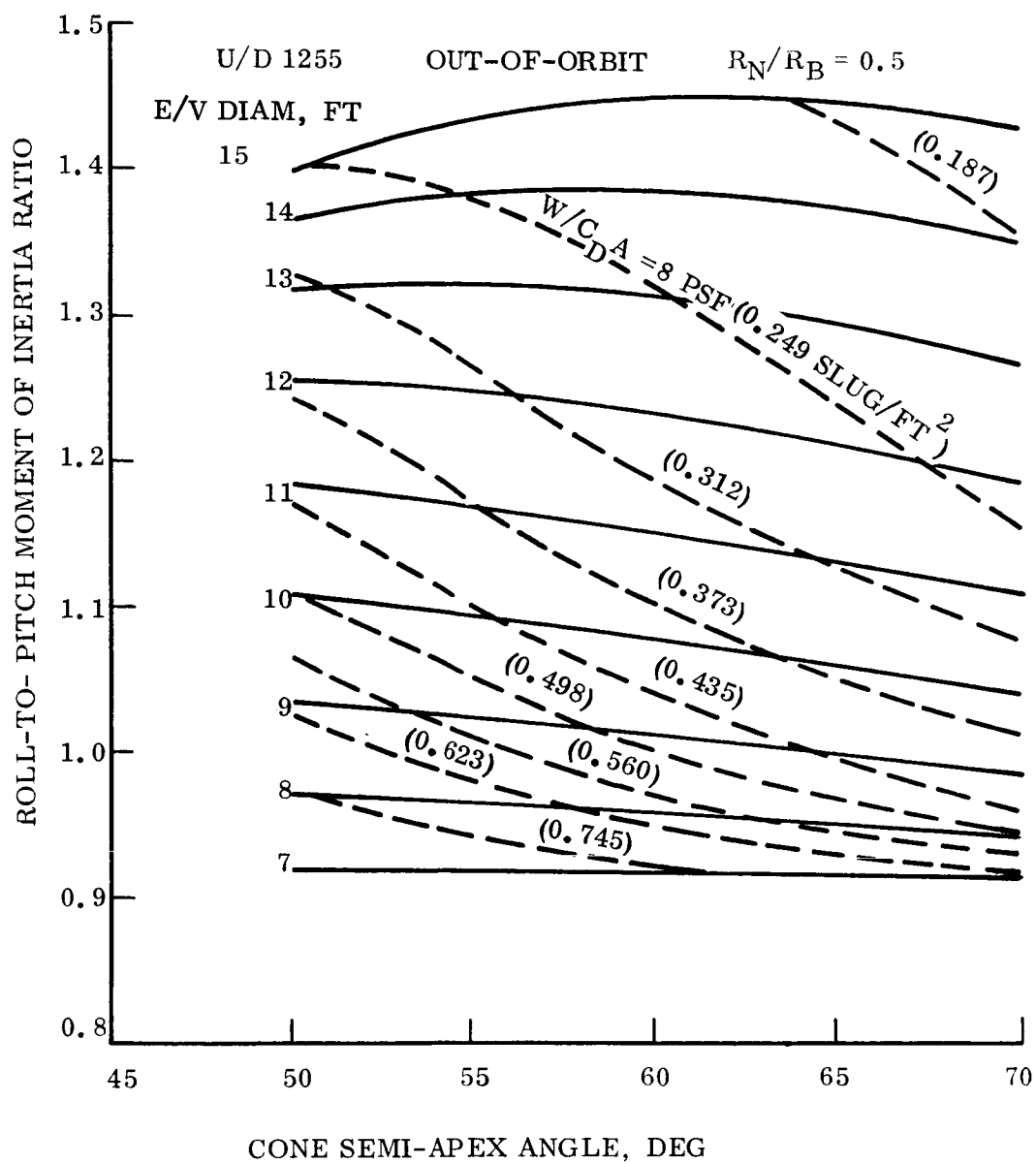


Figure 3.1.2-127. Roll-to-Pitch Inertia Ratio vs Cone Angle

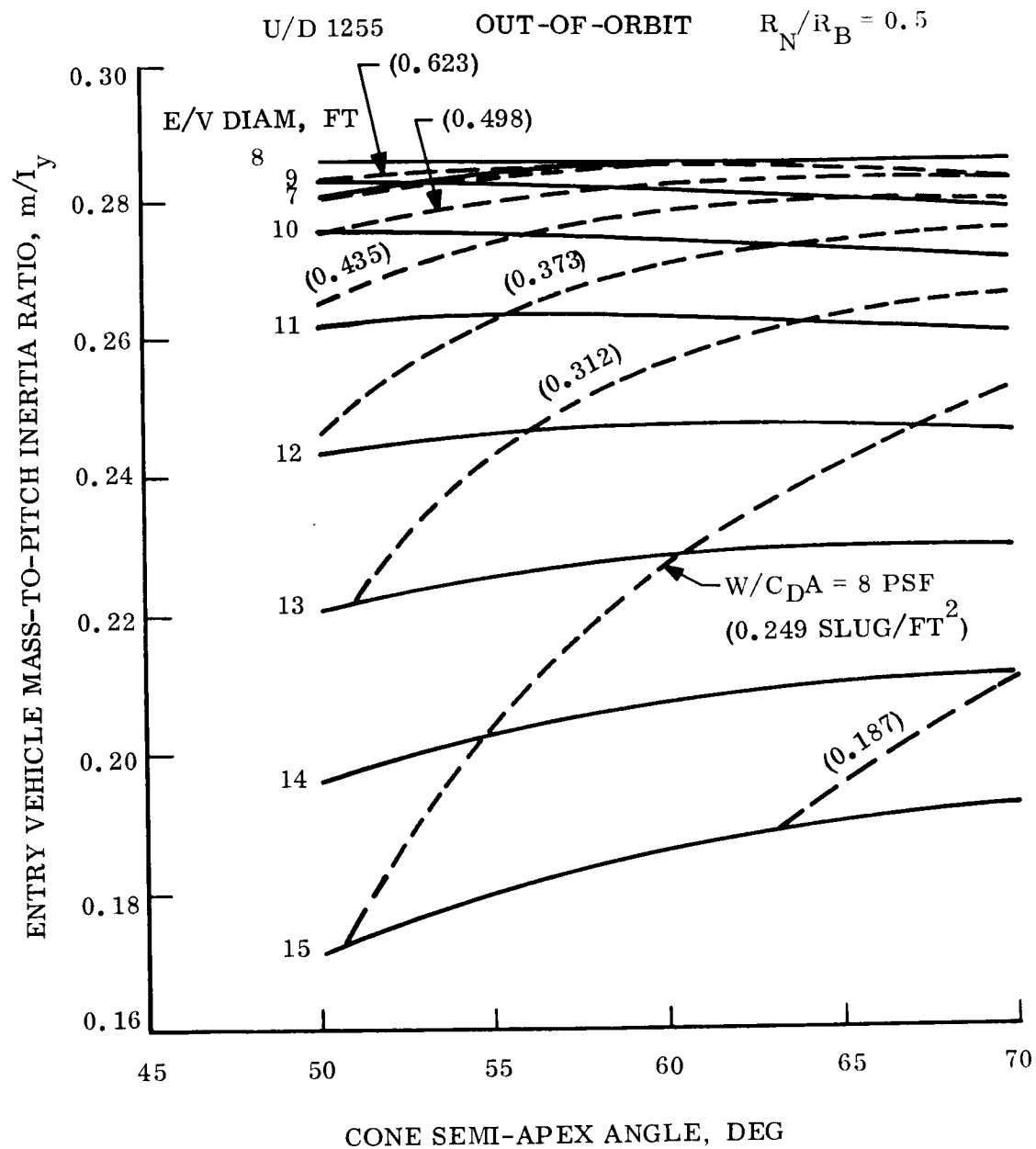


Figure 3.1.2-128. Mass-to-Pitch Inertia Ratio vs Cone Angle

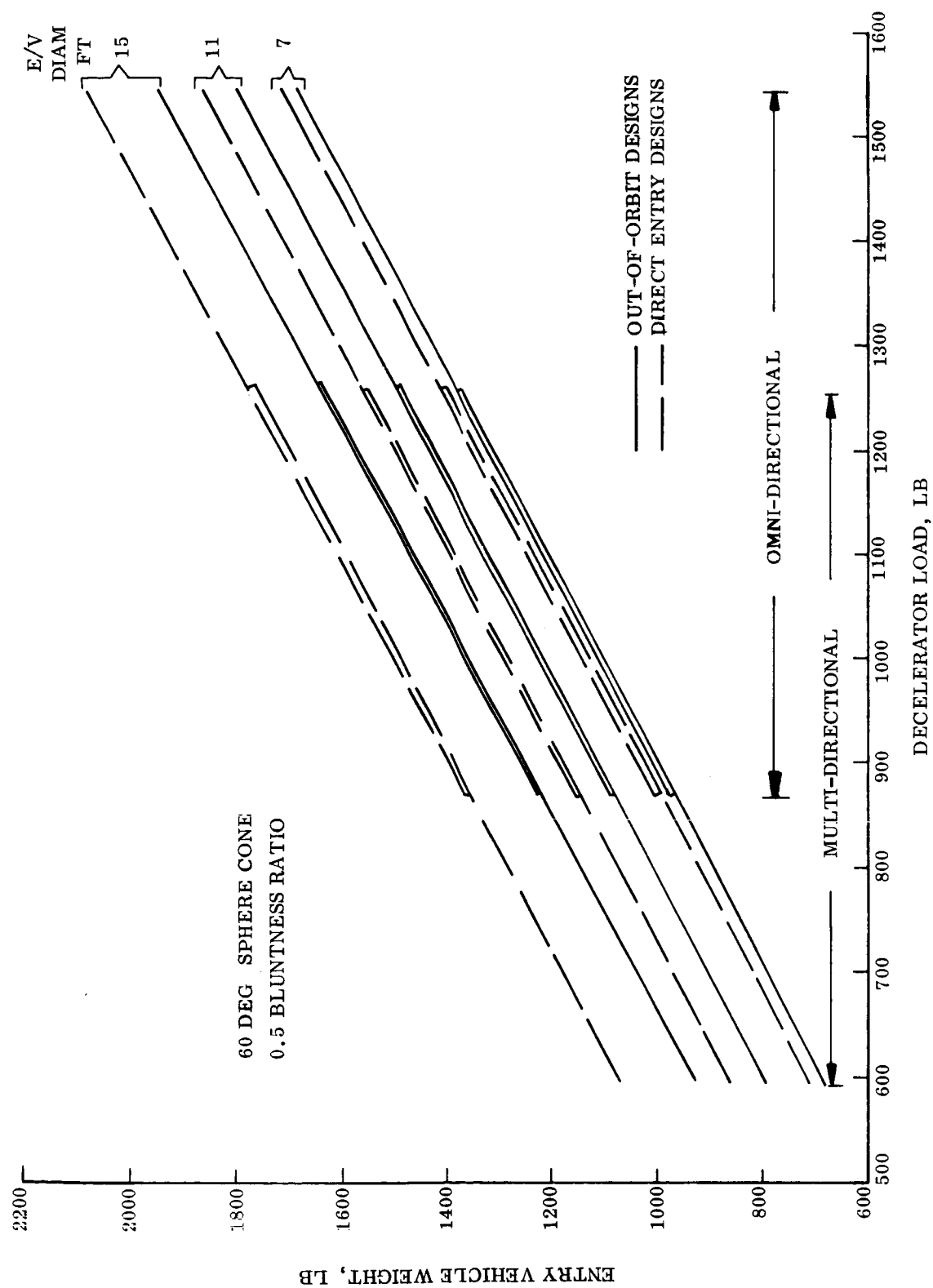


Figure 3.1.2-129. Entry Vehicle Weight vs Decelerator Load

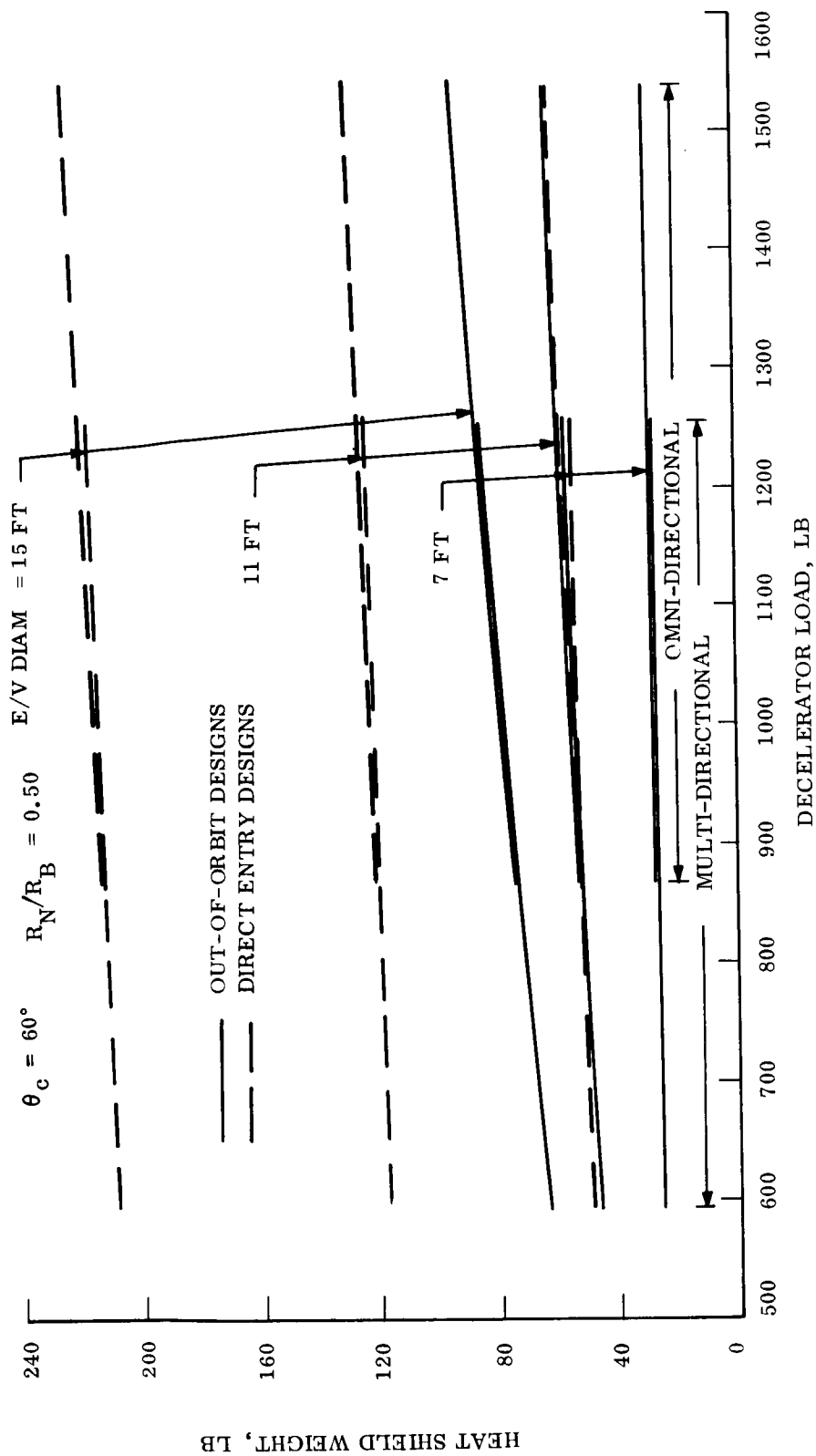


Figure 3.1.2-130. Heat Shield Weight vs Decelerator Load

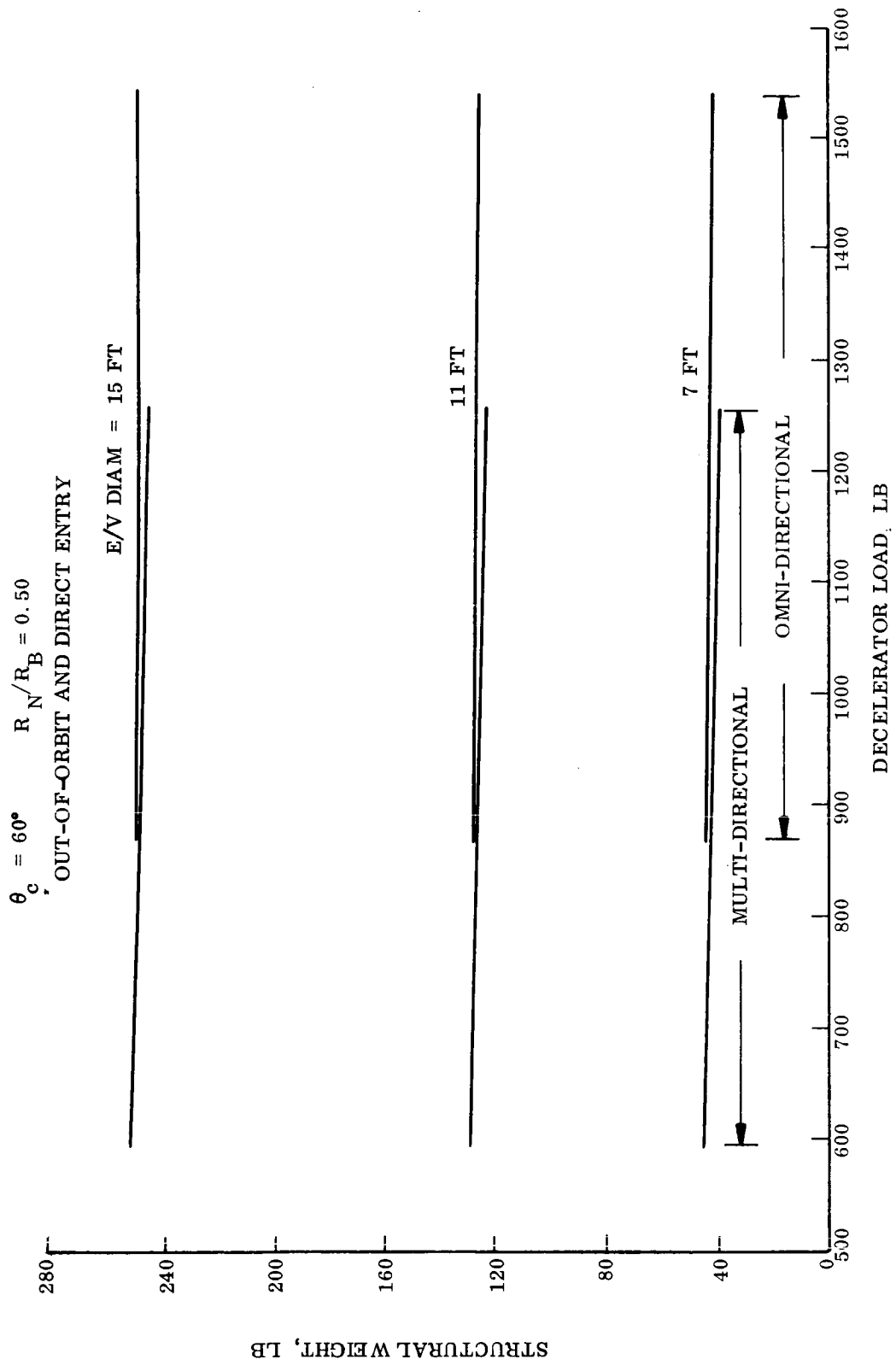


Figure 3.1.2-131. Stability Margin vs Bluntness Ratio

The 5 to 6 lb structural weight difference between Lander concepts is due to numerical approximations required for substantiation of existing CONSEP logic, in order to permit treatment of the aft-cover design approach.

Fig. 3.1.2-132 presents center of gravity location trends. Examination of the constant 7 ft diameter data at a given decelerator load, would indicate an anomaly. The larger diameter omni-directional package results in a vehicle c.g. farther forward than for the smaller multi-directional package. Since the corresponding aeroshell weights are approximately equivalent, the larger package fits farther into a given Capsule, as was confirmed by inspection of the geometric characteristics of decelerator loads M/D 1255 and O/D 1292, in fig. 3.1.2-111.

The point of tangency between the aeroshell and the forward lip of the M/D Lander occurs at a radius several inches greater in magnitude than for the O/D package. This is obviously due to the smaller radius of curvature, and lower center of curvature radial distance values of the O/D decelerator load. Since this positioning difference is constant in dimensional magnitude, increased diameter has the effect of reducing the X_{cg}/R increment. Lander tangency occurs within the nose cap for 15 ft diameter vehicles with a R_N/R_B of 0.5, thereby eliminating this effect. Increased decelerator load size, or reduction in Capsule diameter, both result in increased decelerator load protrusion from the aeroshell, thereby increasing $\frac{X_{cg}}{L}$ levels. The difference between constant diameter curve slopes is due to the influence of the normalizing Capsule forebody length.

The slightly more forward c.g. location of the direct entry Capsule is due to the influence of the heavier aeroshell.

The stability margin variations, shown in fig. 3.1.2-133, reflect the trends illustrated by the X_{cg}/L data. Stability margin, for the given cone angle and bluntness ratio, shows little sensitivity with either Lander concept or path angle selection.

Capsules having diameters of 7, 11, and 15 ft will display corresponding stability margin variations of 33 to 42, 49 to 54, and 56 to 59 percent of diameter.

Figs. 3.1.2-134 and -135 present the ratios of Capsule mass and roll moment of inertia with pitch moment of inertia respectively. Both ratios show comparative insensitivity to path angle design influences. A striking variation is noted with Lander concept, especially at low diameters. The mass-to-pitch inertia ratio data of fig. 3.1.2-134, indicates sensitivity to both decelerator load and Lander concept at the 7 ft diameter value. This is basically due to the m/I_y characteristics of the decelerator loads, shown in table 3.1.2-1. For both concepts, the decelerator load m/I_y ratio decreases with increasing size and weight. The two decelerator load concepts exhibit similar m/I_y values at decelerator loads near and above 1200 lb. This ratio, for a omni-directional load of 900 lb, is 62 percent greater than the corresponding multi-directional concept.

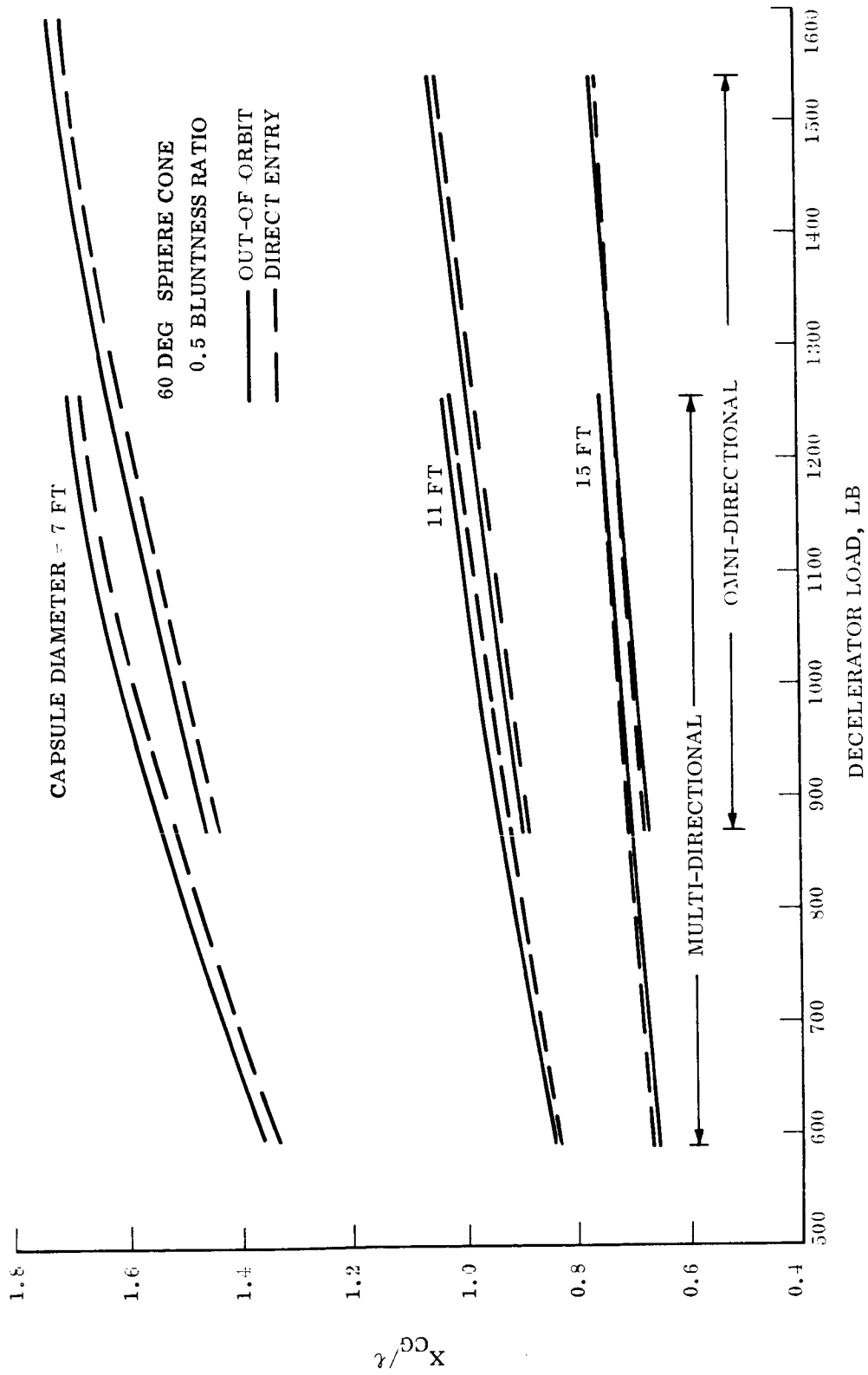


Figure 3.1.2-132. C.G. Location vs Decelerator Load

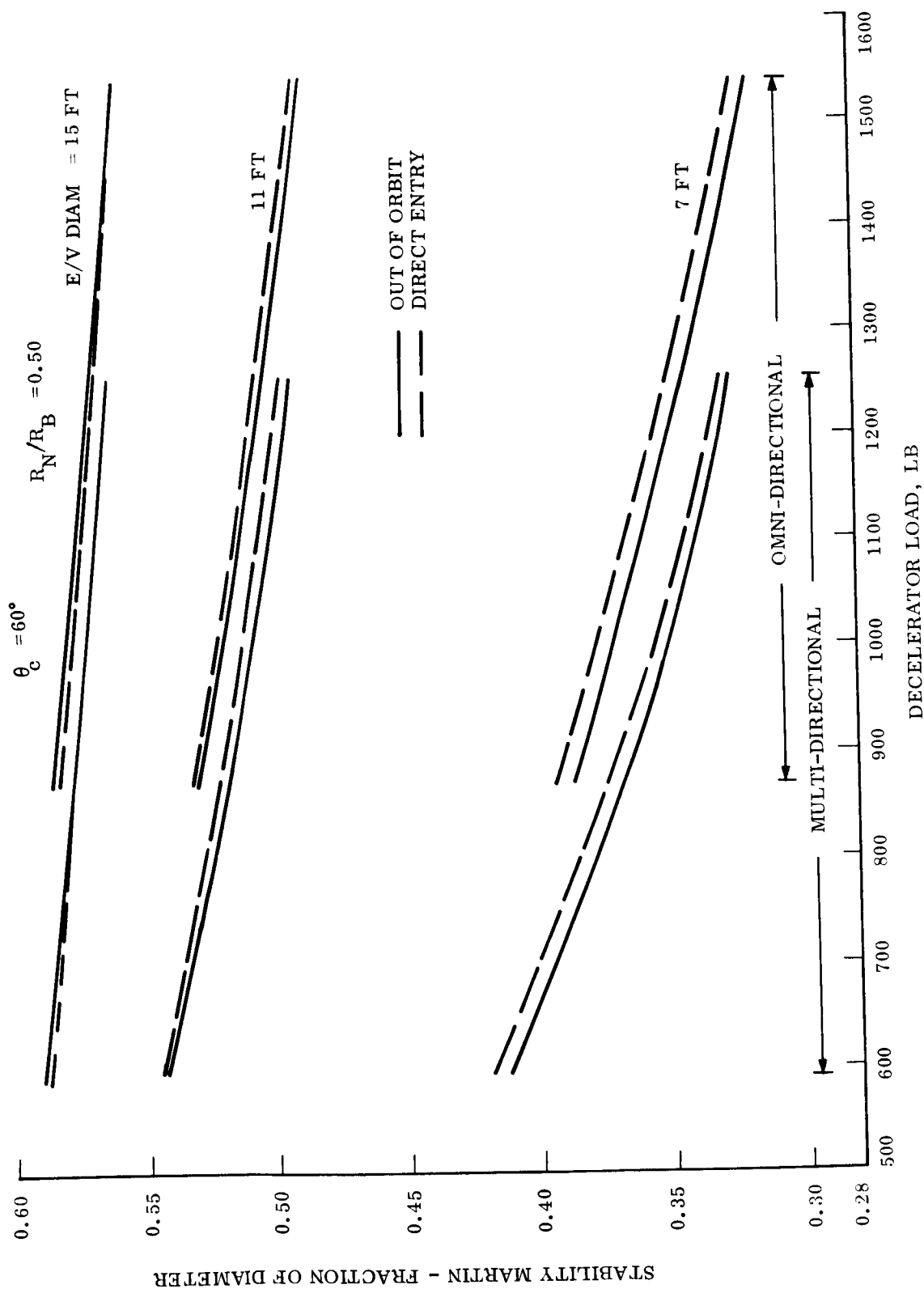


Figure 3.1.2-133. Stability Margin vs Decelerator Load

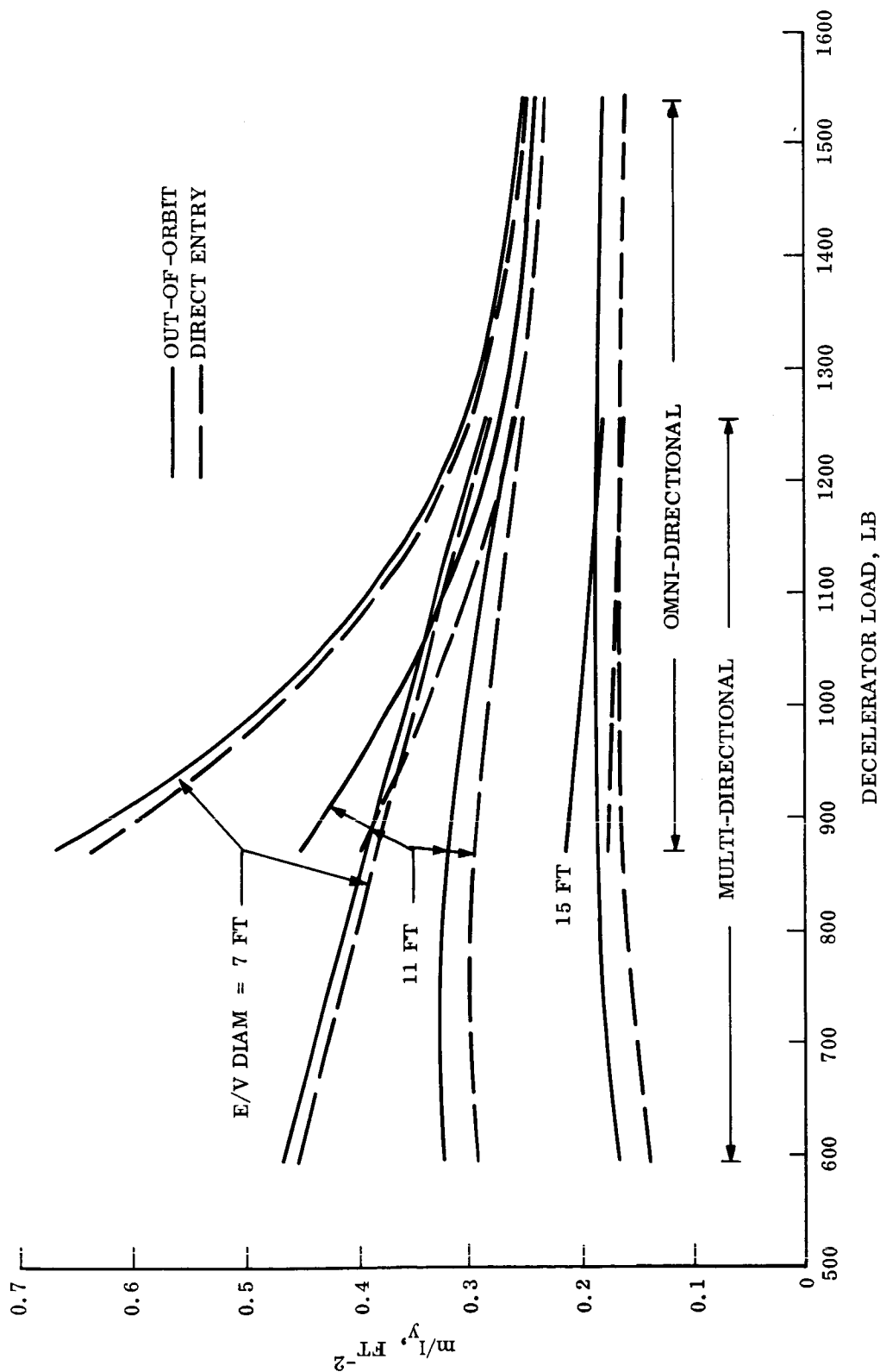


Figure 3.1.2-134. E/V Mass-to-Pitch Moment of Inertia Ratio vs Decelerator Load

This is directly reflected in the total Capsule m/I_y curves. Increased Capsule diameter results in increased aeroshell contribution to weight, with even larger attendant contributions to pitch moment of inertia. This tends to reduce the Capsule m/I_y level and lessens the influence of the decelerator load contributions.

The same comments apply to the inertia ratio curves of fig. 3.1.2-135. Greatest sensitivity to decelerator load occurs at the lowest diameter. Considerable variation with Lander concept is still noted for decelerator loads near 900 lbs (even for 15 ft diameter vehicles). This again is due to the basic decelerator load characteristics.

Awareness of these inertia ratios upon both exospheric and atmospheric attitude control would indicate that Capsule design at diameters on the order of 9 feet is desirable; i. e., such diameters provide I_x/I_y values slightly in excess of unity for exospheric attitude control, without incurring excessive roll rate limit penalties during final phases of entry.

3.1.2.2.2 Heat Shield and Structural Trade-Off Studies

A design parameter sensitivity study was performed to investigate 1) the influence of design backface temperature specification upon heat shield weight characteristics, and 2) the relative effect of structural materials and design concept upon structural weight increments. The nominal out-of-orbit and uncontrolled direct entry flight conditions utilized in the basic parametric design study were also incorporated in this effort; they are out-of-orbit at 15,300 fps and 16° and direct entry at 20,800 fps and 25° .

The resultant data are presented for decelerator loads incorporating the multi-directional Attenuator design load over a weight range from 594 to 1255 lbs. The vehicles have a 60° semi-apex angle, a bluntness ratio of 0.50, and base diameters of 7, 11, and 15 ft. This permits comparison with the data of the preceding sections.

A. Heat Shield Variation with Backface Temperature

The nominal design backface temperature specification was 600° F. This study employs values of 300° and 800° . The heat shield weight contributions for all three temperature levels are presented in fig. 3.1.2-136 as a function of decelerator load for both nominal flight paths. A common ordinate is used to permit flight environment comparisons.

The same influence of vehicle size and decelerator load on heat shield weights noted in the previous section are immediately apparent. The specification of an 800° F backface temperature results in a mean value heat shield weight decrease (from the 600° designs) of 7, 19, and 28 lbs for 7, 11, and 15 ft diameter out-of-orbit vehicles, respectively. The corresponding percentage changes are 26, 36, and 37 percent. The increased insulation requirements at the 300° level causes weight growths of 13, 37, and 64 lbs, over the 7 to 15 ft vehicle diameter range (48, 68, and 83 percent increase over the 600° requirement, respectively).

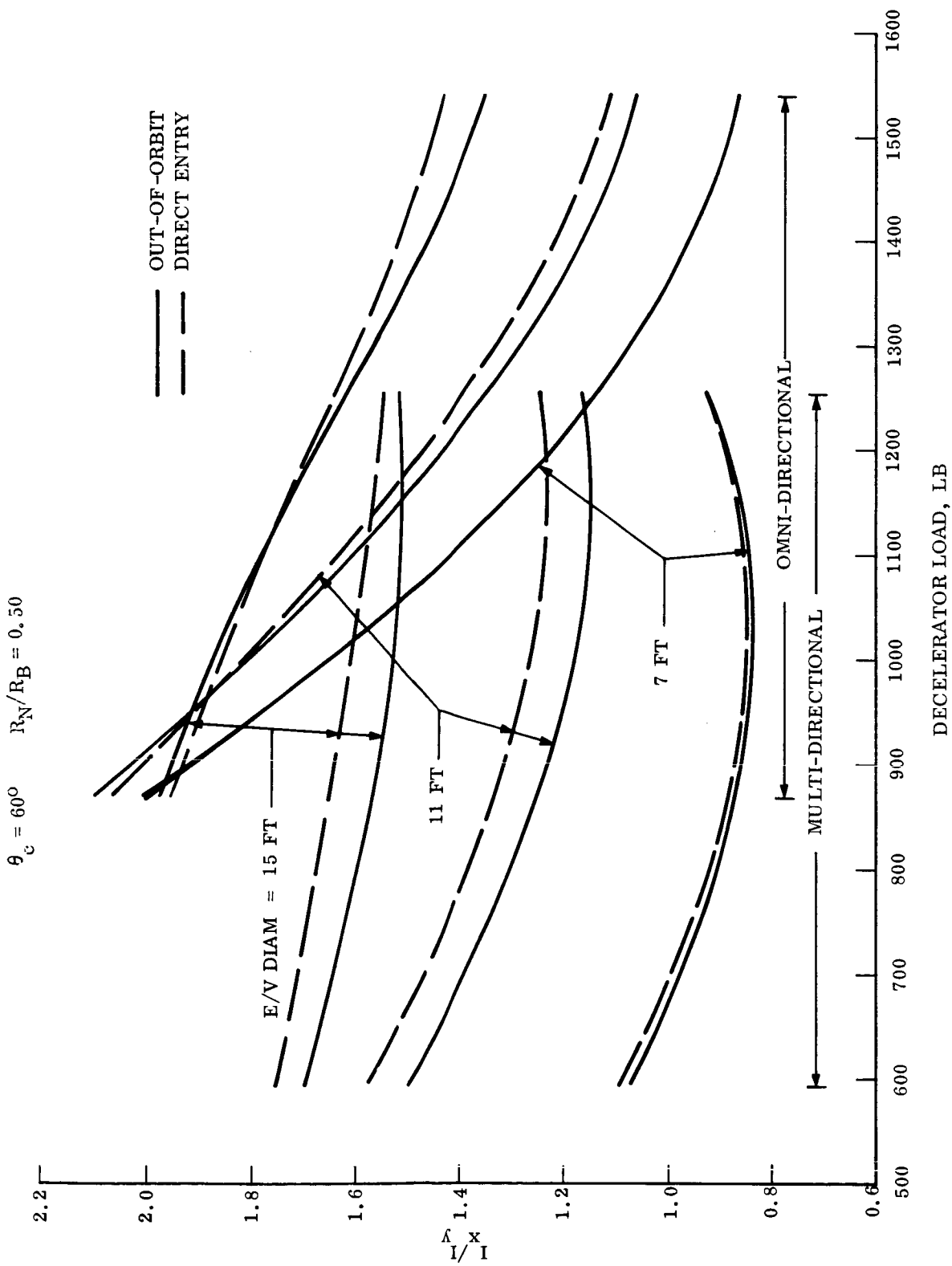


Figure 3.1.2-135. Roll-to-Pitch Moment of Inertia Ratio vs Decelerator Load

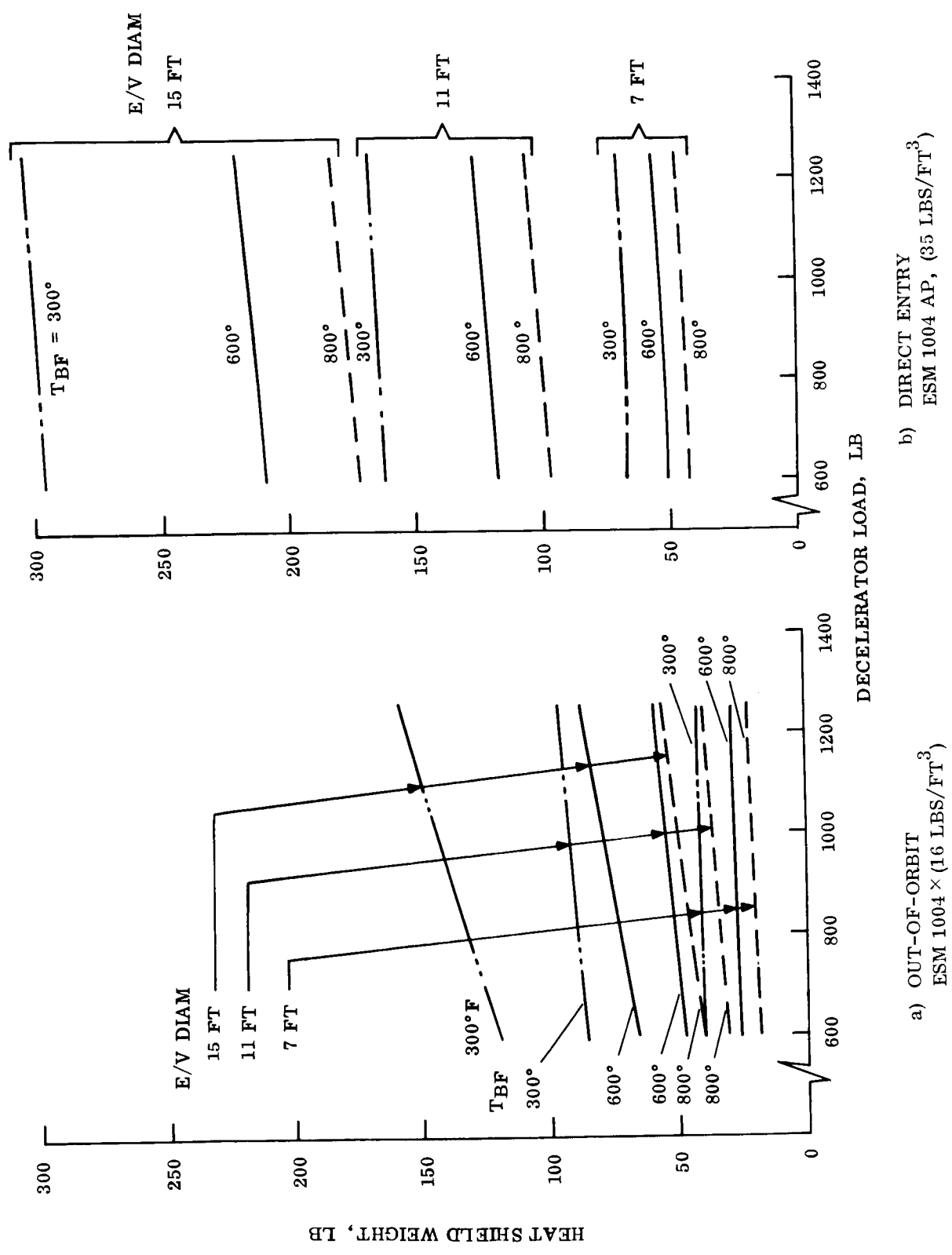


Figure 3.1. 2-136. Heat Shield Weight vs Decelerator Load

Inspection of the direct entry heat shield weight variations indicates that the changes in backface temperature (or insulation requirements) results in the same level of incremental weight variations as noted for the out-of-orbit case. The 1004 AP material is, however, about 2.2 times as heavy as the 1004 X ESM. This would indicate the direct entry insulation thickness variances are less than half those of the out-of-orbit case, despite the poorer insulative properties (due to increased density) of 1004 AP. This is due to the much-decreased heat pulse duration of the direct entry environment, which apparently offsets the higher heating rates.

A variation of design backface temperature from 800 to 300° F yields a 2 to 7 percent mean variation in heat shield weight fraction for both out-of-orbit and direct entry designs. This variation is not overly significant in terms of vehicle design weight, in view of analytical uncertainties. This analysis did not, however, include the effects of backface temperature on the structure. The use of an 800° specification with an aluminum structure, for example, would create considerable structural weight penalties.

B. Structural Weight Variation with Materials Selection

The initial design trend studies utilized a constant surface density, stainless-steel, minimum gage honeycomb structure. Subsequent structural analyses revealed that bending effects due to decelerator load inertia and thermal effects require more conservative design techniques. The structural design techniques available within CONSEP were therefore utilized for this portion of the study, together with the application of a 1.25 confidence factor.

This structural study treats stainless steel, titanium, and aluminum in separate combinations with either honeycomb or ring-stiffened shell. Material properties were those at a temperature of 300° F, in view of the long duration heat pulses characteristic of Mars entry. The structural weight results are shown in figs. 3.1.2-137 and -138 for out-of-orbit and direct entry designs.

The CONSEP procedure applies conservatism in both structural loads determination and in design techniques. Secondary weight variations would require more detailed analysis. CONSEP does, however, show the suitability of minimum or near minimum gage honeycomb structure for out-of-orbit designs. The 7 ft diameter design values are virtually identical to the direct entry designs. This indicates the applicability of near minimum gage design to smaller vehicles, even though a much more stringent loading condition exists:

$$V_o = 20,800 \text{ at } \gamma_e = 33^\circ \text{ and } W/C_D A = 20 \text{ (0.625 slug/ft}^2\text{)}$$

The 11 and 15 ft designs, however, exhibit structural weight increases on the order of 150 and 170 percent, respectively, over the minimum gage designs. These increases are primarily due to the application of differing confidence factors, which indicate a considerable departure from minimum gage design even though the vehicle $W/C_D A$ is considerably reduced from that for the 7 ft diameter design. It should be remembered that the design conservatisms inherent in CONSEP procedures have undoubtedly exaggerated this weight growth.

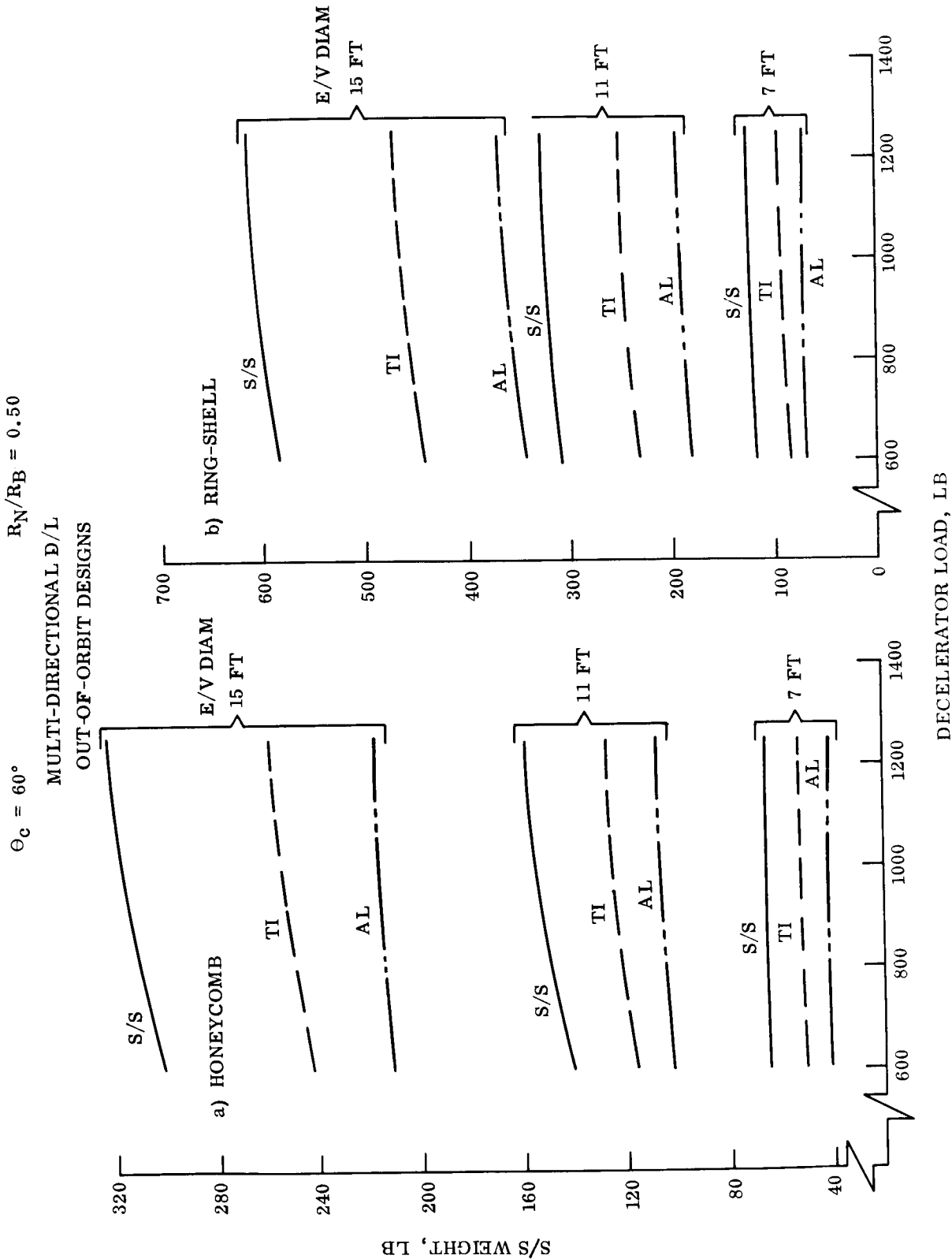


Figure 3.1.2-137. Structural Weight vs Decelerator Load, Out-of-Orbit Designs

MULTI-DIRECTIONAL D/L DIRECT ENTRY

$$\theta_c = 60^\circ \quad R_N/R_B = 0.50$$

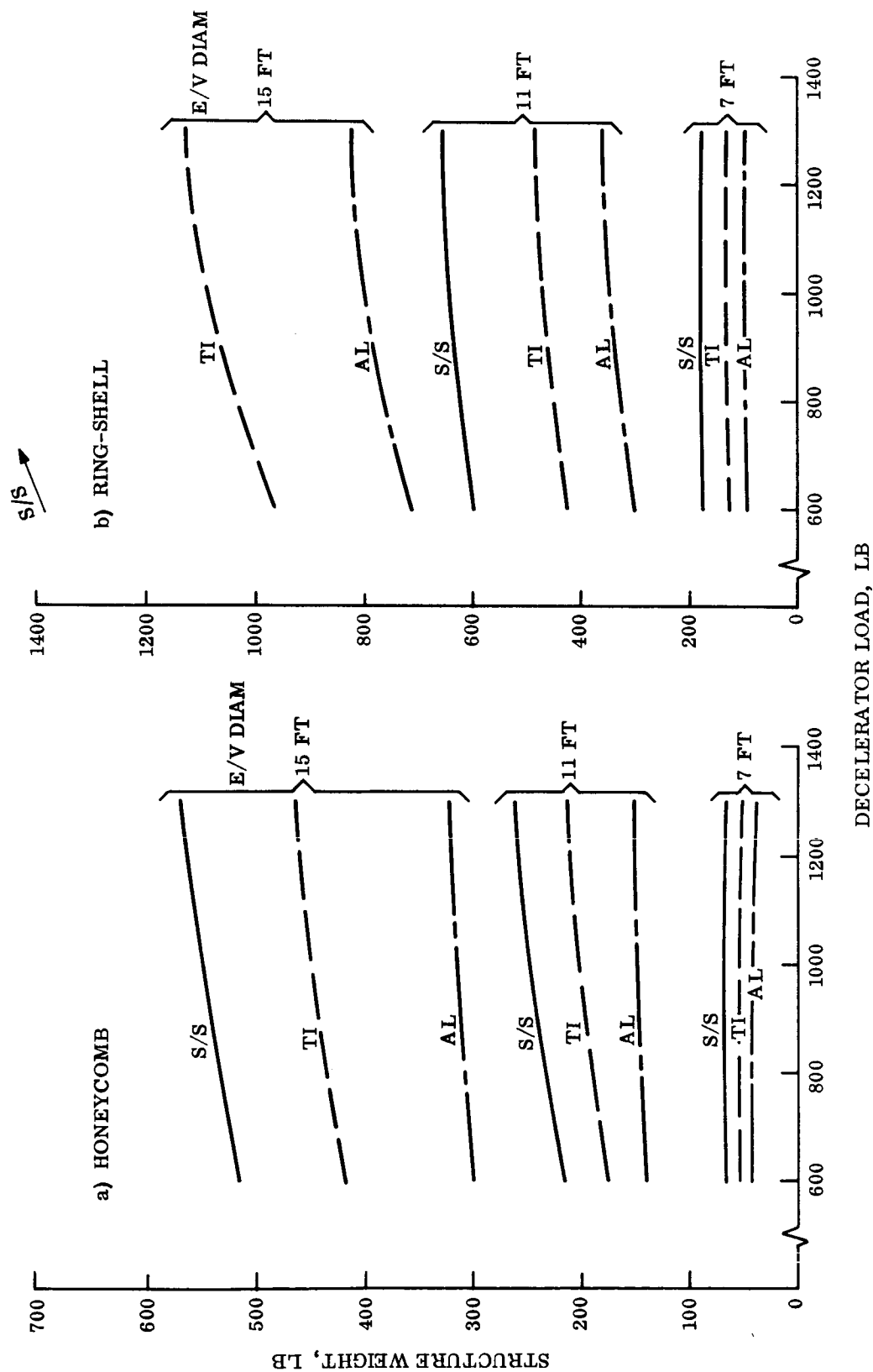


Figure 3. 1. 2-138. Structural Weight vs Decelerator Load, Direct Entry Designs

The trends however, indicate that selection of applied confidence factors and attention to ring-optimization techniques can be of significance for large diameter direct entry designs. This area has been investigated with further refinement of structural analysis techniques as described in the section on Structural Performance.

Comparison of Honeycomb and Ring-Shell Structural Concepts - Figs. 3.1.2-137(b) and -138(b) illustrate the ring-stiffened shell weight variations. The forebody structural weight contributions shown in fig. 3.1.2-137(b) for the direct entry designs are twice the forebody weight of the corresponding material honeycomb concepts. The direct entry data of fig. 3.1.2-138 indicates that the increased aerodynamic load results in even more substantial weight increases up to a factor of three over the corresponding honeycomb designs. It is therefore concluded that the ring-stiffened shell type of structural concept is nearly competitive with the honeycomb concept only for small diameter, out-of-orbit designs where the incurred weight penalty is a small percentage of total vehicle weight.

3.1.2.2.3 Effects of Entry Conditions Upon Heat Shield Design

The design parameter studies resulted in vehicle designs for nominal out-of-orbit and direct entry flight path characteristics, with the nominal path parameters given in table 3.1.2-10 for each of the two flight path types and associated re-entry angle tolerances felt to be typical of total system performance. This sensitivity study illustrates the effect of selected variations in re-entry velocity, and the indicated entry angle tolerances, upon heat shield weight requirements. The condition perturbations are summarized in table 3.1.2-14.

TABLE 3.1.2-14. ENTRY CONDITION VARIATIONS

Entry Mode	NOMINAL		VARIATIONS	
	V_e (fps)	γ_e (H/S) (deg)	V_e (fps)	γ_e (H/S) (deg)
Out-of-Orbit	15,300	14	15,300	16
			15,300	18
			14,000	14
			16,000	14
Direct Entry	20,800	17	20,800	25
			20,800	33
			19,000	17
			21,500	17

The entry angles referred to in this table are the values input to the CONSEP program for heat shield design purposes. The nominal values reflect maximum design practice requirements as they represent the minimum entry angle expected for each entry mode. Thus, the increased entry angles would be expected to show less required heat protection. Figs. 3.1.2-139(a) and -140(a) show that increased entry angles do result in decreased thermal protection. The velocity perturbations at constant entry angle also show predictable influences upon heat shield requirements.

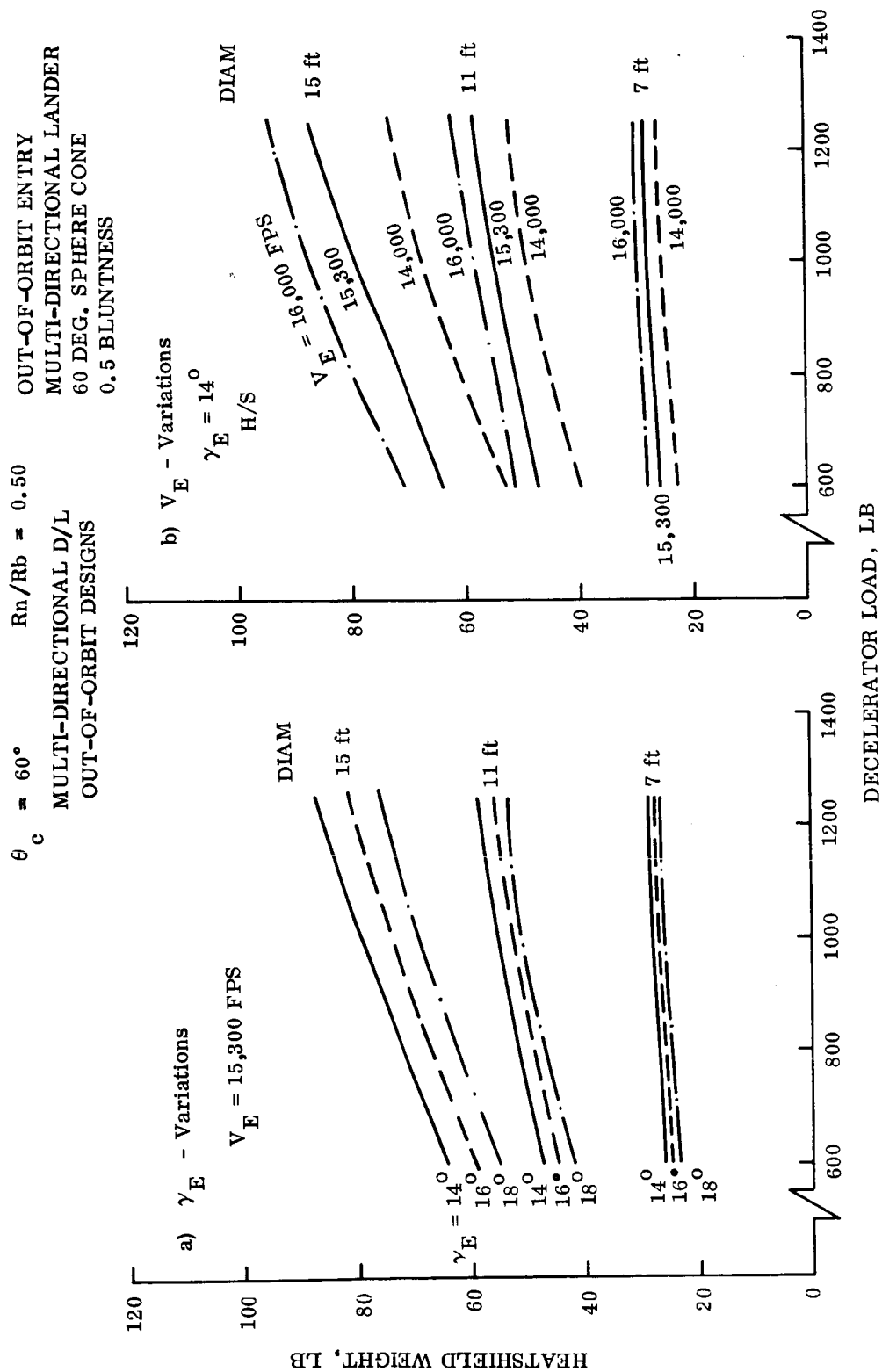


Figure 3.1.2-139. Heat Shield Weight vs Decelerator Load, Out-of-Orbit Designs

$$\Theta_c = 60^\circ \quad R_N/R_B = 0.50$$

MULTI-DIRECTIONAL D/L DIRECT ENTRY DESIGNS

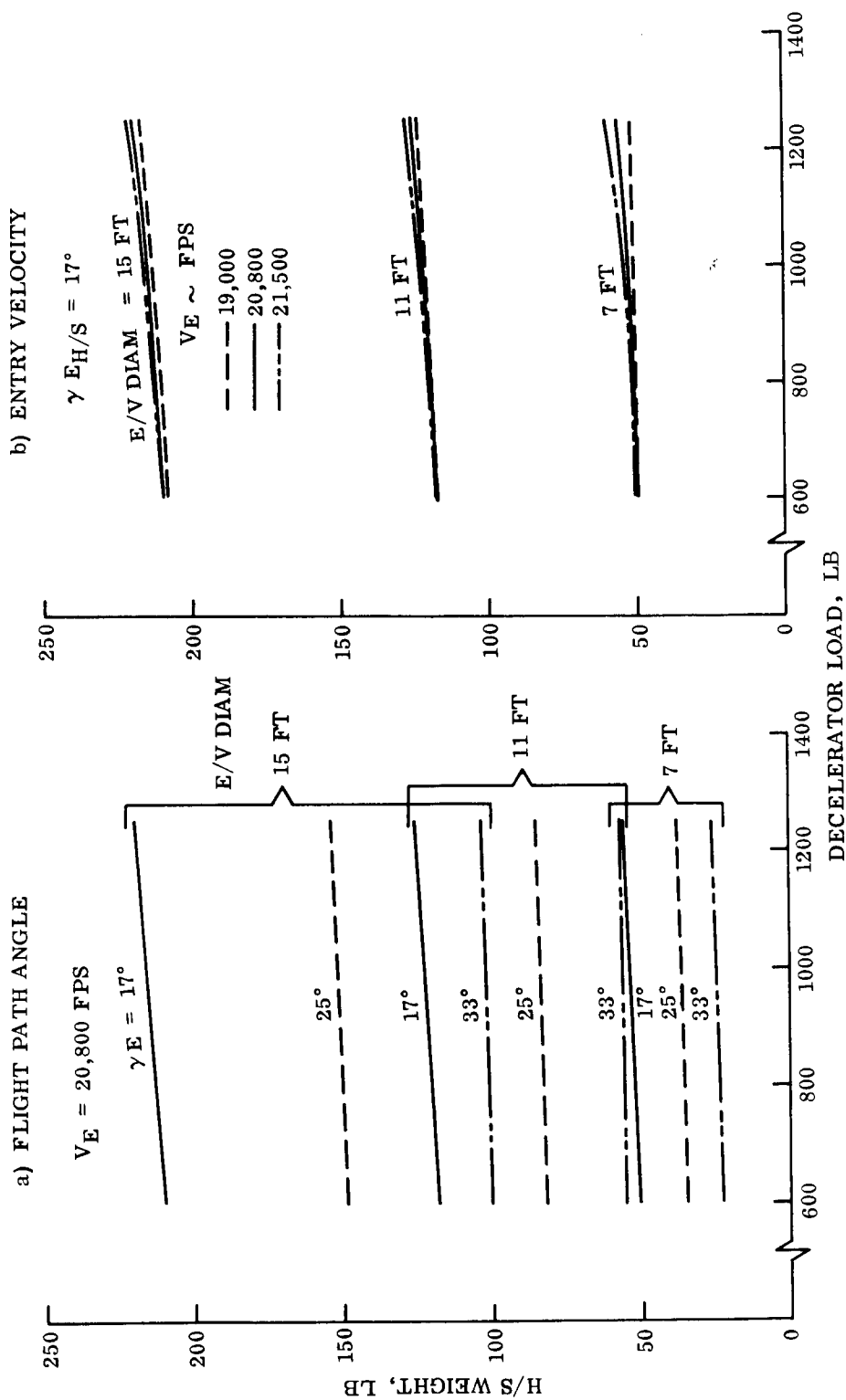


Figure 3.1.2-140. Heat Shield Weight vs Decelerator Load, Direct Entry Designs

The representative velocity perturbations result in greater heat shield weight increments than the entry angle variations for out-of-orbit designs. The converse is true for the direct entry designs. This is primarily due to the large entry angle tolerance characteristic of direct entry. That further entry velocity increase for direct entry designs above 21,500 fps, at an entry angle of 17° , would result in near skip trajectories. This in turn, results in much more substantial heat shield weight increases than indicated in fig. 3.1.2-140(b).

The trends shown in these two figures indicate that out-of-orbit designs, in terms of total entry vehicle weight variations due to thermal protection requirements, are not significantly sensitive to representative variations in entry conditions. The direct entry design heat shield results suggest a greater degree of sensitivity (still less than 10 percent of E/V weight), and justify the necessary use of minimum entry angles during heat shield design.

This flight environment sensitivity study was originally intended to include associated effects upon aeroshell structural design trends. The results of the previous para (B) indicate that further structural analysis, and associated improvement of CONSEP procedure, is necessary before the accuracy of the results would fall within the range of the expected variations.

3.1.2.3 Inflatable Afterbody Trade-off Studies

3.1.2.3.1 General Approach

From paras 3.1.2.1 and 3.1.2.2, it can be seen that direct entry using an aeroshell which fits within the 10 ft shroud limit ($D_0 = 8.34$ ft) permits only small entry weights in order that aerodecelerator requirements are met (i.e., Mach = 2 at a reasonable altitude). For direct entry at the maximum flight path angle of 40° , the allowable ballistic coefficient is on the order of 5.6 psf (0.175 slug/ft²). This combined with a diameter of 8.34 ft, results in an entry weight of 500 lb (assuming $\theta_c = 60^\circ$). This weight is less than the weight of a multidirectional design Lander carrying the minimum science package.

Therefore, in an attempt to achieve more desirable entry weights without exceeding the 10 ft shroud limit, inflatable afterbodies for the sphere cone configurations have been investigated. Both inflatable afterbodies deployed before entry and during entry at $M = 5$ have been considered. The pre-entry concepts have received the major emphasis due to their greater potential for increasing the entry weight and decelerator load. Fig. 3.1.2-141 presents the allowable entry weight as a function of inflated diameter and ballistic coefficient; considering a maximum allowable $\beta = 5$ psf (0.175 slug/ft²) illustrates the gain in entry weight possible for direct entry at $\gamma_e = 40^\circ$. In order to evaluate the weight penalty associated with the use of an inflatable afterbody, a subcontract was awarded Goodyear Aerospace Corporation for preparation of designs of an inflatable afterbody based on their Airmat concept. In addition, work was initiated in-house utilizing technology developed as part of the GE-MOOSE and LIFERAFT studies. The following sections describe the inflatable concepts and present parametric design data. A preliminary assessment of the capability of an inflatable ballute (tucked-back type) deployed after entry but at high supersonic Mach numbers is also presented.

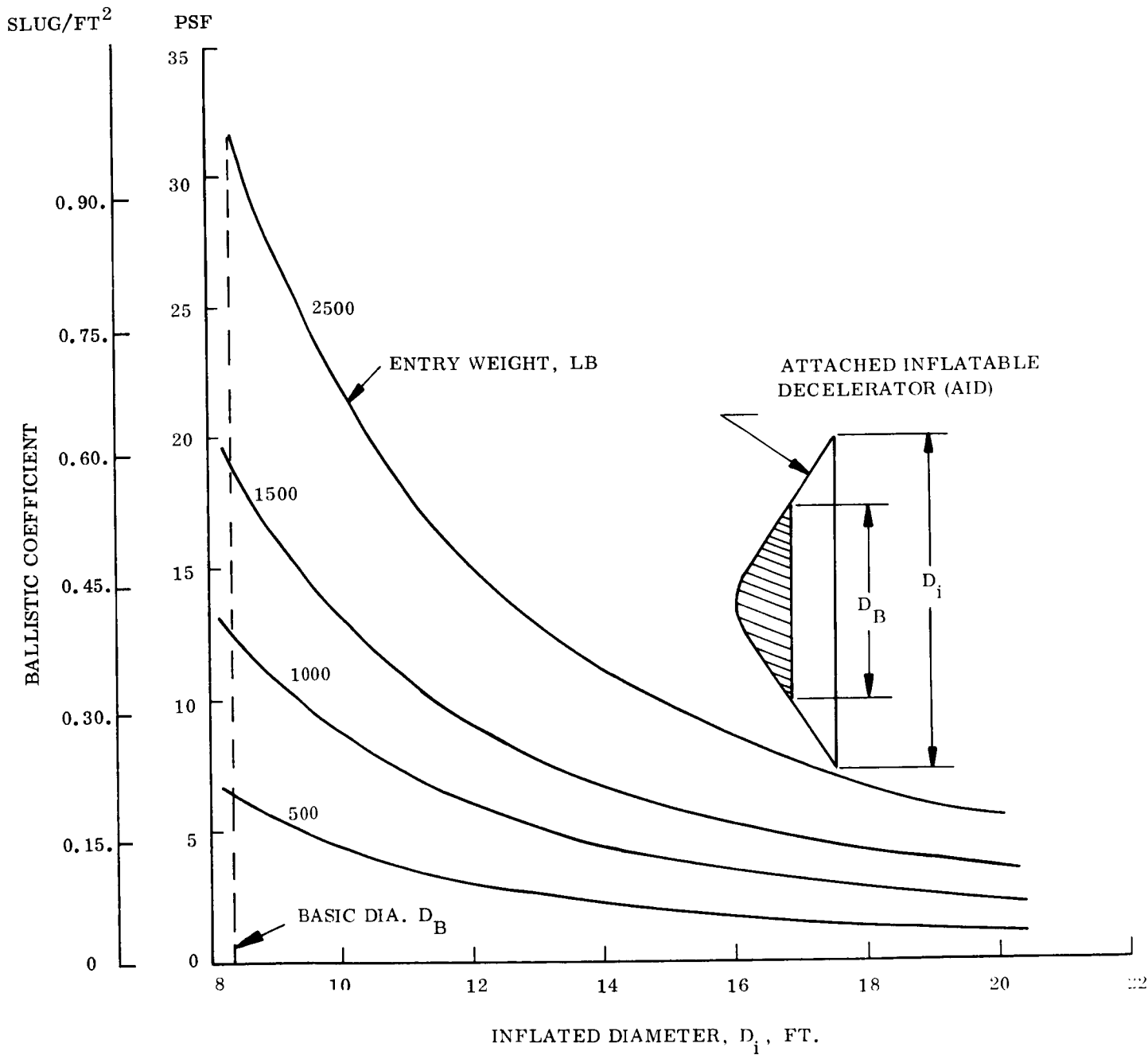


Figure 3.1.2-141. Auxiliary Decelerator, Ballistic Coefficient Variation with Inflated Afterbody Diameter for Various Entry Weights

3.1.2.3.2 Goodyear Airmat Concept

This sub-section presents the results of preliminary parametric analyses for an attached, inflatable Airmat cone decelerator for a Mars entry vehicle as sketched in fig. 3.1.2-142. The purpose of the analyses is to establish the weight requirements for inflatable decelerator subsystems for a range of aeroshell and decelerator diameters. This work was accomplished in accord with agreements made between personnel of the General Electric Company - Re-entry Systems and the Goodyear Aerospace Corporation.

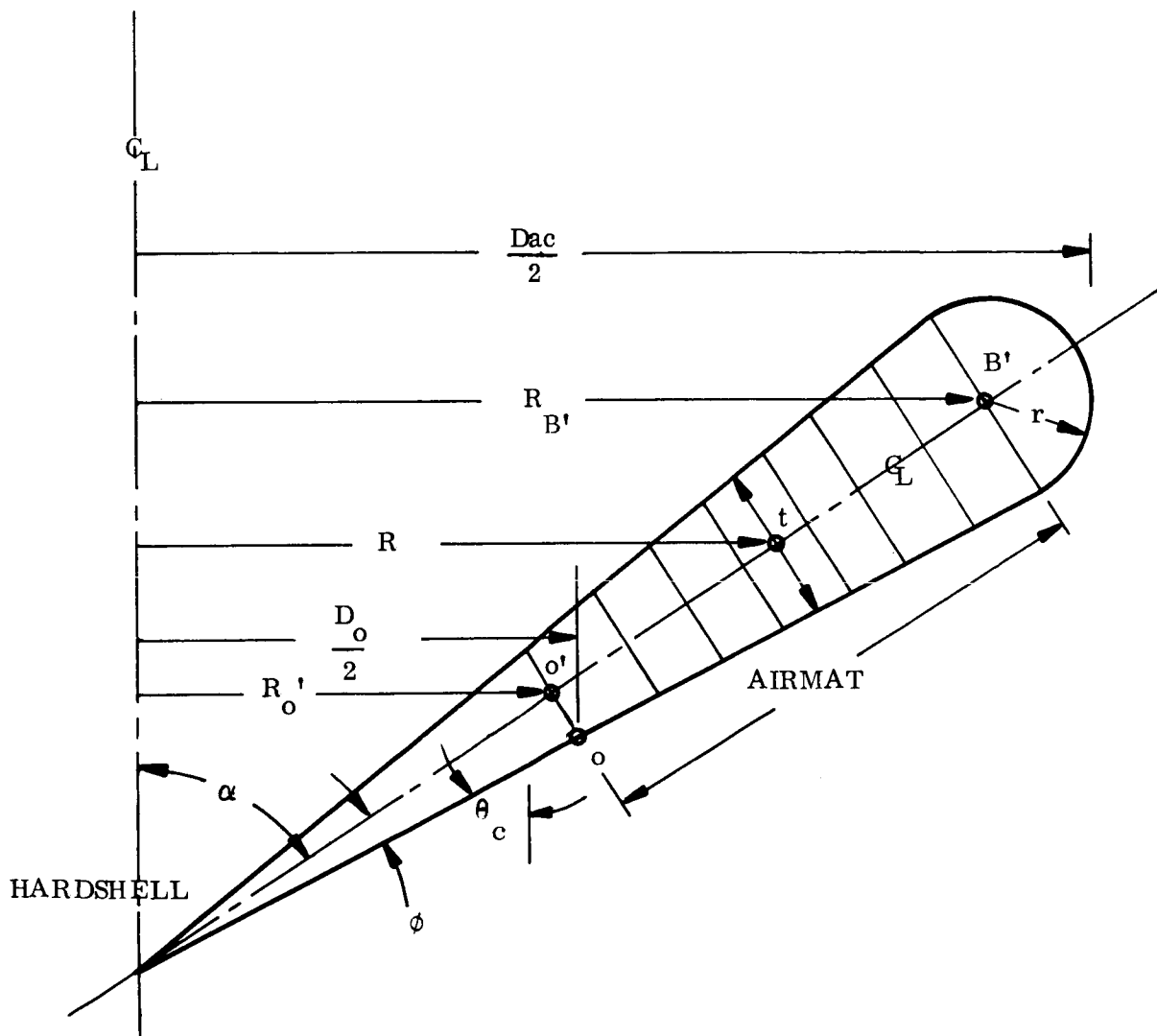


Figure 3.1.2-142. Inflatable Airmat Cone Geometry

The following constraints and requirements were used in developing the inflatable subsystem weight:

1. Direct entry into Mars atmosphere will be considered. An entry velocity of 20,800 ft/sec in conjunction with an entry angle of $30^\circ \pm 10^\circ$ will be assumed. Structural design will be based upon an entry angle of 20° into the VM-3 atmosphere. To account for non-equilibrium and radiative heating to the vehicle, a factor of 1.5 times the convective value shall be used to calculate heat protection requirements.
2. The aeroshell configuration will be a blunt sphere cone having a half cone angle, θ_c , of 60° . The bluntness, nose-to-base radius ratio equals 0.50. The effect of varying the cone half angle (between 50° and 70°) upon total subsystem weight shall be indicated. The aeroshell base diameter (D_o) will be assumed to vary from 5 to 9 ft, and the effective diameter with the decelerator inflated shall vary from 10 to 15 ft (D_{AC}). A vehicle ballistic coefficient of 6.3 lb/sq ft ($m/C_D A = 0.196$ slug/ft²)(decelerator inflated) will be assumed throughout.
3. The decelerator will be inflated before entry into the Mars atmosphere.
4. All material and components required for the decelerator subsystem should be selected for compatibility with the Mars sterilization requirements of 6 cycles of 135°C temperature and long-life packaging.

A. Airmat-Envelope and Inflation System

Figs. 3.1.2-143 and -144 relate the Airmat cone envelope weight to Airmat cone diameter with the "hard" aeroshell diameter and cone half angle as parameters.

From considerations of the basic Airmat cone geometry (fig. 3.1.2-142), external aerodynamic loads and material physical characteristics, the following equation for the Airmat cone weight can be developed.

$$W_{AC} = (S. F.) C_{Pmax} q_{max} \frac{D_{AC}^3 (\pi) K_W \left(\frac{\tan \alpha}{2 \sin^2 \alpha} \right)}{K_M^8 \left(1 + \frac{\tan \phi}{\sin \alpha} \right)^3} \left[(\cot \phi + \pi) \tan \phi - \left(\frac{D_o}{D_{AC}} \right)^2 x \right] \cdot \cdot \cdot \left[\left(\frac{2 K_e}{\cos \phi} \right) + \frac{1}{2} \left(\frac{D_o}{D_{AC}} \right) \right] \quad (10)$$

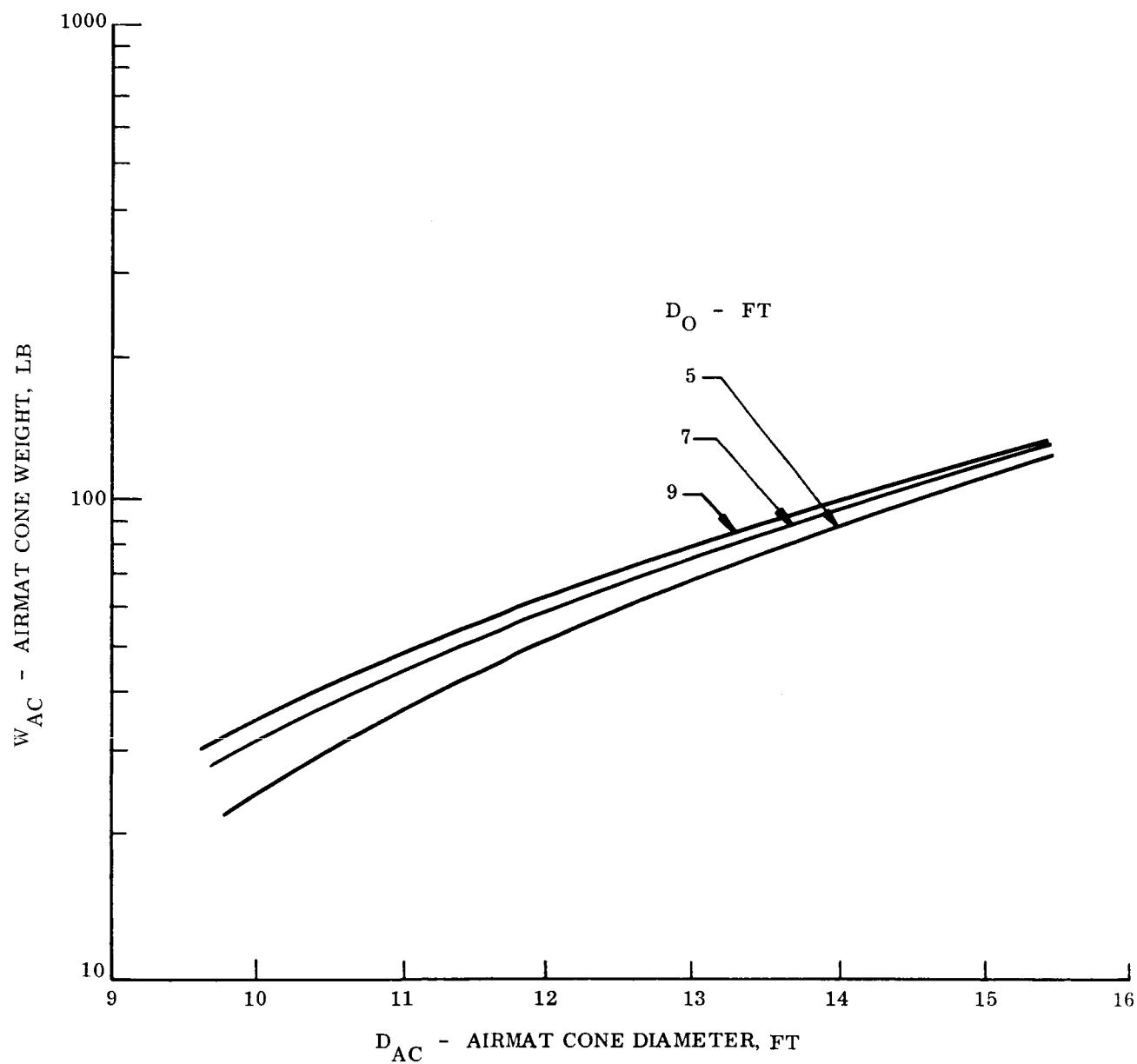


Figure 3.1.2-143. Airmat Cone Envelope Weight vs Diameter for Three Aeroshell Diameters and Cone Half Angle of 60°

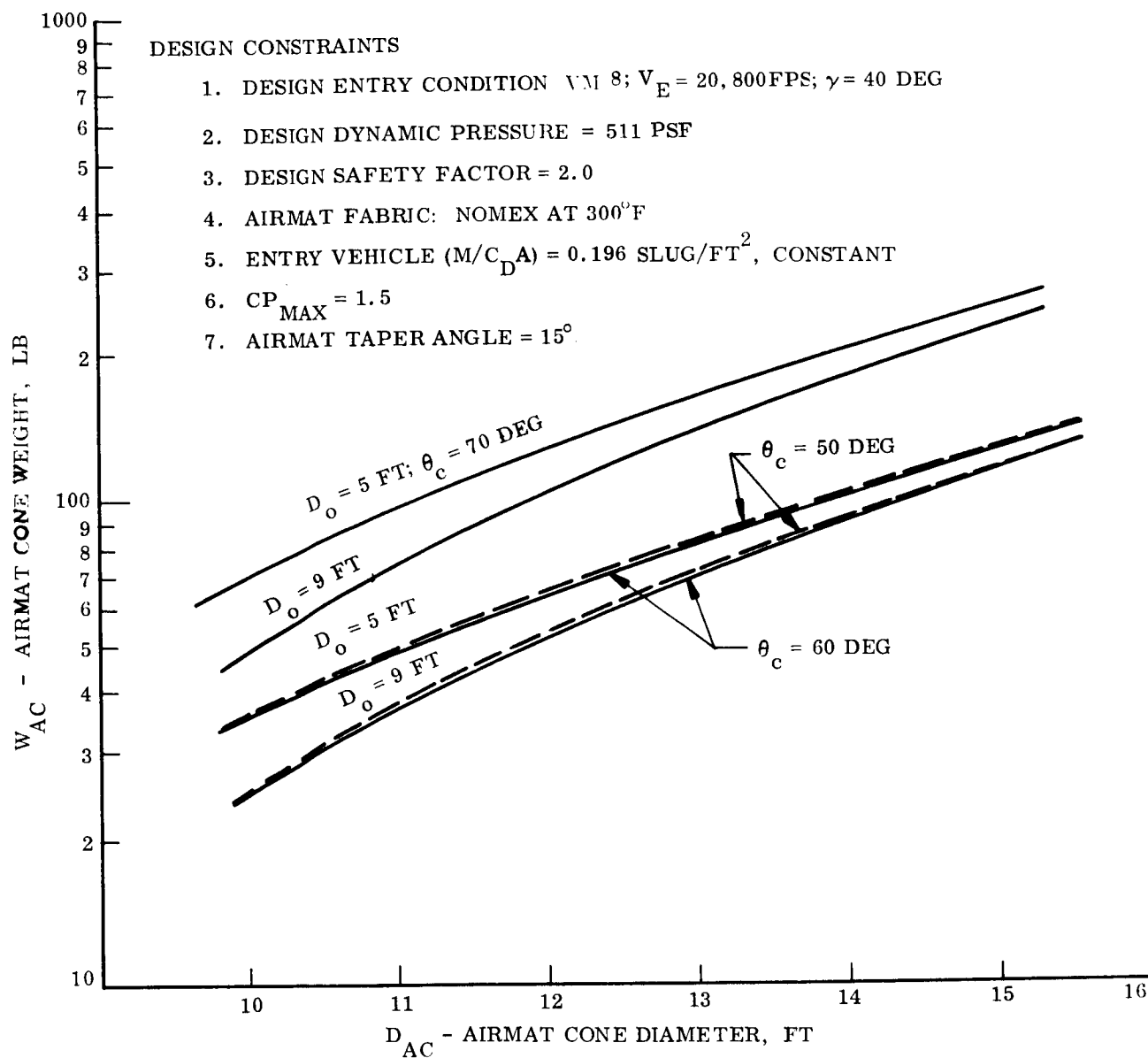


Figure 3.1.2-144. Airmat Cone Envelope Weight vs Diameter for Two Aeroshell Diameters and Two Cone Half Angles

where:

- (S. F.) = overall design safety factor (2.0)
- $C_{P_{max}}$ = surface aerodynamic pressure coefficient (1.5)
- q_{max} = maximum dynamic pressure in entry (511 psf)
- D_{AC} = Airmat cone diameter (10 to 15 ft)
- K_M = Airmat material strength/weight ration
(36,200 ft for coated Nomex fabric at 300°F)
- K_W = design factor to eliminate material wrinkling (1.5)
- K_e = design factor on material elasticity (assumed equal to 1.0)
- D_o = aeroshell diameter (5; 7; 9 ft)
- ϕ = Airmat taper angle (15°)
- θ_c = cone half angle (50°; 60°; 70°)
- α = ($\theta_c - \phi$)

Figs. 3.1.2-143 and -144 are developed by evaluating eq. 10 for the range of parameter values indicated above.

The Airmat cone device is essentially a pressure vessel requiring an auxiliary inflation gas source to retain its design dimensional relationships under an externally-applied, aerodynamic load. Fig. 3.1.2-145 presents the weight requirements for the Airmat cone inflation system weight as a function of Airmat cone diameter with aeroshell diameter and cone half angle as parameters. The inflation system weight is obtained from the following equation:

$$W_{IS} = A_1 A_2 P_I V \left(\frac{M}{18540 T} + \frac{3}{2} \frac{A_3}{K_T} \right) \quad (11)$$

where:

- A_1 = leakage factor (1.0)
- A_2 = factor for valves and lines (1.25)
- A_3 = gas storage bottle design safety factor (2.0)
- K_T = bottle material strength/weight ratio
(83,400 ft for Titanium at 70°F)

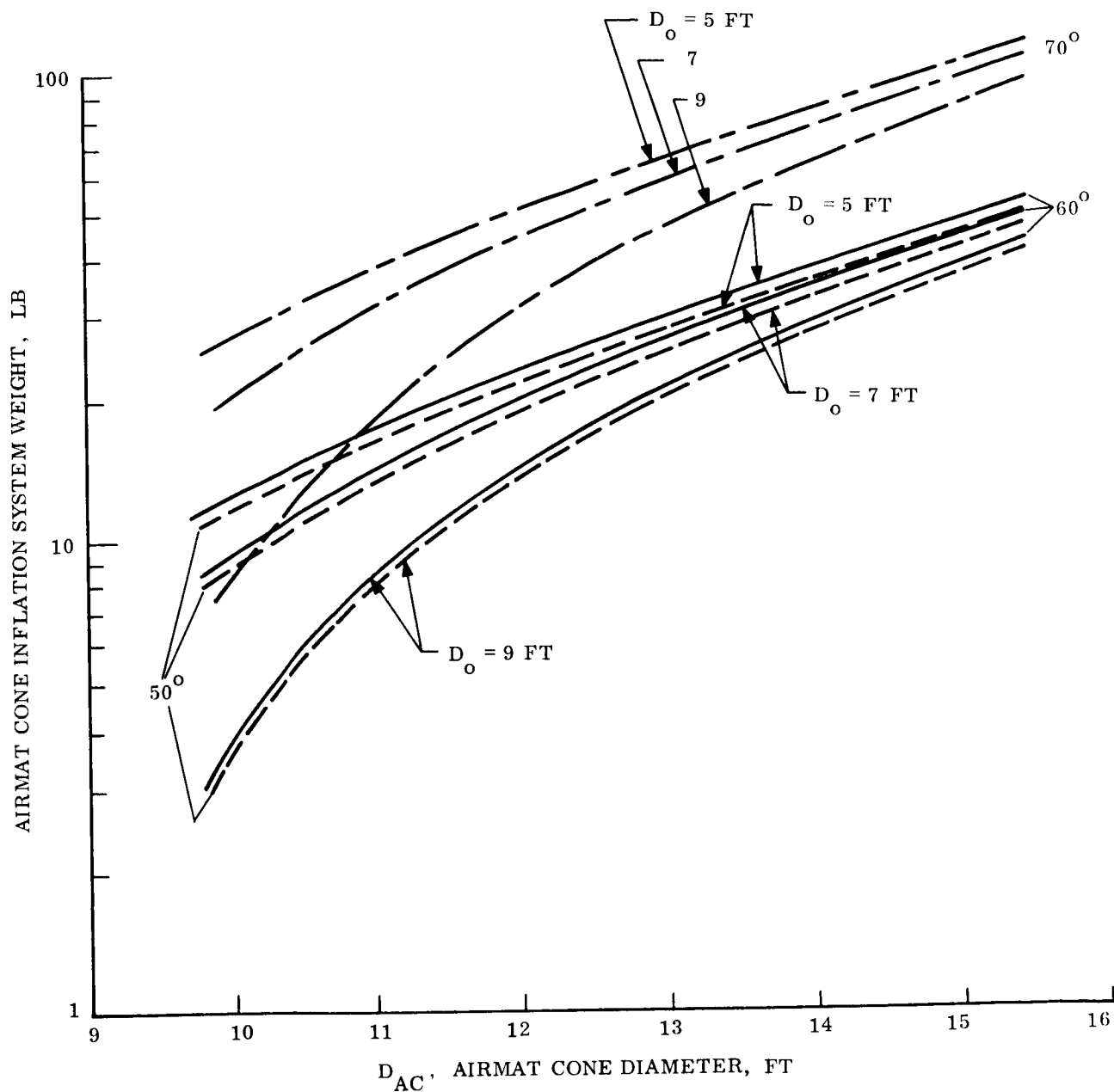


Figure 3.1.2-145. Airmat Cone Inflation System Weight vs Airmat Cone Diameter for Three Aeroshell Diameters and Three Cone Half Angles

#5-

$$\begin{aligned}
 M &= \text{inflation gas molecular weight (29 for nitrogen gas)} \\
 T &= \text{absolute gas temperature (530°R)} \\
 V &= \text{Airmat cone inflated volume, ft}^3 \\
 P_I &= \text{Airmat cone inflation pressure, psf} \\
 &= \frac{K_W C_{P_{\max}} q_{\max} \tan(\theta_c - \phi)}{2 \tan \phi} - (\text{see equation (1)})
 \end{aligned}$$

From figs. 3.1.2-143 through -145, it is indicated that the 50° and 60° half-angle Airmat cones will have a decided weight advantage over the 70° half-angle cone configuration for both envelope and inflation system requirements. There is little difference in the system weight requirements between the 50° and 60° cone configurations as a result of the interaction of geometry and inflation pressure requirements for the range of parameter values and assumed constants as applied in this study.

Figs. 3.1.2-146 and -147 relate the Airmat cone envelope weight and several physical parameters for a 60° half-angle cone as a function of aeroshell-to-Airmat-cone-diameter ratio for Airmat taper angles of 5° and 15°.

B. Heat Shield Requirements

In accord with the requirements and specified criteria set forth for the Airmat cone, preliminary estimates for heat shield requirements have been made. Fig. 3.1.2-148 presents the cold-wall heat flux variation with time from entry. Included on this figure are the estimated heat flux to the Airmat surface. Adjusting the stagnation point heating rate for the appropriate nose radii (corresponding with 5 and 9 ft aeroshell base diameters) and using the 1.5 factor to account for non-equilibrium and equilibrium radiative heating results in estimated heat loads of 1250 Btu's/ft² and 935 Btu's/ft² for the 5 and 9 ft aeroshells, respectively.

These integrated heat inputs may now be put into terms of heat shield requirements. However, due to the comparatively long time of entry (over 120 sec), the heat shield thickness estimate must take into account not only the amount of material ablated, but the amount of material thickness required to insulate the Airmat wall material as well. Three materials were selected to implement the calculation of the heat shield weight. These were:

1. Two types of elastomeric silicone, (GE-RTV-88 and a silicone with a density half again that of RTV-88)
2. A flexible material coating, Dyna-Therm D-65

The amount of material ablated was estimated from the integrated heat input data using the following simplified procedure. Using the term "effective heat of ablation", and the preliminary design equation developed for its use

$$\dot{q}^* = \int (\dot{q} dt) / \rho s, \quad (12)$$

ϕ - AIRMAT TAPER = 5° ———
 15° - - - -

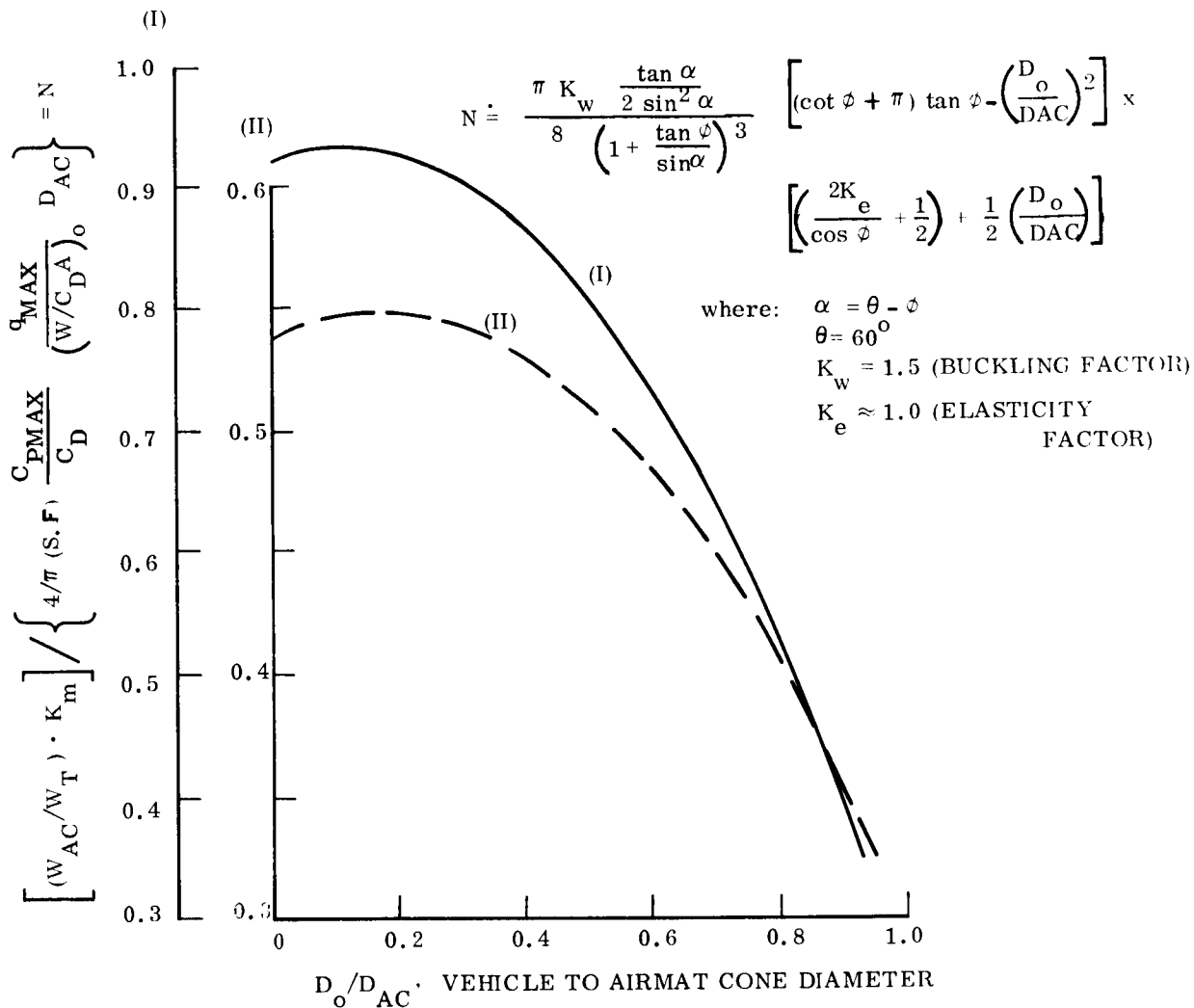


Figure 3.1.2-146. 120° Airmat Cone Weight Relationship
vs Vehicle to Airmat Cone Diameter Ratio

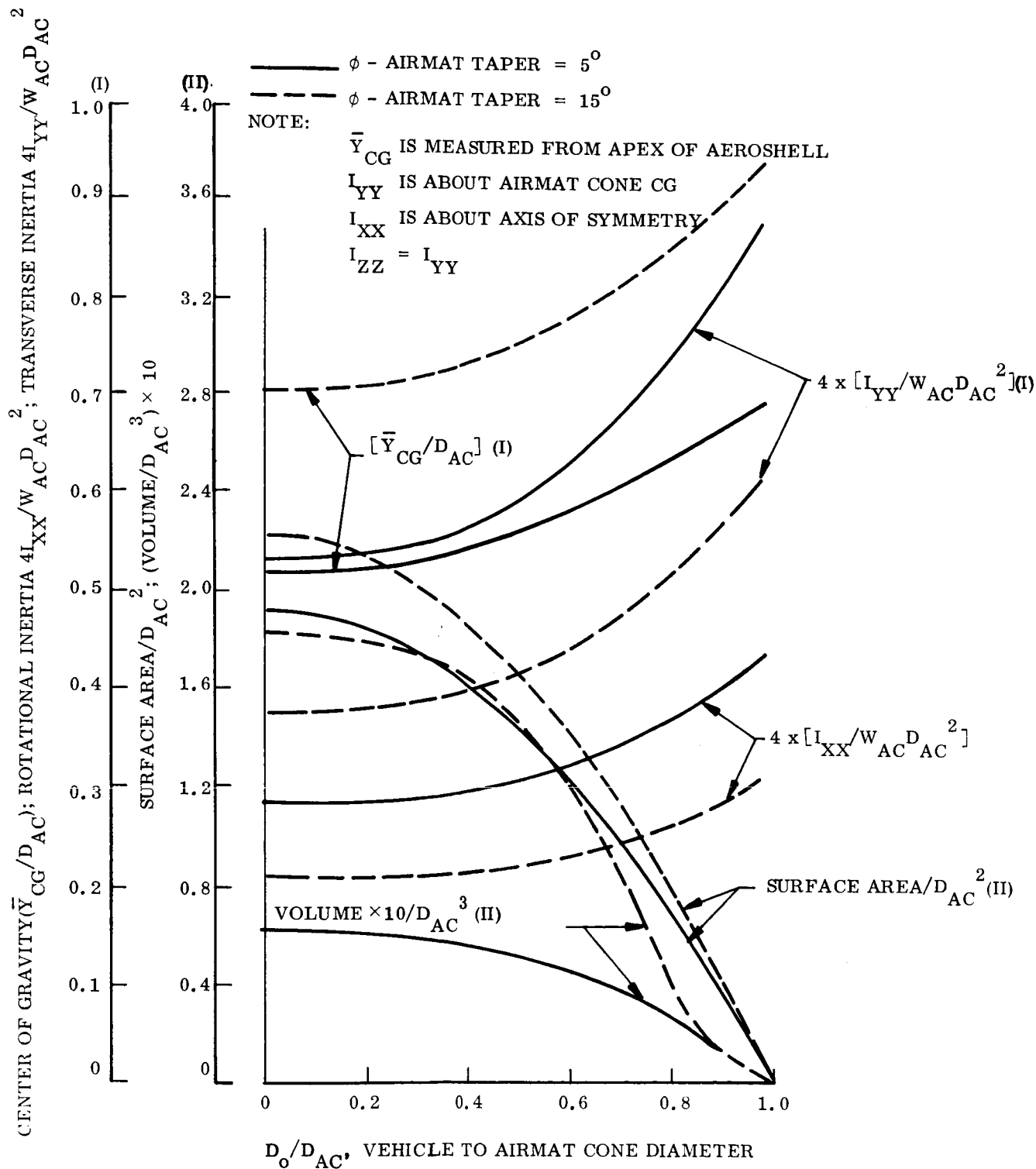


Figure 3.1.2-147. 120° Airmat Cone Physical Parameters vs Vehicle to Airmat Cone Diameter Ratio

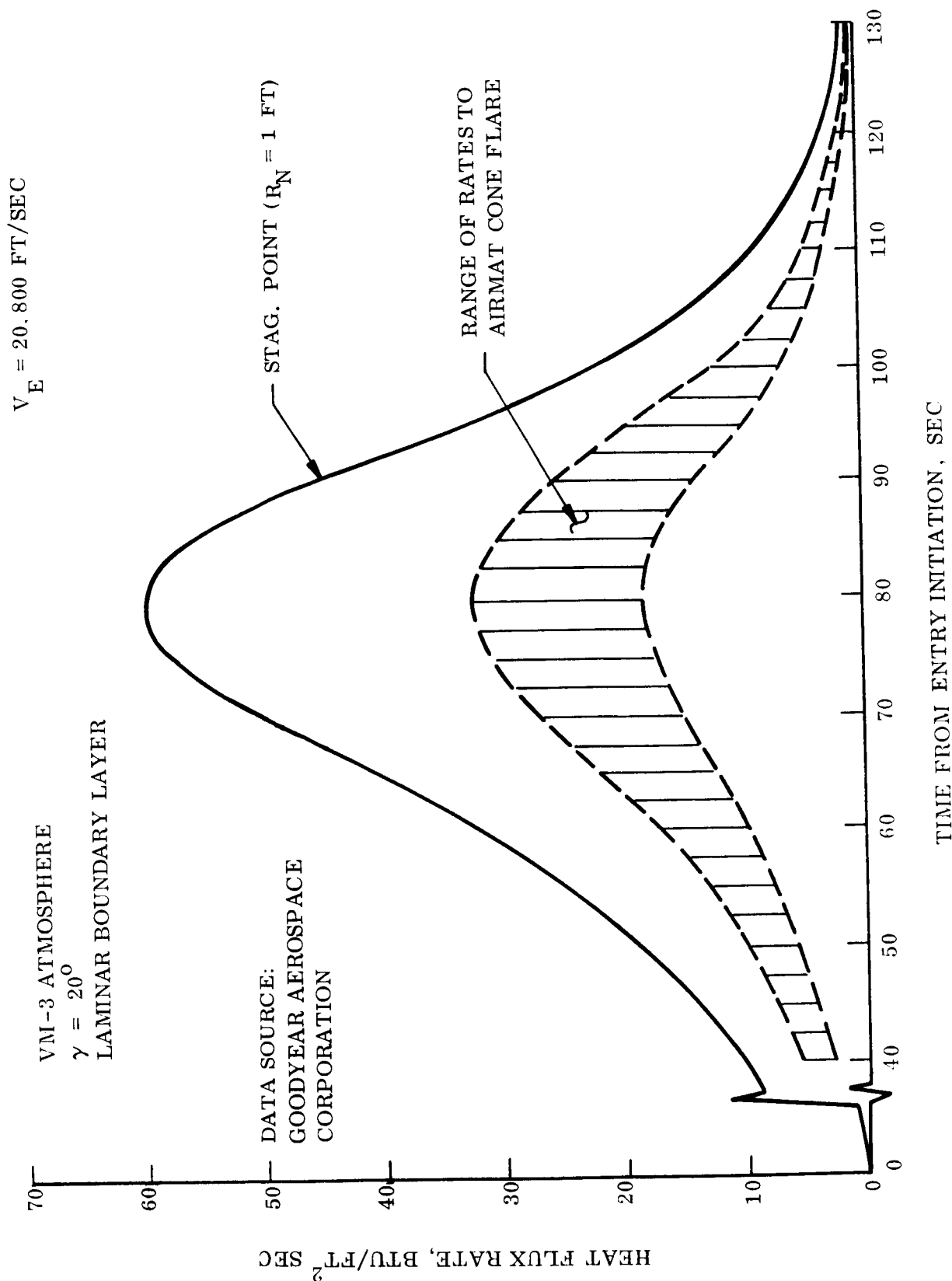


Figure 3.1.2-148. Cold Wall Heat Flux Ratio

an estimate of the material ablated can be calculated. Published data on elastomeric silicones show that the effective heat of ablation for this type of materials range from 10,000 to 20,000 Btu/lb. These values usually were obtained from torch or plasma arc test conditions and quasi-steady heat flux rates. Using the lower value of 10,000 Btu/lb yields an ablation thickness requirement of about 0.1 lb/ft². A value of 5,000 Btu/lb was assumed for the Dyna-Therm material. This yields an ablation thickness requirement of about 0.25 lb/ft² of surface area.

The amount of insulation thickness must now be added to the estimated amount of ablated thickness. These thicknesses were calculated on the basis of transient one-dimensional heat transfer in a slab of finite thickness whose cone surface is raised to a constant elevated temperature while the backface is insulated against heat transfer. Data available on elastomeric silicones indicate that the surface temperature rises to 2500-3000°F and forms a char while the layers immediately under the char experience temperatures of about 1200°F. Limited Dyna-Therm data also show that the surface temperature rises to about 2500°F and a char forms with a similar drop in temperature through the decomposing layers.

On the basis of the preliminary design procedures outlined, two graphs were prepared to show the variation of weight required to shield the conical Airmat flare. The weight of heat shield material required for a 60° half-angle cone is shown in fig.

3.1.2-149. Two base diameters, one 5 ft and the other 9 ft were assumed to form the hard aeroshell vehicle. The weight is presented for the three flexible materials mentioned previously. The weight varies from 70 to 100 lbs for a 10 ft diameter Airmat flare on a 5 ft diameter aeroshell. This weight increases to a range from 180 to 270 lbs for a 15 ft diameter Airmat flare. The comparable weights vary from about 20 lbs at a 10 ft diameter flare to range from 140 to 190 lbs for a 15 ft diameter flare both on a hard 9 ft diameter aeroshell.

The weight data shown in fig. 3.1.2-150 are for a variation in the cone flare angle. This half-angle was varied from 50° to 70°. The weight of heat shield material varied between 20 and 30 lbs for a 10 ft diameter flare to 130 to 220 lbs for a 15 ft diameter Airmat cone flare. The base diameter for the "hard aeroshell" was 9 ft.

The total subsystem weight, with major component breakdown, is shown in fig. 3.1.2-151 as a function of Airmat cone diameter (D_{ac}) for an aeroshell diameter of 8.34 ft (10 ft shroud). The total subsystem weight is modest with respect to the increase in the entry vehicle weight possible as a result of increased diameter. Therefore, the inflated afterbody recommended by Goodyear definitely offers a considerable increase in decelerator load without exceeding the 10 ft shroud limit for direct entry.

3.1.2.3.3 GE MOOSE Concept

In addition to the Goodyear Airmat approach to an erectable skirt for a Mars direct entry Hard Lander, two other concepts have been investigated. The concepts were chosen to provide enough difference in design approach so that a range of erectable structure approaches would be provided. Only non-mechanical erection systems were

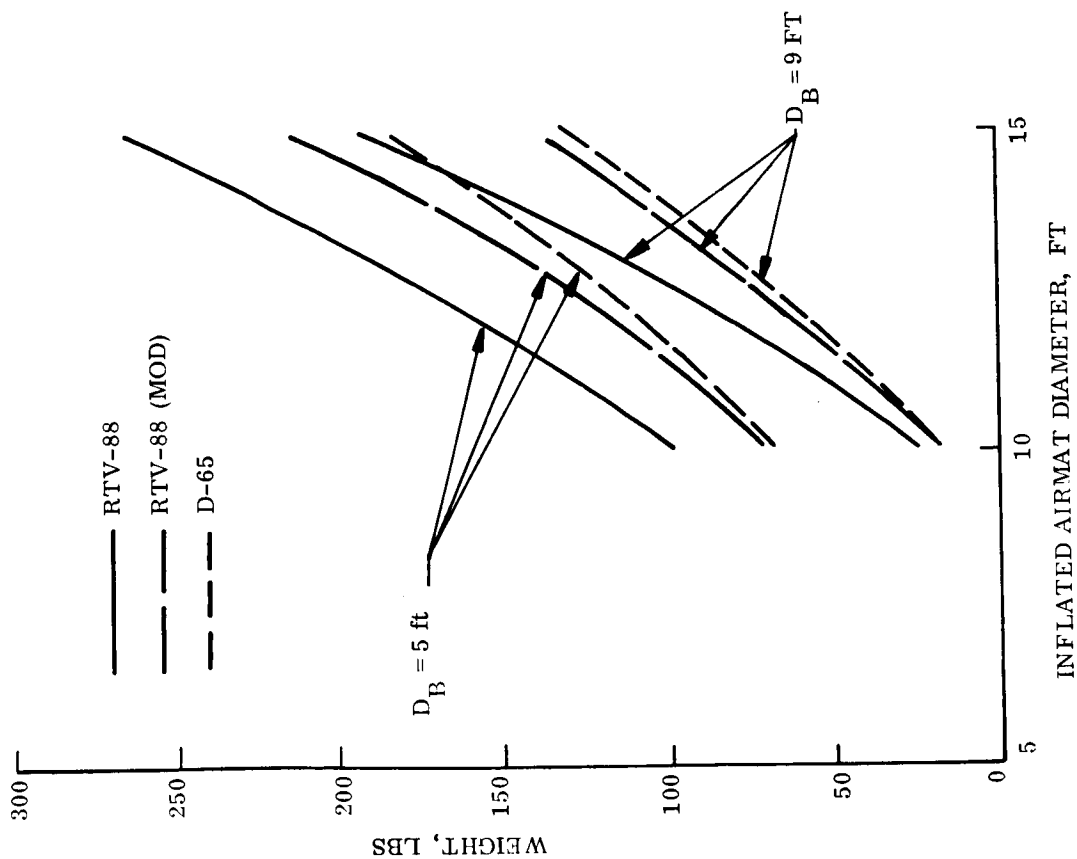


Figure 3.1.2-149. Heat Shield Weight

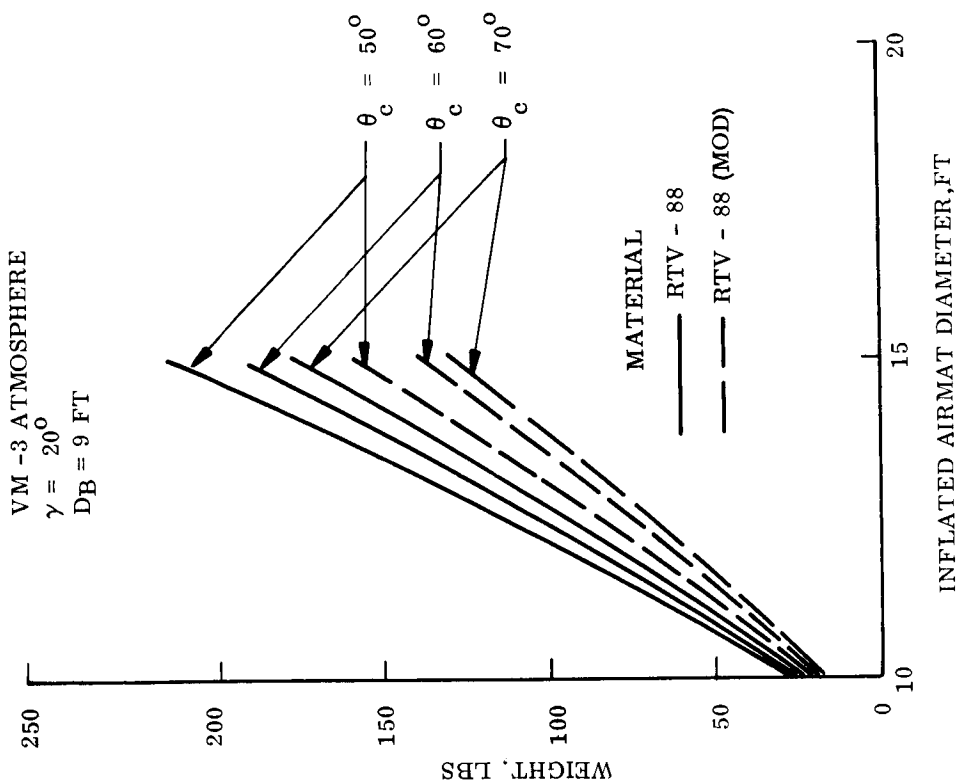


Figure 3.1.2-150. Heat Shield Weight vs Cone Half Angle

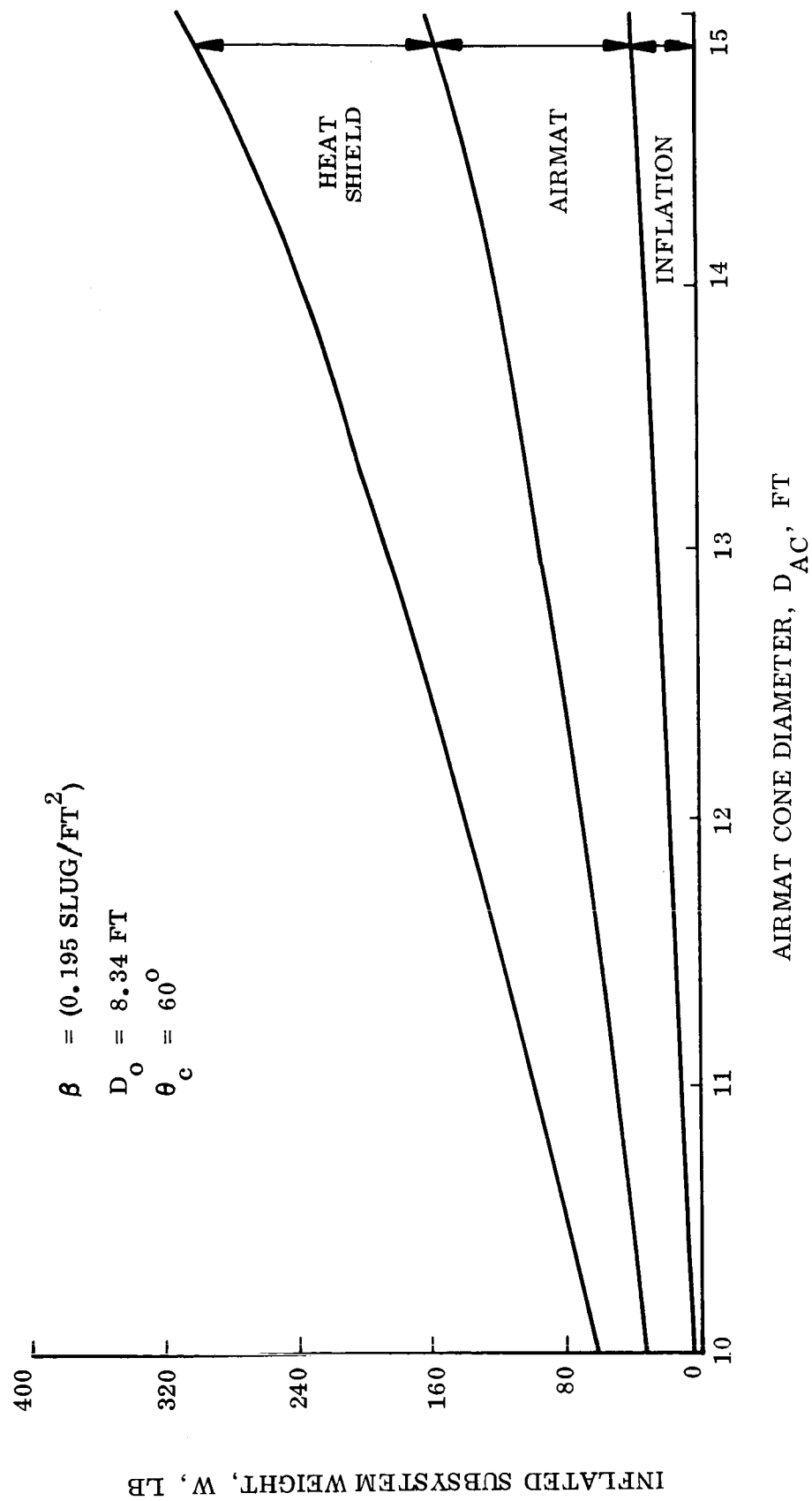


Figure 3.1.2-151. Inflated Subsystem Weight vs Airmat Cone Diameter

investigated because it was felt that these offered the maximum operating reliability. The two concepts chosen for investigation were a foam rigidizable structure and a self erecting elastic structure.

The concepts were based on previous work done in the Manned Orbital Operations Safety Equipment (MOOSE) program. The MOOSE vehicle consisted of a compartmented flexible envelope with a foldable flexible heat shield on a portion of the envelope. The vehicle was formed by inflating the envelope and then filling and rigidizing with foam. The heat shield of the MOOSE was made of a composite of GE BLUE (vinyl silicone elastomer), ESM 1030-1, and 181 glass cloth impregnated with GE BLUE. Five full scale vehicles were constructed along with two full scale heat shields in the MOOSE program.

Three of the vehicles were erectable structures used to determine structural concepts, contour accuracy, and manufacture development. Another vehicle was a full scale foamed vehicle used in water drop tests to determine 'g' loads on the vehicle and on an anthropomorphic dummy. The fifth vehicle was an under water inflatable test vehicle used for ingress-egress testing. The two heat shields made for the MOOSE program were constructed by draping 181 glass cloth over a male mold and coating with GE BLUE.

Foam generators were evaluated in the MOOSE program. The Sonic mixer was selected as the most suitable foam mixer. A foaming system was made which incorporated a Sonic foam mixer. Approximately 1000 lbs of foam were made with the foam generator system.

The techniques used during the MOOSE program are directly applicable to an erectable skirt for the Mars Hard Lander.

Foam Rigidized Structures - The foam rigidizable structure consists of a series of tubular longerons joined together along their sides. The tubes are made of 181 glass cloth coated with GE BLUE. The tubes are faced on both sides with 181 glass cloth impregnated and bonded to the tubes with GE BLUE. A heat shield of cored ESM 1030 is bonded to the outside of the structure and overlaid with a 0.05 in. layer of GE BLUE.

The foam rigidizable structure is packaged around the base of the aeroshell. After the bio barrier is removed, the tubular longerons are inflated and the foamed with rigid polyurethane foam. The two part foam is stored in a toroidal tank at the base of the aeroshell. The foam ingredients are mixed using a Sonic mixer, piped to a plenum chamber to which the tubular longerons are connected. The Sonic mixer consists of a vibrating reed which has one end fixed and other end positioned near the inlet orifice. The fluid motion through the mixer causes the reed to vibrate, thus mixing the foam. Nominal pressure drop through the mixer is 60 psi.

The weight of the foam structure and foam equipment for the point design of an 8.3 ft aeroshell base diameter, 12 ft skirt diameter, is 206 lb.

Self Erecting Elastic Structure - A self erecting elastic structure was investigated to a limited degree. At the present time it appears heavier than either the Airmat or foam rigidized structure. It is likely, however, that additional design and analysis would make it competitive.

A. Design Analysis

Trajectories - Two different trajectories were considered for the Mars Hard Lander expandable skirt studies; both were of the direct entry type. The requirement for a low $W/C_D A$ (6.3 PSF or ~ 0.2 slug/ft²) which the expandable skirts provide, exists only for direct entry.

The two trajectories used were:

<u>Velocity</u> <u>(fps)</u>	<u>Entry Angle</u> <u>(degrees)</u>	<u>W/C_DA</u> <u>(lb/sq ft)</u>	<u>Atmosphere</u>
20,800	20	6.3	VM-3
20,800	40	6.3	VM-8

The trajectories are presented in parametric form in para 3.1.2.1. The $\gamma = 20^\circ$ trajectory was used for the entry heating analysis since this represents the most severe total integrated heating and heat soak period while the second trajectory, $\gamma = 40^\circ$, was used for the structural analysis since this represents the higher structures loads.

B. Heat Shield Thermal Analysis

A thermal analysis has been performed for the erectable skirt design concept for the Mars Hard Lander. The heat shield thickness requirements were determined for the most severe entry heating condition, into the VM-3 model atmosphere. For the specific case examined, the entry condition resulted in the maximum backface temperature at the shield/structure bondline. The resultant total heating loads, both convective and radiative, are shown as a function of axial length on the erectable skirt. The corresponding heat shield thickness of the GE BLUE/ESM 1030 composite are given as a function of both maximum allowable backface temperature and axial length from the stagnation point.

Heating - The thermal design analysis was made for the following conditions:

$$V = 20,800 \text{ ft/sec}; \gamma = 20^\circ;$$

$$W/C_D A = 6.3 \text{ lb/ft}^2 (0.196 \text{ slug/ft}^2); \text{ VM-3 model atmosphere.}$$

The total heat load to the frustum included both the convective heating and the radiative heating (which was not negligible). The radiative heat pulse was made up of both equilibrium and non-equilibrium radiation. The total heat load to the frustum

is shown in fig. 3.1.2-152 as a function of axial length, varying from 720 Btu/ft² to 615 Btu/ft². The heating rates are for cold wall conditions ($T_W = 0$) and zero angle-of-attack. The total heat load distribution was obtained using the current techniques of the CONSEP program which included up to date estimates of the equilibrium and non-equilibrium radiative heating for both direct and out-of-orbit entry modes.

Heat Shield Thermal Performance - No surface melting is expected in the relatively mild thermal environment which results from the entry conditions of this study. Thus the heat shield must act only as a charring insulator. The materials selected for this study were a composite of GE BLUE ($\rho_v = 74 \text{ lb/ft}^3$) and GE ESM 1030 ($\rho_v = 18 \text{ lbs/ft}^3$). Using the total heat loads shown in fig. 3.1.2-152 together with correlations of thermal response for the selected materials in these environments, heat shield thicknesses were obtained as a function of maximum allowable bondline temperature. Fig. 3.1.2-152 shows the thickness required for the GE BLUE/ESM 1030 composite to maintain the bondline to several maximum temperature at two axial stations. Fig. 3.1.2-153 shows the variation of shield thickness requirements as a function of axial length along the frustum for selected maximum backface temperature. The heaviest shield results from designing to a maximum backface temperature of 300°F for the end of skirt location of the 9 ft diameter vehicle, the thickness requirements being 0.05 in. GE BLUE and 0.485 in. ESM 1030 cored out to 25 percent of the original weight. Fig. 3.1.2-153 indicates very little variation in the shield requirements along the erectable skirt. The total shield thickness was increased to approximately 0.56 in. to provide a safety margin of about 25°R.

Materials - The materials, structure, and thermal performance properties for the foam rigidized structure have been obtained as reported in ref. 3-52. The thermal performance of the ESM 1030-1 heat shield material is shown in table 3.1.2-15. Manufacturing, decontamination, and sterilization are described in their respective sections.

Heat Shield Material - The heat shield consists of ESM 1030-1 overlayed with 0.05 in. of GE BLUE. The overlay of GE BLUE over the ESM 1030-1 prevents possible splits of the ESM 1030-1 surface and also enables the ESM 1030-1 to be cored to reduce weight by providing an ablation surface. The ESM 1030-1 is foamed modified epoxy silicone. The GE BLUE is a vinyl silicone. The density of the ESM 1030-1 is 18 lb/ft³. The density of the GE BLUE is 74 lb/ft³. The ESM 1030-1 exhibits a large endotherm when heated. The GE BLUE material shows excellent thermal stability. At 860°F only a 5 percent weight loss is indicated. The differential thermal analysis curve shows little if any change up to 1000°F in the absence of air.

Manufacturing - Fabrication procedures for GE BLUE/glass composites have been developed for the preparation of laboratory test samples and full scale thermal shields for the MOOSE escape system. Although curing of the full scale thermal shields was carried out at room temperature, laboratory tests have confirmed that the cure time can be effectively reduced by curing at elevated temperatures. Because of the large size of the MOOSE thermal shields and the extendable skirt for the Mars Hard Lander, the low cure shrinkage of GE-BLUE is a distinct advantage. Fabrication of the MOOSE thermal shields also demonstrated that the drape characteristics of the glass cloth material selected permits shaping to doubly curved surfaces such as sphere-cone configurations, without requiring seams or double lap folds.

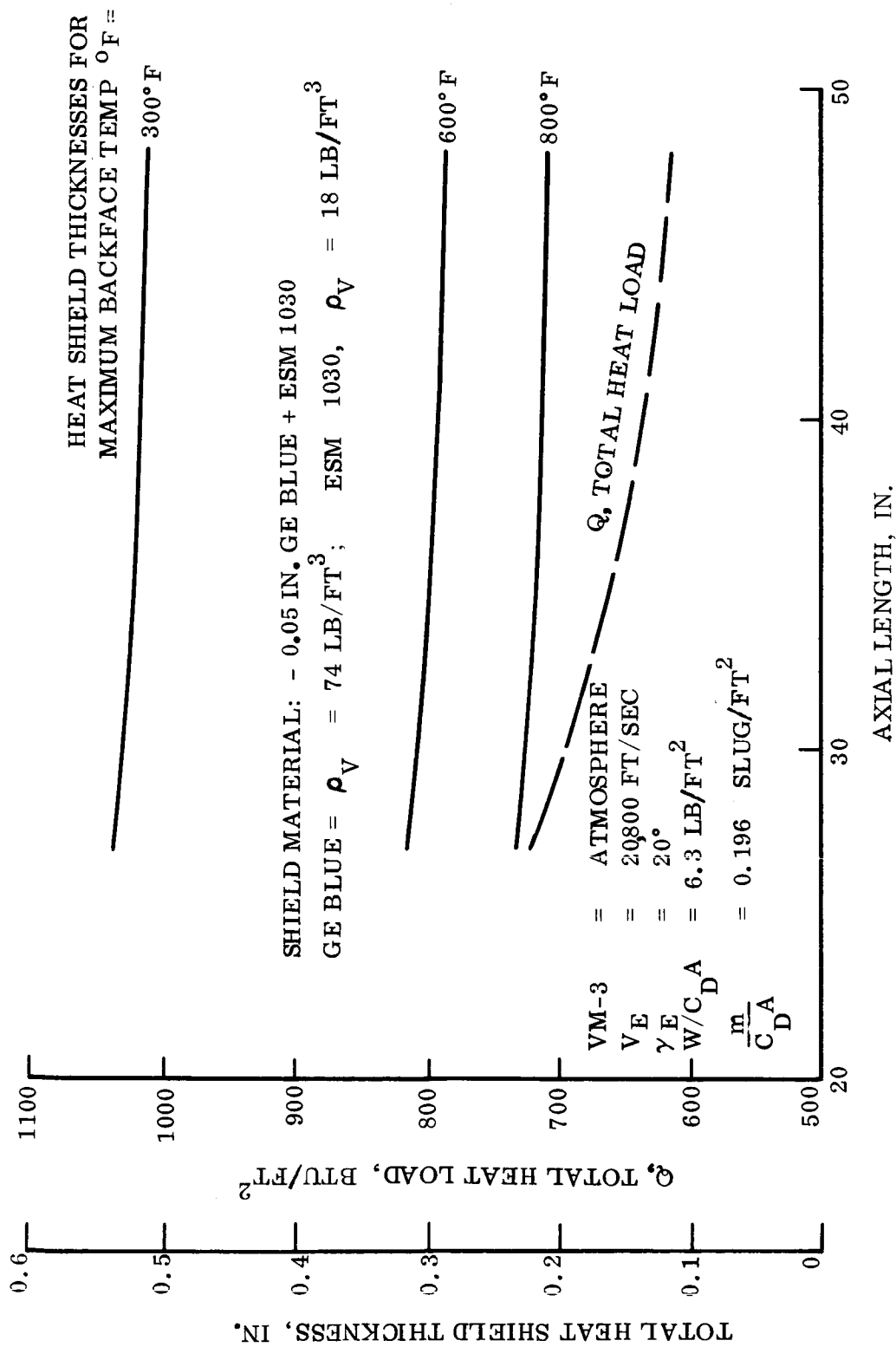


Figure 3.1.2-152. Total Heating and Heat Shield Thickness as a Function of Maximum Backface Temperature

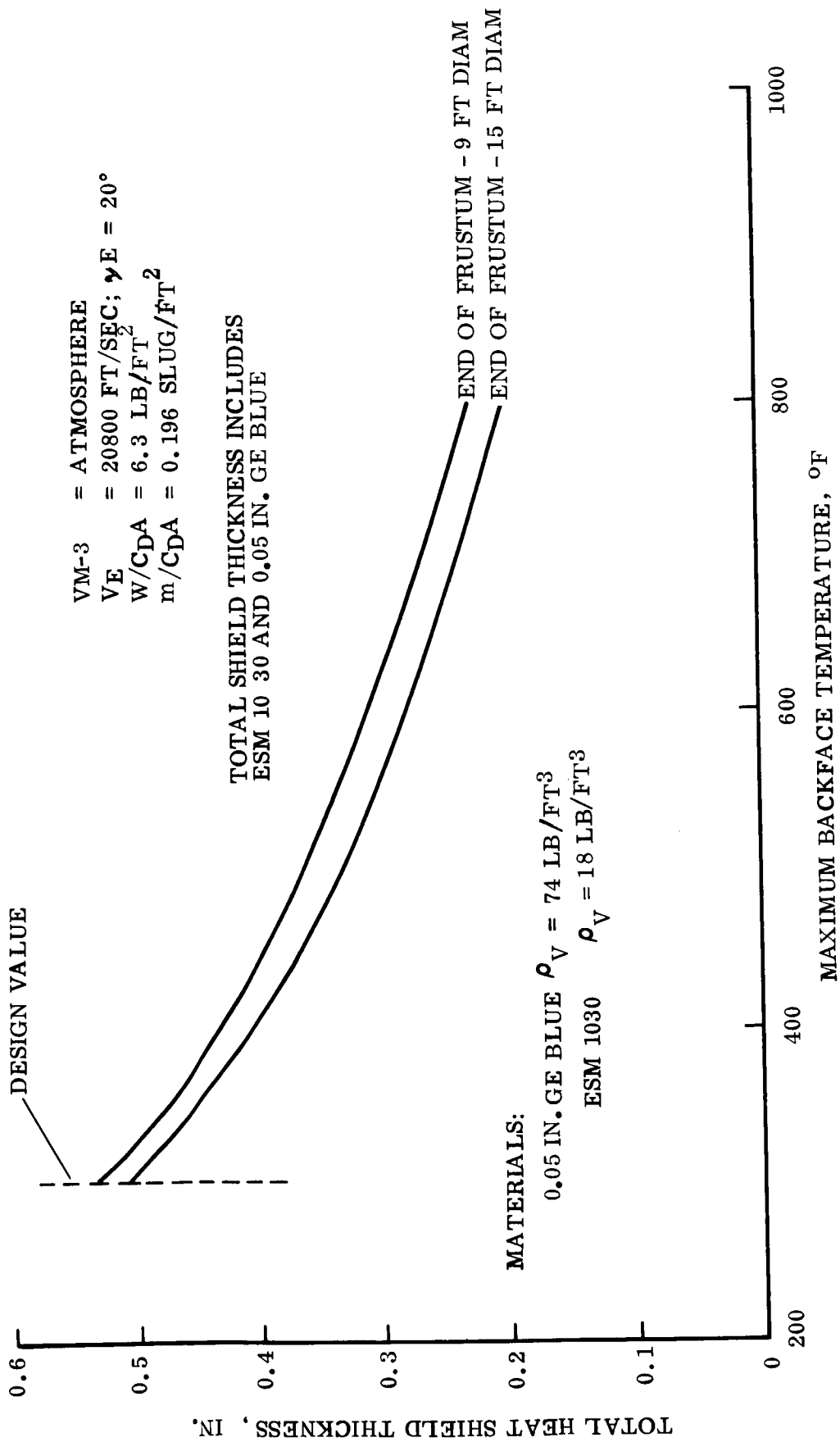


Figure 3.1.2-153. Total Heat Shield Temperature as a Function of Maximum Backface Temperature

TABLE 3.1.2-15. PROPERTIES OF CANDIDATE HEAT SHIELD MATERIAL

PROPERTIES		ESM 1030-1		
1. Density (pounds/foot ³)		18.0 ± 2.0		
2. Specific Heat (Btu/pound-°F)	0°F	0.30		
	100°F	0.35		
	300°F	0.44		
	500°F	0.49		
3. Dynamic Enthalpy		Large Endotherm (Air, 0.01 atm, 750°F)		
4. Thermal Conductivity x 10 ⁵ Btu/ft-sec-°F	1 atm, Air	0°F	1.12	
		100°F	1.24	
		300°F	1.48	
	Vacuum	0°F	0.79	
		100°F	0.84	
		300°F	0.95	
5. TGA - 5% Weight Loss	In Air	700°F		
	Onset of Degradation	700-800°F		
	50% Weight Loss	975°F		
	Residual Weight	15%		
6. Weight Loss in Vacuum, 10 ⁻⁵ mm Hg, Room Temperature, 5 days		0.62%		
7. Ablation - Time to Backface Temperature, 500°F (Testing in CO ₂ N ₂ at pressure)	7 Btu/ft ² -sec - 1700 Btu/1 bm	276,372 sec		
	12 Btu/ft ² -sec - 5800 Btu/1 bm	405,405 sec		
	24 Btu/ft ² -sec - 5700 Btu/1 bm	134 sec		
8. Optical Properties - Total Solar Absorptance		0.376		
	Total Hemispherical Emittance	0.85		
9. Electrical Properties - Room Temperature	Dielectric Constant	1.47-0.011		
	Loss Tangent	0.007		
	Attenuation			
	Coefficient, cm ⁻¹	0.00889		
Maximum Change in Signal - During Ablation - Transmission		-1.4 dB		
Maximum Change in Signal - During Ablation - Reflection		-3.5 dB		
10. Tensile Properties		Tens	Elong	Mod
		Str	%	PSI
		PSI		
	-150°F	199	(2)	22,300
	-100°F	125	(2)	---
	- 50°F	33.6	56	60
11. Coefficient of Thermal Exp. (Δ DL/L) /°F, (in/in/°F)	75°F	20.5	46	45
	300°F	10.1	35	29
	+75 to -75°F	1.06 x 10 ⁻⁴		
	+75 to -150°F	-----		
	-200 to -250°F	3.3 x 10 ⁻⁵		

In both the foam inflated and self erecting skirt designs, bonding of the GE BLUE/glass composites to the metal supports and the plastic thermal shield can be carried out using GE BLUE as the adhesive. Particular emphasis is placed on cleaning and priming the metal surfaces to assure complete wetting and adhesion. Post-curing of ESM 1004AP aeroshell thermal shield material prior to bonding will eliminate the possibility of under-cure of the GE BLUE adhesive due to the presence of unreacted components in the ESM 1004AP.

Either glass or quartz threads can be used to sew sections of GE BLUE/glass composites together to form either lap seams or tubular arrays. The strength of the GE BLUE/glass composite material at the sewn joints is expected to far exceed the stresses developed during folding, erection, and aerodynamic loading.

A technique for coring the ESM 1030-1 material has been developed and demonstrated on a laboratory scale. Both top and bottom edges are smooth and the cores are of uniform diameter through the thickness of the ESM 1030-1 material. No special problems are expected in producing large cored sheets of ESM 1030-1. Mechanization of the coring procedure is possible and should be investigated. In addition, consideration should be given to forming the cores by the use of a mold during fabrication of the ESM 1030-1 material.

Bonding of the GE BLUE/glass composite to the cored and shaped ESM 1030-1 will require the use of male and female bonding fixtures. It should be possible to bond the GE BLUE/glass composites to the ESM 1030-1 before or after curing of the GE BLUE coating.

Decontamination and Sterilization - The GE BLUE/glass composites and ESM 1030-1 should be capable of withstanding the decontamination and sterilization cycles for the Mars Hard Lander system without deleterious effects on performance and reliability. Preliminary laboratory tests have demonstrated that mechanical properties of ESM 1030-1 remain unchanged after exposure to the ethylene oxide-freon-humidity decontamination environment. Both GE BLUE and ESM 1030-1 materials can be post-cured at temperatures above that (125°C) required for sterilization without deleterious effects on their properties.

No data are currently available on which to judge the ability of the polyurethane foam precursors to withstand the thermal sterilization cycles. Because of the required reactive properties of the polyurethane foam precursors, special techniques may have to be developed to assure that they meet the bacteriological standards without affecting their reactivities, cure time, and the final foam properties. One possible approach is to investigate the conditions under which these materials are manufactured; if they are free of bacterial contamination in their final stage of production due to their thermal and/or chemical environment, they can be injected under sterile conditions directly into thermally insulated, sealable pre-sterilized containers. If it is necessary to protect the precursors, thermal insulation or other thermal control techniques would be selected to limit the temperature rise of the polyurethane foam precursors in the containers during thermal sterilization exposure.

Foam Rigidized Structure Design - The foam rigidized structure consists of a cone frustum shell to which a heat shield is bonded on the outside surface and a GE BLUE impregnated 181 glass cloth on the inside surface, fig. 3.1.2-154.

The heat shield is a sandwich construction that is made up of GE BLUE, ESM 1030-1 cored to 25 percent of the original weight and 181 glass cloth. The base and apex, of the cone are toruses, fig. 3.1.2-154. The two toruses are joined together by thin cone frustums, that are tangent to each other, fig. 3.1.2-154. The apex torus, which is located at the aft surface of the vehicle, serves as a plenum chamber for distribution of liquid foam to all the thin cone frustums during the foaming process.

Vehicle Attachment - There are three proposed methods of attaching the foam inflated skirt to the basic vehicle structure, namely:

1. Common Heat Shields: One continuous heat shield for the vehicle and skirt. This would eliminate the need for attachment points at the edge of the vehicle (fig. 3.1.2-155).
2. Scarfig: Same reason as item one (ref. fig. 3.1.2-155).
3. Bolted Skirt: This will require fasteners on the outside edge and inside edge (ref. fig. 3.1.2-155).

Further analysis of vehicle design is required in order to determine which method would be most acceptable.

Parametric Analysis - A parametric analysis was performed on the foam rigidized structures. Figs. 3.1.2-156 and -157 provide data to determine skirt weight for skirts up to twelve feet in diameter with an eight ft diameter vehicle and skirts up to 15 ft in diameter with a 9 ft diameter vehicle. The foam inflated skirt consists of GE BLUE: 181 GLASS CLOTH: Polyurethane foam (2; 4; 6; 8; of 10 lb/ft³ density) and ESM (0.51 thick: 25, 50, and 75 percent cored). Fig. 3.1.2-158 shows weights of the foaming system, heat shield and structure for a 8.3 ft base diameter aeroshell with a skirt diameter of 9 to 15 ft.

The example illustrates the use of fig. 3.1.2-158 in determining the skirt weights for an 8.3 foot aeroshell with a 12 ft skirt diameter.

I. Foam Inflated skirt:

- a. Weight of Foaming System = 22 lb
- b. Weight of structure = 125 lb
- c. Weight of heat shield = 59 lb
- d. Design Weight = $22 + 125 + 59 = 206$ lb total weight at design point.

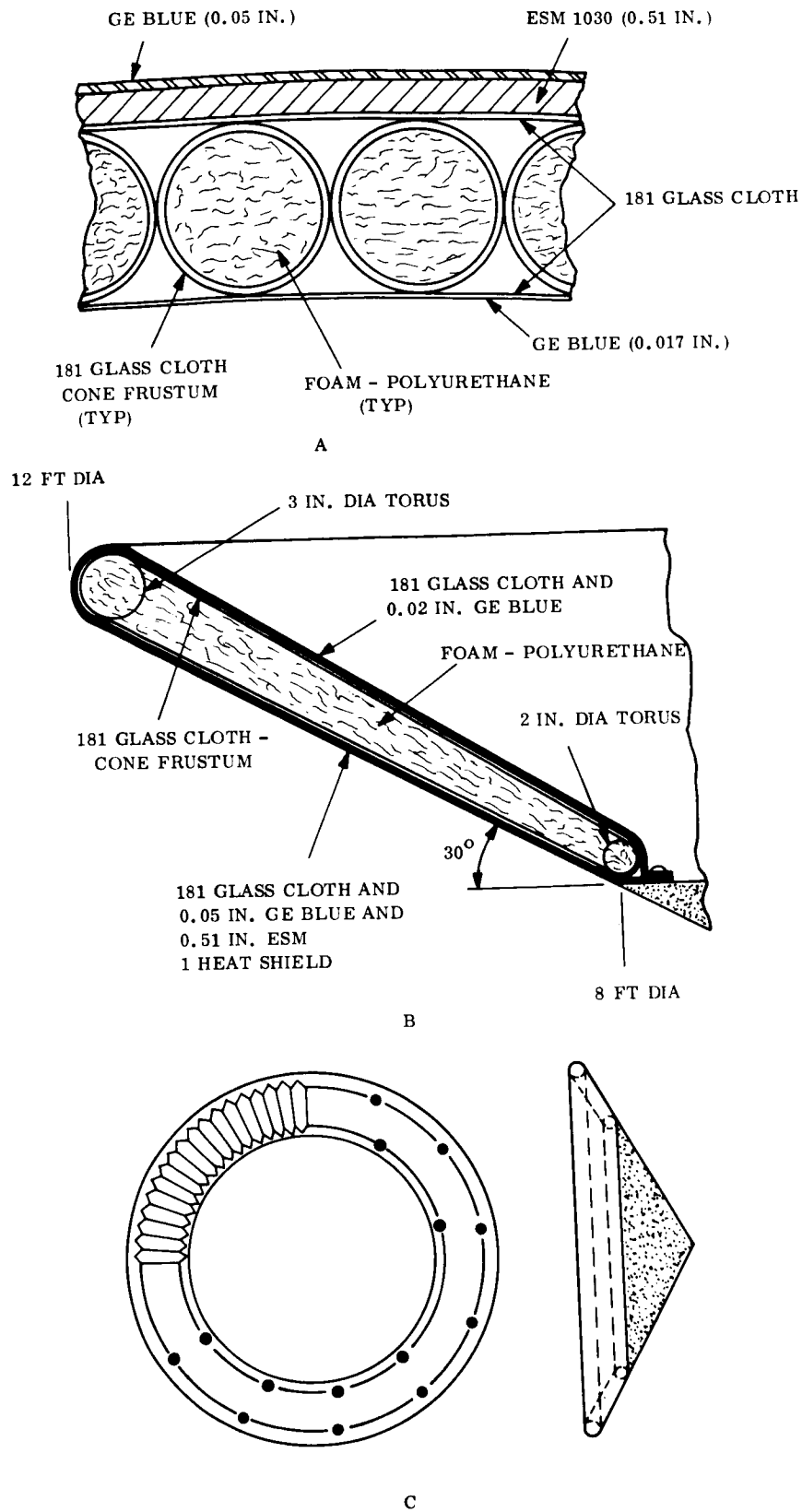
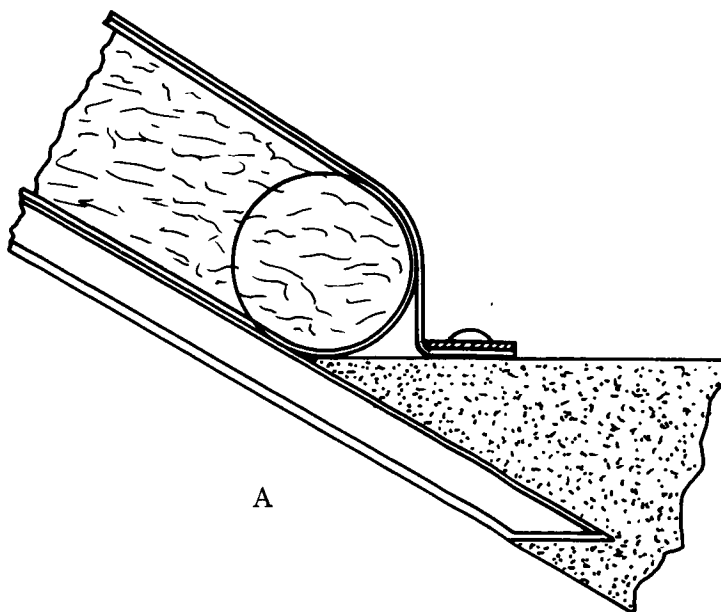
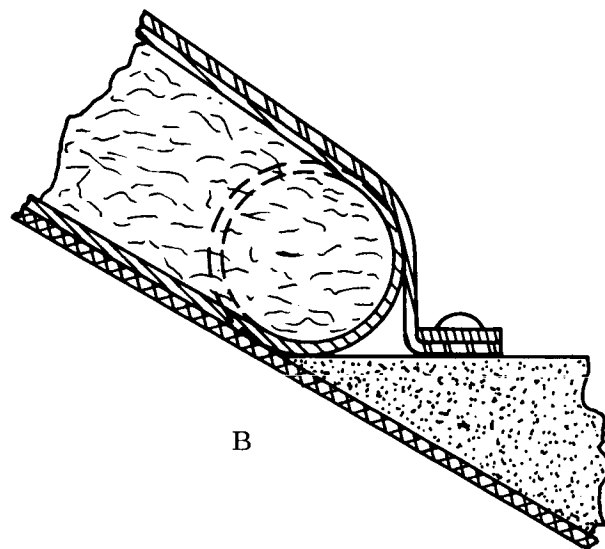


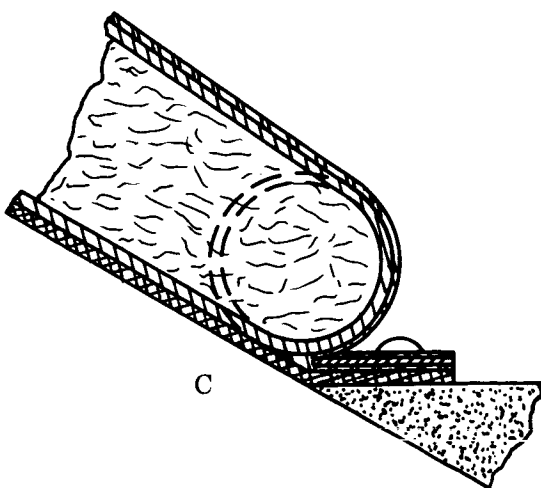
Figure 3.1.2-154. Foam Rigidized Skirt, Structural Design



A



B



C

Figure 3.1.2-155. Foam Rigidized Skirt, Vehicle Attachment

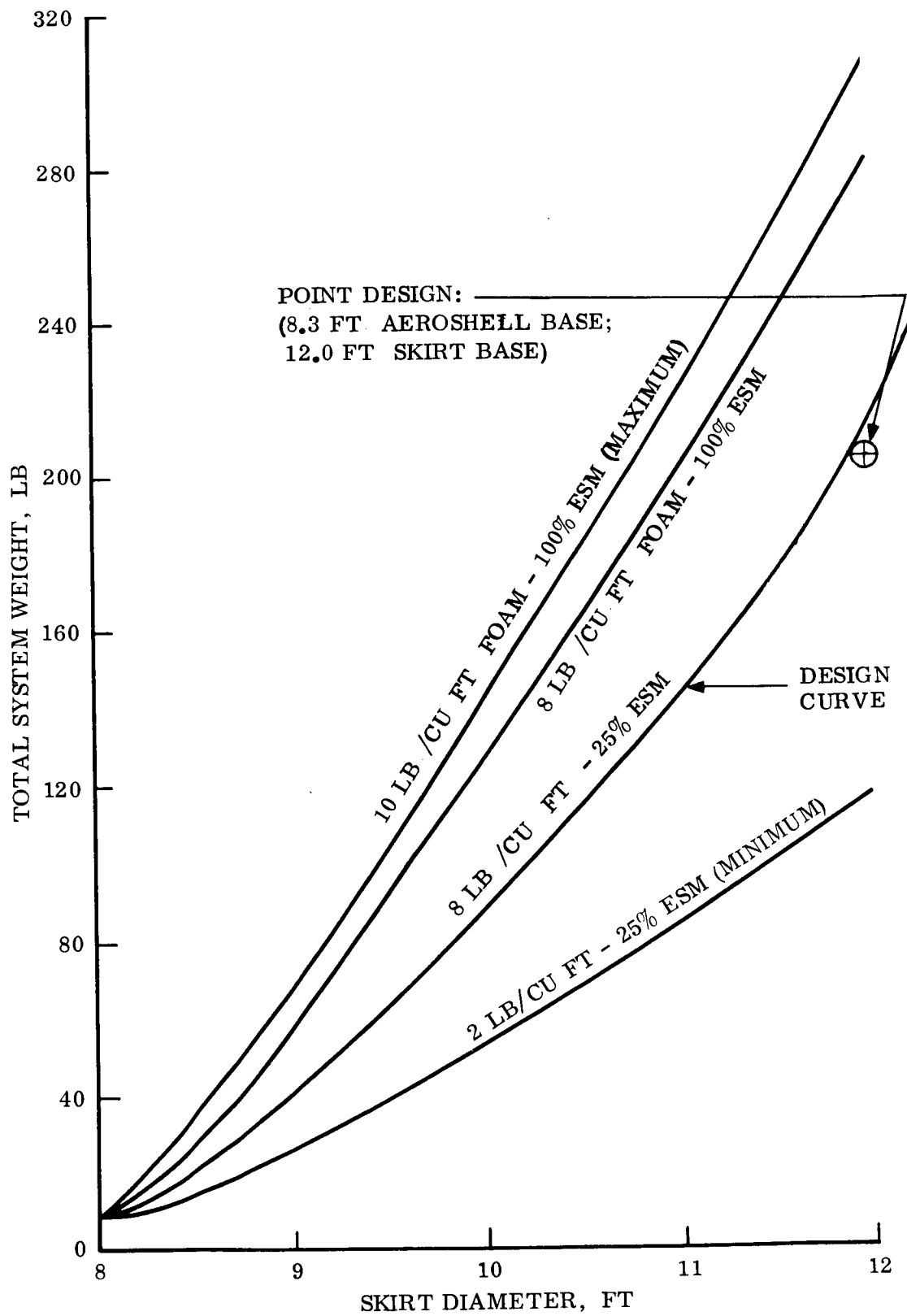


Figure 3.1.2-156. Foam Erectable Skirt - 8 ft Vehicle

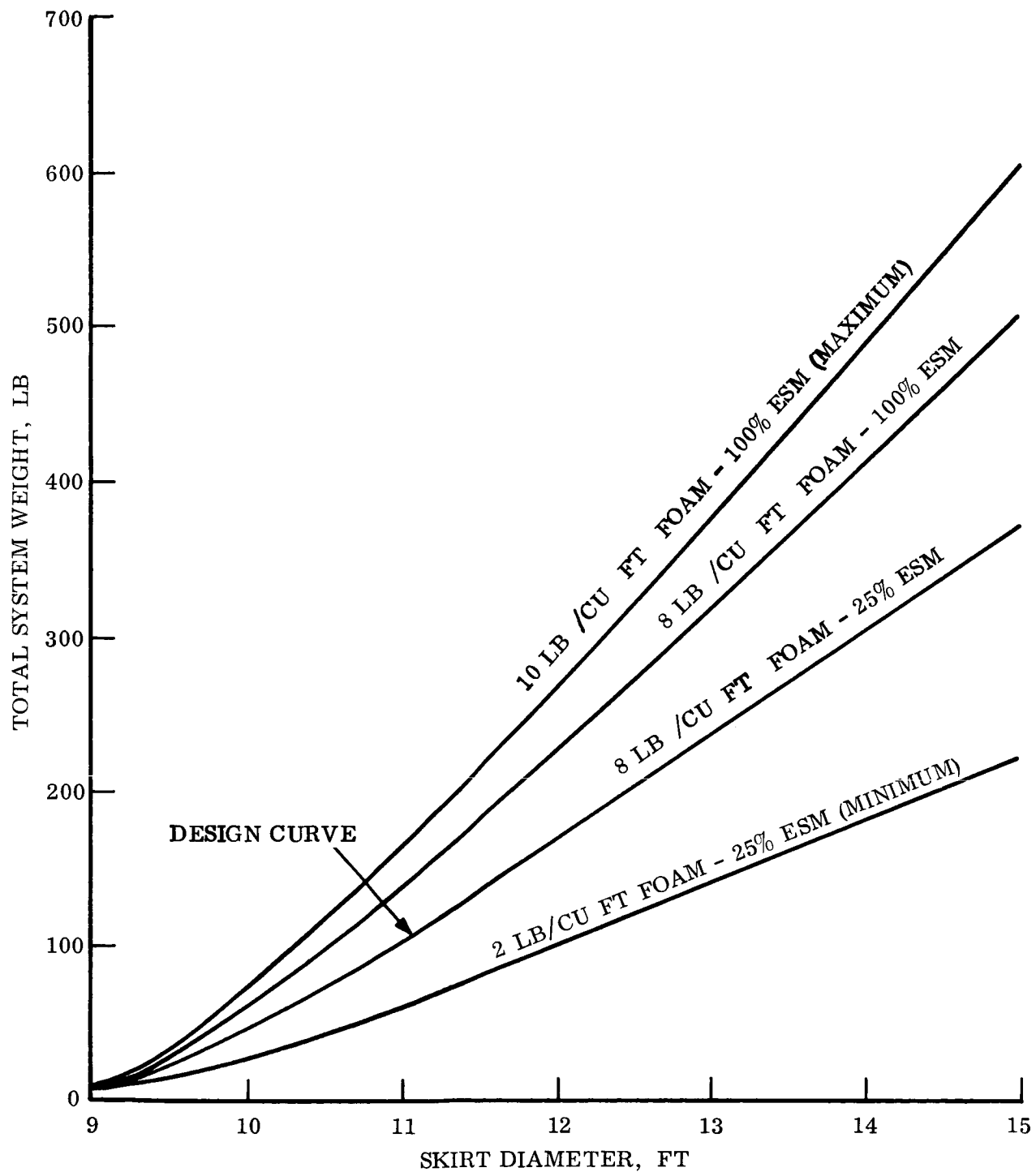


Figure 3.1.2-157. Foam Erectable Skirt, 9 ft Vehicle

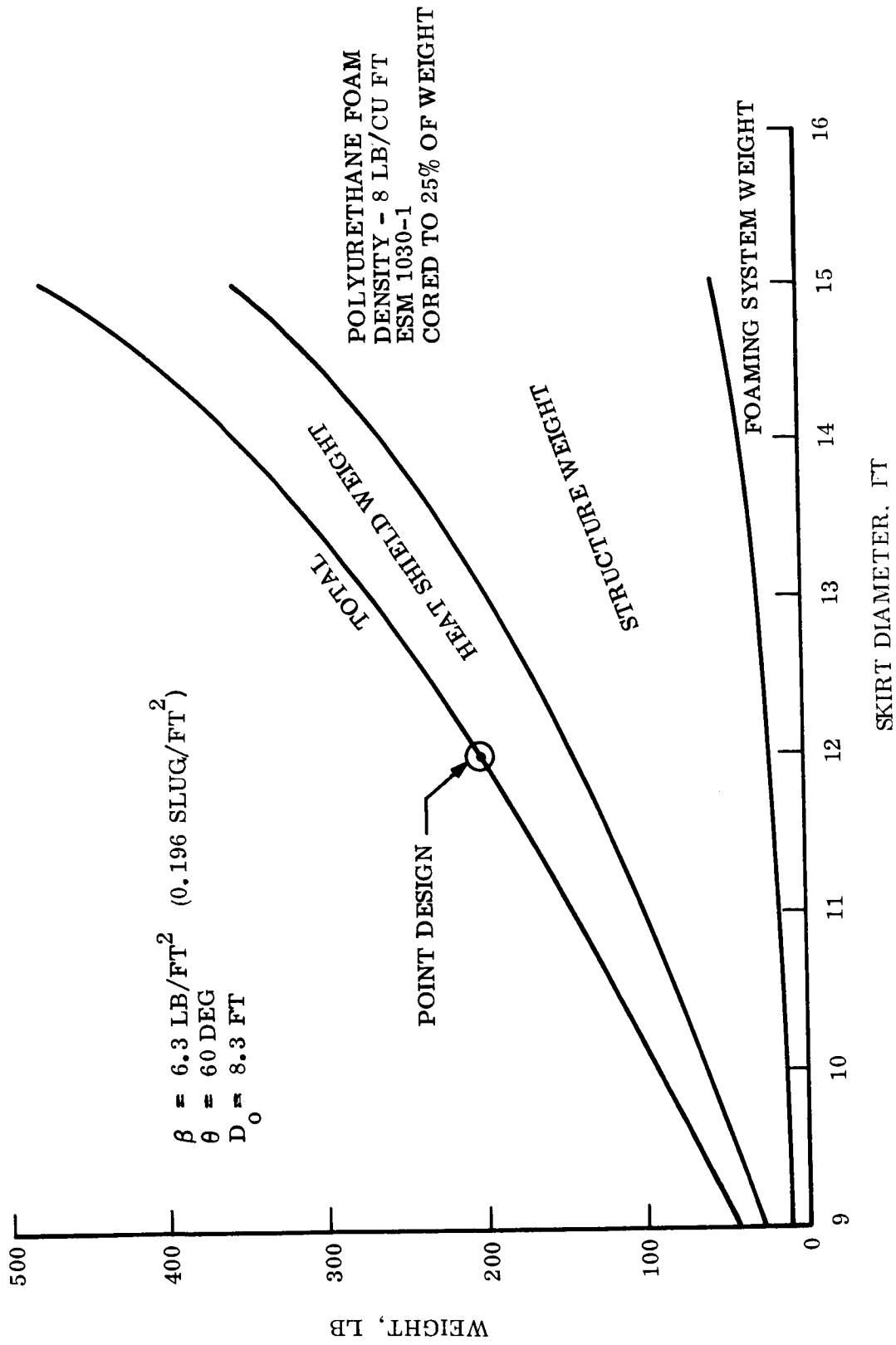


Figure 3.1.2-158. Foam Erectable Skirt, Total and Subsystem Weight vs Skirt Diameter

Structural Requirements and Criteria - For an erectable aeroshell skirt design, the critical structural requirement is the provision of two simultaneous conditons: first, in the stowed state, the structure must have sufficient flexibility so that it may be packaged within a given configuration; second, when deployed it must attain sufficient rigidity and stiffness to sustain the loading during atmospheric entry. The erectable structure must therefore contain means to rigidize the skirt after deployment.

The foam rigidized erectable concept uses polyurethane foam to rigidize a tubular structure formed by layers of fiberglass fabric and GE BLUE. Thermal protection is provided by a composite structure of GE BLUE, ESM 1030-1, and fiberglass cloth.

The interface of the afterbody with the bare aeroshell requires special consideration. The interface structure must be designed for adequate strength and stiffness to transfer aerodynamic drag loads through the aeroshell aft ring. Increased foam density as well as thicker fiberglass cloth are required. A stiffened torroidal ring is also necessary at the rearmost face of the structure.

Structural Loading Analysis - The erectable afterbody is subjected primarily to the aerodynamic pressure loading and aerothermodynamic heating. The critical loading conditions are same as those specified for ballistic rigid aeroshell configurations as summarized in table 3.1.2-16.

TABLE 3.1.2-16. DESIGN CONDITIONS

γ_E Path Angle (deg)	40
β (lb/ft ²)	6.3
V_E (1000 fps)	20,800
G_x (earth 'g's)	81
Altitude (1000 ft)	55,000
Stagnation pressure (lb/in. ²)	3.55
Skirt pressure (lb/in. ²)	2.78

The foam rigidized structure is analyzed by analogy to sandwich construction. Fiberglass cloth layers are assumed to carry in-plane loading, resisting flexure, and foam is assumed to provide stiffness to prevent wrinkling of the glass cloth. The foam density is chosen to give sufficient compressive and shear strength and modulus of the composite are estimated. The foam depth and glass cloth thickness are calculated to obtain required effective properties to sustain the aerodynamic loading. The structural performance characteristics of foam cored, single-and double-ply glass cloth faced sandwich construction are shown in fig. 3.1.2-159. This indicated the foam thickness required to satisfy the aerodynamic pressure loading for various skirt sizes with forward and aft diameters of the cone varying from 5 to 9 and 10 to 15 ft

respectively. For the design trajectory (Table 3.1.2-16) the critical buckling pressure is 3.5 psi (skirt pressure times a safety factor of 1.25). The required shear strength of the polyurethane foam is 200 psi which requires a foam with a density of 8 lb/ft³.

Foam Rigidible Structure System Operation - The foam rigidible structure is folded in a series of accordian pleats along the conical frustum. These pleats are then rolled up into a toroidal section at the base of the aeroshell. The foam generating equipment is packaged around the base of the toroidal section. The foam rigidible structure is first pressurized with nitrogen to inflate and erect it after the bio barrier has been removed. After the structure has been inflated, the explosive activated valves are opened which pressure the foam tanks and open the inlet to the Sonic mixer. The foam ingredients are mixed in the Sonic mixer and injected into the flexible plenum chamber at the base of the aeroshell.

The foam then distributes itself into the vertical tubes and fills the tubes with foam. In approximately 10 minutes the foam is rigidized. Relief valves are provided for venting escaping foam and gas.

3.1.2.3.4 Post Heating Inflatable

The use of a supersonic aerodecelerator stage is considered for the application to the Mars Hard Lander application. Numerous past parametric studies have been performed to evaluate the merits of a decelerator deployed after the entry thermal and loading cycle. Such studies have been performed with a particular Lander philosophy in mind, in this case for application to the present Hard Lander concept.

Trajectories - The key trajectory parameters, velocity, Mach number, and deceleration, are shown in figs. 3.1.2-160 through -162 for a typical application of the post-heating inflatable afterbody. The trajectory is the steep direct entry reference ($V_e = 20,800$ fps and $\gamma_c = 40^\circ$ at $h = 220$ km) into the VM-8 model atmosphere, which represents the design condition for this type of drag modulation, with deployment of the system made at Mach 4.

Fig. 3.1.2-163 shows the actual tradeoff information for decision making on the design/usefulness of a post heating inflatable afterbody. This curve shows the altitude at Mach 2 as a function of the hypersonic ballistic coefficient and the area modification ratio at Mach 4. That is, at Mach 4 the $C_D A$ value is changed by inflating an afterbody which increases the area. The most direct comparison can be made by assuming that Mach 2 must be achieved by some particular and fixed altitude since the effect on flight path angle (the other key ballistic trajectory parameter in determination of deployment requirements) is negligible.

As can be seen in fig. 3.1.2-163 the advantage of increasing area decreases as the increment increases. The $C_D A$ ratio of infinity, in fact, represents the altitude at Mach 4.

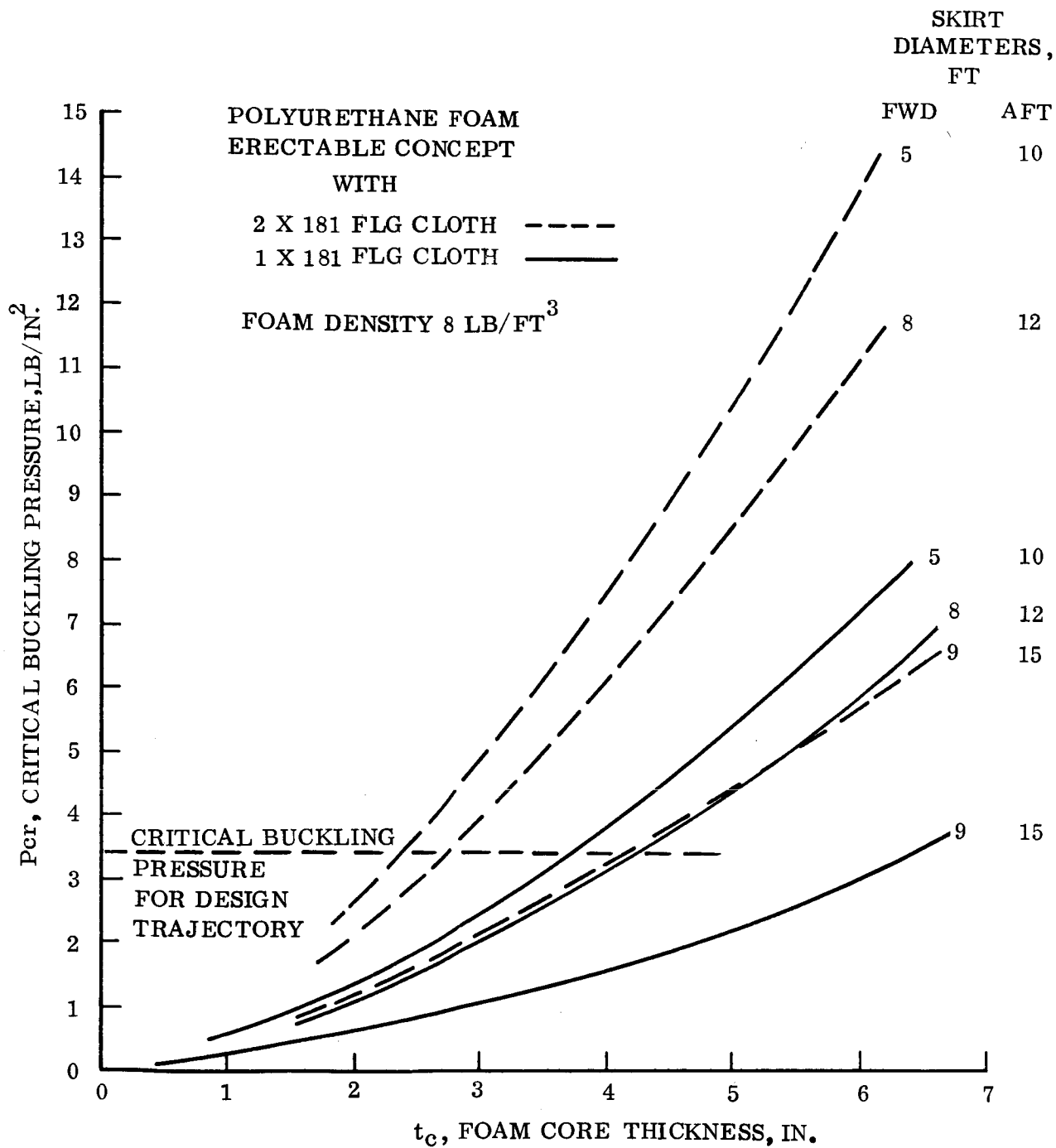


Figure 3.1.2-159. Foam Erectable Skirt, Structural Performance

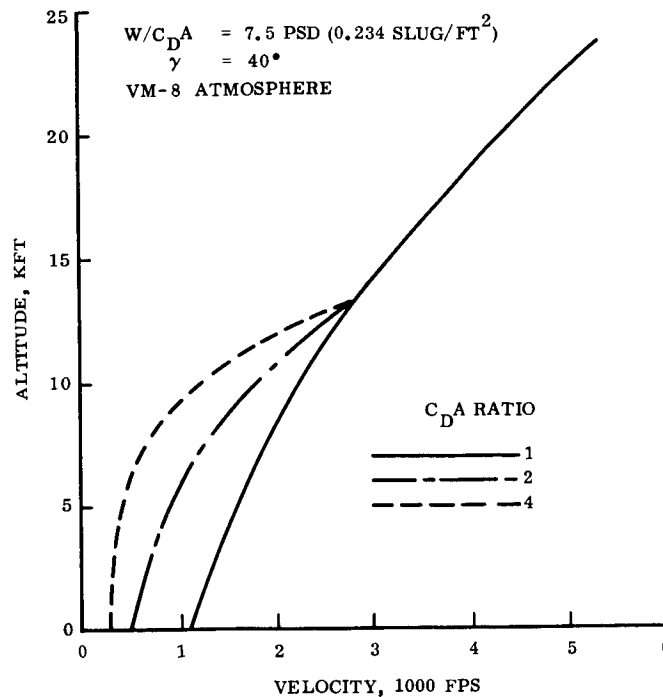


Figure 3.1.2-160. Inflatable Aft Body Performance Velocity Histories-Inflated at Mach 4

The increased performance to be achieved by inflating the afterbody at Mach 5 or 6 can be seen (in the limiting case) in fig. 3.1.2-164.

In the studies reported in ref. 3-53, the use of various types of attached and trailing inflatable decelerators are parametrically treated and compared for application to Mars entry vehicle deceleration. In these studies of ref. 3-53, the decelerator deployment is considered to occur at Mach numbers of the order of 5 or less. This is contrasted with the systems described in the previous sections of the report where the attached inflatable (or extendable) devices are considered to be deployed prior to atmospheric entry. As a consequence, the latter devices are exposed to the thermal and loading environments of the entry trajectory. The work was based upon vehicle deceleration to a target Mach number of 1. The attached tucked-back ballute appeared to represent the lowest weight fraction of the entry vehicle weight of the types considered. This fraction amounted to 0.175 for an inflated C_DA to vehicle C_DA ratio of about 2.5. Reference to fig. 3.1.2-163 indicates that at a Mach 2 altitude of 10,000 ft at which the allowable β is 6.25 psf (0.194 slug/ft²) for the basic aeroshell geometry, an area ratio of 2.4 will provide a ballistic coefficient of 7.25 psf (0.225 slug/ft²). Hence, an entry vehicle weight ratio of 7.25/6.25, or 1.16 represents an allowable entry vehicle weight increase of 16 percent. This may be approximately compared to the 17.5 percent weight fraction represented by the tucked-back ballute required to furnish the required drag area.

Although more detailed comparisons are required to establish the merits of the post-entry heating decelerator applications, this cursory comparison indicates that the advantage of such a device in the hard Lander application is marginal.

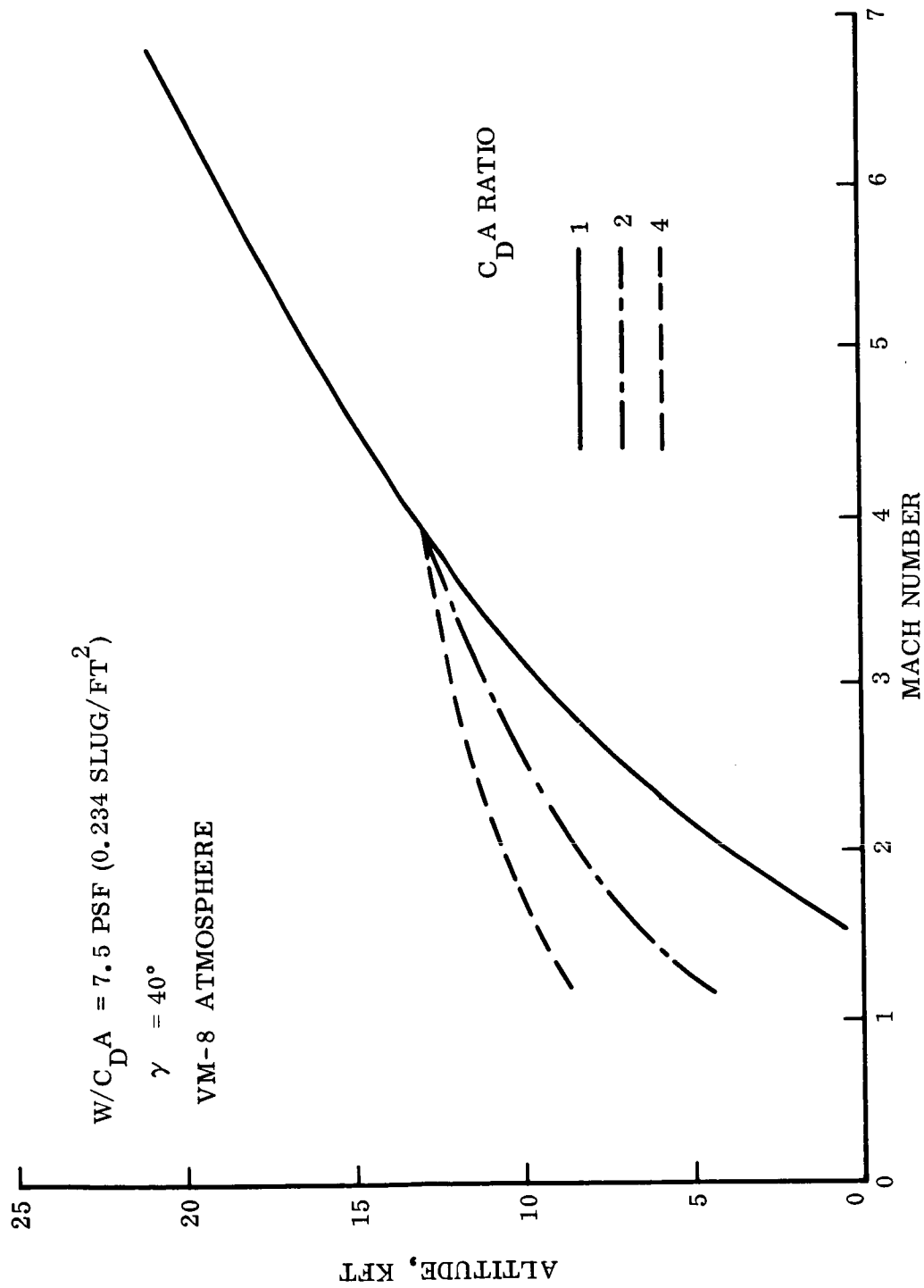


Figure 3.1.2-161. Inflatable Aft Body Performance Mach Histories-Inflated at Mach 4

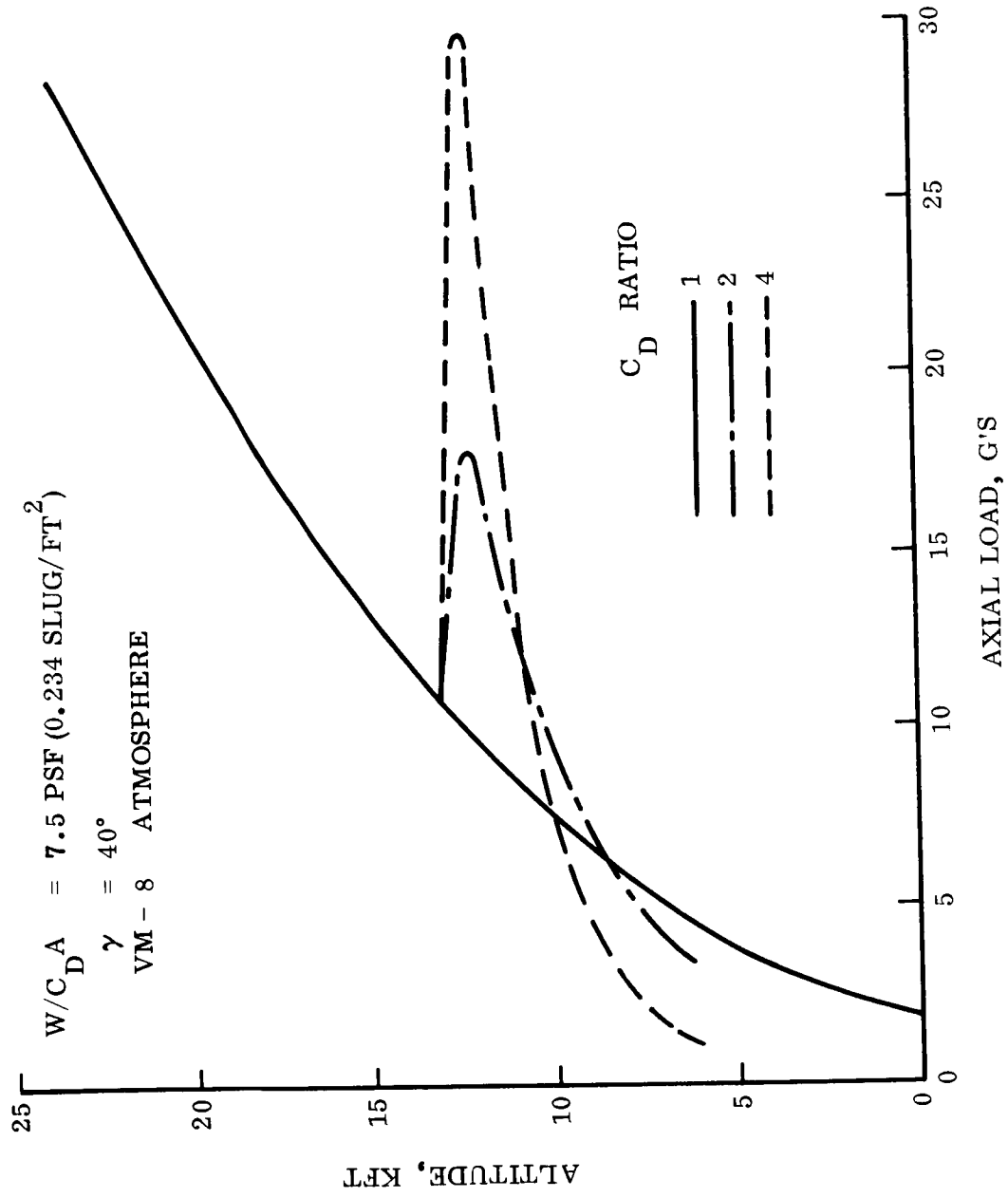


Figure 3.1.2-162. Inflatable Aft Body Performance Axial Load Histories-Inflated at Mach 4

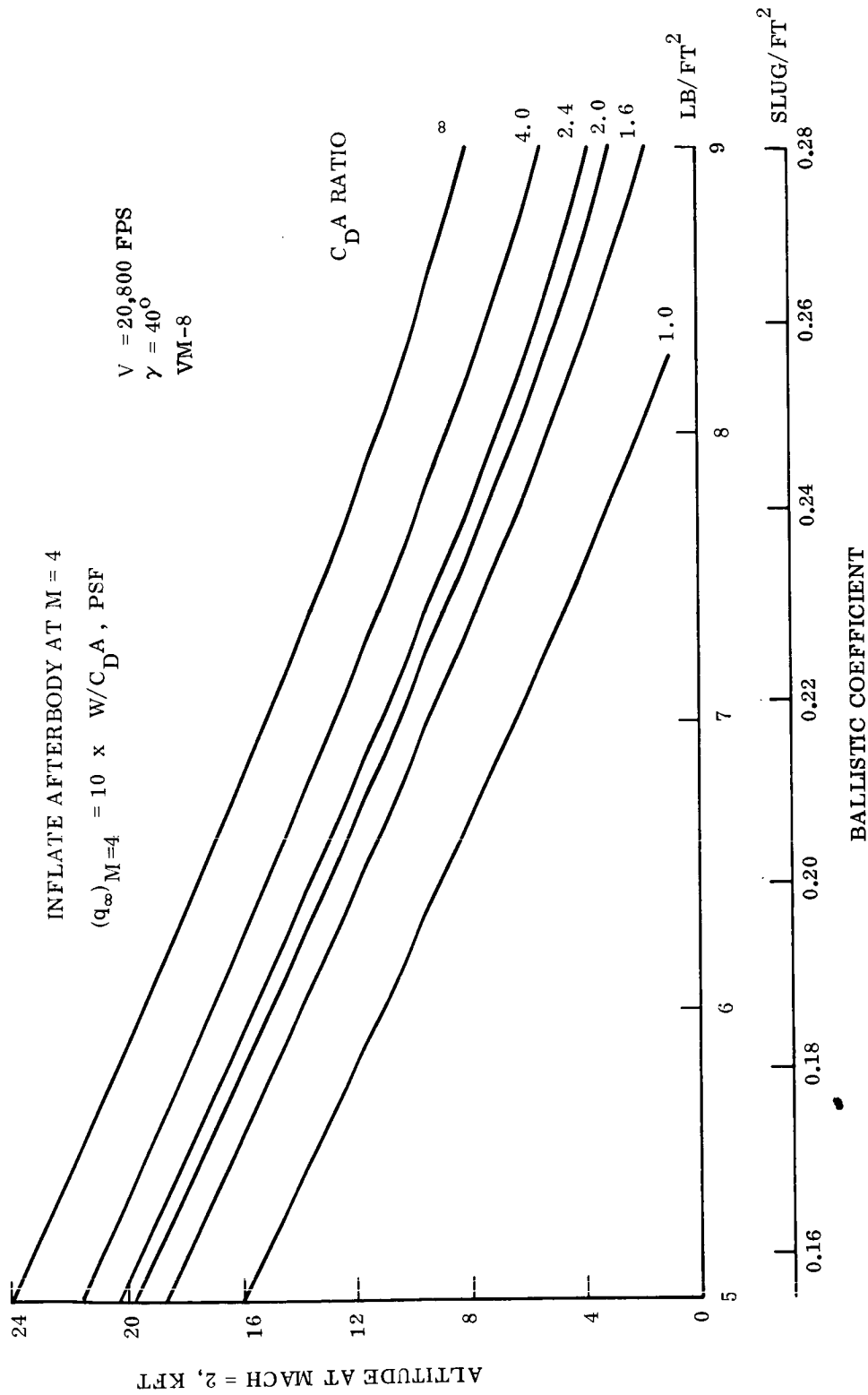


Figure 3.1.2-163. Inflatable Afterbody Performance

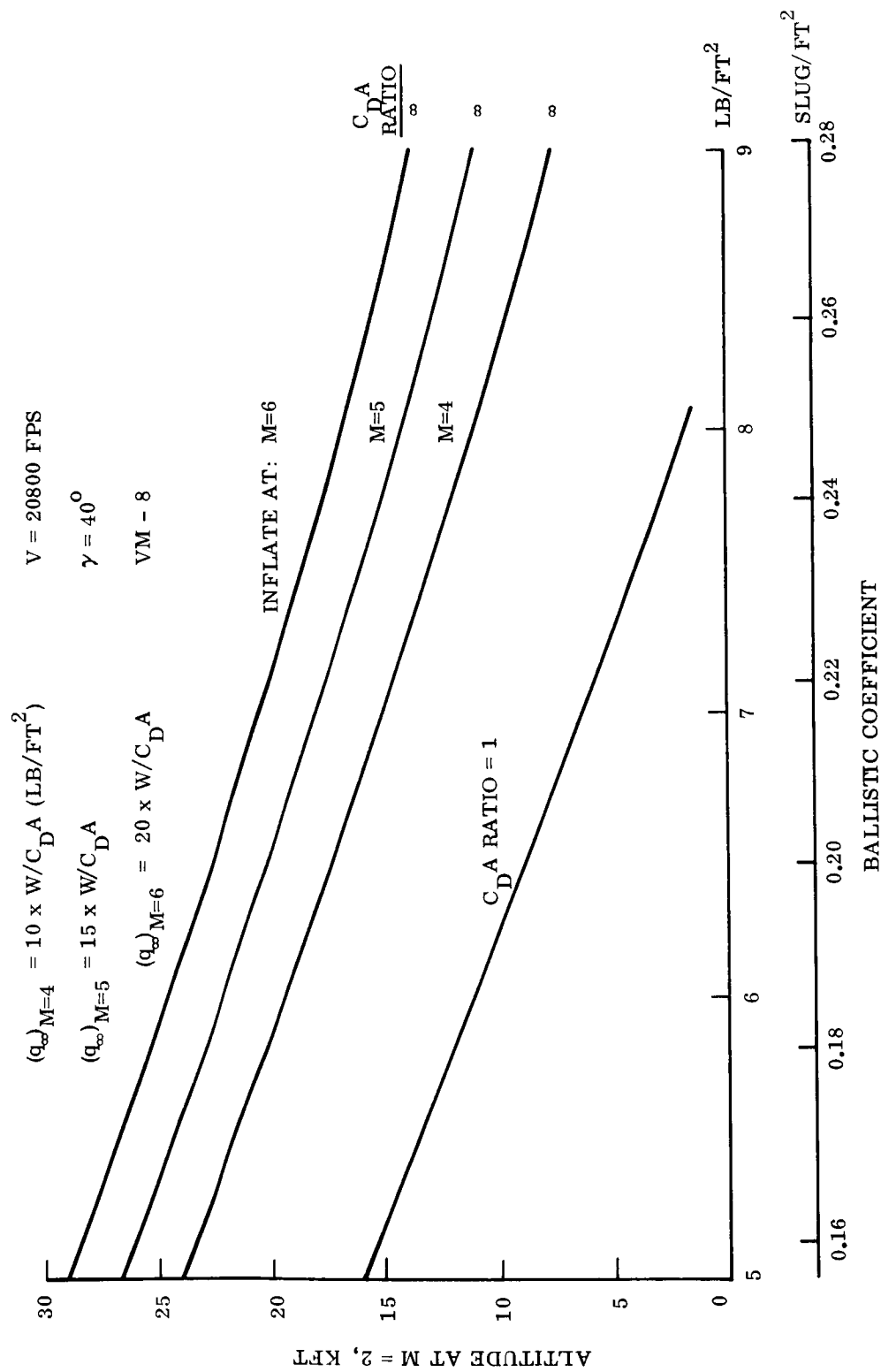


Figure 3.1.2-164. Inflatable Afterbody Performance

3.1.2.4 Extendable Flap Trade-off Studies Approach

An alternate concept of increasing aeroshell drag area before entry without exceeding the 10 ft shroud limit is to provide foldable flaps which would be stored within the 8.34 ft diameter constraint before separation from the Spacecraft. The approach taken in this investigation was to consider four flaps symmetrically placed around the aeroshell base with variable flap length and width. The four flaps are assumed to open to the cone angle of the aeroshell ($\theta_c = 60^\circ$) which has a basic diameter of 8.34 ft. The drag coefficient (Newtonian) of a flat plate at 60° angle-of-attack was corrected to the vehicle reference area and then added to the vehicle C_D to determine the overall drag history.

Figs. 3.1.2-165 and -166 show the variation of ballistic coefficient with flap length for a range of entry weights for a vehicle with four flaps, each two or four ft wide. Note in fig. 3.1.2-165 that for a ballistic coefficient of 0.171 slug/ft^2 ($\gamma_e = 40^\circ$ direct entry) and an entry weight of 1,000 lb, the flap length would have to be about nine ft (two ft wide flap). This length of flap, unless containing multiple hinge points, would present a difficult packaging problem. For the four ft wide flap, fig. 3.1.2-166, the corresponding length is approximately 4.7 ft which is a more reasonable packaging length. Even so, the extendable flaps do not package as well as the inflatable afterbody discussed in the previous sections.

Figs. 3.1.2-167 and -168 show the variation of flap force with length for the same range in entry weights as considered in figs. 3.1.2-165 and -166. The maximum dynamic pressures were determined by choosing a flap length, entering fig. 3.1.2-165 or -166 to get a corresponding ballistic coefficient, and then entering fig. 3.1.2-169 which shows $q_{\max}$ vs β for a direct entry into VM-8 at $\gamma = 40^\circ$ and $V_e = 20,800 \text{ fps}$. The flap force is then given by:

$$\begin{aligned} F &= C_N qS \\ C_N &= 2 \sin^2 \alpha = 1.5 \\ S &= \text{flap area} \\ q &= q_{\max} \text{ (from above)} \end{aligned}$$

Fig. 3.1.2-170 shows the variation of flap hinge moment coefficient with length for the two and four ft flap widths. The center of pressure of a flap has been shown from other programs to act at about 65 percent of the flap length and, hence, that value was assumed in this analysis. The hinge moment coefficient is defined by:

$$C_m = C_N - \frac{X_{CP}}{L_{REF}} \quad (13)$$

where:

$$C_N = 1.5$$

$$\frac{X_{CP}}{L} \times \frac{L}{L_{Ref.}} = 0.65 \quad \frac{L_{FLAP}}{L_{REF}} \quad (14)$$

$$L_{Ref.} = \text{flap width}$$

Structural Loading and Design Considerations - The variation of flap aerodynamic loading and hinge moments for 2 ft and 4 ft wide flaps with length varying up to 24 ft are shown in table 3.1.2-15 and fig. 3.1.2-171.

The alternate structural configurations to sustain these loads are: a box section with at least two spars, with a honeycomb sandwich trailing edge and ribs spaced at about 1 to 2 ft spacing. In addition, two or more struts would be required for each flap to extend and support the surface prior to and during Martian atmospheric entry. An alternate concept of honeycomb sandwich panel flaps with a single main spar and a single strut offers a simpler and lighter design. The former concept will be required for higher aerodynamic pressures corresponding to high ballistic coefficients combined with larger and longer flaps. The latter concept satisfies the requirements of lower ranges of these same parameters and is considered here in further detail.

With heat shield to structure interface temperature assumed to be in a lower range, tolerable for use of aluminum as the structural material, at least minimum gauge sizes would be required in honeycomb sandwich construction which optimizes the panel skins to give maximum buckling strength.

Structural weight densities were obtained for honeycomb sandwich construction using minimum face sheets and single spar and strut design. The depth of core was varied to increase load capability, strength, deflection and buckling criteria being applied. The results are presented in figs. 3.1.2-172 to -175. Figs. 3.1.2-172 and -175 indicate that for aerodynamic pressure loading of the order of 10 lb/in² (1440 lb/ft²), minimum face thickness (0.010 in.) sandwich with 0.5 in. core would provide adequate stiffness for 2 ft x 2 ft panels. The core density is kept constant at 6.1 lb/ft³ throughout. Figs. 3.1.2-174 and -175 present parametric data for a single spar and strut respectively for various flap lengths and aerodynamic pressures.

In addition to strength requirements, it is anticipated that aeroelastic stiffness requirements would be critical for the flap design and must be investigated for any final analysis.

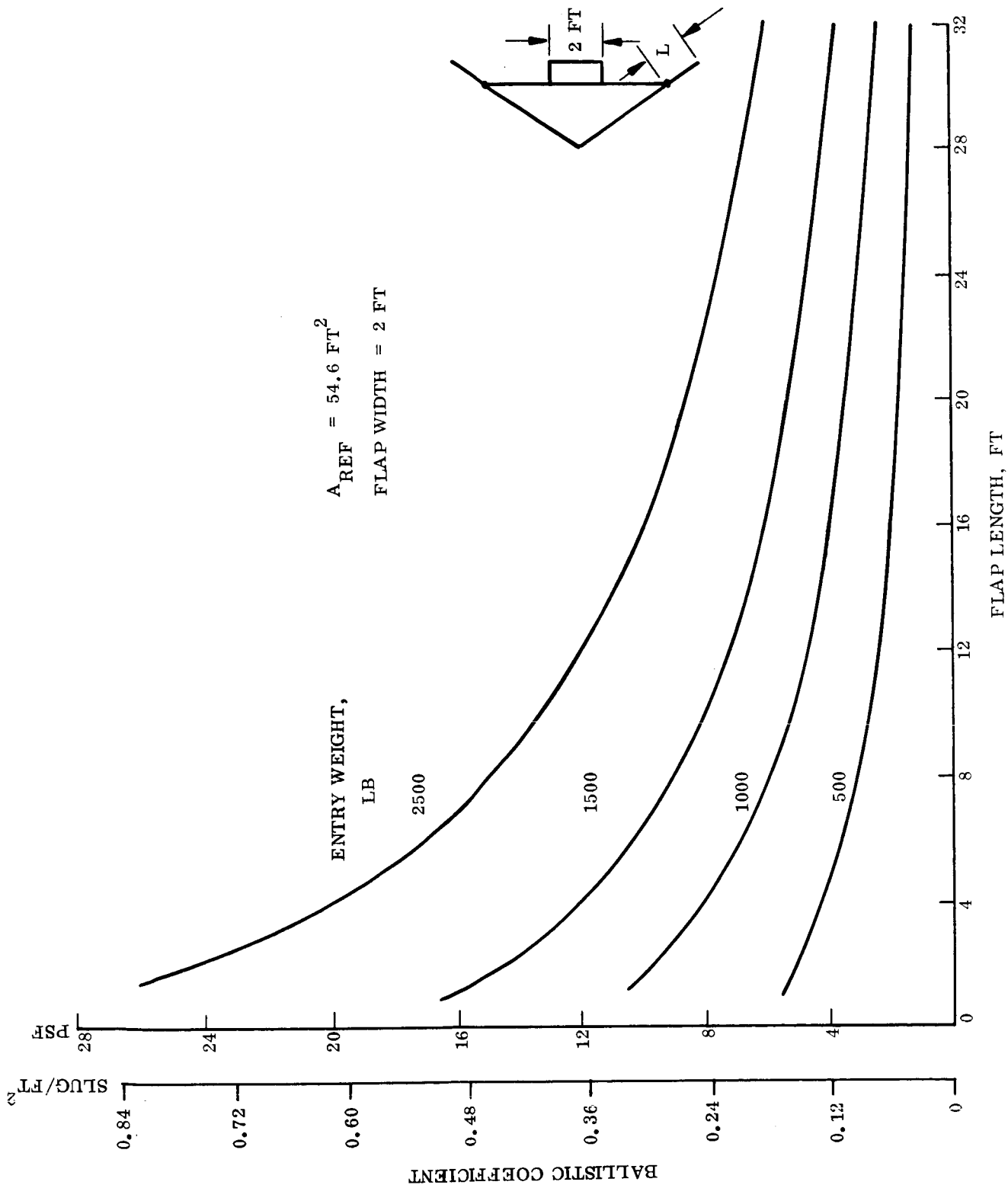


Figure 3.1.2-165. Ballistic Coefficient Variation with Flap Length

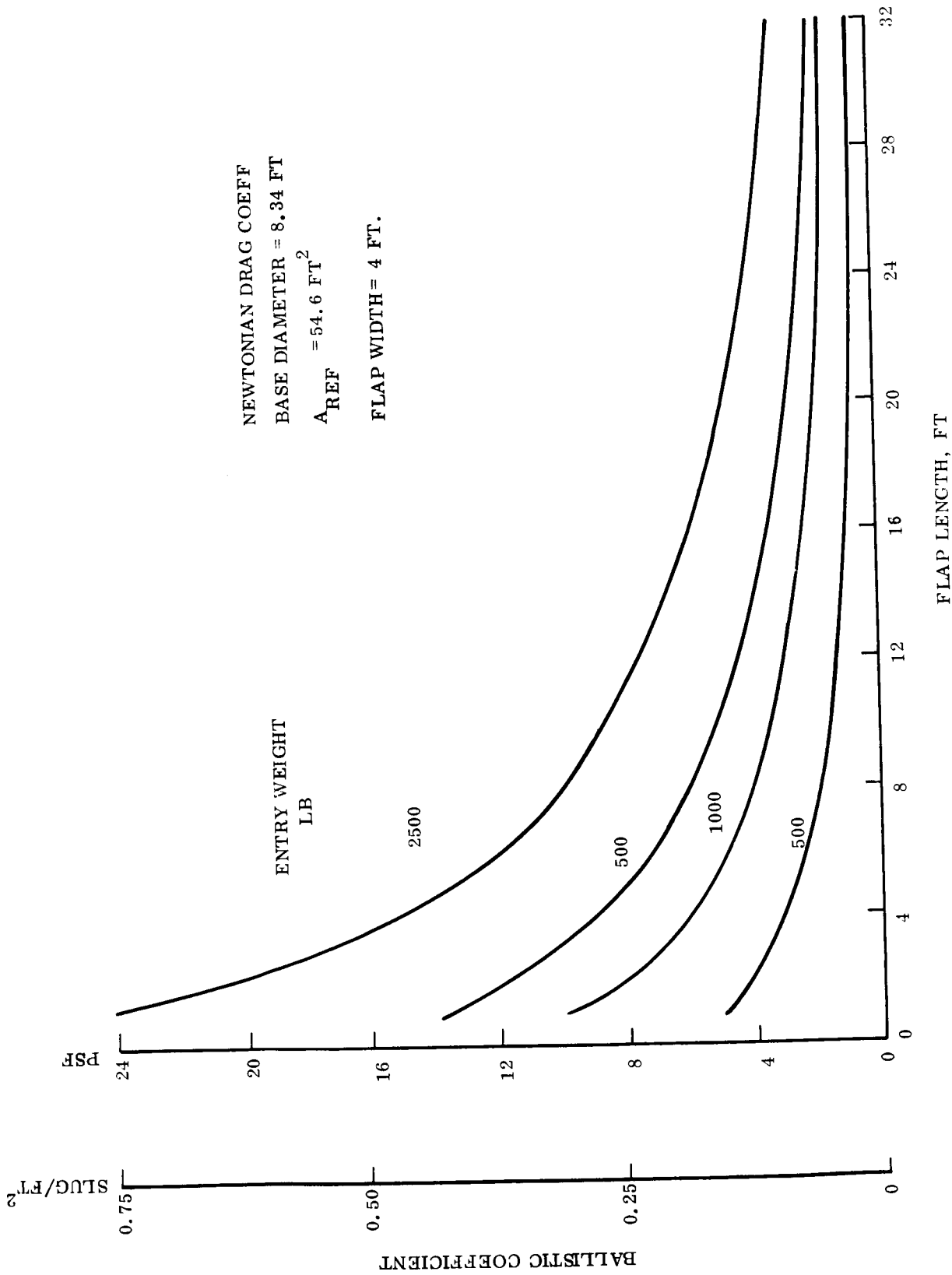


Figure 3.1.2-166. Ballistic Coefficient Variation with Flap Length

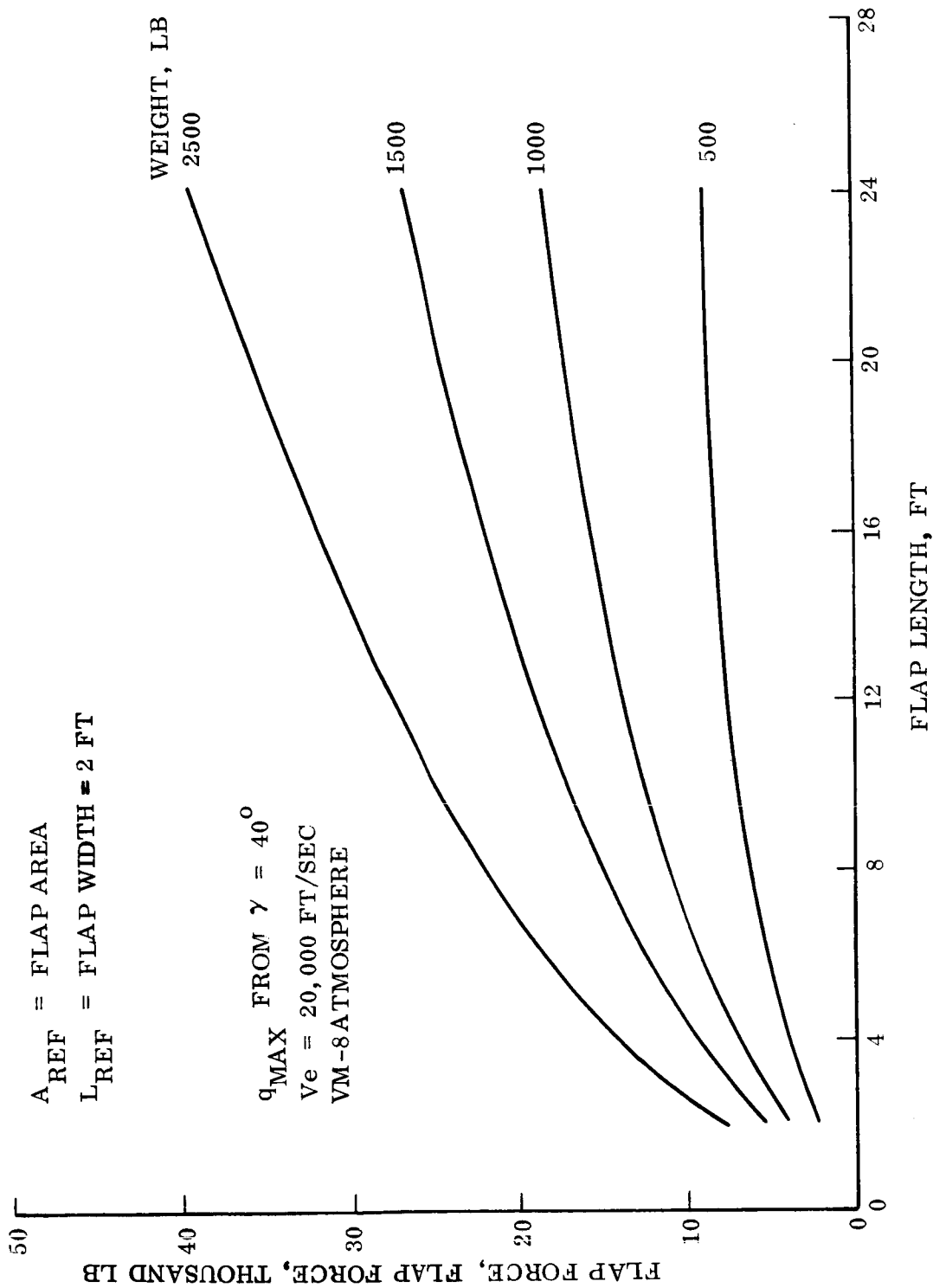


Figure 3.1.2-167. Variation of Flap Force with Flap Length

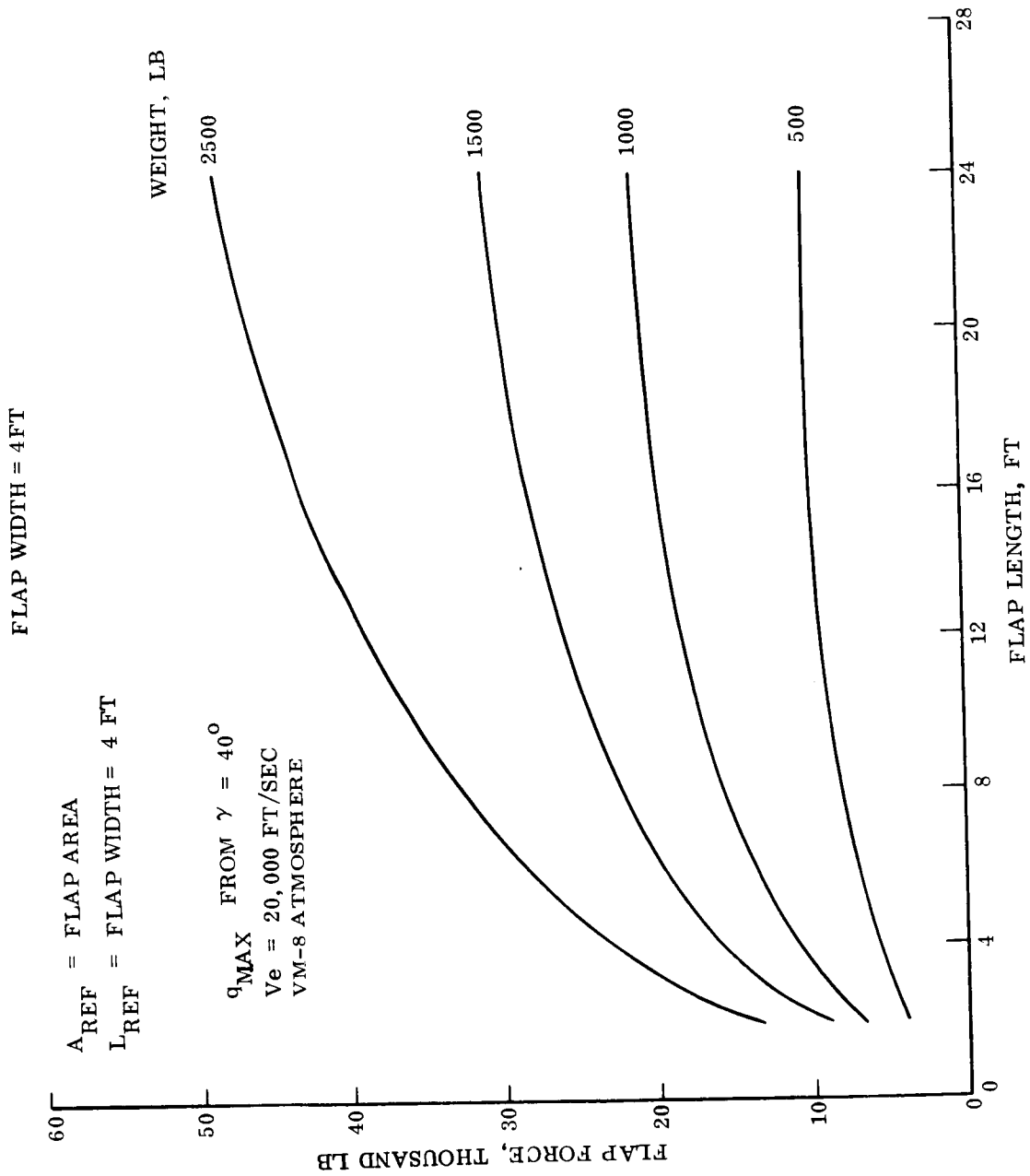


Figure 3.1.2-168. Variation of Flap Force with Flap Length

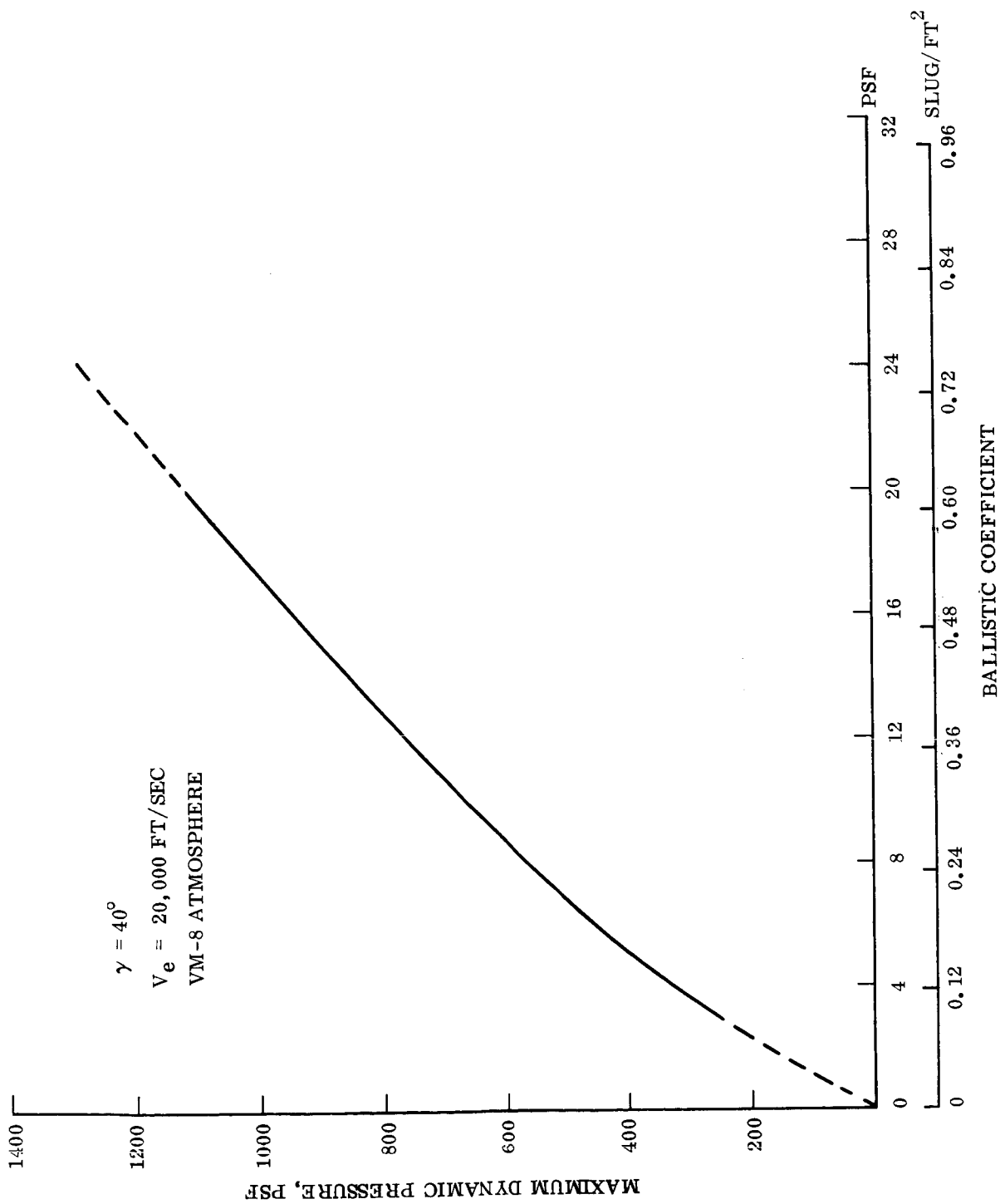


Figure 3.1.2.169. Variation of Max Q with Ballistic Coefficient

A_{REF} = FLAP AREA

L_{REF} = FLAP WIDTH

$$C_M = C_N \left(-\frac{X_{CP}}{L_{REF}} \right)$$

$$C_{N_{FLAP}} = 1.5$$

q_{MAX} FROM $\gamma = 40^\circ$

$V_e = 20,000$ FT/SEC

VM-8 ATMOSPHERE

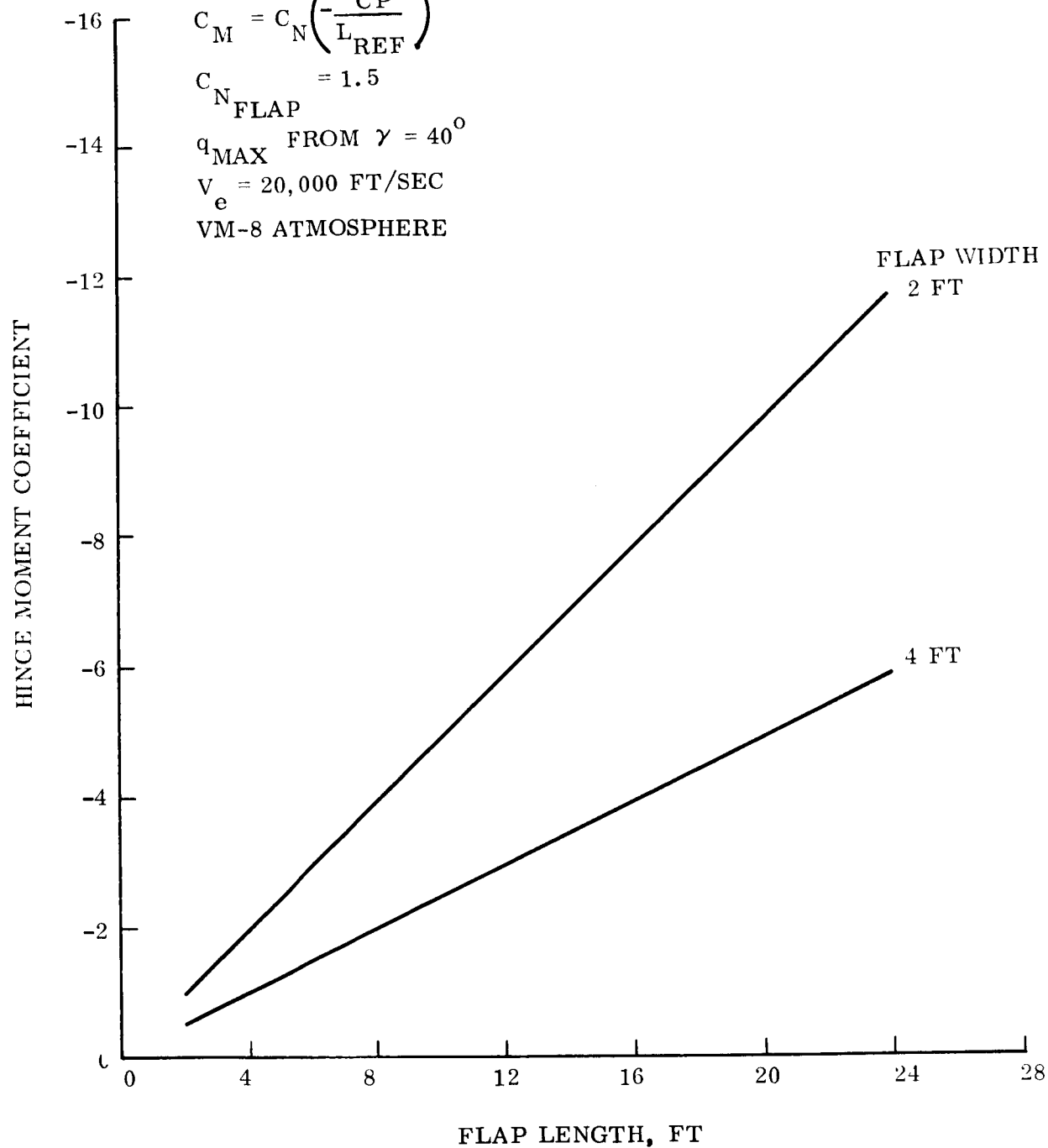
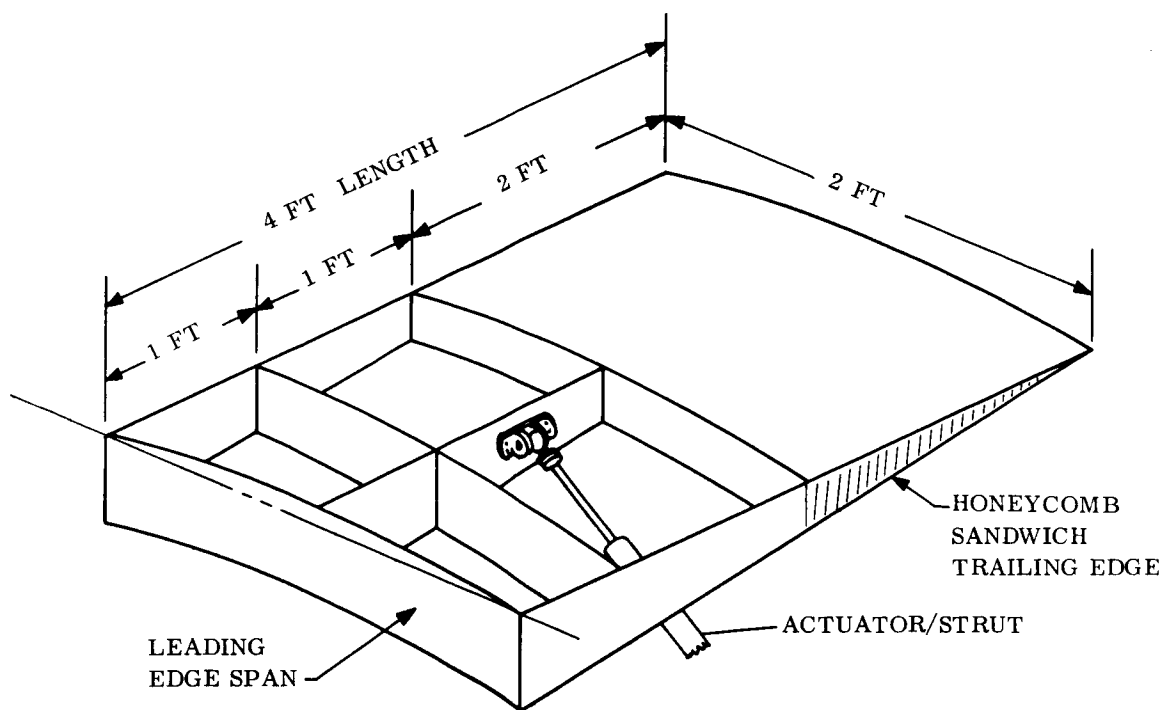
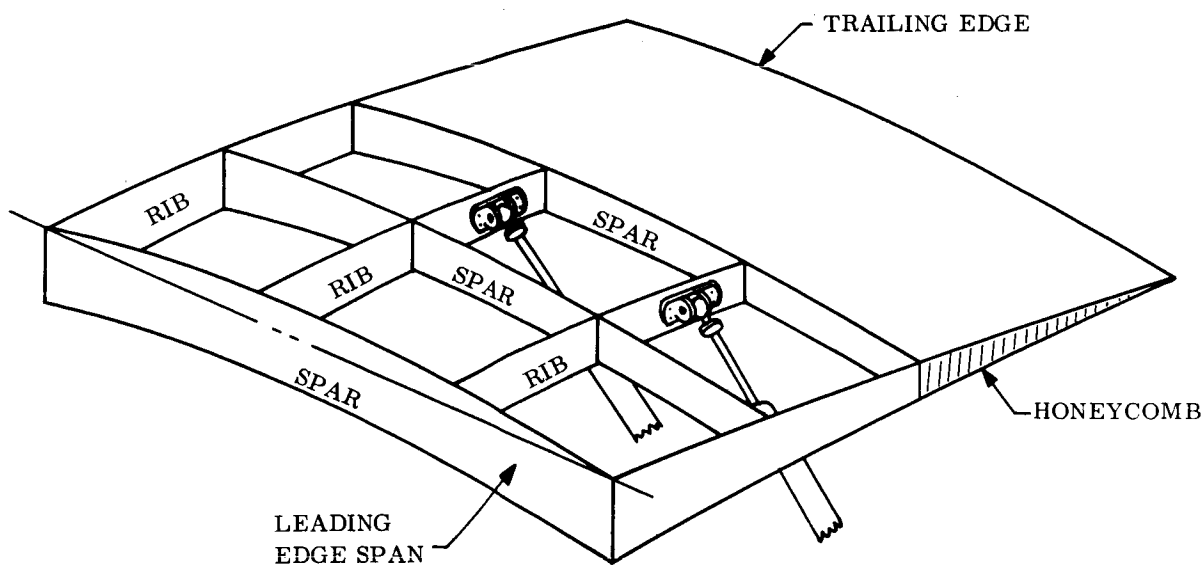


Figure 3.1.2-170. Variation of Hinge Moment with Flap Length



a) Two ft wide Single Span Configuration



b) Four ft Wide Twin Spar, Twin Actuator/Strut Configuration

Figure 3.1.2-171. Extendable Flap Design Concept

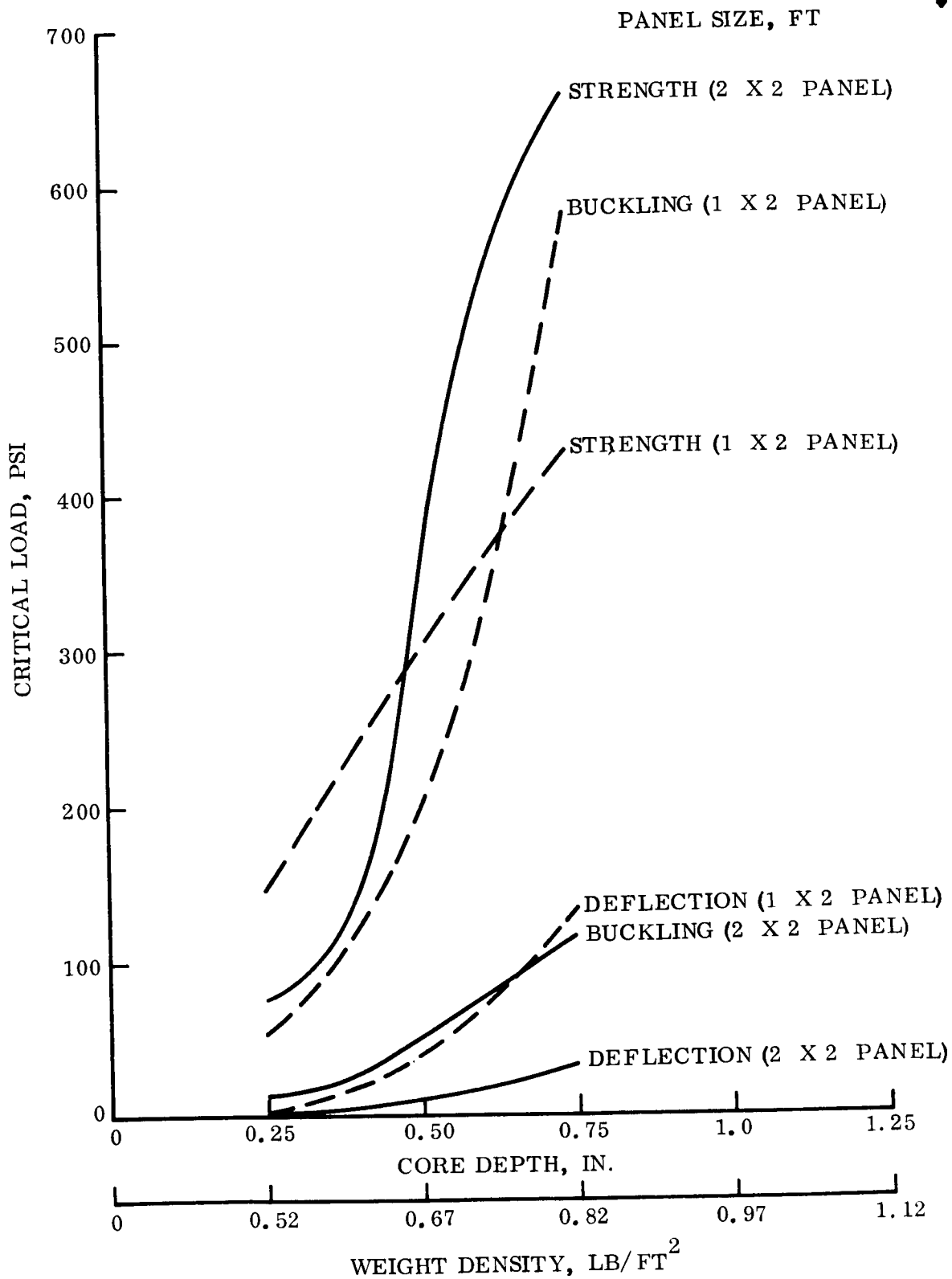


Figure 3.1.2-172. Panel Allowable Loading vs Core Depth and Panel Weight Density

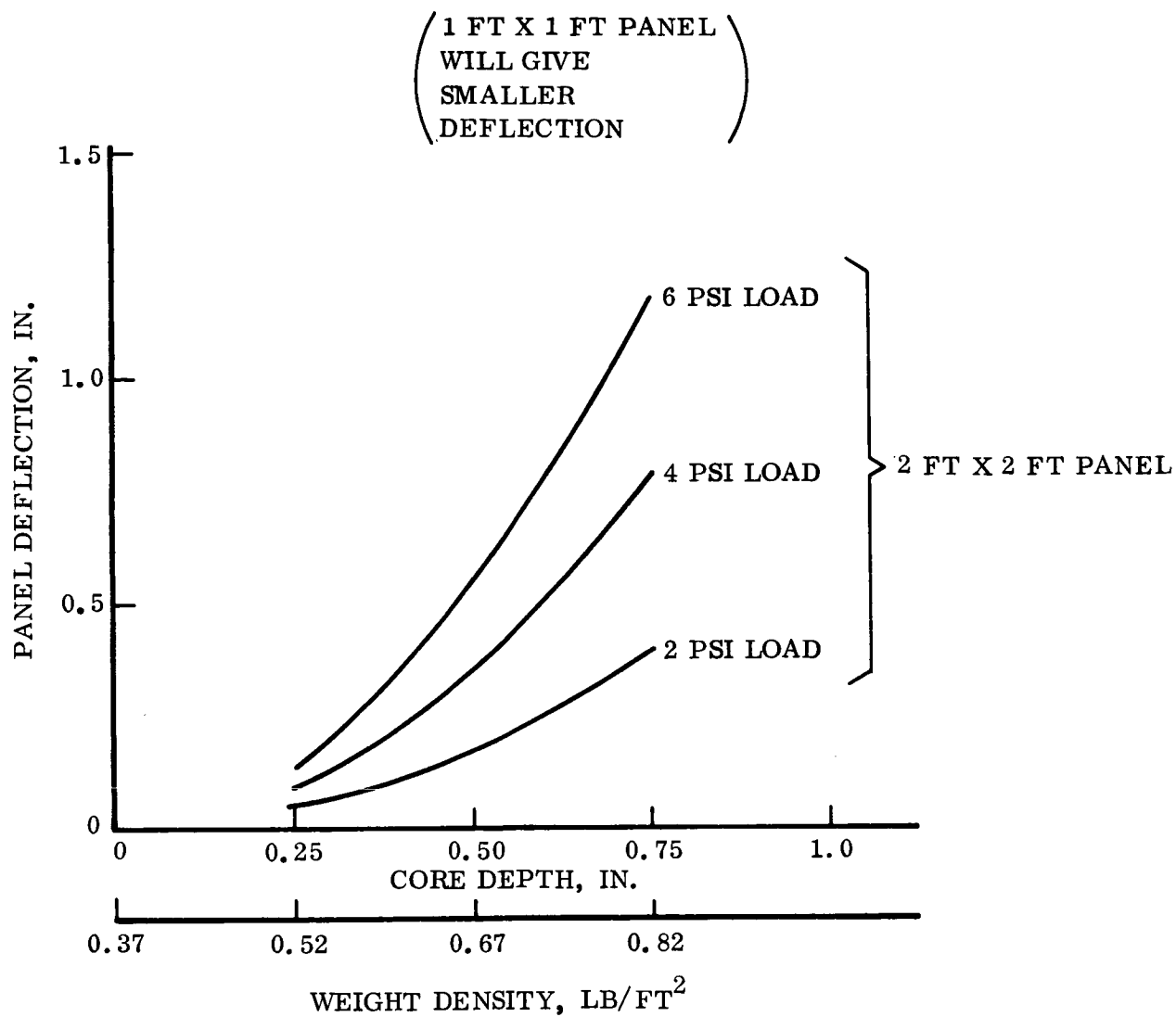


Figure 3.1.2-173. Panel Deflection vs Core Depth and Panel Weight Density

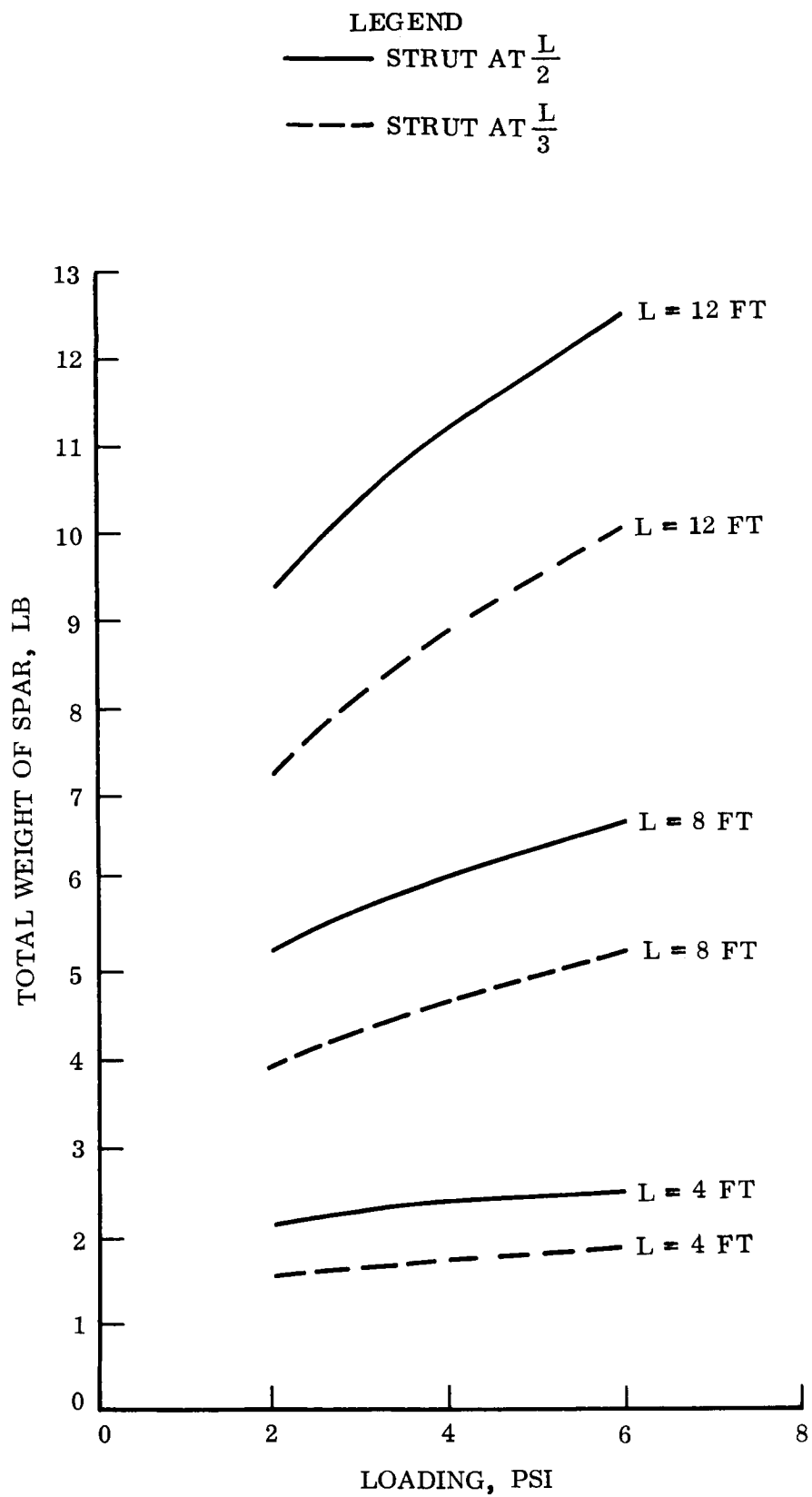


Figure 3.1.2-174. Total Spar Weight vs Panel Loading

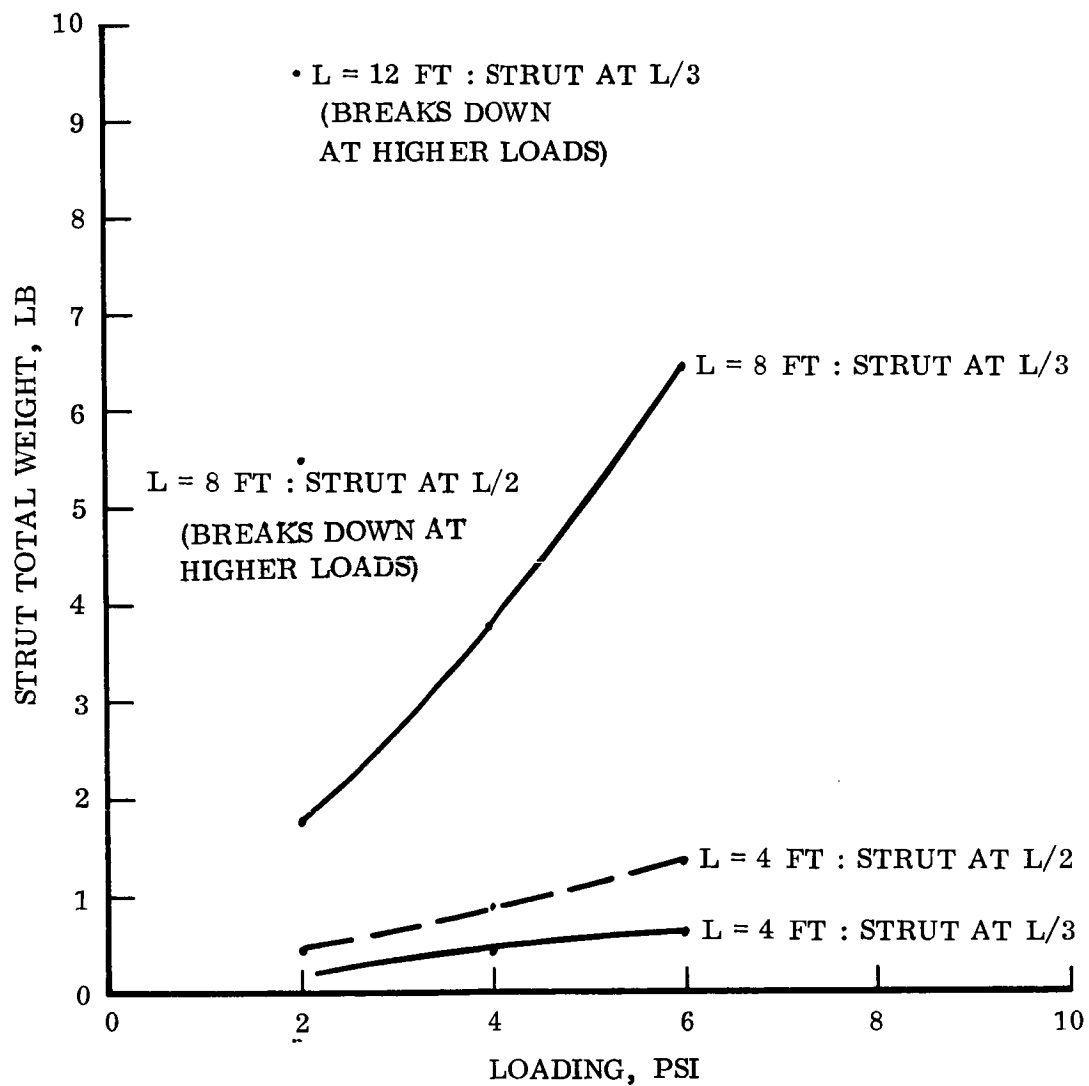


Figure 3.1.2-175. Total Weight of Strut vs Loading

3.1.3 LIFTING ENTRY CONFIGURATION PERFORMANCE

This section presents the results of an interdisciplinary effort on lifting entry configurations which are very similar to their ballistic counterparts save for necessary modifications to provide: (1) offset centers-of-gravity, (2) appropriate heat shields, (3) adequate structure, and (4) sufficient sustained attitude control power. Of these modifications, the attitude control system undergoes the most significant change from the ballistic counterpart due to much higher maximum and integrated roll torques during entry. Overcoming such torque, to provide particular vehicle attitudes for selected lift-to-drag ratios, typically requires a large gas supply, an increase in duty cycle, and control-jet nozzle re-sizing. The 0.3 L/D system has an integrated roll torque approximately 8.2 times that for the ballistic entry, with corresponding changes in the roll control system. However, the gas weight increases only from 4 to 32.8 lb, so that the weight penalty for a modest L/D is minimal.

This section presents in sequence the trajectory performance trade-offs and rationale; the aerodynamic force, moment, and static stability data; the aerodynamic roll torque, attitude error, and control requirements; the heat shield performance data; the structural materials data; and the overall configuration trade-off results. Many of these items are such relatively minor iterations of the comparable ballistic discussions of paragraphs 3.1.2.1 and 3.1.2.2 that only the separate lift-crested data is included in this section with a ballistic base point, the reader being referred to the ballistic writeup for that material otherwise duplicated. Thus, the major results of this section are those of an aerodynamic and trajectory nature.

Further, although both the Apollo Command Module shape and a raked-cone-cylinder shape were chosen to compare with a sphere-cone aeroshell shape for their suitability in performing a lifting entry into the Mars atmosphere, the bulk of the results reported here deal only with the aeroshell. The other two shapes do not have a comparable ballistic counterpart, and analysis was consequently more abbreviated.

3.1.3.1 System Constraints

The lifting entry configurations considered during this study derive their lifting capability through the use of offset centers-of-gravity which force the vehicles to aerodynamically trim in a lifting condition. Each lifting configuration is a modification of its ballistic counterpart, with changes in internal packaging to provide particular offset center-of-gravity locations for specific lift-to-drag ratios. In addition, a fully active roll control system is required to maintain proper vehicle attitude during entry. Small increases in heat shield weight also were found necessary due to the changed heating pulses.

The lifting configurations were restricted to sizes within an essentially cylindrical volume with a 100 in. diameter, whose exact shape is defined by the 10 ft shroud design. CONSEP studies fixed the parametric variations of $W/C_D A$, entry vehicle weight, and decelerator load, for various L/D ratios. Individual variations

(with L/D) of heat shield and control system weights were separately determined for specific point designs.

The results of these designs, studies, and analyses show that an L/D of 0.15 or greater is required for direct entry. Such an L/D represents some 500 lbs decelerator load, which is near the minimal such value that includes a minimum science package.

The following sub-sections detail the major results of the lifting configuration performance activities, principally in the aerodynamic, thermodynamic, and structural engineering disciplines. The results reported here are not expected to change by any substantial amount as a result of further lifting configuration studies.

3.1.3.2 Trajectory Performance Trade-offs

The use of lift has been studied for possible application to the direct entry vehicle design. The expression "lift" is used although the designs are not true lifting vehicles in the maneuverable sense, the intent rather is to perform a slight pullup to allow the speed to decay. The controlling parameter for the vehicle design is the altitude/path angle combination at Mach 2 in the VM-8 atmosphere which defines the minimum altitude for parachute deployment.

Values of L/D from zero to 0.3 have been used in this study. Comparisons in this section have been made primarily on the basis of altitude at Mach 2 in the VM-8 atmosphere model. Fig. 3.1.3-1 shows the effect of L/D on path angle at Mach 2, and the variation in flight path angle shown can have an effect on the minimum altitude of deployment consistent with terminal requirements; however, evaluation requires detail analysis of trajectories incorporating parachute sequencing. These factors have been included in the aerodecelerator analysis (Section 3.2).

Figs. 3.1.3-2 through -4 show the variation of altitude at Mach 2 as a function of ballistic coefficient for L/D values of 0, 0.15, and 0.30. As can be seen, the value of lift augmentation is most pronounced at the steep path angles, and in fact a trend reversal occurs at 20° entry path angle where increasing the L/D from 0.15 to 0.30 actually decreases the altitude and increases the path angle at Mach 2. The explanation for this lies in the roller-coaster-like altitude history of the high lift shallow entry case. This phenomenon is illustrated in the time histories of entry parameters shown in figs. 3.1.3-5 and 3.1.3-6 for the 20° path angle case, L/D = 0.15, and in figs. 3.1.3-7 and -8 for the 20° path angle case, L/D = 0.30.

Parametric maps of the peak dynamic pressures encountered in the VM-8 atmosphere model are presented in figs. 3.1.3-9 through 3.1.3-11 for 20°, 30° and 40° path angles, respectively. For comparative purposes, fig. 3.1.3-12 shows the peak pressures for a 20° path angle into the VM-3 model atmosphere. Parametric data on the integral of pressure is shown in fig. 3.1.3-13 for entry into the VM-8 model atmosphere. As was shown in the non-lifting section, the integral of pressure is essentially independent of path angle and, furthermore, is but slightly dependent on atmosphere model.

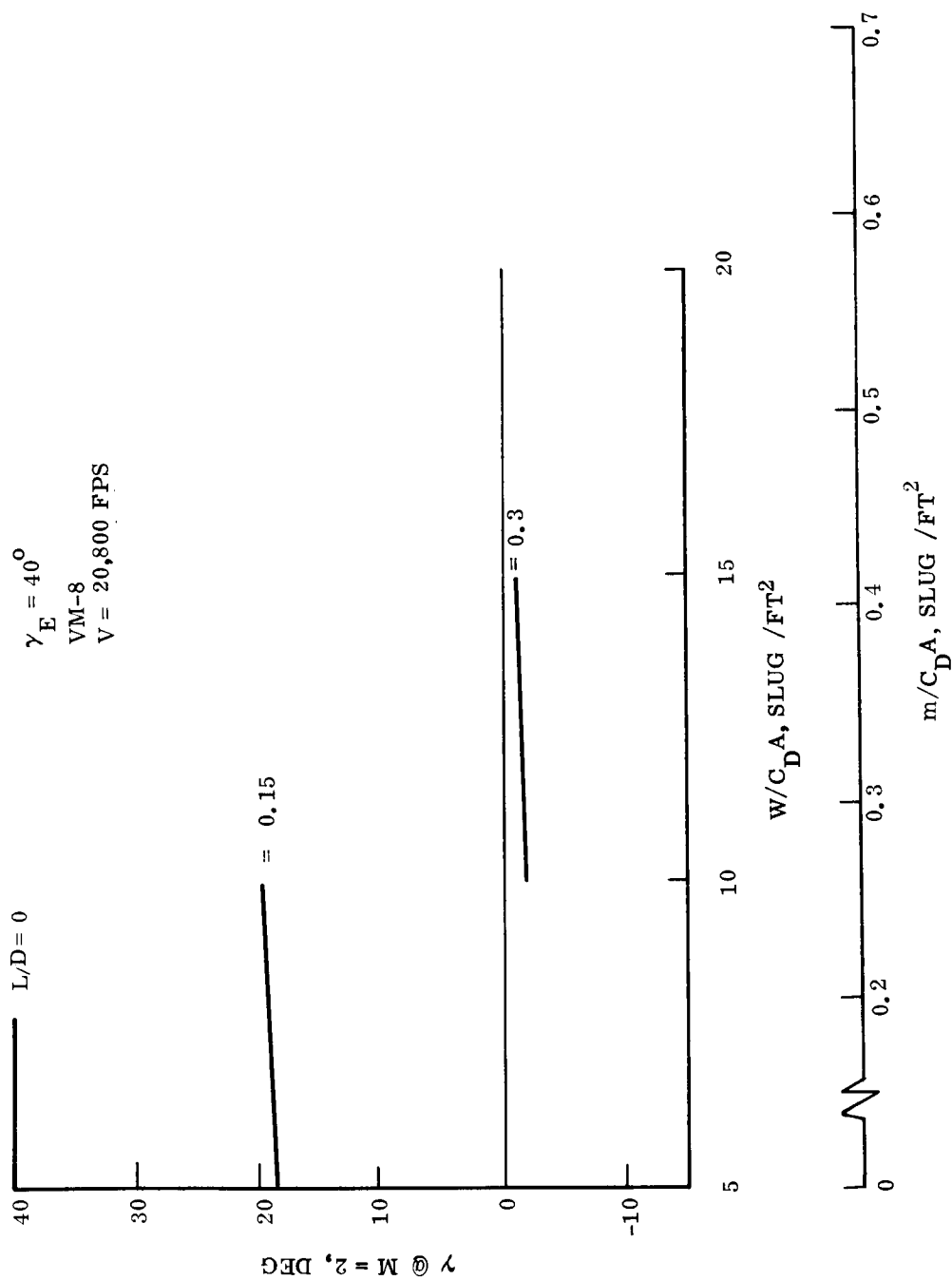


Figure 3.1.3-1. Path Angle at Mach 2

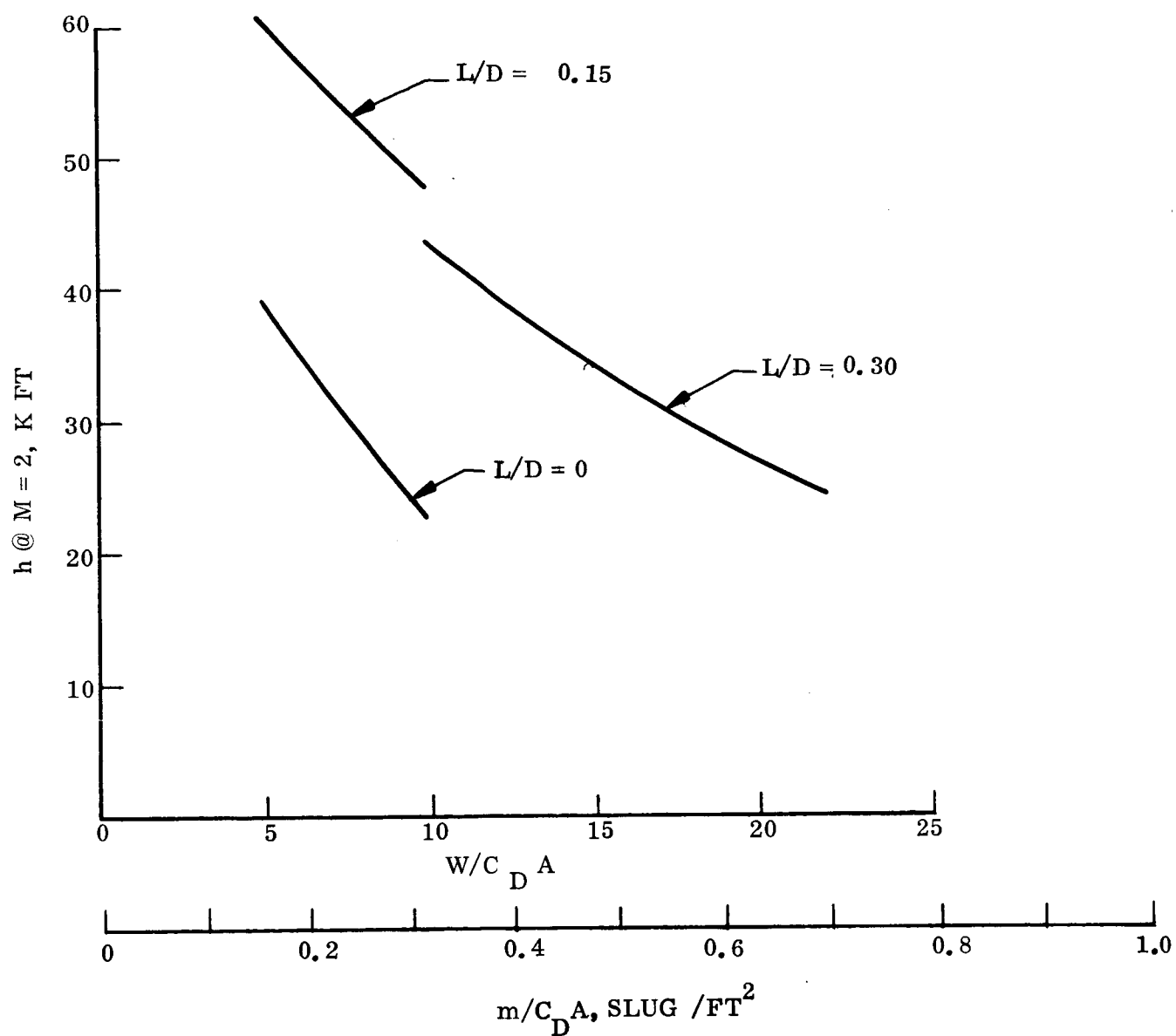


Figure 3.1.3-2. Altitude at Mach 2, $\gamma = 20^\circ$, $V = 20.8$ kfps, VM-8

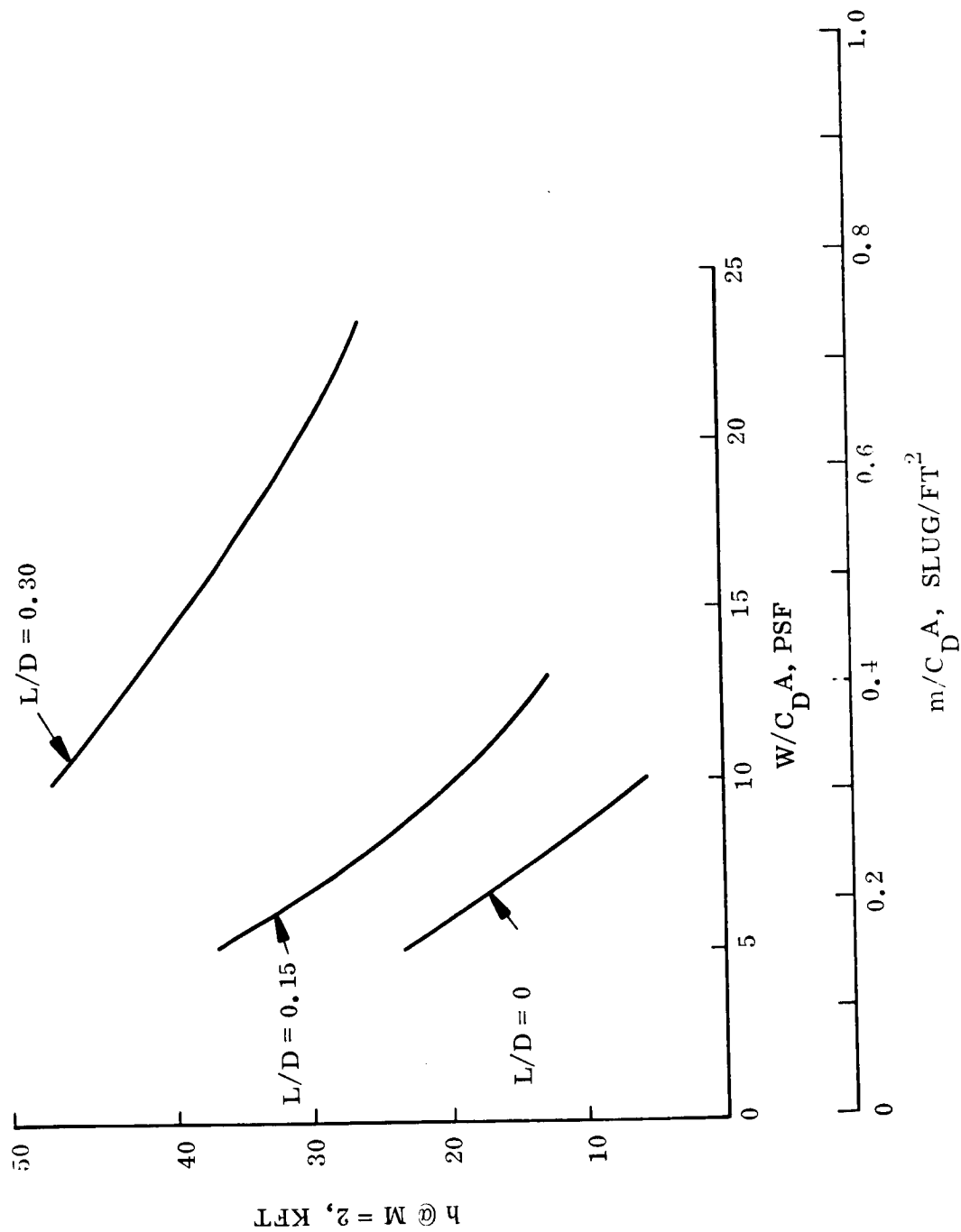


Figure 3.1.3-3. Altitude at Mach 2, $\gamma = 30^\circ$, $V = 20.8 \text{ kfps}$, VM-8

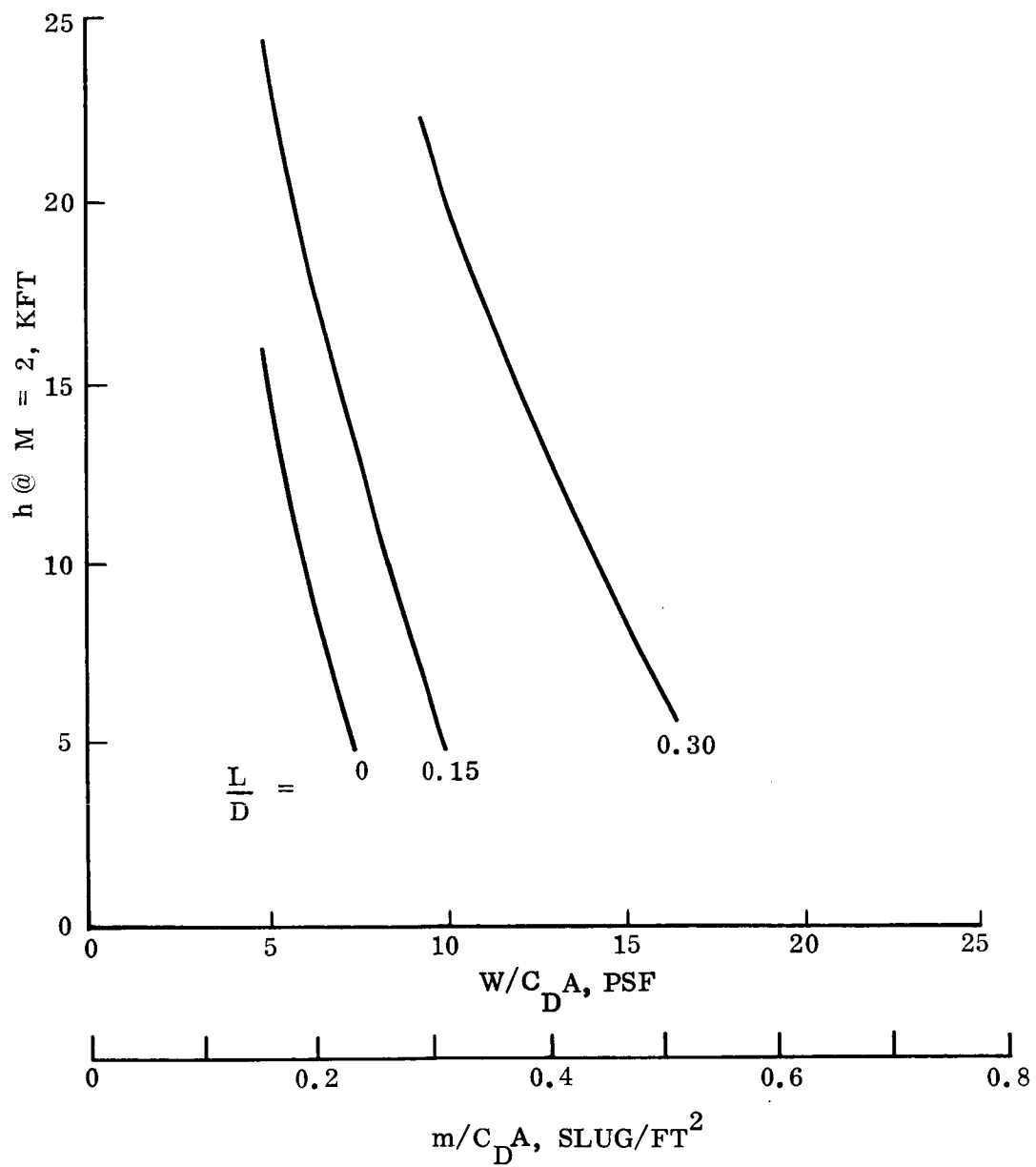


Figure 3.1.3-4. Altitude at Mach 2, $\gamma = 40^\circ$, $V = 20.8$ kfps, VM-8

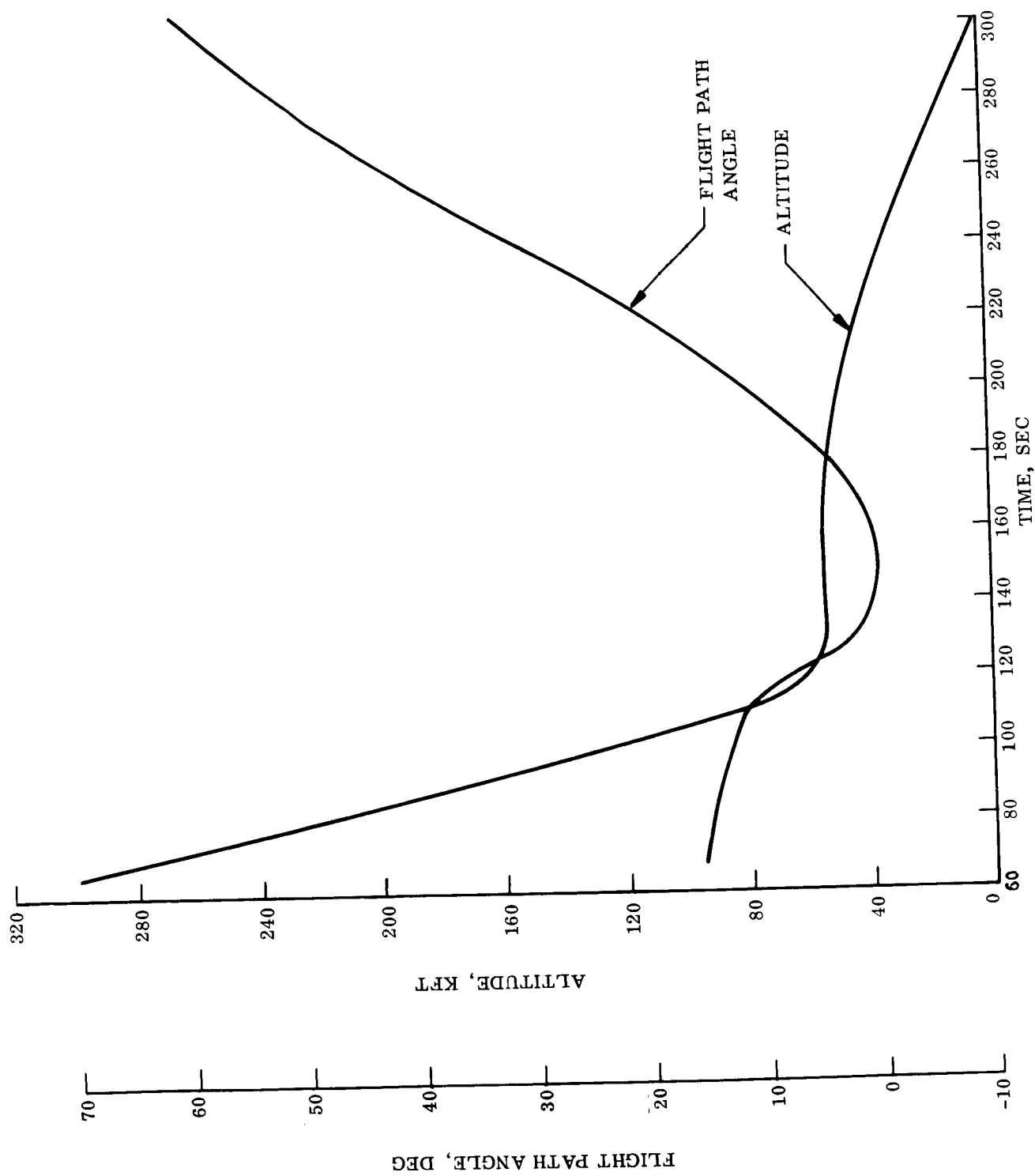


Figure 3.1.3-5. Entry Time History for h and γ at $L/D = 0.15$, $\beta = 0.312$ slug/ft², $\gamma_E = 20^\circ$, VM-8

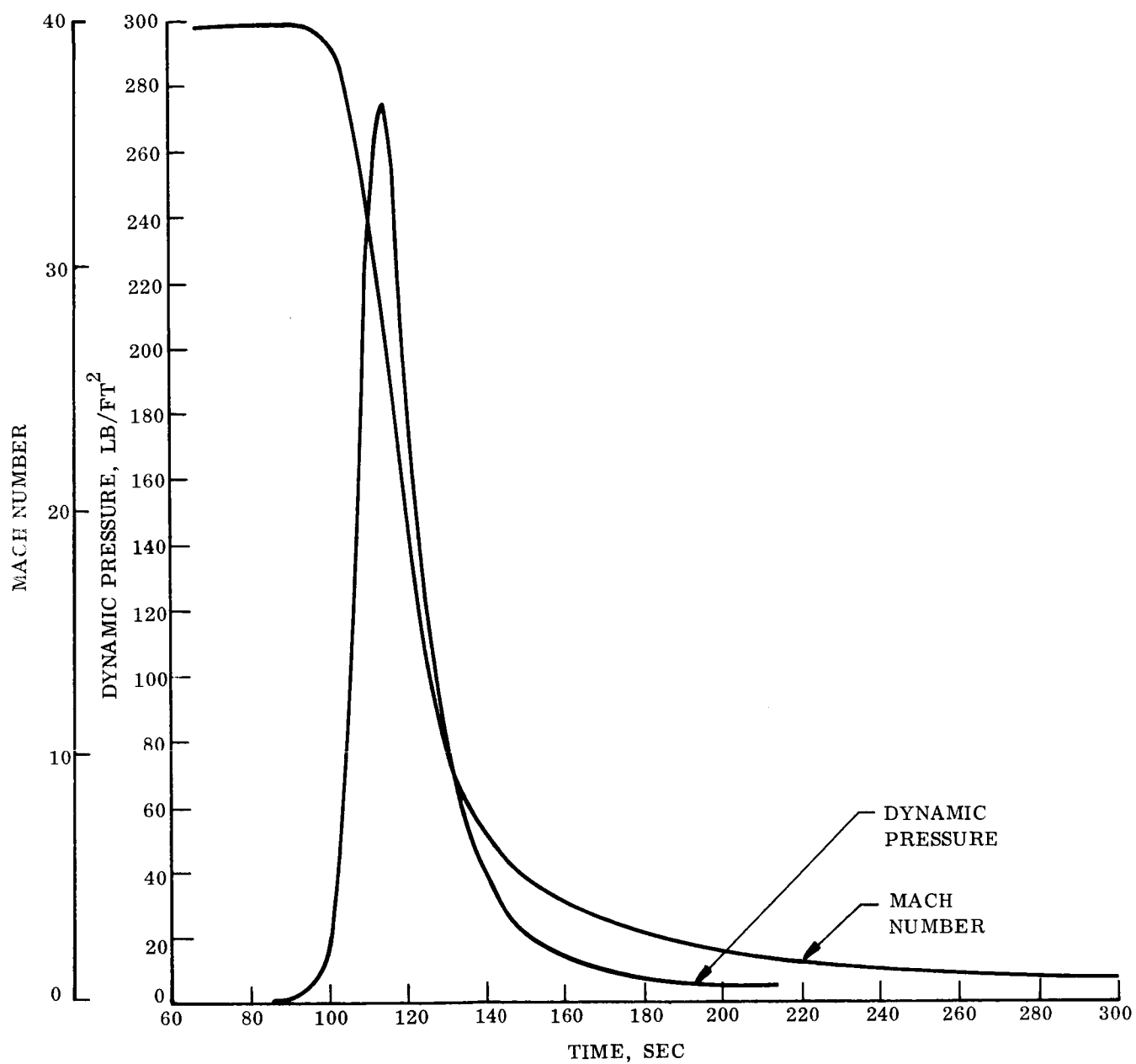


Figure 3.1.3-6. Entry Time History for M and q at $L/D = 0.15$, $\beta = 0.312$ slug/ft², $\gamma_E = 20^\circ$, VM-8

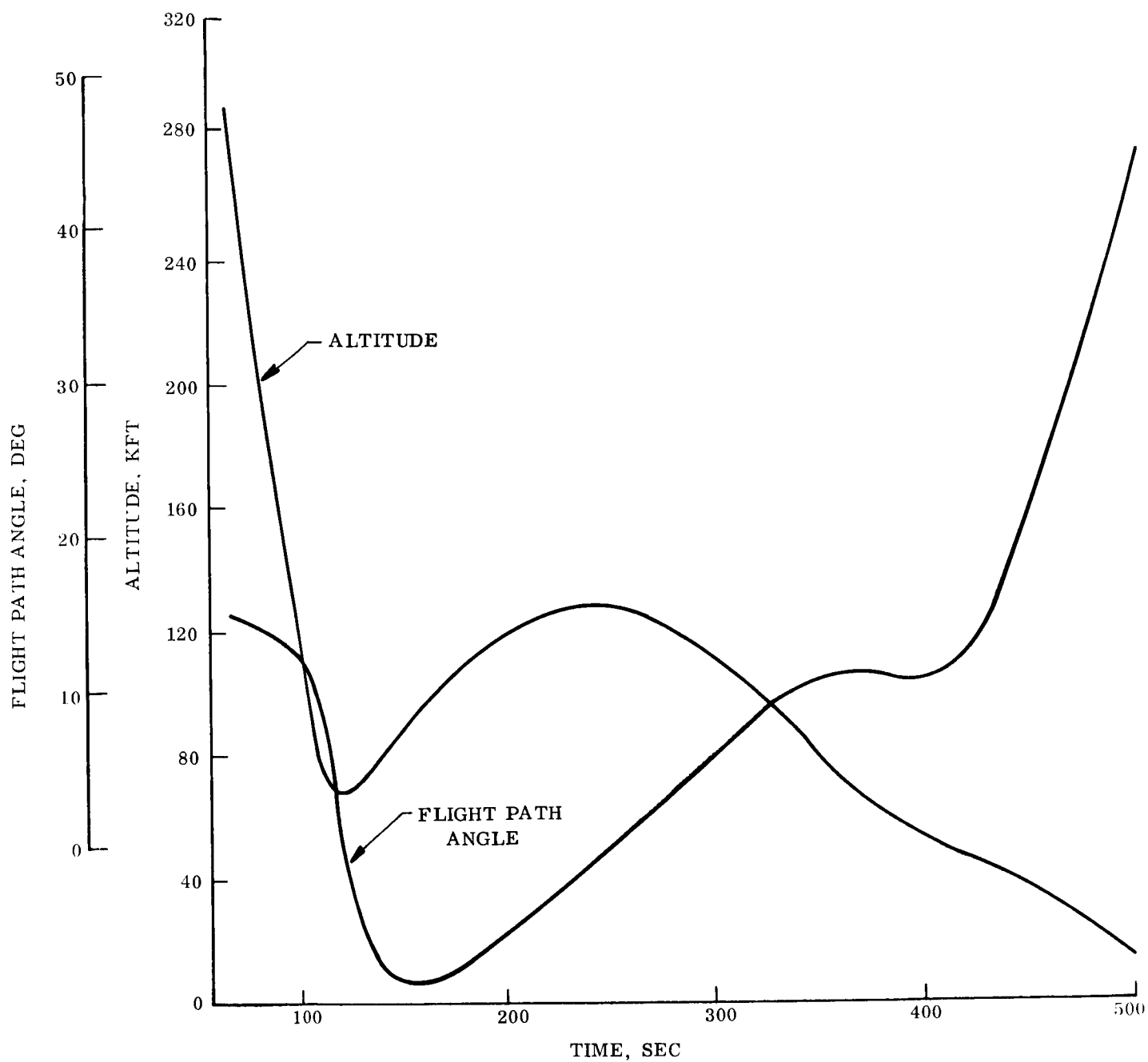


Figure 3.1.3-7. Entry Time History for h and γ at $L/D = 0.3$, $\beta = 0.312 \text{ slug/ft}^2$, $\gamma_E = 20^\circ$, VM-8

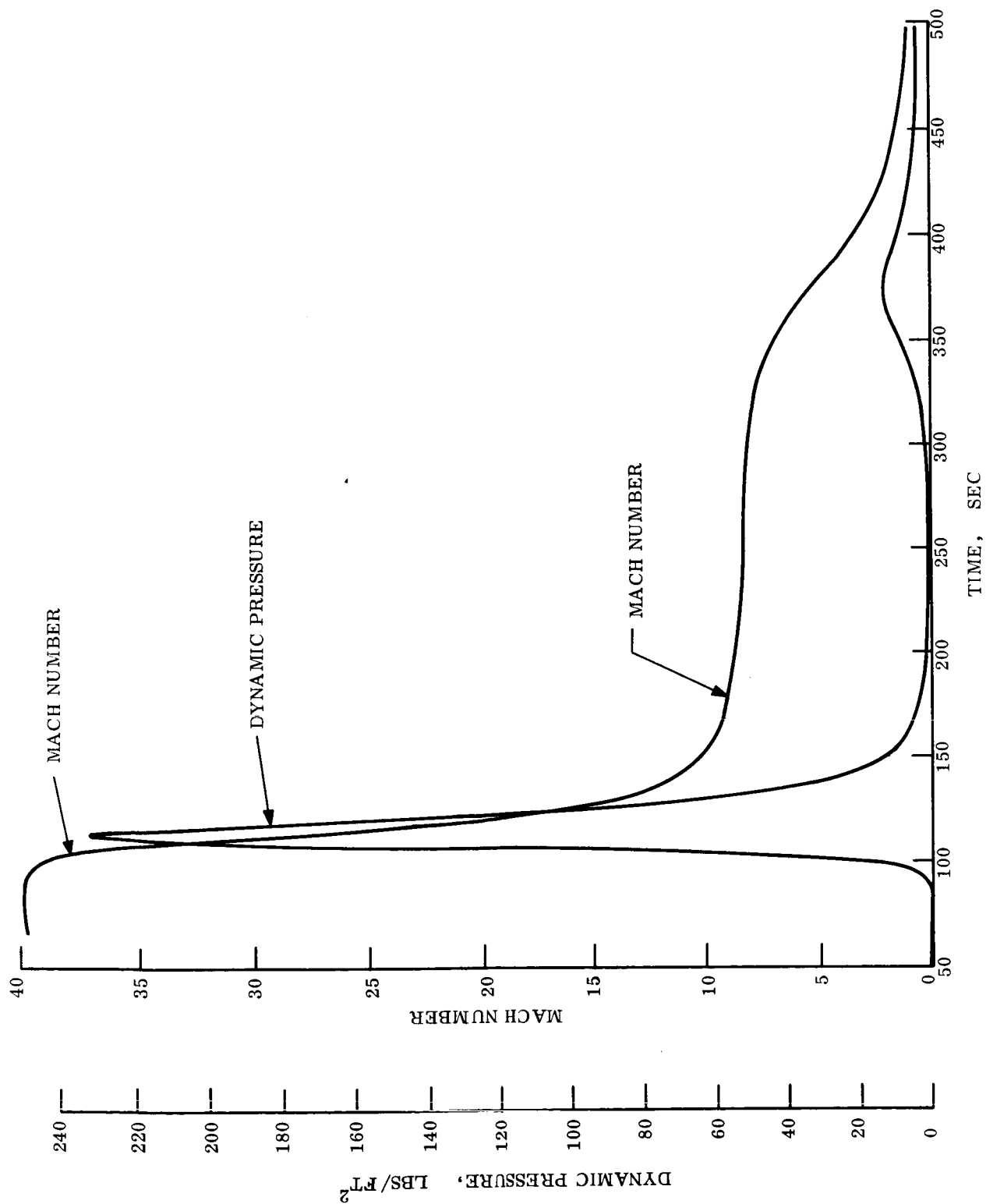


Figure 3.1.3-8. Entry Time History for M and q at $L/D = 0.3$, $\beta = 0.312 \text{ slug/ft}^2$, $\gamma_E = 20^\circ$, VM-8

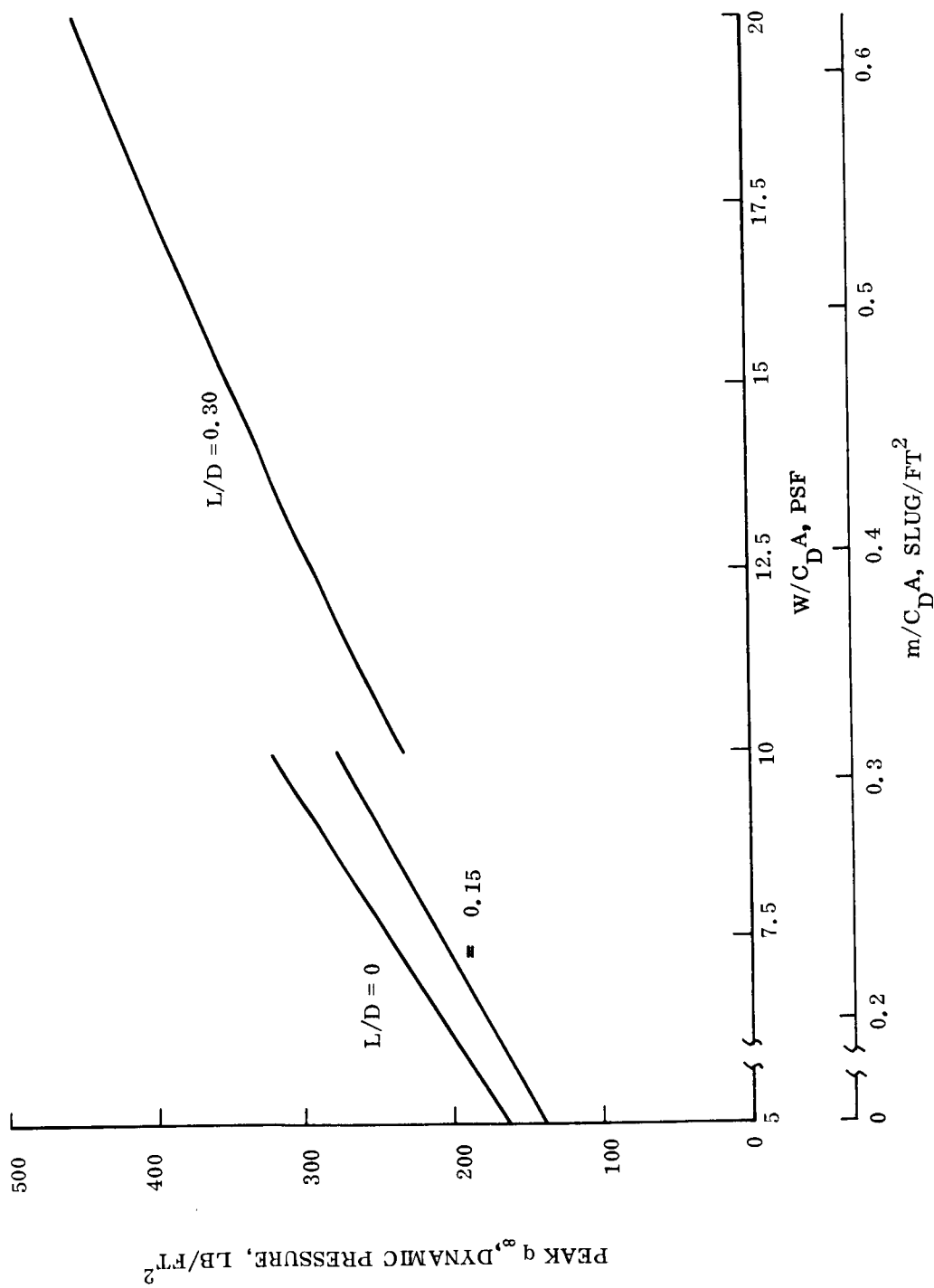


Figure 3.1.3-9. Peak Dynamic Pressure Map, $\gamma = 20^\circ$, $V_E = 20.8$ kfps, VM-8

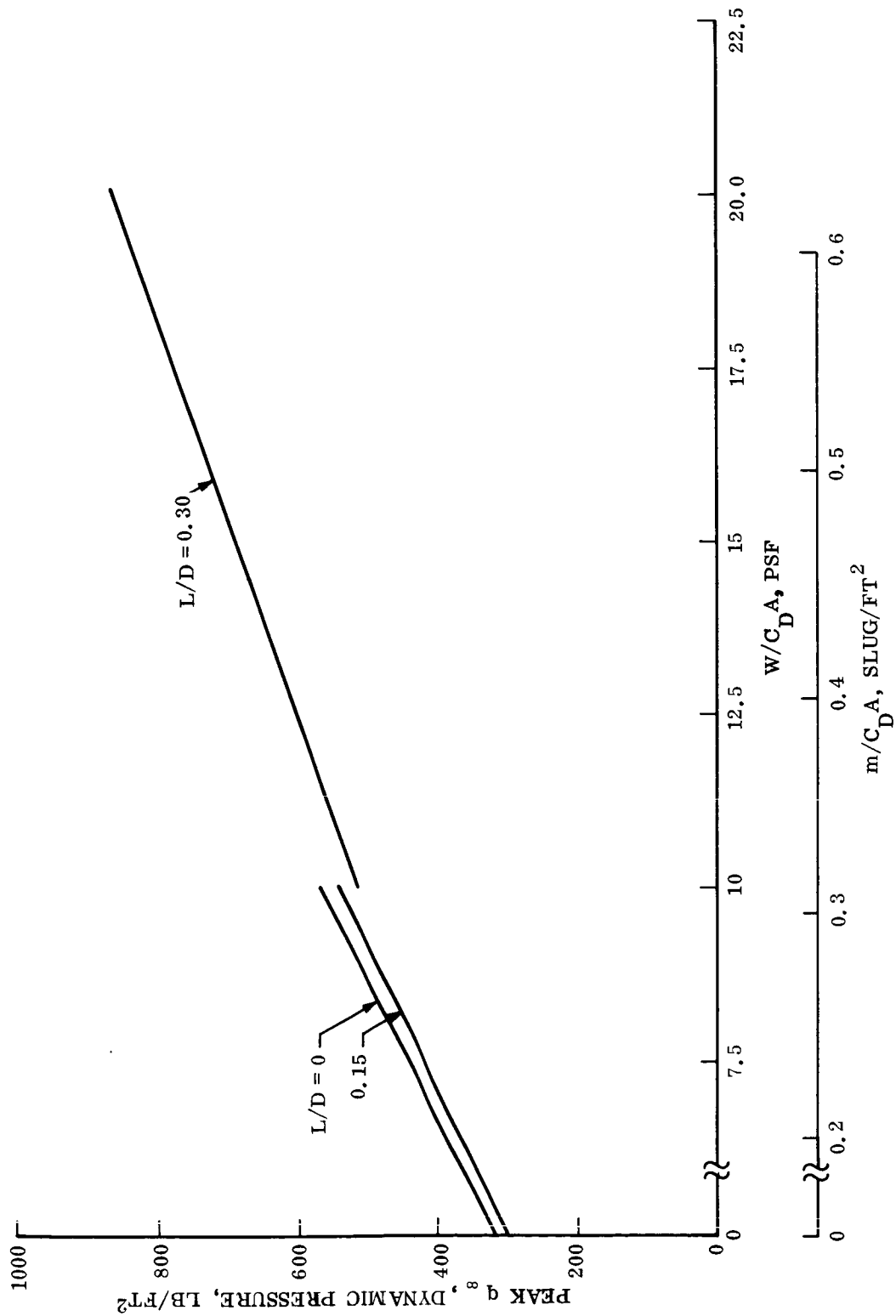


Figure 3.1.3-10. Peak Dynamic Pressure Map, $\gamma = 30^\circ$, $V_E = 20.8$ kfps, VM-8

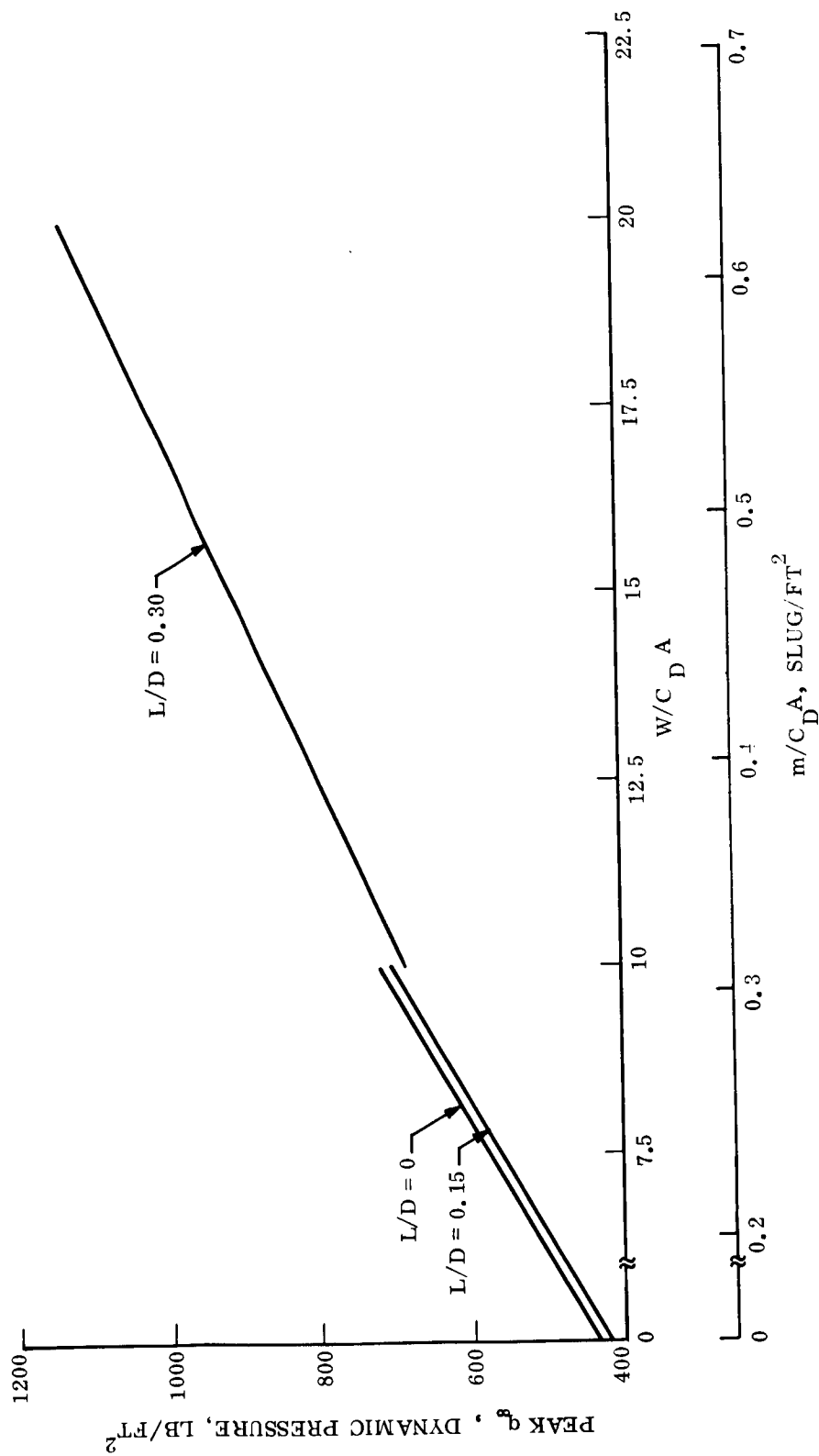


Figure 3.1.3-11. Peak Dynamic Pressure Map, $\gamma = 40^\circ$, $V_E = 20.8 \text{ kfps}$, VM-8

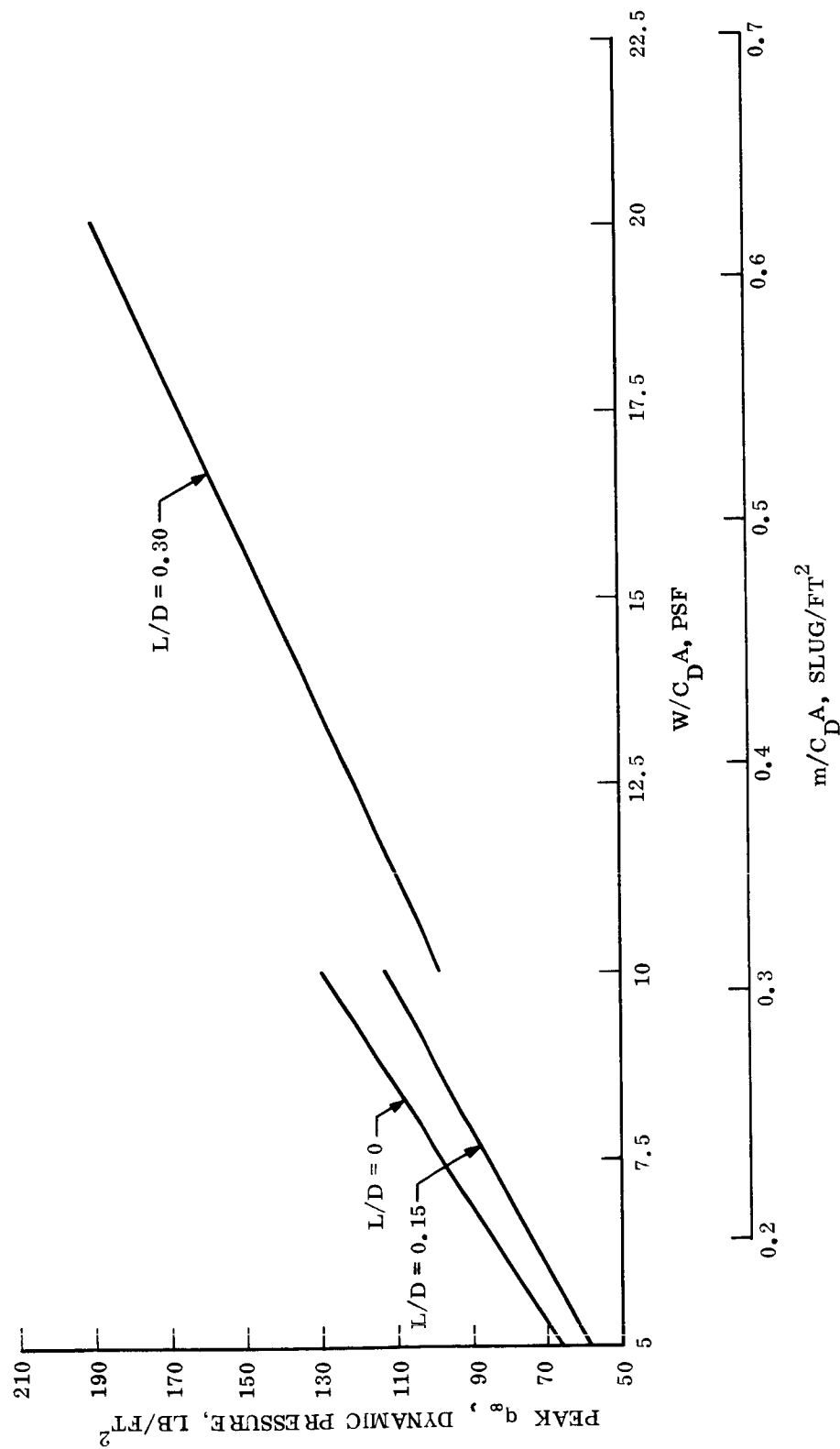


Figure 3.1.3-12. Peak Dynamic Pressure Map, $\gamma = 20^\circ$, $V_E = 20.8$, VM-3

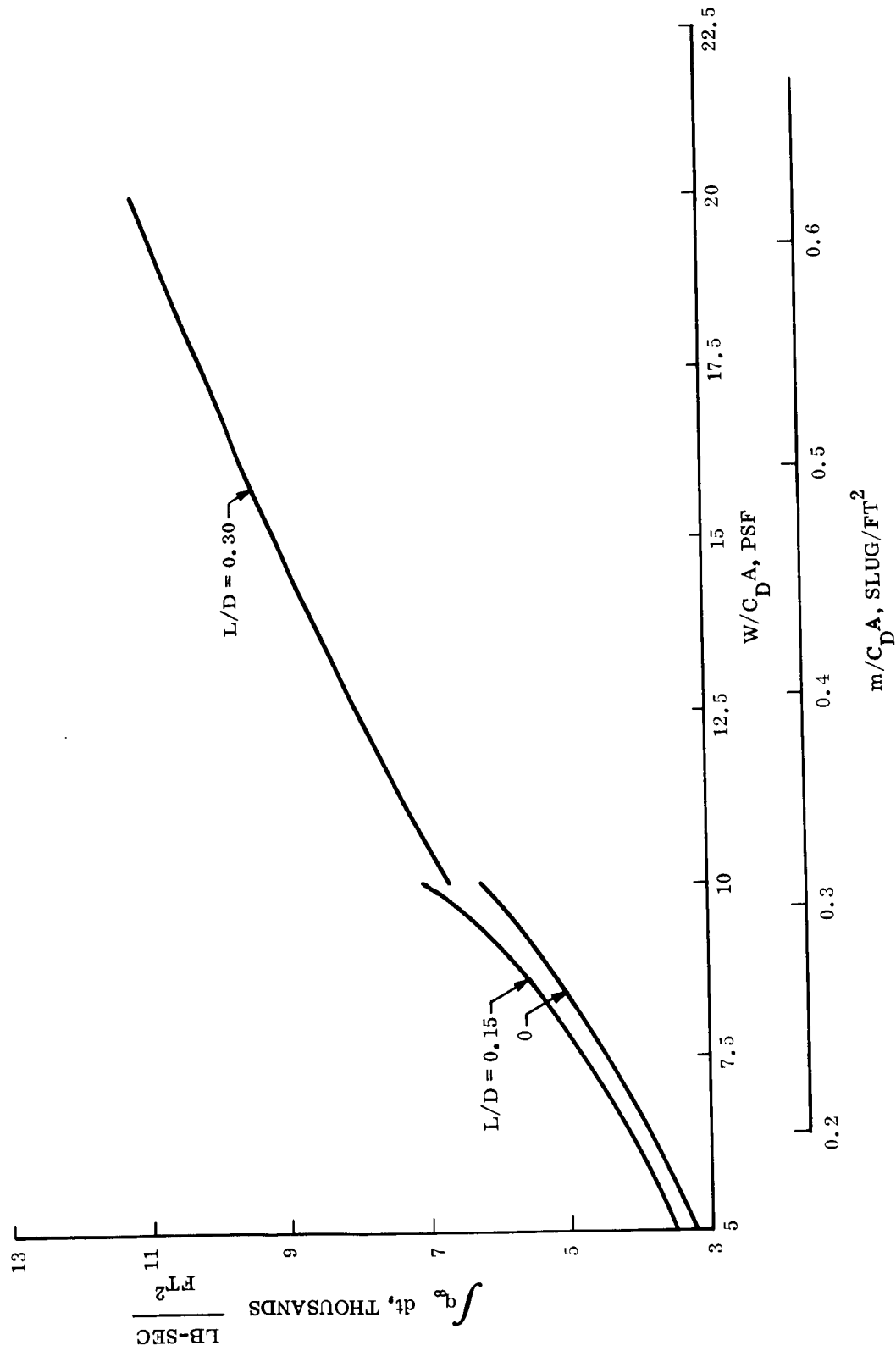


Figure 3.1.3-13. Integrated Dynamic Pressure Map, $V_E = 20.8$ kfps, VM-8

Point mass trajectory data for the limiting conditions are presented in figs. 3.1.3-14 through 17. The trajectories are at maximum ballistic coefficient for each L/D value, in the VM-8 atmosphere, for a 40° path angle.

3.1.3.3.3 Aerodynamic Performance and Stability

3.1.3.3.1 Force Coefficient Data

The lifting characteristics of the 60° sphere-cone aeroshell, sliced 60° sphere cones, and Apollo Command Module-shaped vehicles were generated with the corresponding requirements for zero moment. Fig. 3.1.3-18 gives the L/D variation with angle-of-attack for a 60° sphere-cone aeroshell with a bluntness ratio of 0.2 as obtained from ref. 3-54. The corresponding center-of-gravity location is indicated in fig. 3.1.3-19 and was generated using the methods of ref. 3-55. No change from these presented characteristics is expected for an increase in bluntness ratio to 0.5 or so.

Fig. 3.1.3-20 presents the variation of L/D with angle-of-attack for an Apollo Command Module shape which was obtained using the aerodynamic characteristics of ref. 3-55. The c. g. positions given in fig. 3.1.3-21 were obtained from refs. 3-54 and 3-55.

Fig. 3.1.3-22 presents the variation of L/D with slice angle for a 60° sphere cone at zero angle-of-attack. The required axial and radial c. g. position for a zero trim angle are given in fig. 3.1.3-23 for a slice angle of 73°. These results were generated from refs. 3-54 and 3-56. The 73° slice shape was chosen as the raked-cone-cylinder for further analysis since it yields an L/D just above 0.31.

The L/D value for each of these three shapes was selected at approximately 0.30, allowing for some viscous effects and analytic simplicity by choosing the next higher even values of pitch angle or rake angle to arrive at the comparative performance numbers presented in table 3.1.3-1.

It should be noted that the c. g. positions are those at which the vehicle will have no moments acting and hence is in aerodynamic trim. Such positions do not indicate the static stability of the vehicle, which is discussed next.

3.1.3.3.2 Stability Criteria

The general equation used to determine the c. g. positions required for trim at some given angle-of-attack is

$$C_{m_{cg}} = C_{m_{NOSE}} + C_N \frac{X}{D} - C_A \frac{Z}{D} \quad (15)$$

where the C_m 's are non-dimensional moments with the 'c. g.' subscript denoting a moment reference at the c. g., and 'nose' at the vehicle apex. X/D is measured axially from the vehicle apex, and Z/D vertically, plus aft and down, respectively. The longitudinal stability parameter was obtained by differentiating this equation with respect to C_N and evaluating the derivative at α_{TRIM} :

$$\left(\frac{\partial C_{mcg}}{\partial C_N} \right)_{\alpha_T} \cong \left(\frac{C_{m_{NOSE}}}{C_N} \right)_{\alpha_T} + \left(\frac{X}{D} \right)_{\alpha_T} - \left(\frac{\partial C_A}{\partial C_N} \frac{Z}{D} \right)_{\alpha_T} \quad (16)$$

A more complete analysis than presented here would include the effects of:

$$\left(\frac{\partial C_{mcg}}{\partial C_A} \right)_{\alpha_T} \& \left(\frac{\partial C_{mcg}}{\partial C_R} \right)_{\alpha_T} \quad (17)$$

for better definition of the vehicle center of pressure.

Figs. 3.1.3-24 through -26 present the variation in longitudinal stability parameter ("neutral point") with c. g. position. Both the 60° sphere-cone aeroshell and Apollo Command Module shapes appear to have adequate longitudinal stability near the forward c. g. limits but the stability decreases rapidly as the c. g. offset distance approaches zero. The raked cone-cylinder shape ($\delta = 73^\circ$) has a positive longitudinal stability value at the forward c. g. limit of only 1 percent which is substantially lower than that for the other shapes.

TABLE 3.1.3-1. COMPARATIVE LIFTING PERFORMANCE

Shape	Cone Semi-Vertex Angle θ , deg.	Pitch Angle-of-Attack α , deg.	Lift-to-Drag Ratio L/D
Aeroshell	60	25	0.310
Apollo Command Module	33	20	0.317
Raked Cone-Cyl.	60	0*	0.315
*Rake Angle = 73°			

3.1.3.4 Control Requirements

3.1.3.4.1 Roll Torques

The two principal sources of roll torques are the product asymmetry $C_2 \alpha_T$ and the pure roll torque coefficient C_1 . For non-lifting vehicles, the pure roll torque (caused primarily by ablation characteristics and therefore very uncertain in value and/or consistency) dominates since the C_2 is an uncertainty value and is therefore small; however, on lifting vehicles, the angle-of-attack is achieved by offsetting the center of mass and C_2 (center of mass offset) can be quite large.

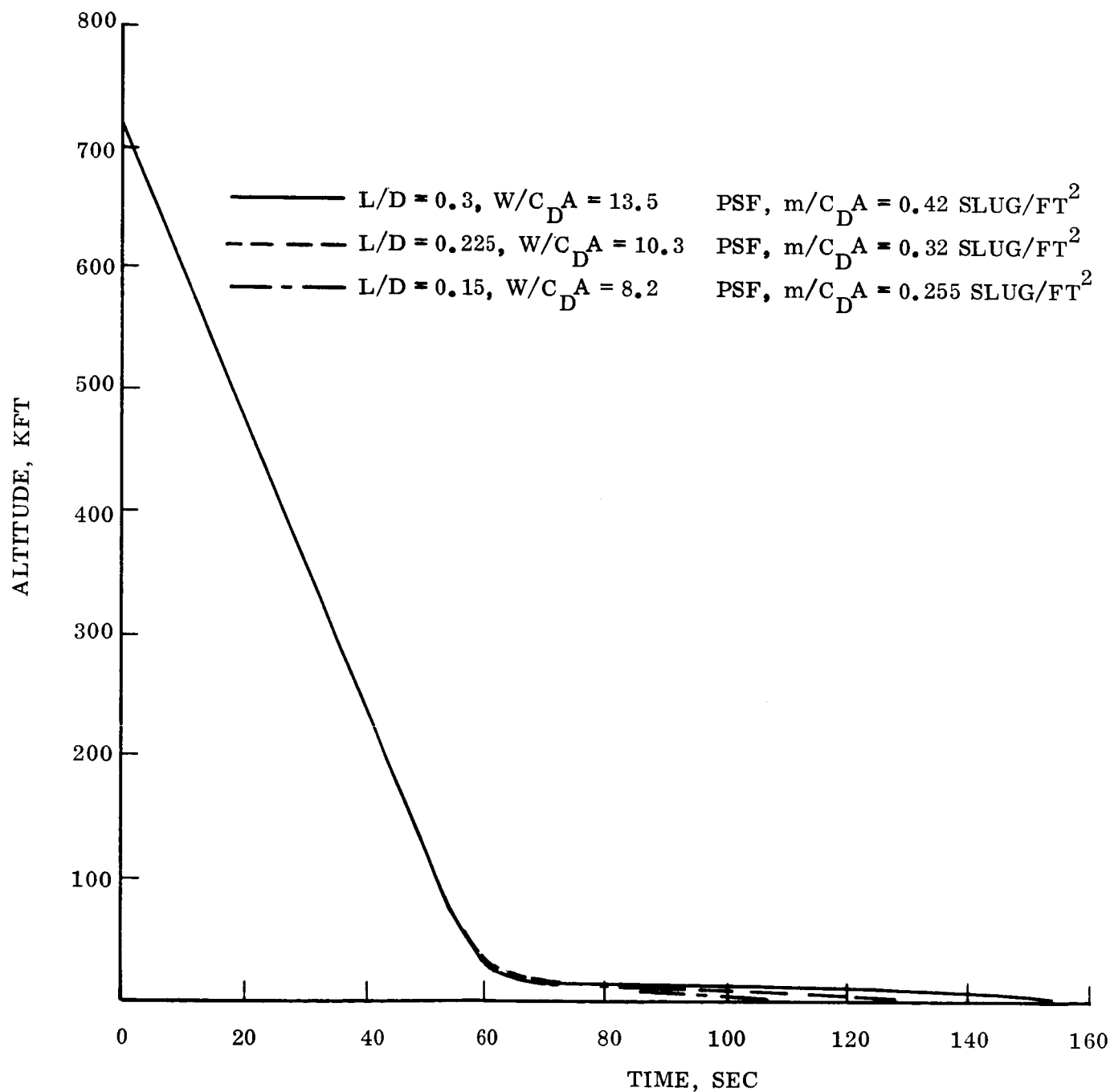


Figure 3.1.3-14. Altitude Histories for Small L/D , $\gamma_E = 40^\circ$, VM-8

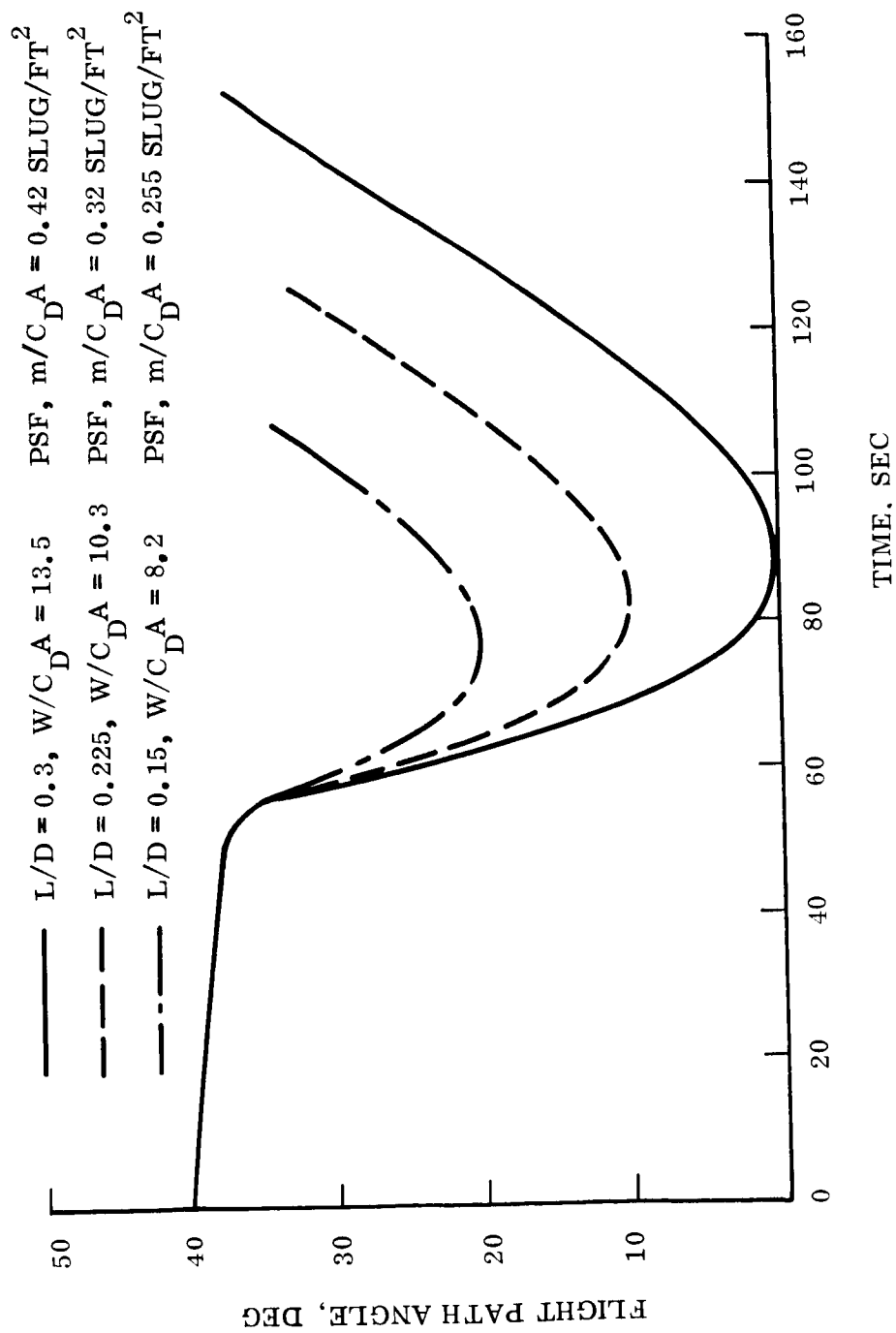


Figure 3.1.3-15. Flight Path Histories for Small L/D , $\gamma = 40^\circ$, VM-8

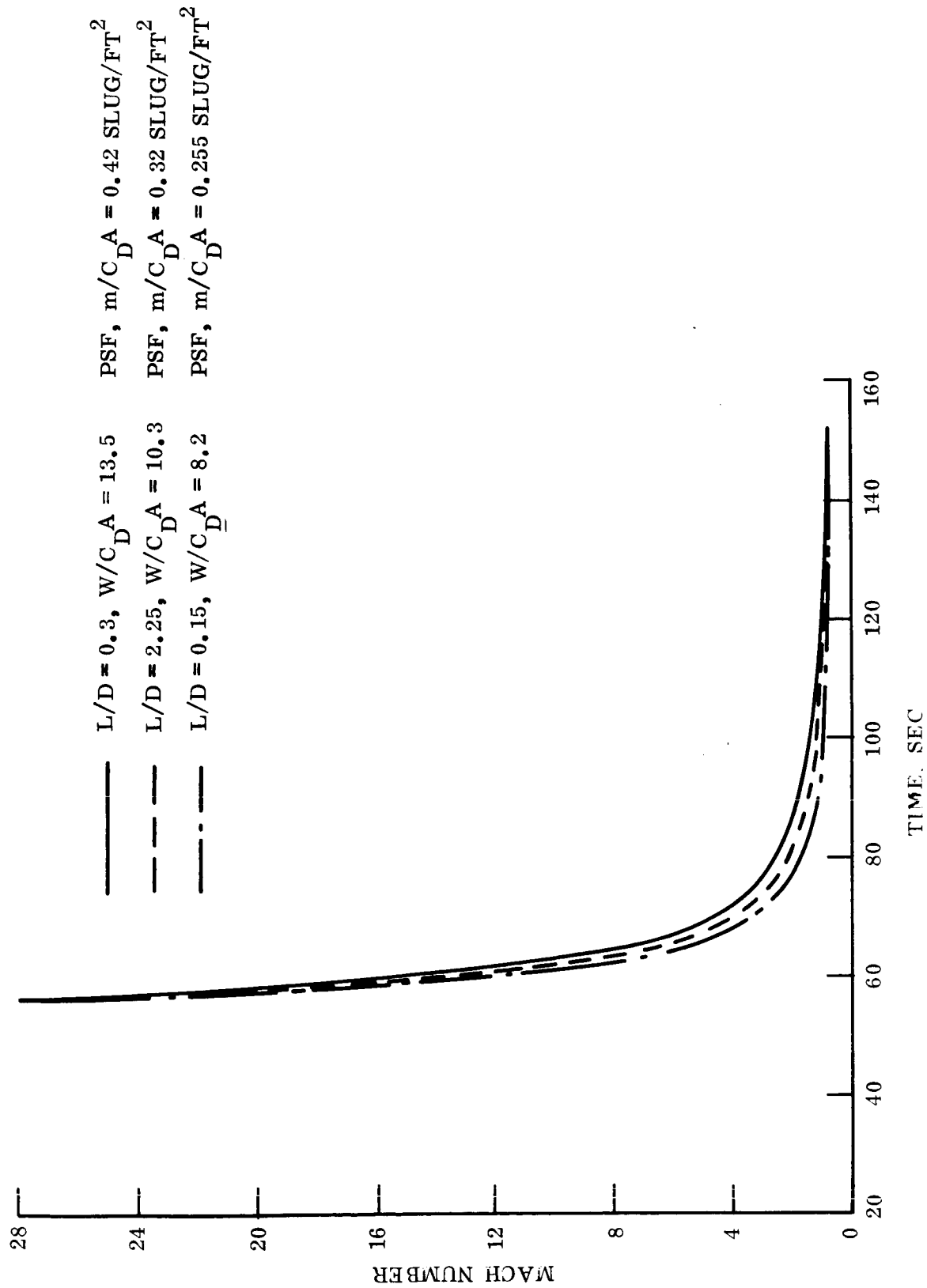


Figure 3.1.3-16. Mach Number Histories for L/D , $V_E = 40^\circ$, VM-8

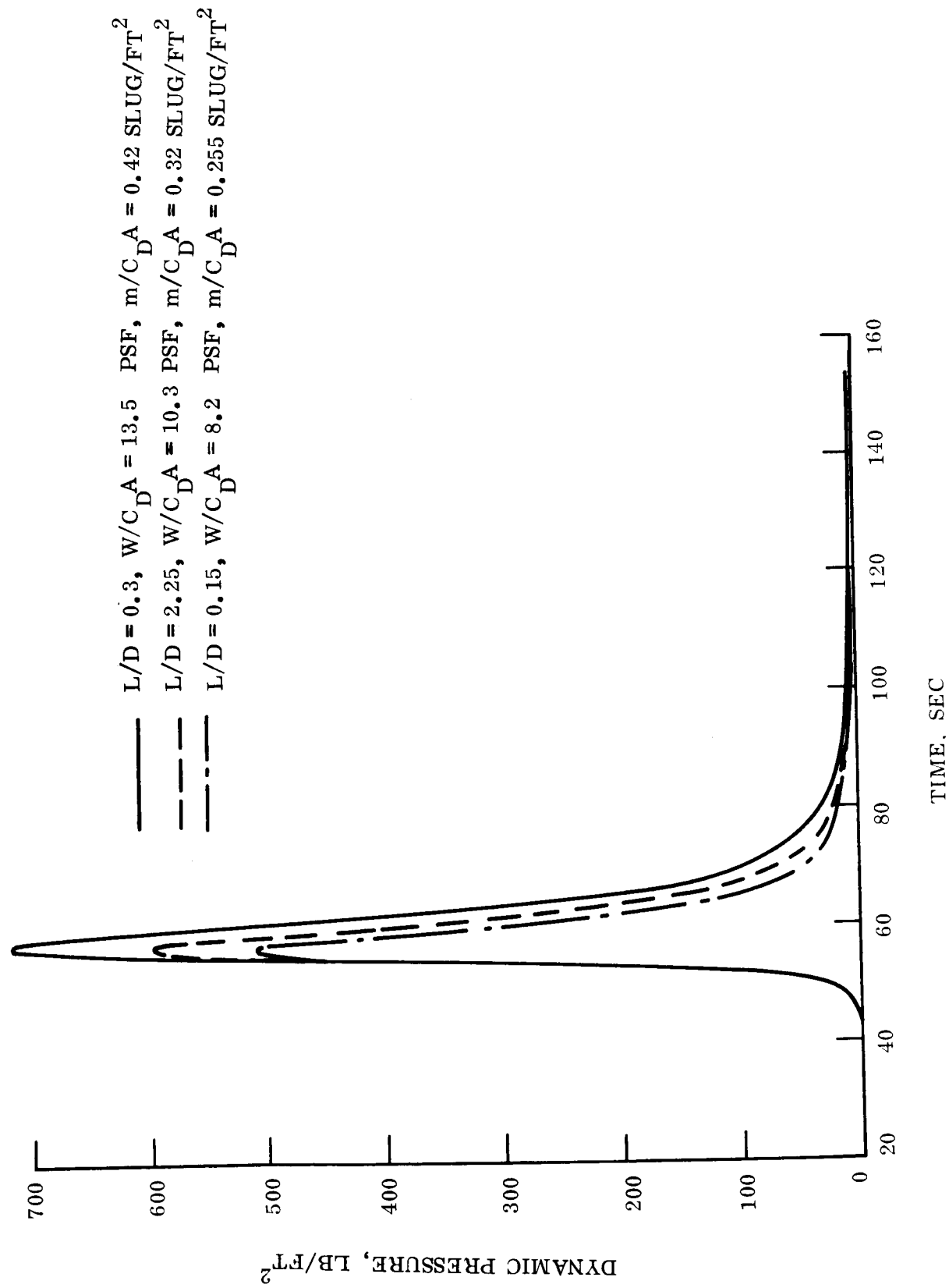


Figure 3.1.3-17. Dynamic Pressure Histories for Small L/D , $\gamma = 40^\circ$, VM-8

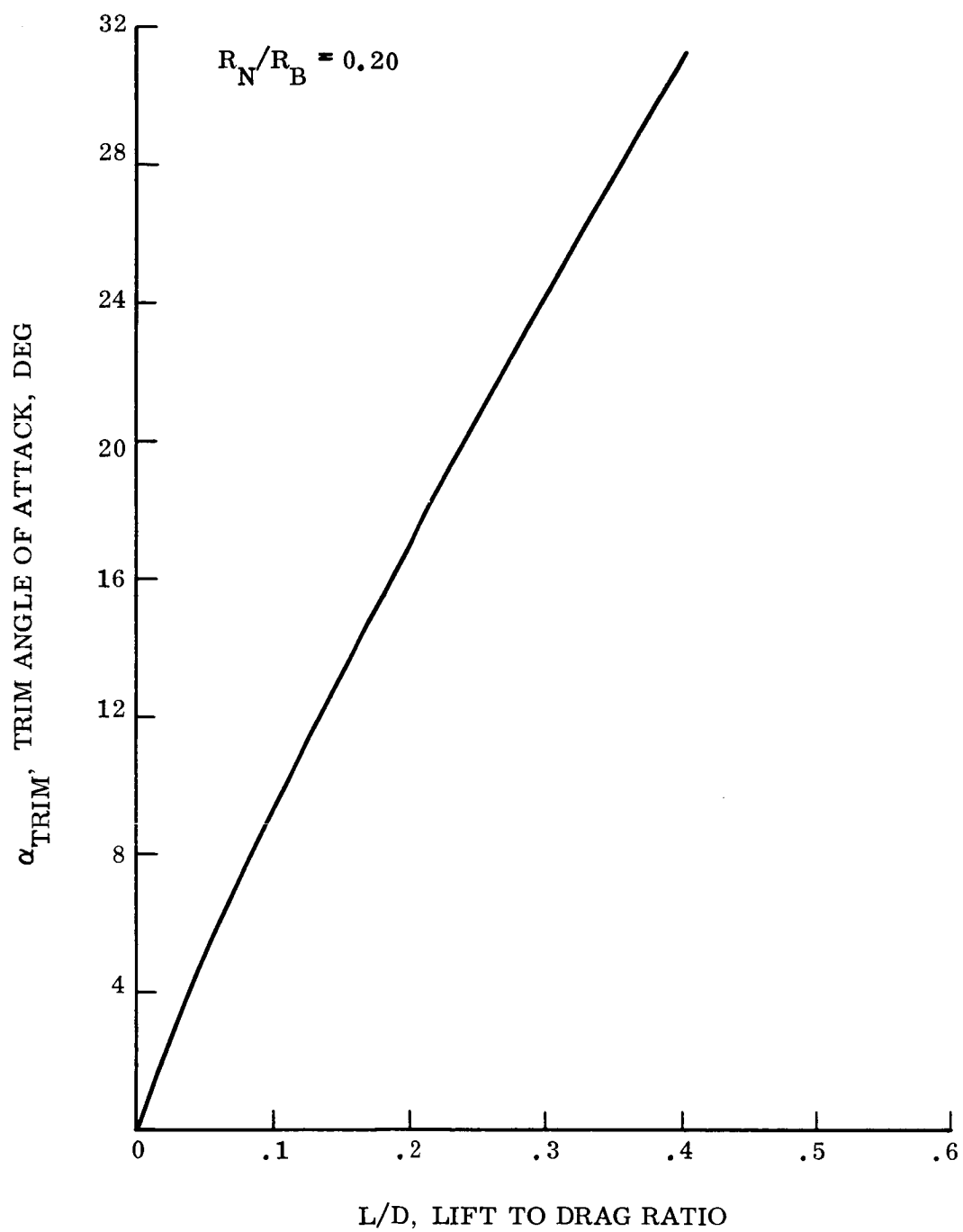


Figure 3.1.3-18. Lift/Drag Ratio Variation with Trim Angle-of-Attack, 60° , 0.2 Blunted Cone

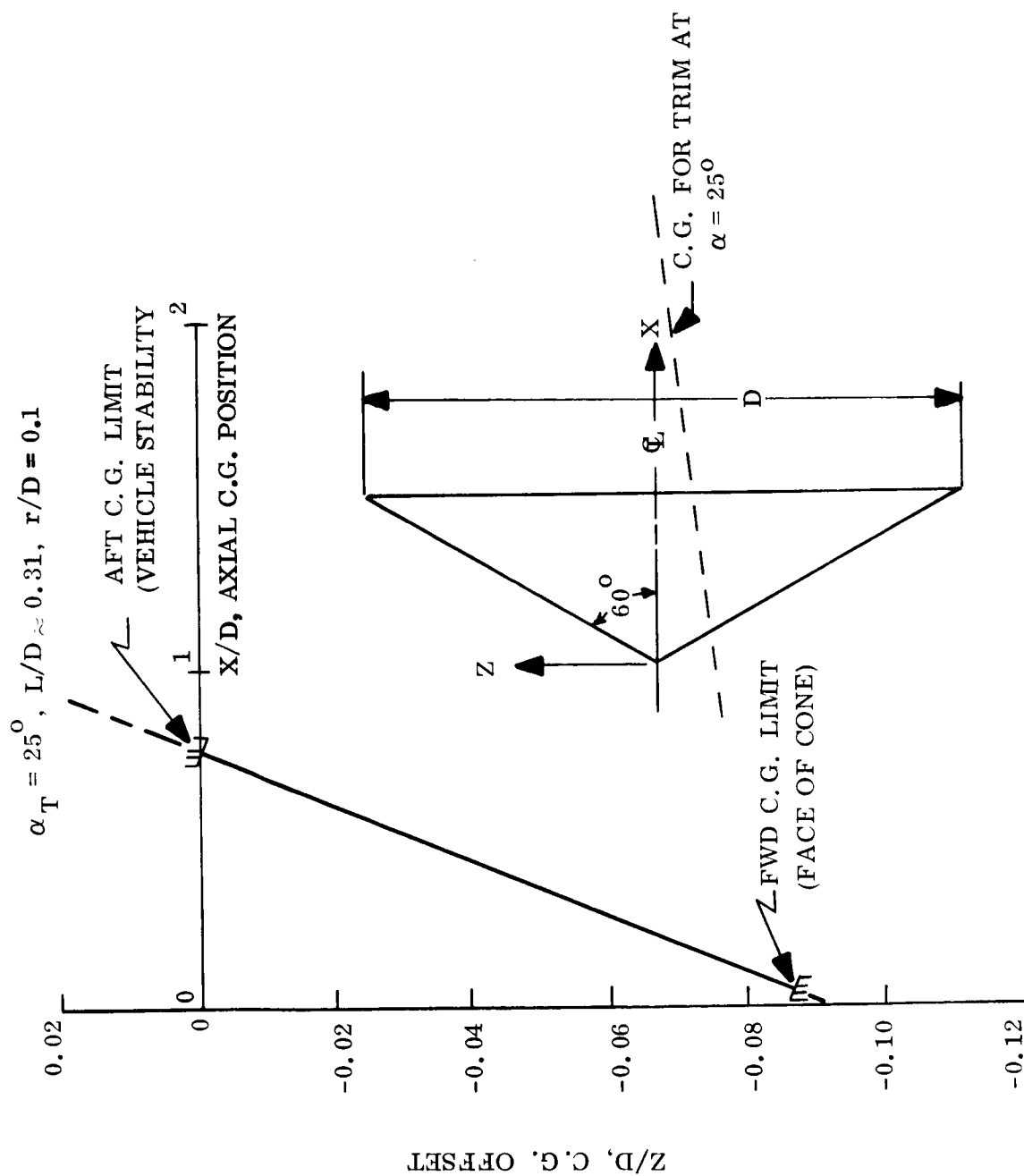


Figure 3.1.3-19. C.G. Offset Variation with Axial c.g. Position,
 $60^\circ, 0.2$ Blunted Cone

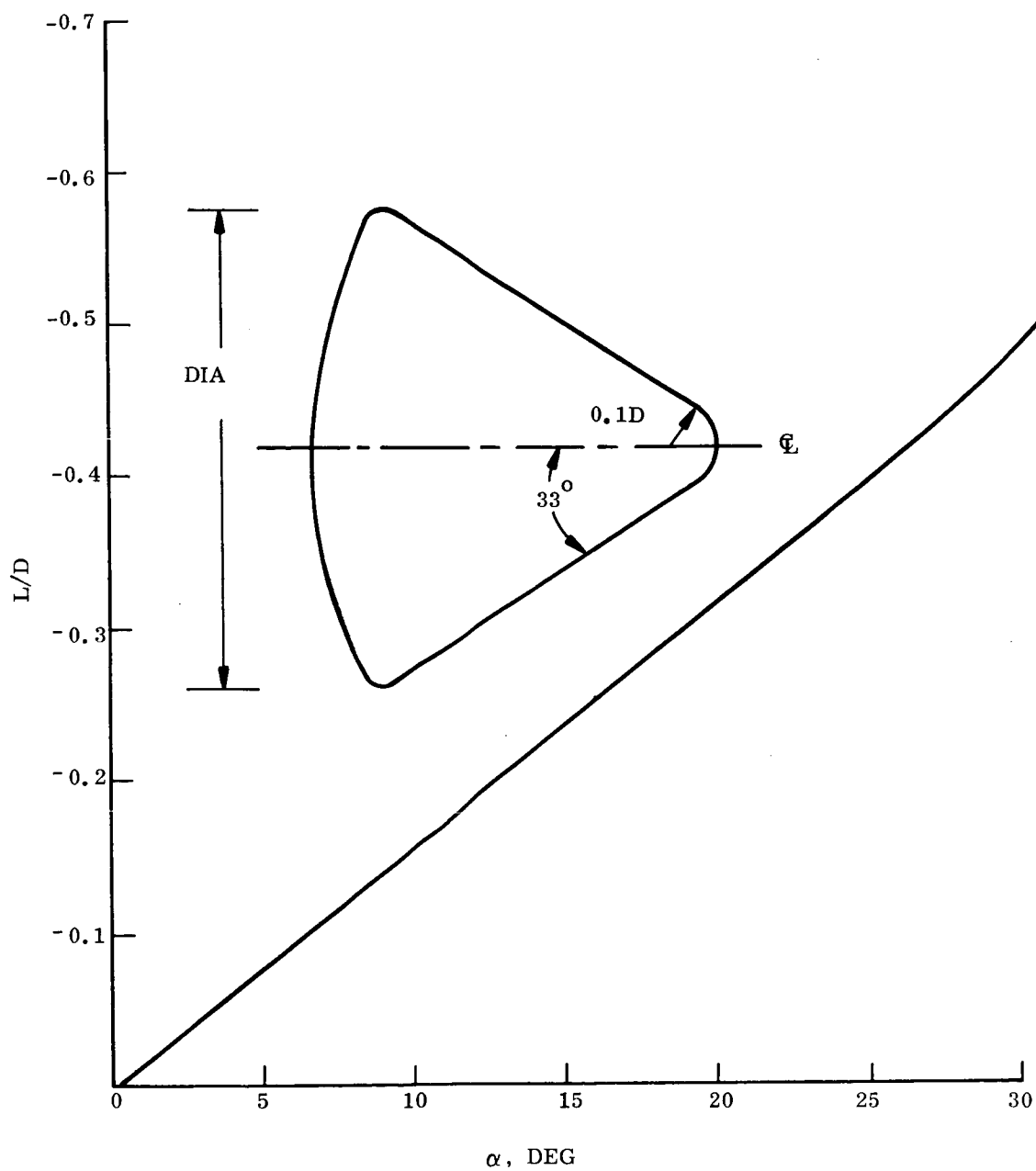


Figure 3.1.3-20. L/D vs Angle-of-Attack, Apollo CM - Type Vehicle

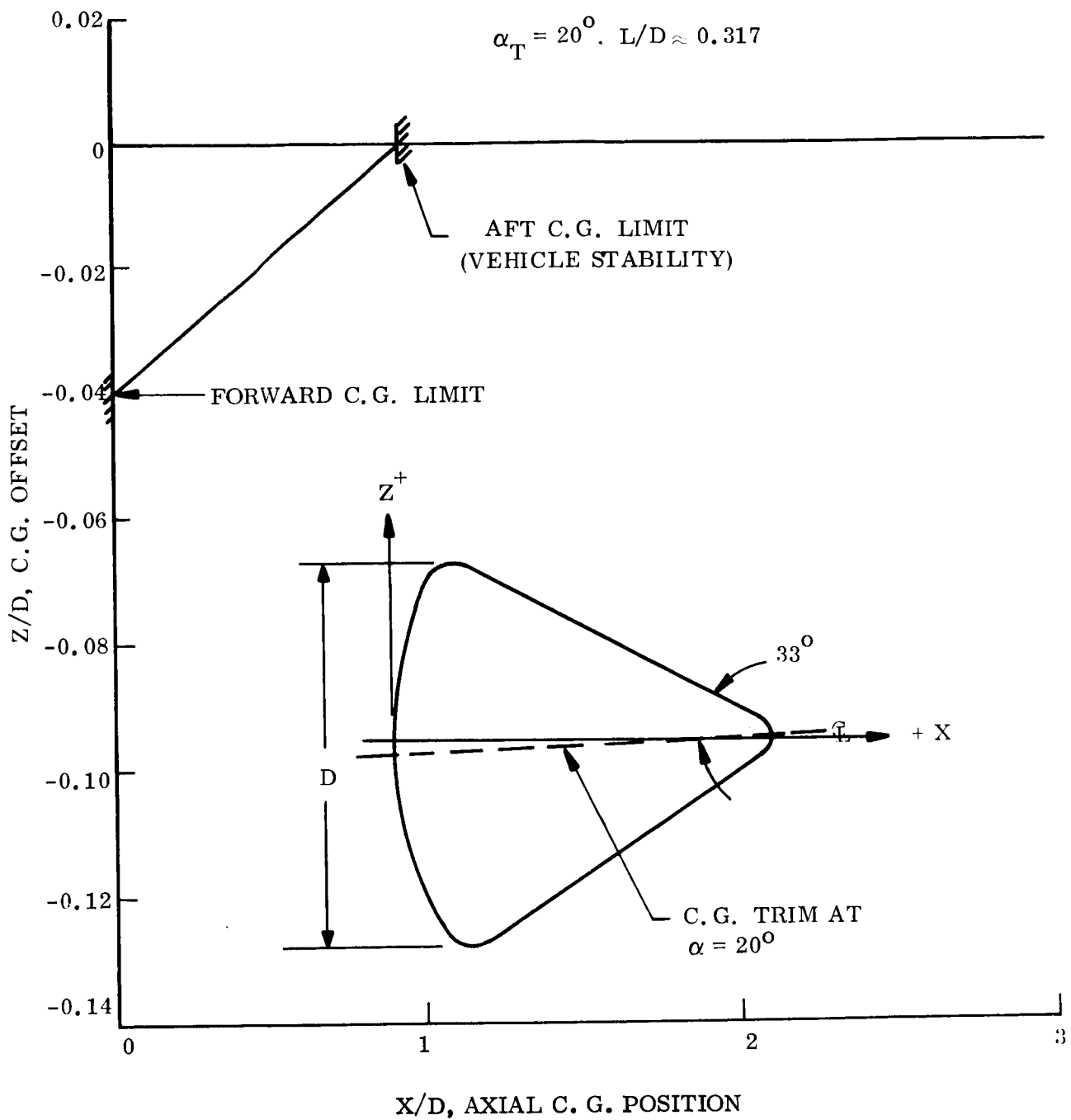


Figure 3.1.3-21. C.G. Offset Variation with Axial c.g. Position,
Apollo C.M. Shape

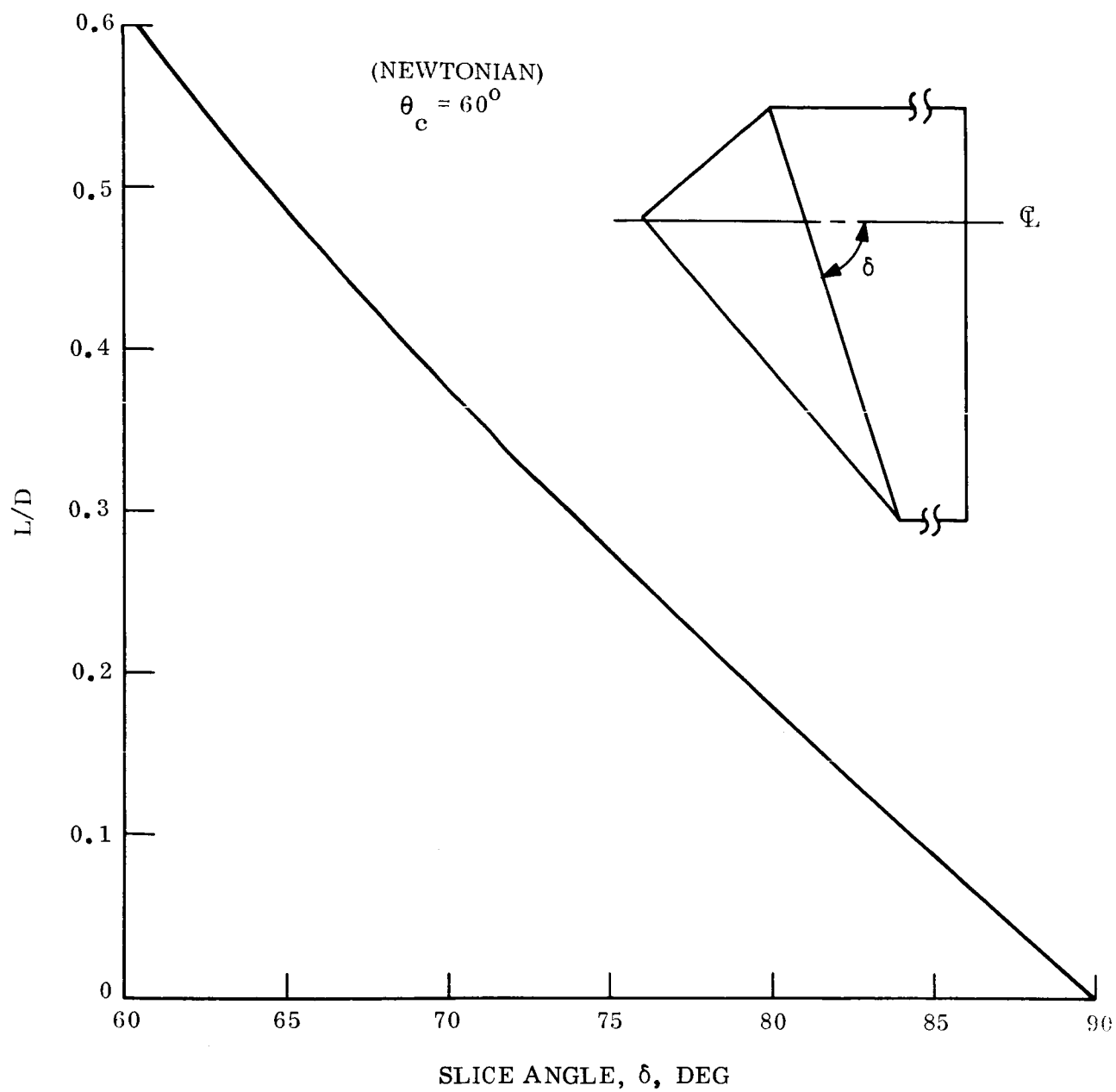


Figure 3.1.3-22. L/D Variation with Rake Angle, 60° Raked Cone Cylinder

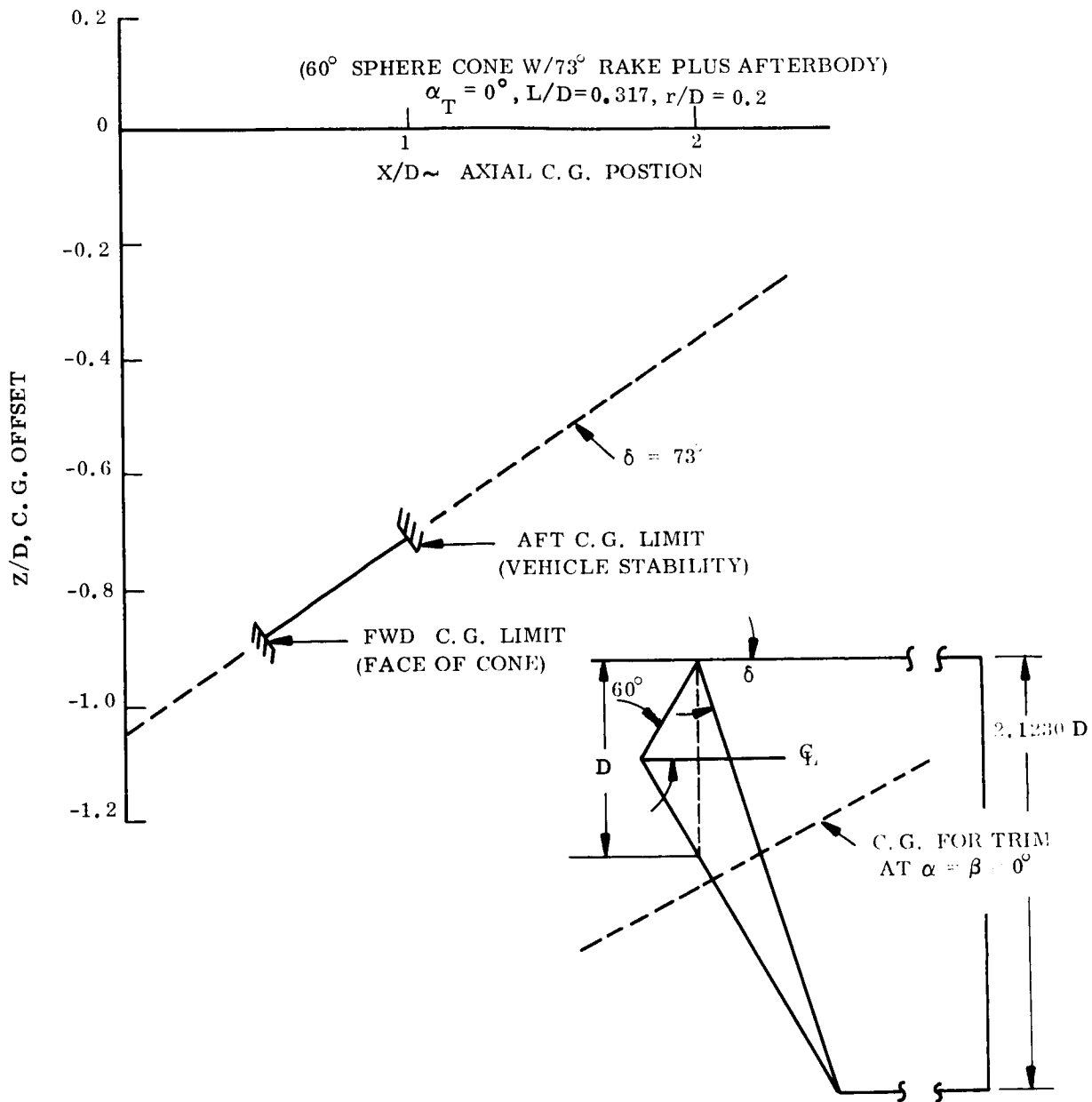


Figure 3.1.3-23. C.G. Offset Variation with c.g. Position, Raked Cone Cylinder

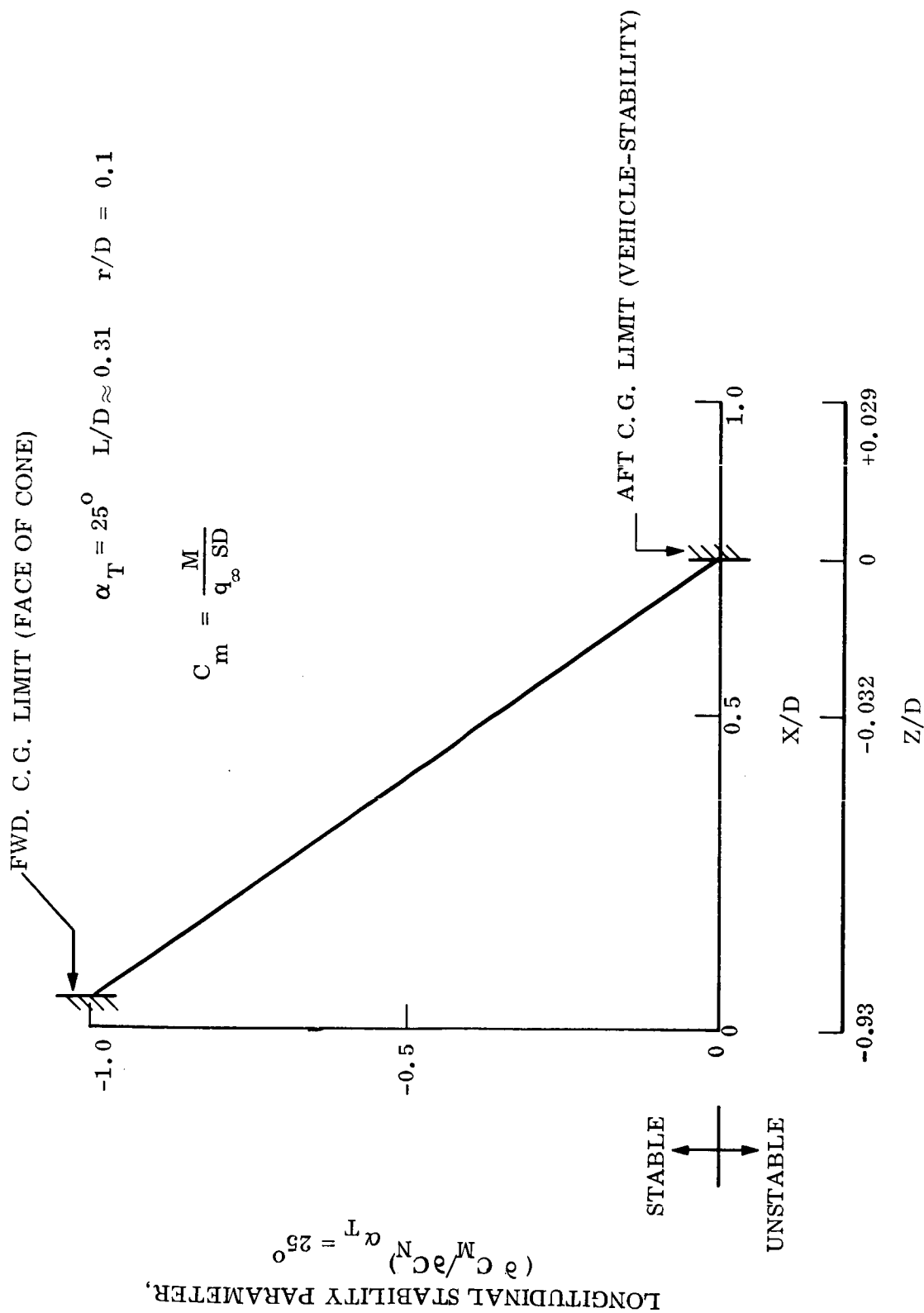


Figure 3.1.3-24. Longitudinal Stability Parameter Variation with c.g. Location, 60° Sphere Cone

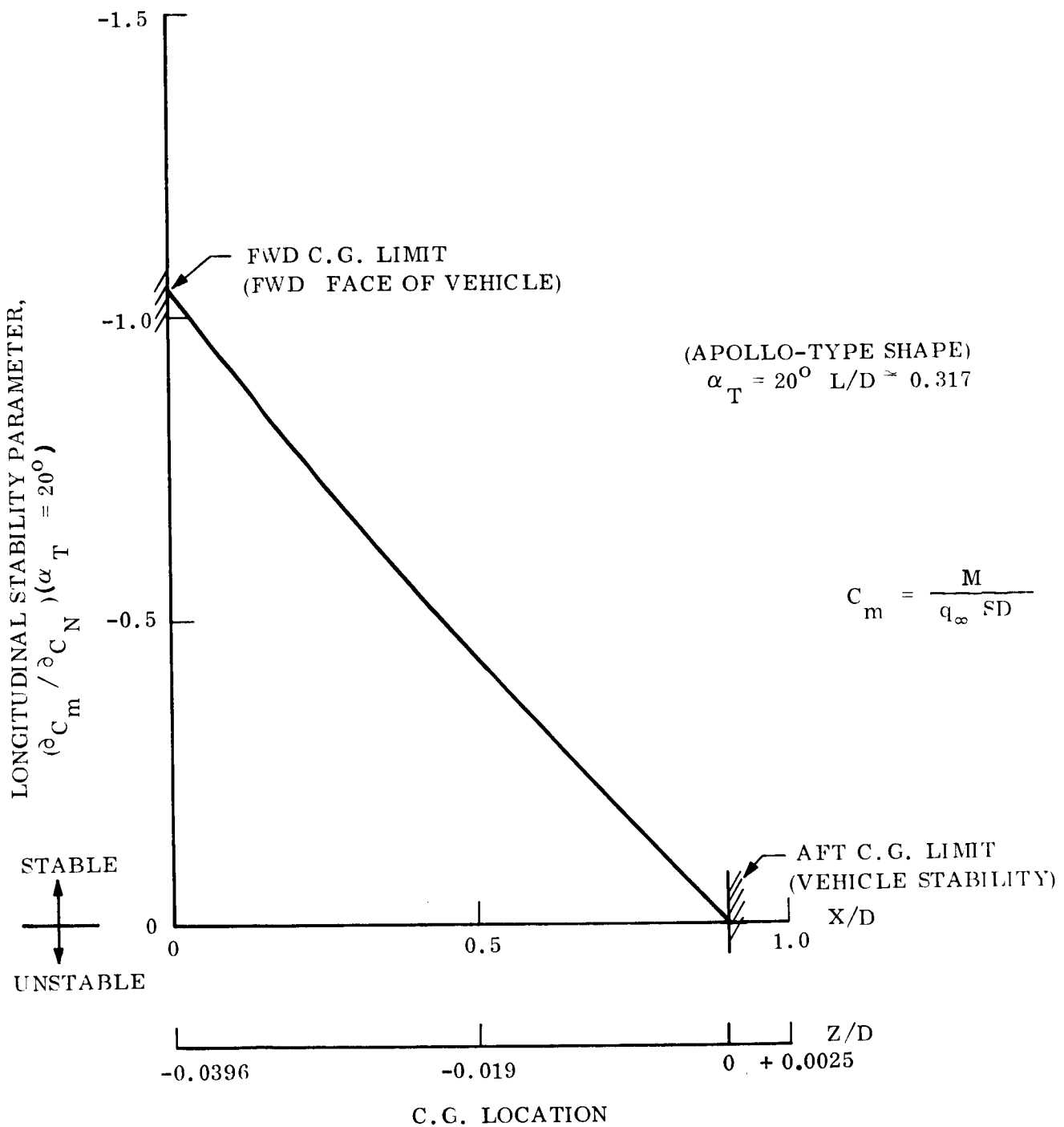


Figure 3.1.3-25. Longitudinal Stability Parameter Variation
 with c.g. Location, Apollo CM Shape

#6

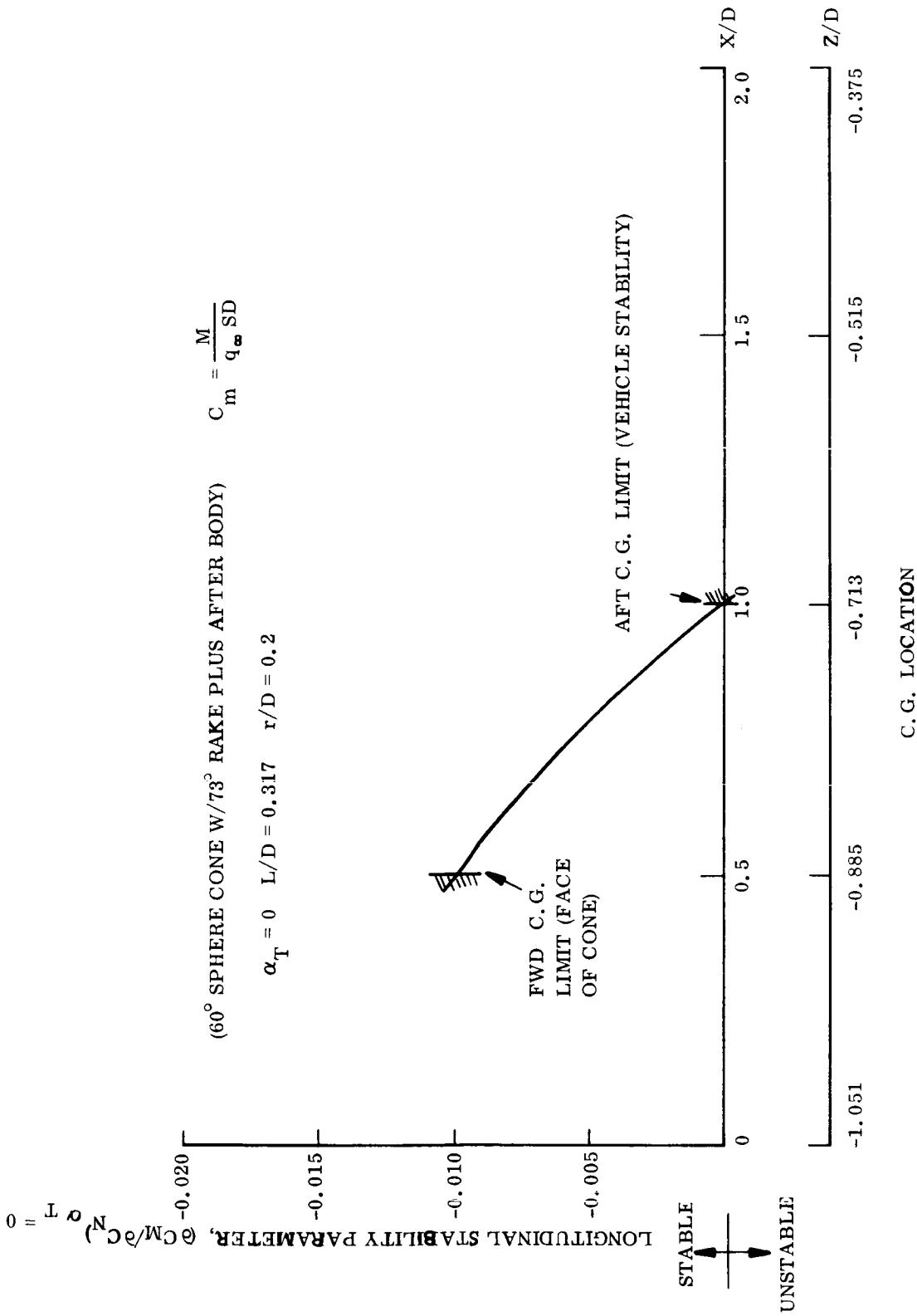


Figure 3.1.3-26. Longitudinal Stability Parameter Variation with c.g. Location, Raked Cone Cylinder

The roll torques and integrated roll torques for lifting vehicles are based on a 100 inch diameter and on the following reference conditions for direct entry:

$$V = 20,800 \text{ fps}$$

$$\gamma_e = 30 \pm 10^\circ$$

$$h = 722,000 \text{ feet (220 km)}$$

The roll torques due to $C_2 \alpha_T$ are from the expression: $\tau_1 = C_{N\alpha} \alpha_T C_2 qS$. Assuming $\alpha_T = 1^\circ$, $C_2 = 5 \text{ in.}$ (for $L/D = 0.30$), and $C_{N\alpha} = 0.007 \text{ per deg}$, the roll torque becomes, $\tau_2 = 0.0023 qd^2$. Similarly, the pure roll torque which might be developed via ablation asymmetries is: $\tau_2 = 0.00008 qd^3$. Based on the 100 in. diameter reference designs,

$$\begin{aligned} \tau &= \tau_1 + \tau_2 = [0.0023 (69.4) + 0.00008 (577)] q \\ &= 0.194 q \end{aligned} \quad (18)$$

The comparable roll torque expressions for (L/D) 's of zero and 0.15 are $\tau = 0.050 q$ and $(\tau) 0.15 = 0.118 q$, respectively. These calculations are all for the 60° sphere-cone aeroshell vehicle.

For the Apollo Command Module shape, the $C_{N\alpha}$ is 0.0028 per deg and C_2 becomes 3 in. (for an L/D of 0.3). The resulting roll torque expressions are $\tau = 0.04 q$, $\tau = 0.056 q$, and $\tau = 0.071 q$, for L/D values of 0, 0.15, and 0.30, respectively.

For both the Apollo and aeroshell configuration, L/D values of 0, 0.15, and 0.30 correspond to $W/C_D A$ values of 6.3, 8.2 and 13.5 psf (0.197, 0.255, and 0.420 slug/ft²), respectively. The roll torques determined from the foregoing expressions applied to the appropriate flight paths are presented in fig. 3.1.3-27.

3.1.3.4.2 Attitude

The lifting configurations must be roll attitude controlled to insure that the lift vector is upward and approximately in the trajectory plane. The vehicles are designed with a specified center-of-mass offset to passively provide the desired trim angle-of-attack necessary to achieve the design L/D ratio.

The firm requirement on allowable roll error is a trade-off item to be weighed against the L/D ratio, ballistic coefficient, and trajectory path angle. Figs. 3.1.3-28 and -29 present data for this trade-off. The first of these curves shows the variation in altitude at Mach 2 as a function of roll attitude error for the limit ballistic coefficients for L/D 's of 0.15, 0.225, and 0.30. The second figure shows the variation in flight path angle at Mach 2 for the same variables. Both of these parameters affect the decelerator deployment requirement and performance. The variation in flight path angle at Mach 2 is probably slight enough to be ignored in a preliminary analysis even for very large roll angle errors (30° or more).

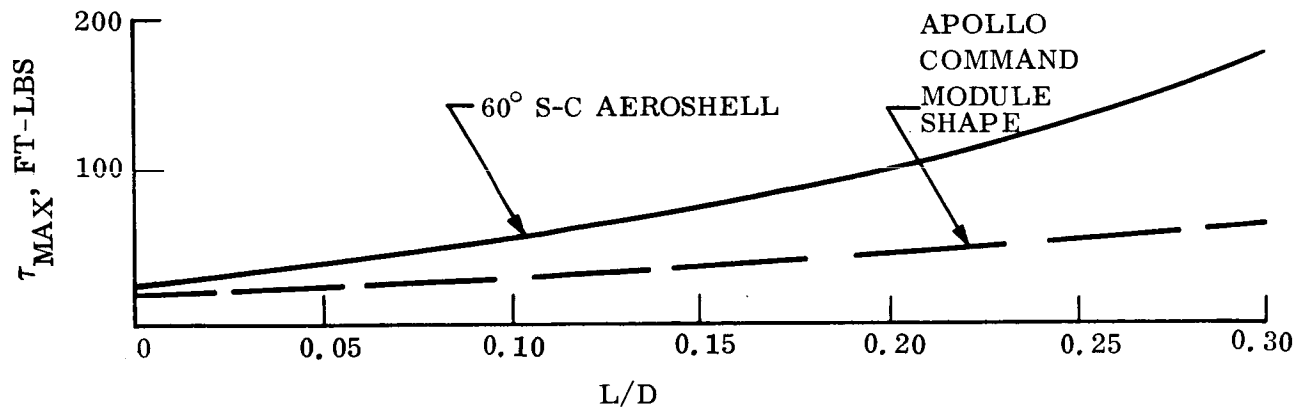
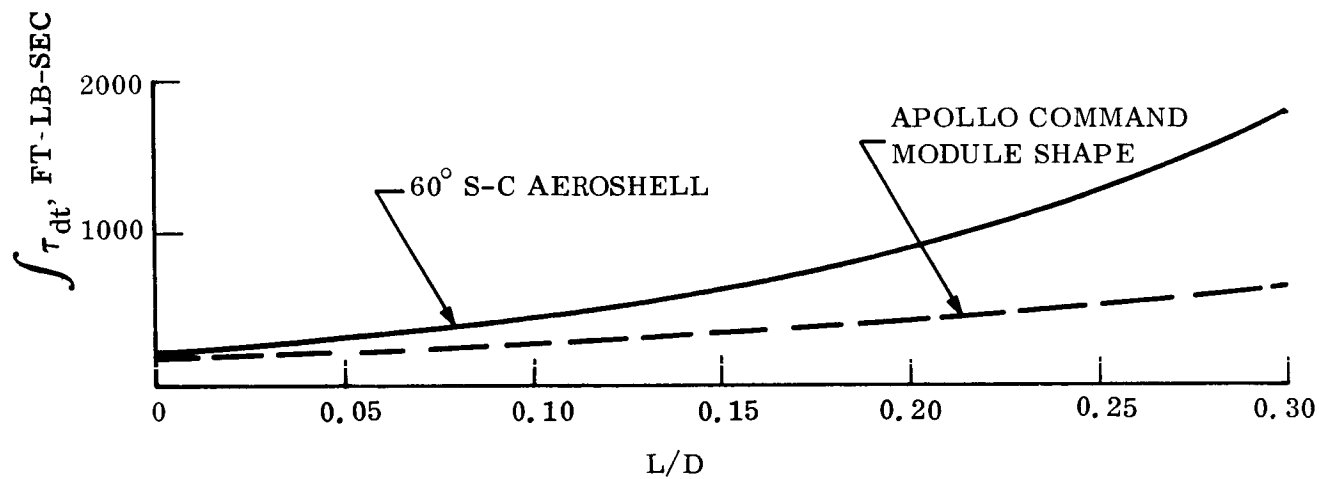
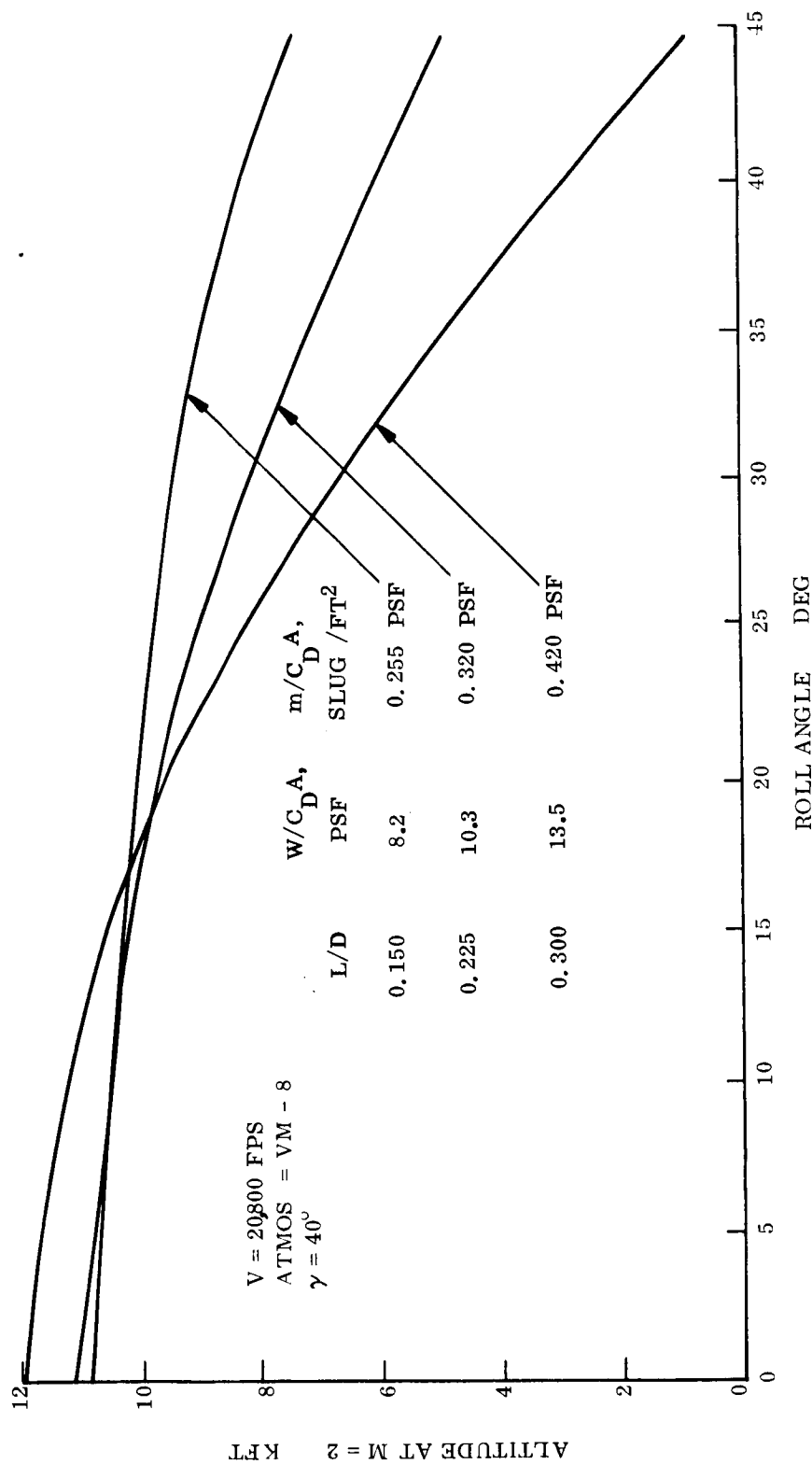


Figure 3.1.3-27. Maximum and Integrated Roll Torques

Figure 3.1.3-28. Altitude at Mach 2 vs Roll Angle for Selected Small L/D 's

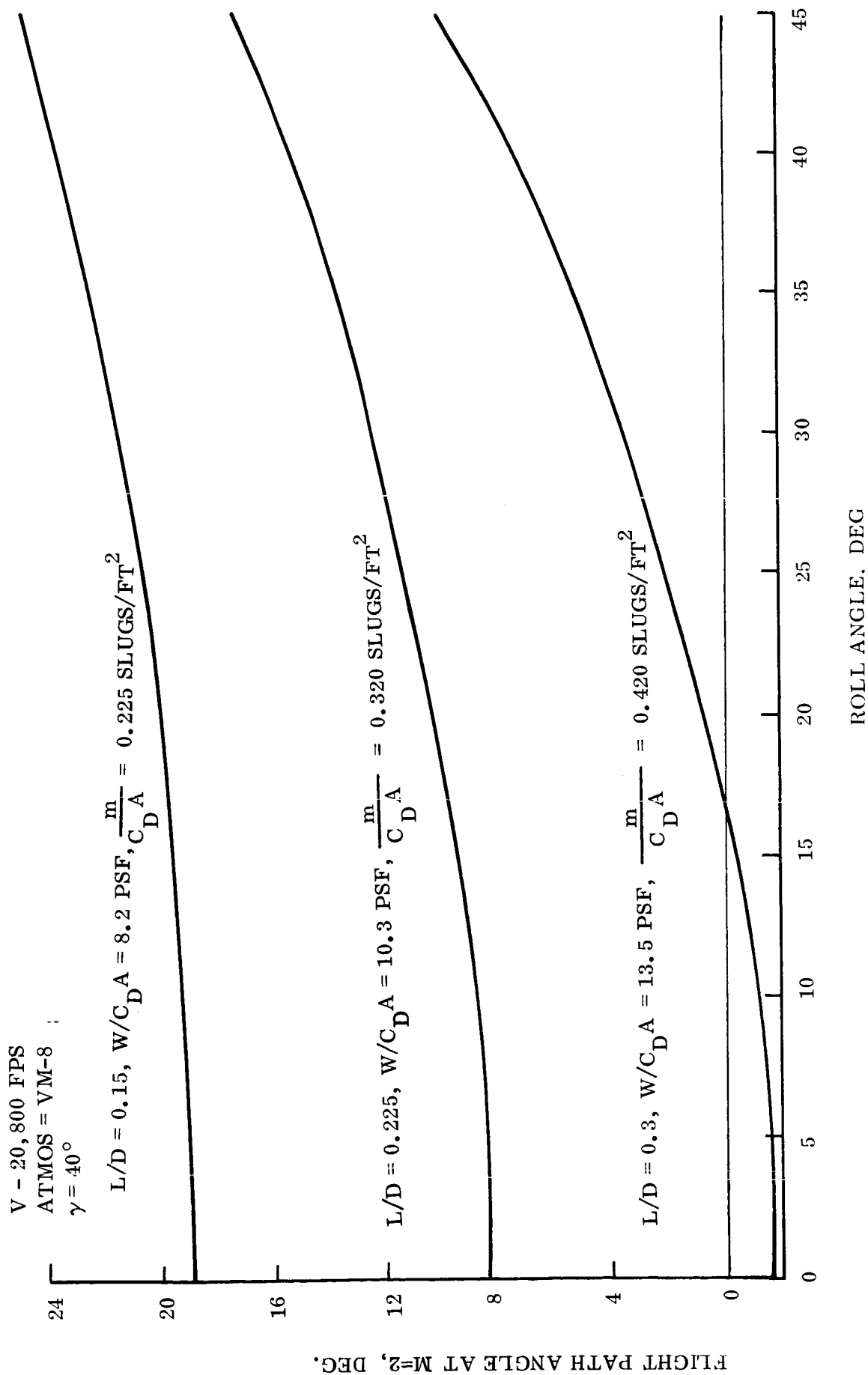


Figure 3.1.3-29. Flight Path Angle at Mach 2 vs Roll Angle for Selected Small L/D's

The loss in altitude at Mach 2 for a given roll angle is more pronounced on the higher L/D trajectories. Similarly, the trajectory shown ($\gamma_e = 40^\circ$) is the one for which roll errors have the biggest effect. For L/D = 0.3, a roll angle error of 10° to 15° can be tolerated at little cost in maximum ballistic coefficient. Similarly for an L/D = 0.15, a roll angle error of 20° to 25° could be tolerated with little loss in maximum $W/C_D A$.

These curves were generated using a constant roll angle error throughout the trajectory so if a control system were to develop such roll errors during the flight, the losses would be proportionately less.

3.1.3.5 Heat Shield Performance

A study has been made to determine parametrically the total heat shield requirements at several lifting entry conditions for the sphere-cone aeroshell. The entry conditions were:

$$V_e = 20,800 \text{ ft sec}$$

$$\gamma_e = 20^\circ$$

$$VM-3 = \text{atmosphere}$$

The vehicle has a 100 in. base diameter, bluntness ratio $R_N/R_B = 0.5$, and a cone semi-vertex angle $\theta_c = 60^\circ$. Three lifting entry conditions were considered: L/D = 0, 0.15, and 0.3, with corresponding ballistic coefficients of 7.5, 10, and 15 lb/ft² (0.233, 0.312, and 0.467 slug/ft²), respectively.

The heat flux histories developed for these lifting entry vehicles, using the techniques described in para 3.1.2.1, are quite similar to those for ballistic entry. Total heat loads for three body locations, stagnation point, sphere-cone tangency point, and end of skirt, are presented as a function of L/D in fig. 3.1.3-30.

It can be seen that the larger L/D results in the larger heat load due to the increased heating time and larger ballistic coefficient. The total heat load includes convective, equilibrium, and non-equilibrium radiative heating.

Fig. 3.1.3-31 shows the increased total shield weight requirement with increasing values of L/D, using ESM 1004 AP = 35 lb/ft³ as the shield material. The variation in total shield weight is not large, varying between 48.7 lbs at L/D = 0 and 53.2 lbs for an L/D = 0.3. No safety factor is embodied in the presented data for the purpose of this parametric study.

The calculations here are for forebody heat protection only. The protection required on the afterbody appears minimal from a cursory analysis.

3.1.3.6 Structural Performance

For a sphere-cone configuration for lifting entry with ballistic coefficients similar to ballistic entry vehicles, the structural requirements and criteria are the same as

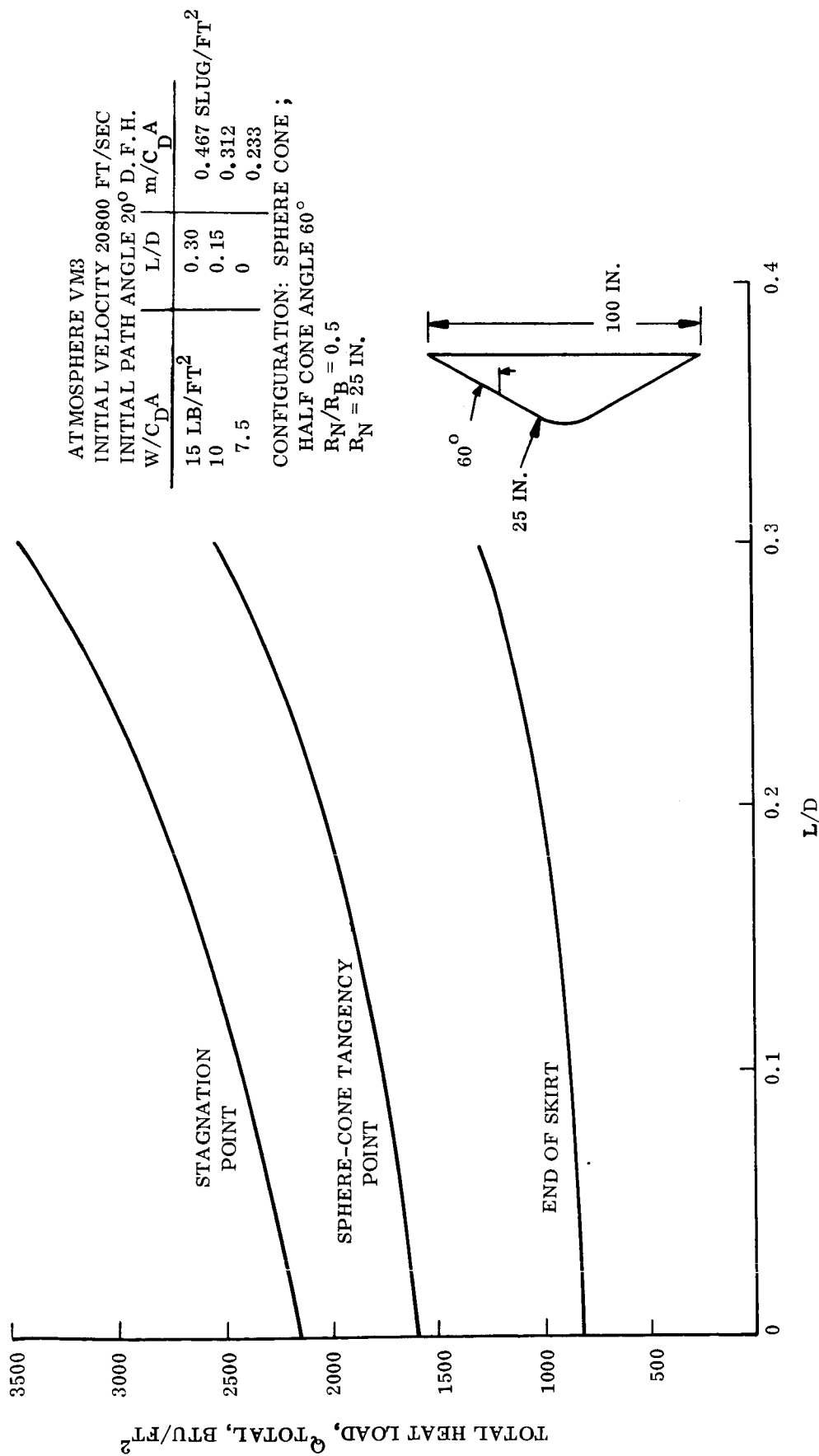


Figure 3.1.3-30. Total Heating as a Function of L/D

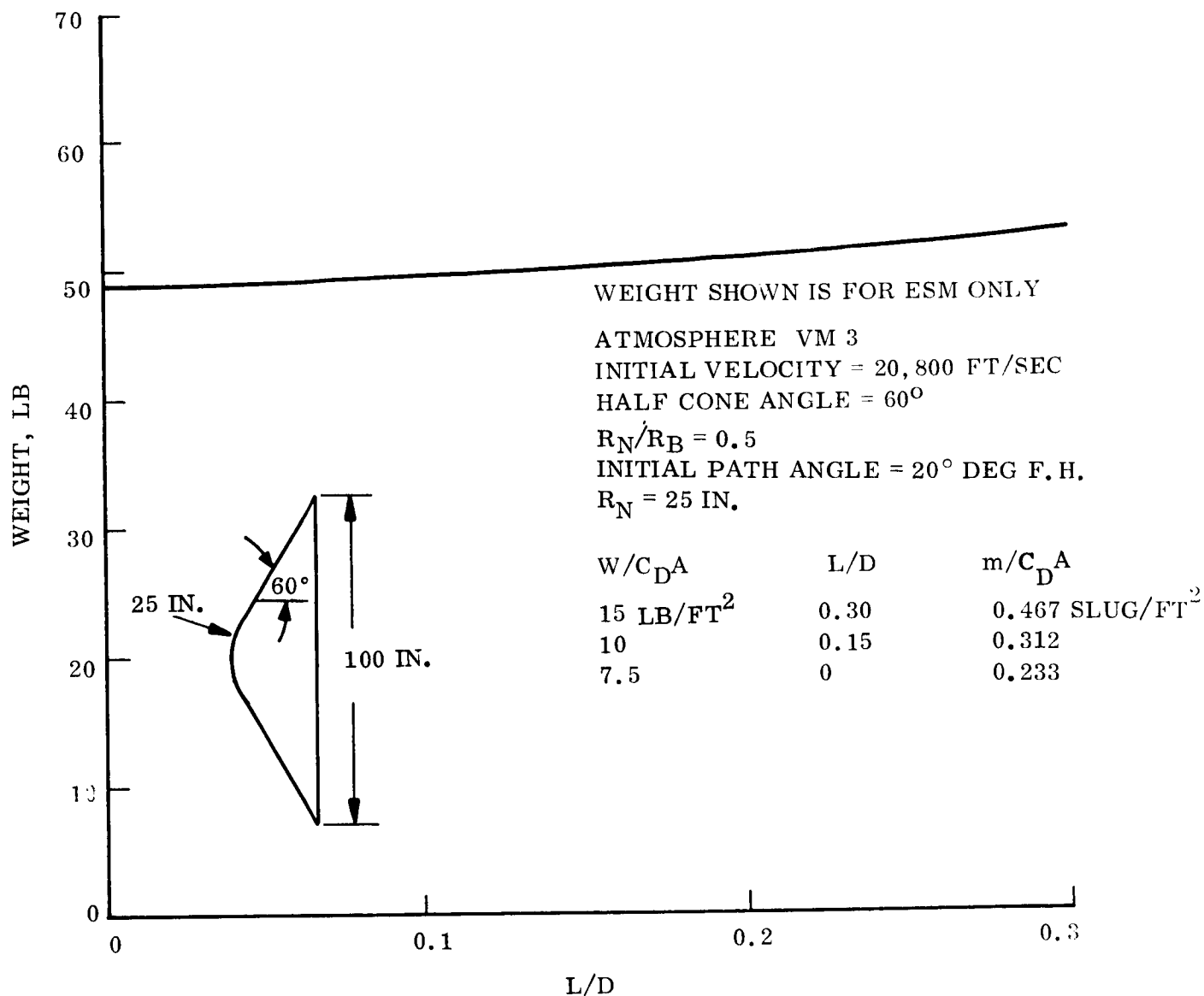


Figure 3.1.3-31. ESM 1004 AP Shield Weight as a Function of L/D

those already described in Section 3.1.2. Windward and leeward variations of aerodynamic pressure distributions have been analyzed and are within the structural capabilities of the vehicles designed for ballistic entry.

Vehicles with higher ballistic coefficients, $W/C_D A$, of 20 and 60 psf (0.625 and 1.83 slug/ft²) were considered for lifting entry and compared with a ballistic entry vehicle. All vehicles are of 60° half-cone angle and 0.5 bluntness ratio, and are detailed in table 3.1.3-2.

The weight of each vehicle was assumed distributed along the structure. The distributed weight was proportioned to the vehicle radius or station. The resultant structure weight density is listed in table 3.1.3-3 and is given for the spherical nose and conical skirt separately. For all vehicles, the minimum weight nose structures are titanium honeycomb (0.44 lb/ft²) or stainless steel honeycomb (0.70 lb/ft²). Both honeycombs are based on minimum gauge face sheets of 0.005 in. and core thickness of 0.125 in. These honeycomb weights for the nose apply for all structure temperatures up to 800°F. Therefore, a minimum weight shield could be used if the nose structure were allowed to reach the greatest temperature consistent with requirements of the components within the vehicle.

TABLE 3.1.3-2. VEHICLES ANALYZED

Entry Condition	Ballistic	Lifting	Lifting
Diam (ft)	15	9	9
γ - Path Angle (deg)	40	40	25
ρ lb/ft ²	20	20	60
β (slugs per sq ft)	0.625	0.625	1.86
V_e (kfps)	24	24	24
G_x	82	62	29
G_N	0	7.6	3.5
Weight (lb)	5,140	1,840	5,530
Length (ft)	3.75	2.25	2.25
Stagnation			
Pressure (psi)			
Windward	21.0	14.9	20.4
Leeward	21.0	14.9	20.4
Skirt Pressure (psi)			
Windward	16.5	17.8	24.9
Leeward	16.5	6.7	9.5
Note: In Table 3.1.3-3, the structural weight densities include fabrication factors of 1.05, 1.25, and 1.20 for monocoque, honeycomb, and ring stiffened construction types, respectively.			

TABLE 3.1.3-3. STRUCTURAL DENSITY VARIATION (LB/FT³)

Type of Entry		Ballistic				Lifting				Lifting			
Entry Angle Deg		40				20				25			
ρ , lb /ft ²		20				20				60			
ρ , slug/ft ²		0.625				0.625				1.86			
Base Diameter Ft		15				9				9			
Temp. Deg F	Mat'l	Nose		Skirt		Nose		Skirt		Nose		Skirt	
		Mono	H/C	H/C	R/S*	Mono	H/C	H/C	R/S*	Mono	H/C	H/C	R/S*
200	Al	1.50	0.58	1.96	3.79	0.87	0.61	1.26	3.65	0.96	0.61	1.72	4.23
	SS	2.79	0.67	0.88	6.17	1.68	0.70	0.70	5.91	1.88	0.70	0.77	6.74
	Ti	2.03	0.43	0.85	4.53	1.21	0.44	0.56	4.40	1.35	0.44	0.74	5.03
	Be	--	--	--	1.74	--	--	--	1.64	--	--	--	1.93
400	Al	1.50	3.47	10.6	4.09	0.89	2.25	7.25	3.90	0.99	3.07	10.1	4.54
	SS	2.93	0.67	1.01	6.23	1.69	0.70	0.70	5.97	1.89	0.70	0.89	6.82
	Ti	2.08	0.43	1.07	4.72	1.24	0.44	0.69	4.60	1.38	0.44	0.93	5.26
	Be	--	--	--	1.79	--	--	--	1.68	--	--	--	1.98
600	Al	1.58	--	--	7.66	0.93	--	--	5.39	1.03	--	--	6.99
	SS	2.93	0.67	1.24	6.33	1.71	0.70	0.82	6.08	1.91	0.70	1.11	6.95
	Ti	2.13	0.43	1.31	4.91	1.28	0.44	0.85	4.79	1.42	0.44	1.15	5.49
	Be	--	--	--	1.86	--	--	--	1.73	--	--	--	2.04
800	Al	--	--	--	--	--	--	--	--	--	--	--	--
	SS	2.93	0.67	1.85	6.50	1.75	0.70	1.23	6.26	1.95	0.70	1.68	7.16
	Ti	2.18	0.43	1.51	3.06	1.31	0.44	0.99	4.94	1.46	0.44	1.34	5.67
	Be	--	--	--	1.96	--	--	--	1.81	--	--	--	2.15
1000	Al	--	--	--	--	--	--	--	--	--	--	--	--
	SS	3.06	2.19	6.91	6.99	1.84	1.48	4.74	6.74	2.05	2.00	6.56	7.73
	Ti	2.23	0.83	2.86	5.32	1.35	0.59	1.90	5.20	1.50	0.77	2.61	5.97
	Be	--	--	--	2.25	--	--	--	2.04	--	--	--	2.43
*6.4 in. ring spacing													
*7.7 in. ring spacing													
*7.7 in. ring spacing													
Note:		Mono - monocoque		Al - Aluminum									
		H/C - honeycomb		SS - Stainless-Steel									
		R/S - ring-stiffened		Ti - Titanium									
				Be - beryllium									

The minimum weight skirt structure is also honeycomb, stainless steel or titanium. However, their dimensions are greater than minimum gauge. Also, their weights must increase as the structures operate at higher temperatures. Figs. 3.1.3-32 through 3.1.3-34 show the minimum weight structures at various operating temperatures. The minimum weight of the structure and shield system must be found by adding the weights from these figures, to the shield weights. For example, combining fig. 3.1.3-33 with the shield of fig. 3.1.3-35 will give the resultant system weight in fig. 3.1.3-36.

The weight densities discussed here do not include the weights of rings at the tangency point and at the base of the vehicle. In order to include the ring weights, the weight densities should be increased by about 0.1 lb/ft².

3.1.3.7 Configuration Trade-off Studies

The trade-off studies on lifting entry vehicles for the Mars Lander mission were limited to three specific shapes: aeroshell (60° right circular cone with a 0.5 bluntness ratio, Apollo Command Module, and raked cone-cylinder (60° right circular cone with a base skewed 73° and with an attached elliptical cylinder). The trade-off studies discussed here are confined to the aeroshell unless otherwise stated.

Fig. 3.1.3-37 presents a matrix of aeroshell sizes, weights, and W/C_{DA} values within which the 100 in. diameter size was selected for trade-off studies to show the effect of L/D . The 100 in. diameter line is shown dotted in that figure. Preliminary estimates of landing impact attenuator and instrumentation package weights revealed that a minimum L/D value of about 0.15 would be required for direct entry. However, the trades discussed here include L/D values from zero to 0.3, but all for the 100 in. limit vehicle diameter defined by available booster shroud space.

Fig. 3.1.3-38 is the variation of W/C_{DA} , entry vehicle weight, decelerator load, and aeroshell weight, all as functions of L/D ratio for the 100 in. diameter aeroshell. This figure shows that the use of L/D permits increasing payload weight fractions with relatively small increases in the aeroshell weight (structure and heat shield). Because the ballistic ($L/D = 0$) aeroshell requires some form of roll control during entry, the penalty for attitude control for the lifting case is minimized since the control jets, control valves, electronic sensing and command units, as well as the gas supply bottles are present; the nominal weight increases are primarily due to larger gas supply requirements.

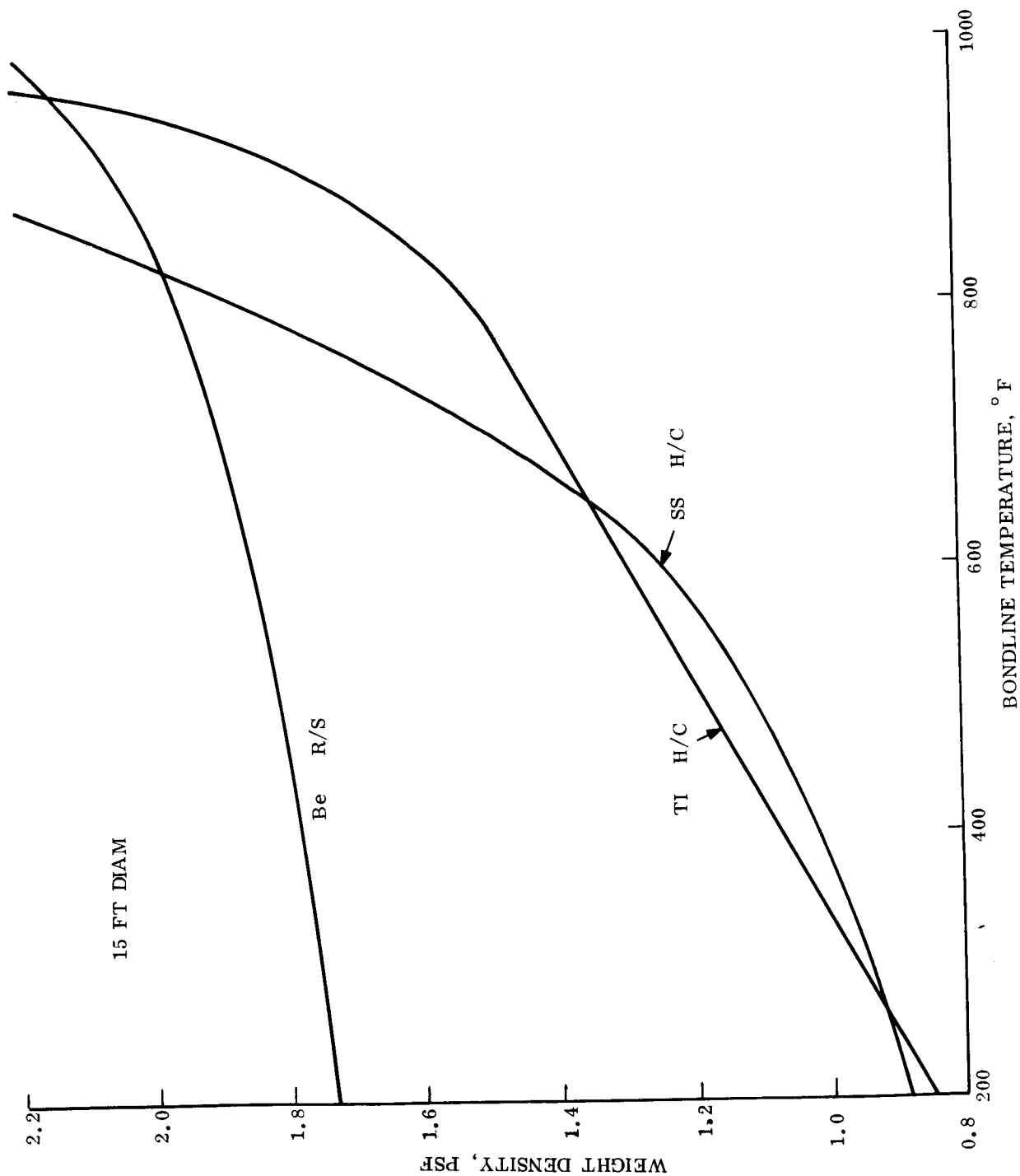


Figure 3.1.3-32. Skirt Weight for Ballistic Entry, $\gamma = 40^\circ$, $\beta = 0.675 \text{ Slug/ft}^2$

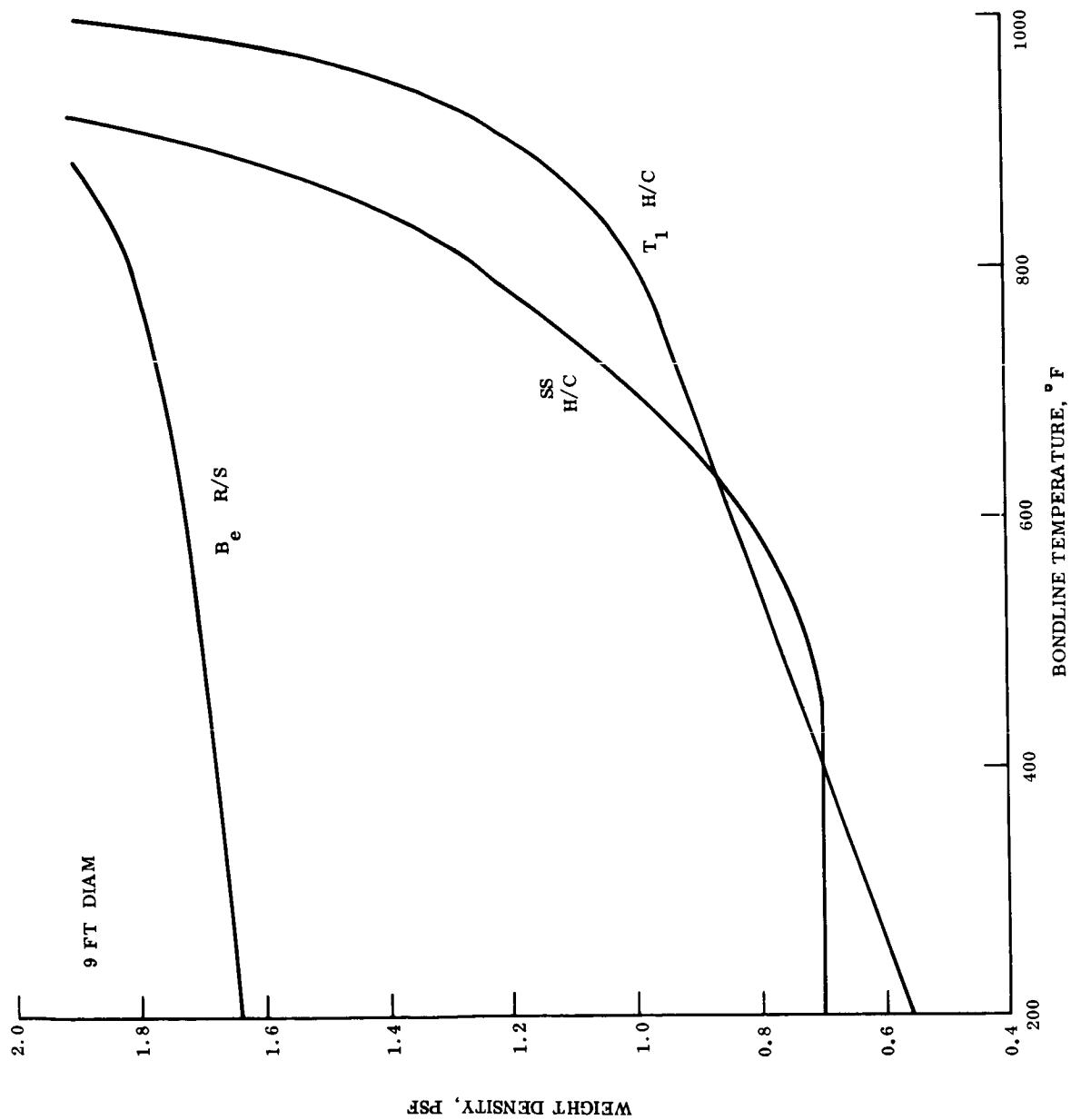


Figure 3.1.3-33. Skirt Weight for Lifting Entry, $\gamma = 40^\circ$, $\beta = 0.625 \text{ Slug/ft}^2$

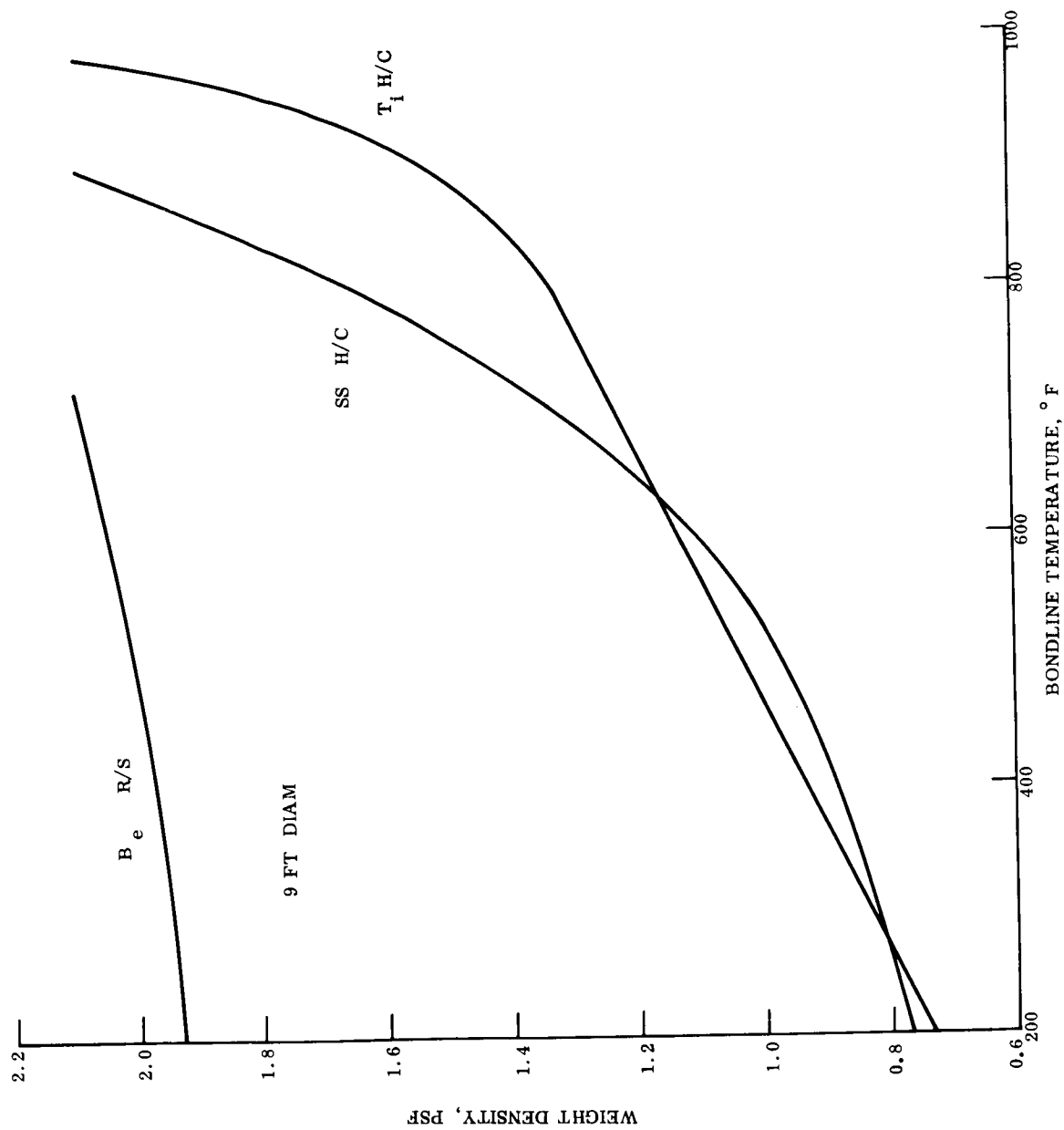


Figure 3.1.3-34. Skirt Weight for Lifting Entry, $\gamma = 25^\circ$, $\beta = 1.83 \text{ Slug/ft}^2$

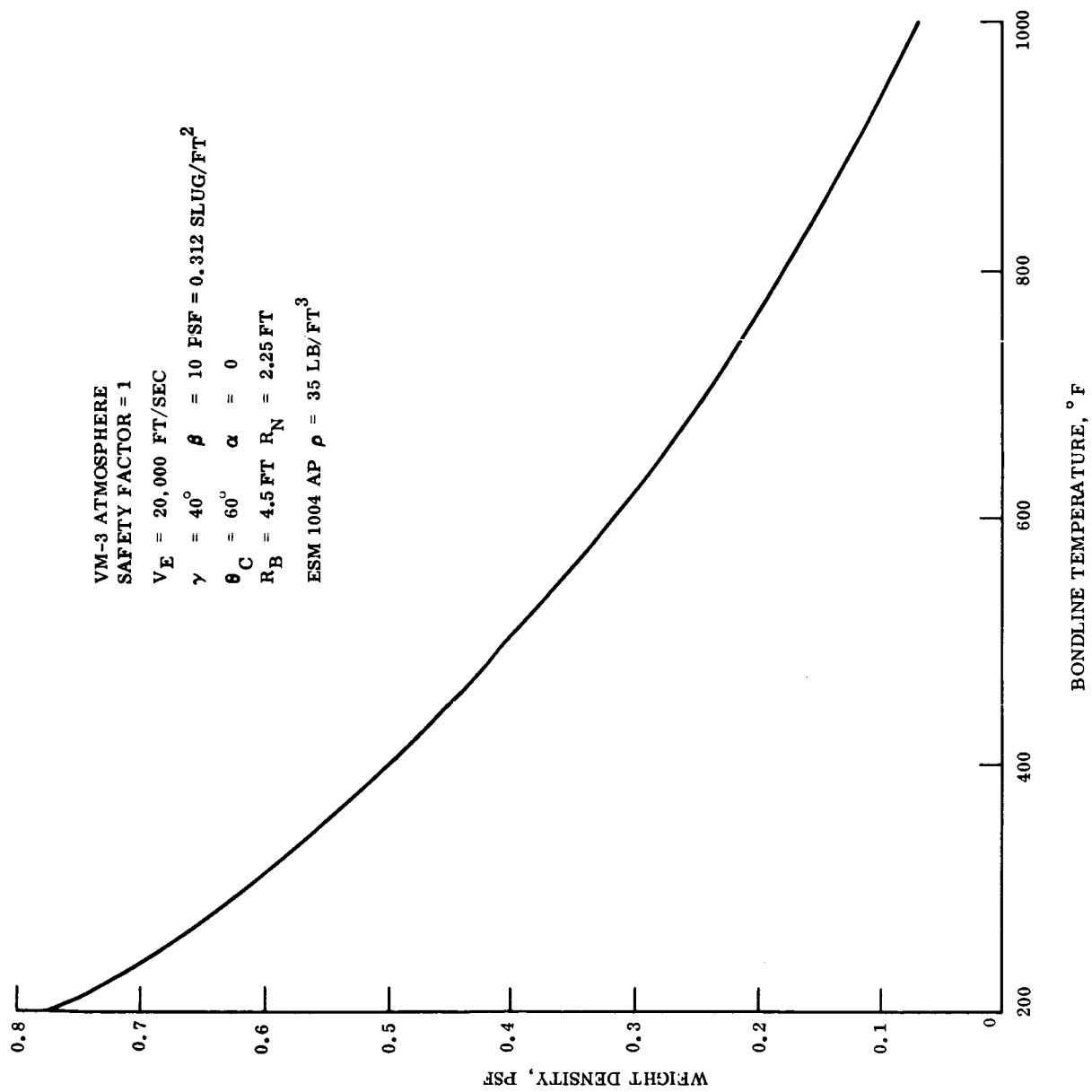


Figure 3.1.3-35. Shield Weight vs Bondline Temperature

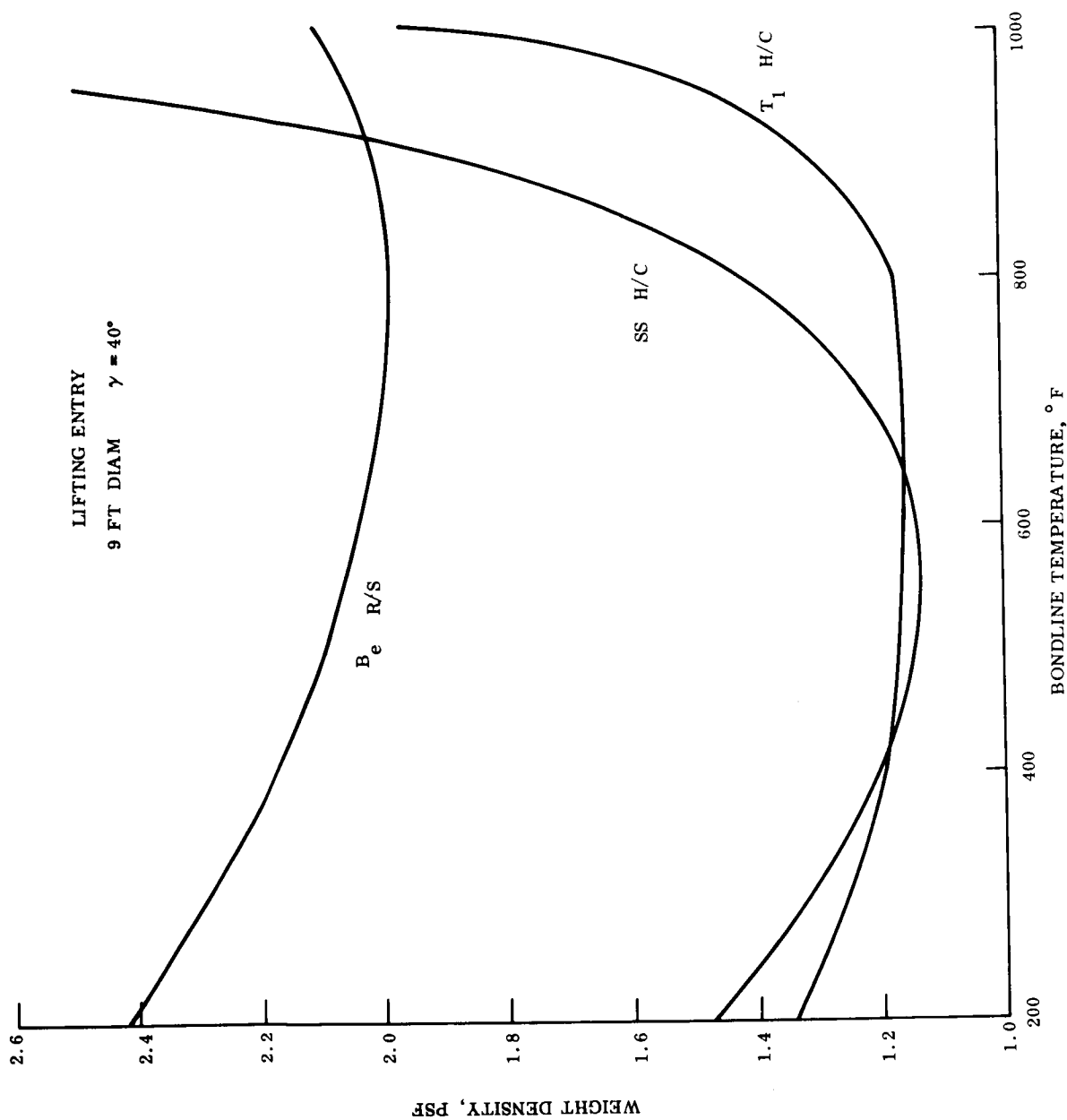


Figure 3.1.3-36. Weight of Structure and Shield

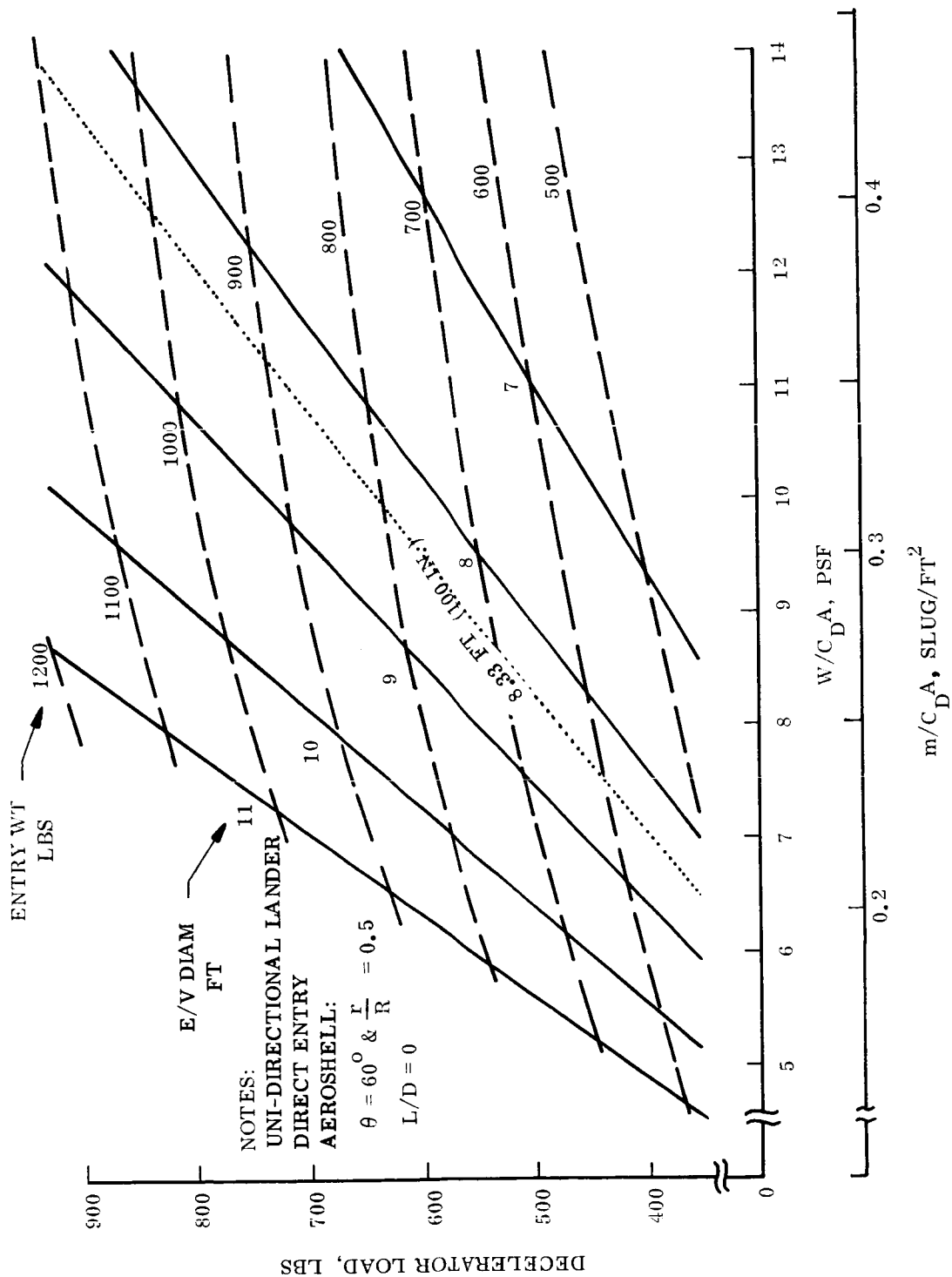


Figure 3.1.3-37. Decelerator Load Versus W/C_D^A

$\frac{\text{SLUG}}{\text{FT}^2}$

PSF

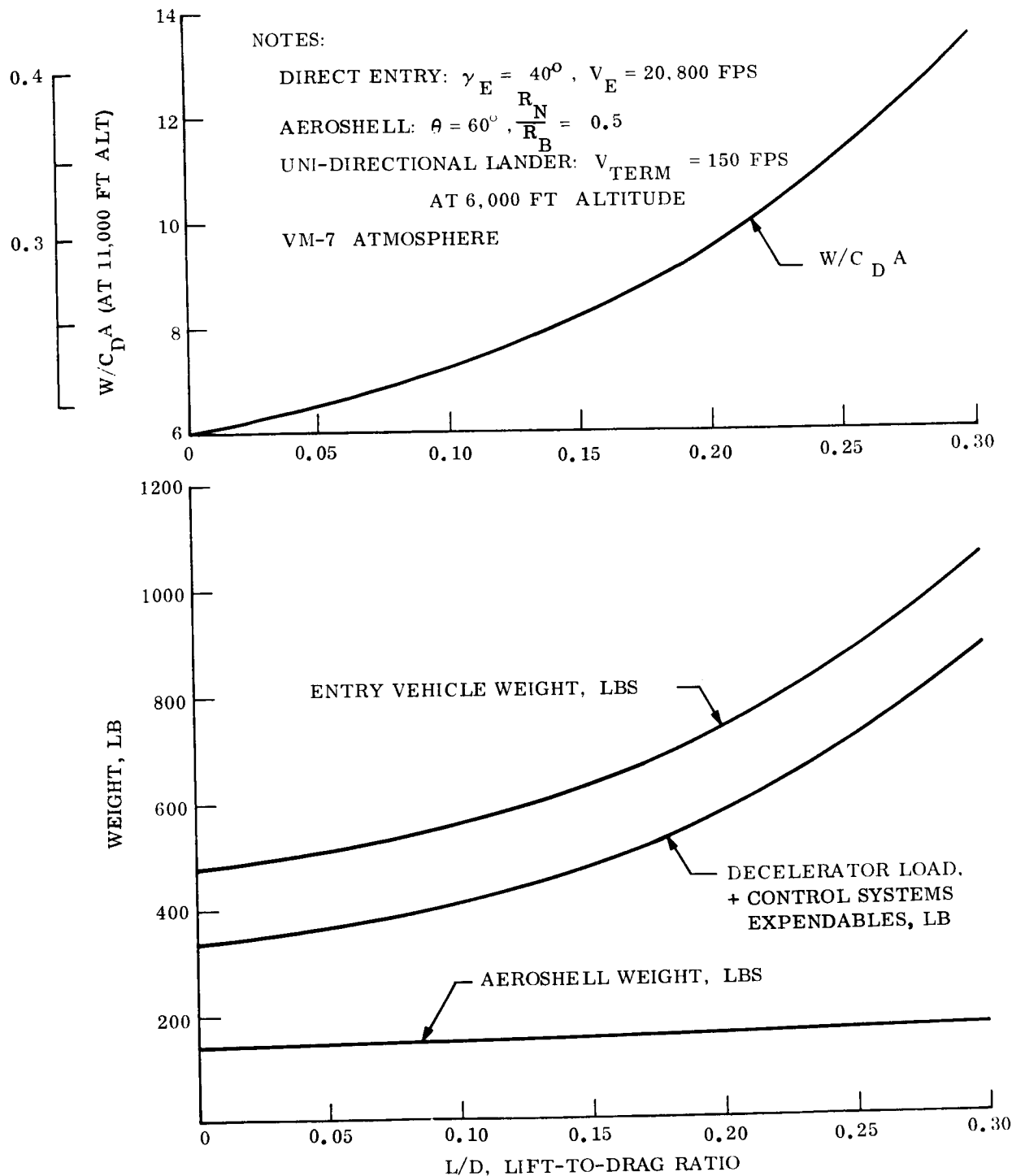


Figure 3.1.3-38. Variations of $W/C_D A$ and Vehicle Weight Fractions with L/D

3.2 PARACHUTE DECELERATOR

3.2.1 REQUIREMENTS AND TECHNOLOGY

3.2.1.1 System Constraints

This section describes parametric parachute performance data for final stage retardation. Environmental conditions at initiation of parachute deployment depend on the entry vehicle retardation capabilities. A parachute deployment limit of Mach 2 is prescribed as a study constraint. This constraint requires that the aeroshell decelerate to Mach 2 at a sufficiently high altitude so that the Lander can be decelerated to some prescribed terminal velocity and altitude. That altitude must conform to the uncertainties of the planet radius and terrain.

Typical cross plots of altitudes at which Mach 2 occurs are shown for the VM-8 model atmosphere in figs. 3.2.1-1, -2 and -3. This atmospheric model is selected since a given Mach number occurs at the lowest altitude in this case, and is thus most critical when using a Mach number limitation as a deployment criterion. Both out-of-orbit and direct entry modes (shallow and steep entry angles) are represented in these plots. Based on Mach 2 at 10,000 ft to 15,000 ft altitude, ballistic coefficients of the order of 12 to 14 psf (0.373 to 0.435 slugs/ft²) can be obtained for the out-of-orbit case. Direct entry at low (25°) angle would allow β 's of 9 to 11 psf (0.280 to 0.342 slugs/ft²) and direct entry at high angle (40°) restricts the maximum β to the range of 5 to 6.5 psf (0.156 to 0.202 slugs/ft²). Techniques and procedures which increase the allowable β 's (and hence entry vehicle weight) are discussed in previous sections of this report.

To perform the parachute subsystem parametric studies, certain requirements which include the Mach 2 deployment limit were assumed. These requirements and/or specifications are considered to be consistent with state-of-the-art, past practice, and current guidelines, listed in table 3.2.1-1.

3.2.1.2 Earth Atmosphere Flight Tests

A significant contribution to the establishment of the feasibility of the use of large parachutes in a Mars-simulated atmospheric environment resulted from the NASA/LRC-conducted Planetary Entry Parachute Program (PEPP). On the basis of these tests (refs. 3-58 through 3-65), several candidate designs performed fairly satisfactorily. The parachutes tested were the Modified Ringsail, the Disk-Gap-Band, and the Cross types with sizes varying from 30 ft to 85 ft in diameter. Both ground launched rocket vehicles and balloon launched vehicles were employed for tests initiated above 100,000 ft altitude. In general, deployment Mach numbers ranged from low supersonic to over Mach 2, with dynamic pressures less than 15 psf. All Ringsail parachutes were designed with 15 percent geometric porosity to provide stability in the low density environment with resulting low drag coefficients. Marginal inflation conditions resulted in some PEPP tests due to the high porosities used. One test of a large (85 ft) modified Ringsail having marginally high porosity was deployed reefed at a dynamic pressure of less than 6 psf and was disreefed at a dynamic pressure of about 2 psf. Fair inflation was realized by this canopy without

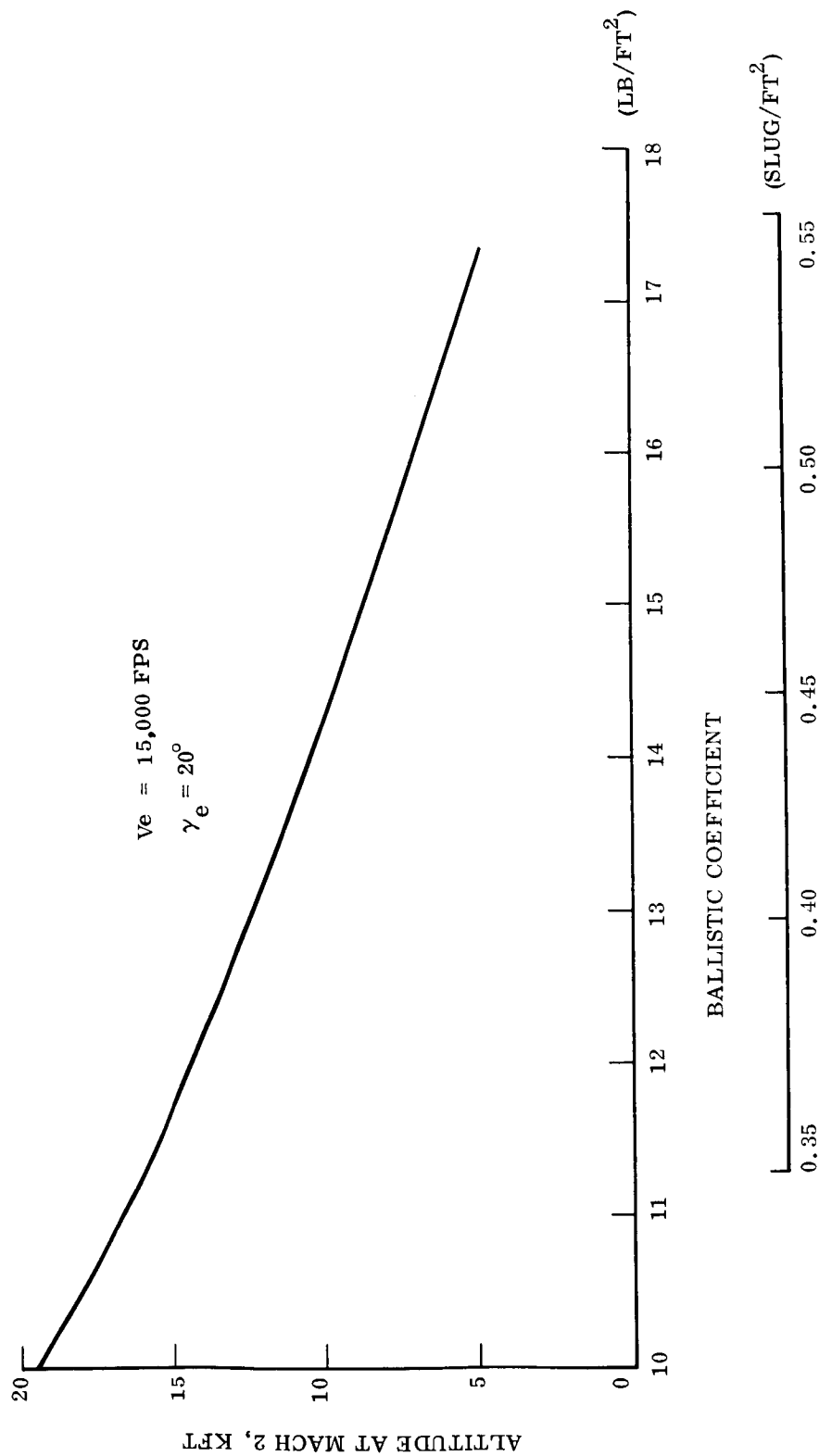


Figure 3.2.1-1. Mach 2 Altitude Variation with Ballistic Coefficient - Out-of-Orbit

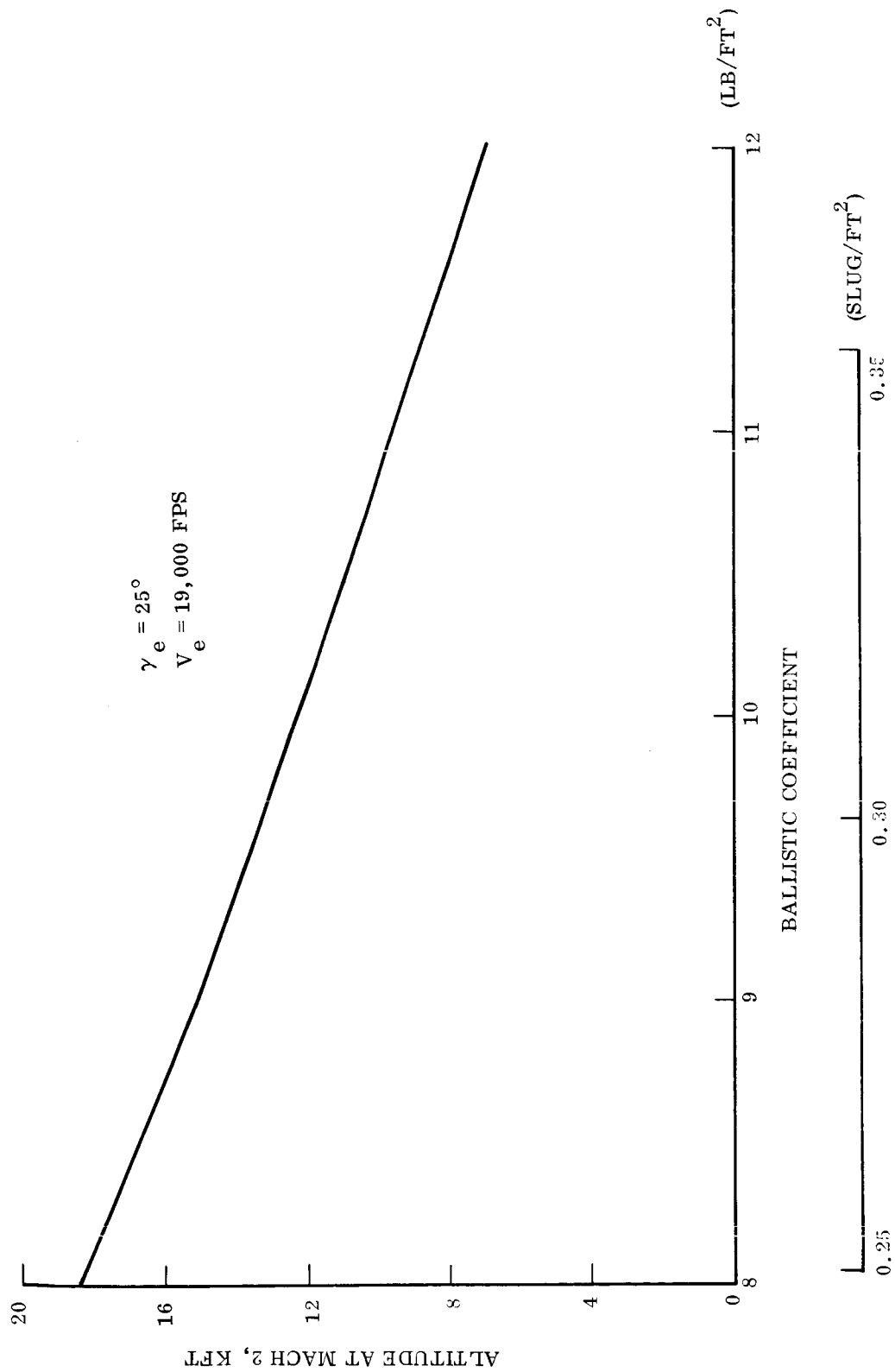


Figure 3.2.1-2. Mach 2 Altitude Variation with Ballistic Coefficient - Low Angle, Direct Entry

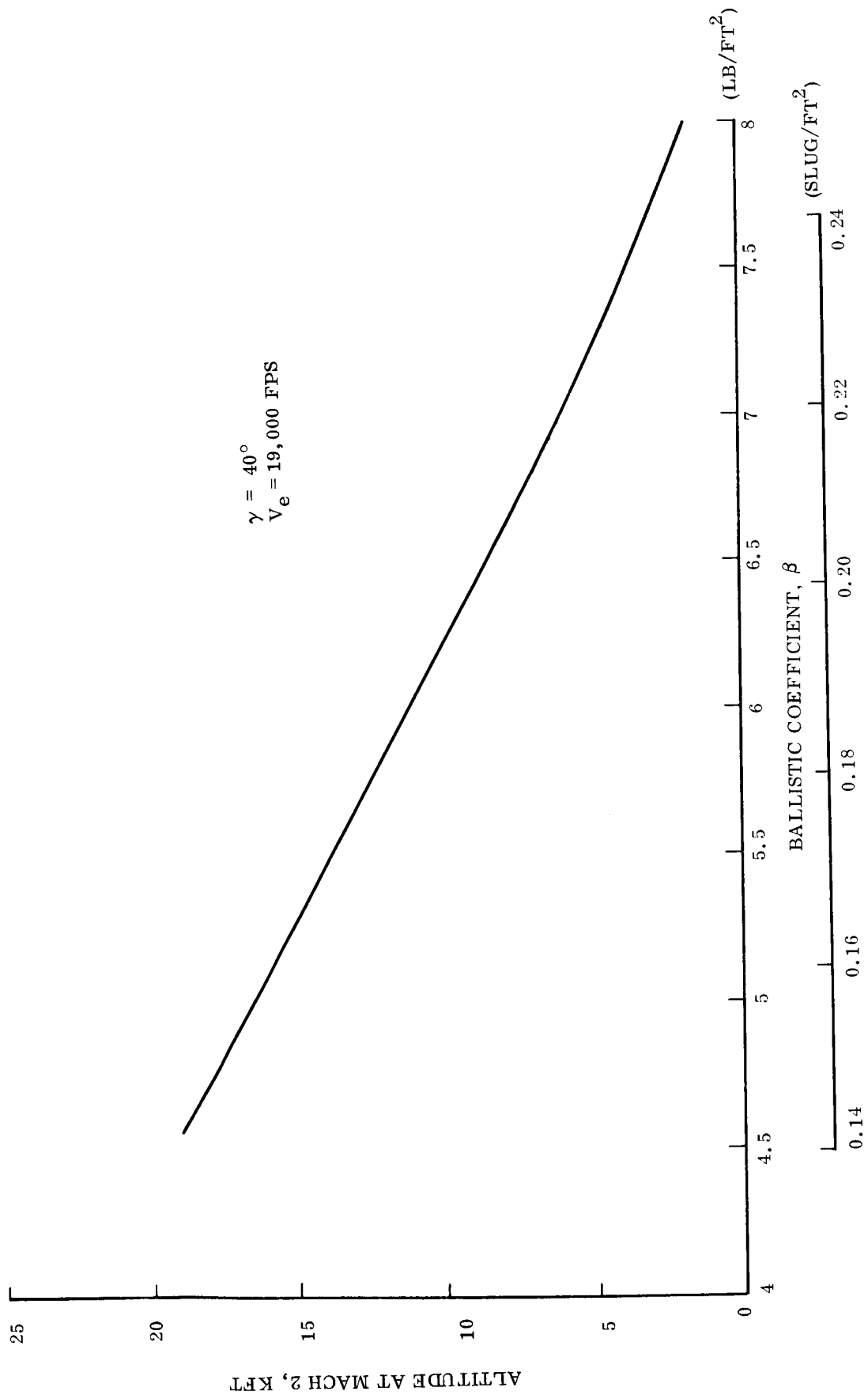


Figure 3.2.1-3. Mach 2 Altitude Variation with Ballistic Coefficient - High Angle, Direct Entry

TABLE 3.2.1-1. PARACHUTE SUBSYSTEM REQUIREMENTS - PARAMETRIC STUDIES

Parameter	Requirement	Basis for Selection
Minimum altitude for terminal velocity on main chute	6000 ft	• Preliminary trajectory calculations • Consideration of surface roughness • Previous experience
Maximum deployment altitude	None other than communication requirements on descent time	• Past studies indicate communications might limit maximum descent time
Minimum deployment Mach number	None other than consideration of vehicle low speed stability	• Large cone angle vehicle angle-of-attack divergence possibility at low Mach number
Minimum dynamic pressure at deployment	2.0 PSF	• High altitude test data
Maximum deployment Mach number	M = 2.0	• Present contract
Parachute type	Modified Ringsail	• Results of NASA LRC PEPP tests, and extensive past use
Number of parachute stages	Maximum of 2 (main and drogue) Main could be cluster and/or reefed	• State-of-the-art • Past experience • Use of drogue for main chute deployment
Parachute materials	Dacron or Nomex	• Sterilization requirement
Deployment sensing	Passive based on entry vehicle trajectory parameters	• Past studies • Previous experience • Lightweight • Reliability
Maximum parachute diameter	100 ft	• Past practice, ease of handling and fabrication
Maximum number of parachutes in a cluster	3	• Quantity of data
Modified Ringsail parachute drag coefficient	0.52	PEPP tests
Parachute filling time, sec.	$\frac{0.64 \lambda g \times 2.0 D_0}{V}$	PEPP tests

the need of an inflation aid. Filling times for these canopies followed, in most cases, the standard filling time expression for geometric porosity parachutes (ref. 3-66).

$$t_f = \frac{0.65 \lambda g D_o}{V}$$

where

t_f = filling time (sec)

λg = geometric porosity (percent)

D_o = nominal diameter

V = initial velocity, FPS

The modified Ringsail parachutes appeared to follow more closely the "performance" curve of ref. 3-66 which provides a higher value. The filling time used in the parametric studies assumed a 15 percent geometric porosity and introduced a factor of two in the filling time expression as suggested by PEPP Ringsail data.

3.2.2 SUBSYSTEM SYNTHESIS

Deceleration of the hard Lander is accomplished in several stages including the aeroshell drag, deployment of a parachute in one or more drag stages and deceleration of the payload at impact by the crush-up impact attenuation subsystem. Within the constraints of loads, booster shroud diameter, current state-of-the-art, Mars model atmospheres and entry trajectories, it is desirable to combine the various deceleration stages in such a way that their weight is minimized and the payload weight is maximized for a given mission.

The range of terminal vertical velocities of the Lander descending on the parachute is varied from 50 to 200 fps in the VM-7 atmosphere. This model has the lowest surface density and represents the highest terminal velocity environment. Since the exact Martian surface level is unknown impact as high as 6,000 ft above the assumed surface is considered.

A parachute deployment limit equal to or less than $M = 2.0$ has been assumed based on NASA RFP L-9088. The purpose of the parachute subsystem is to separate the Lander from the aeroshell and decelerate it to a desired impact velocity. Since the object is to maximize payload weight, it is required that the maximum entry vehicle ballistic coefficient (β) be determined, which is compatible with the parachute deployment limit and vertical impact velocity.

The RESSEP computer program (Appendix A-4) has been used to determine the maximum allowable entry vehicle β as a function of terminal condition and mission mode, i. e., direct entry and out of orbit. Based upon the results of the RESSEP study, the maximum β are plotted in figs. 3.2.2-1, -2 and -3. Fig. 3.2.2-1 presents the maximum

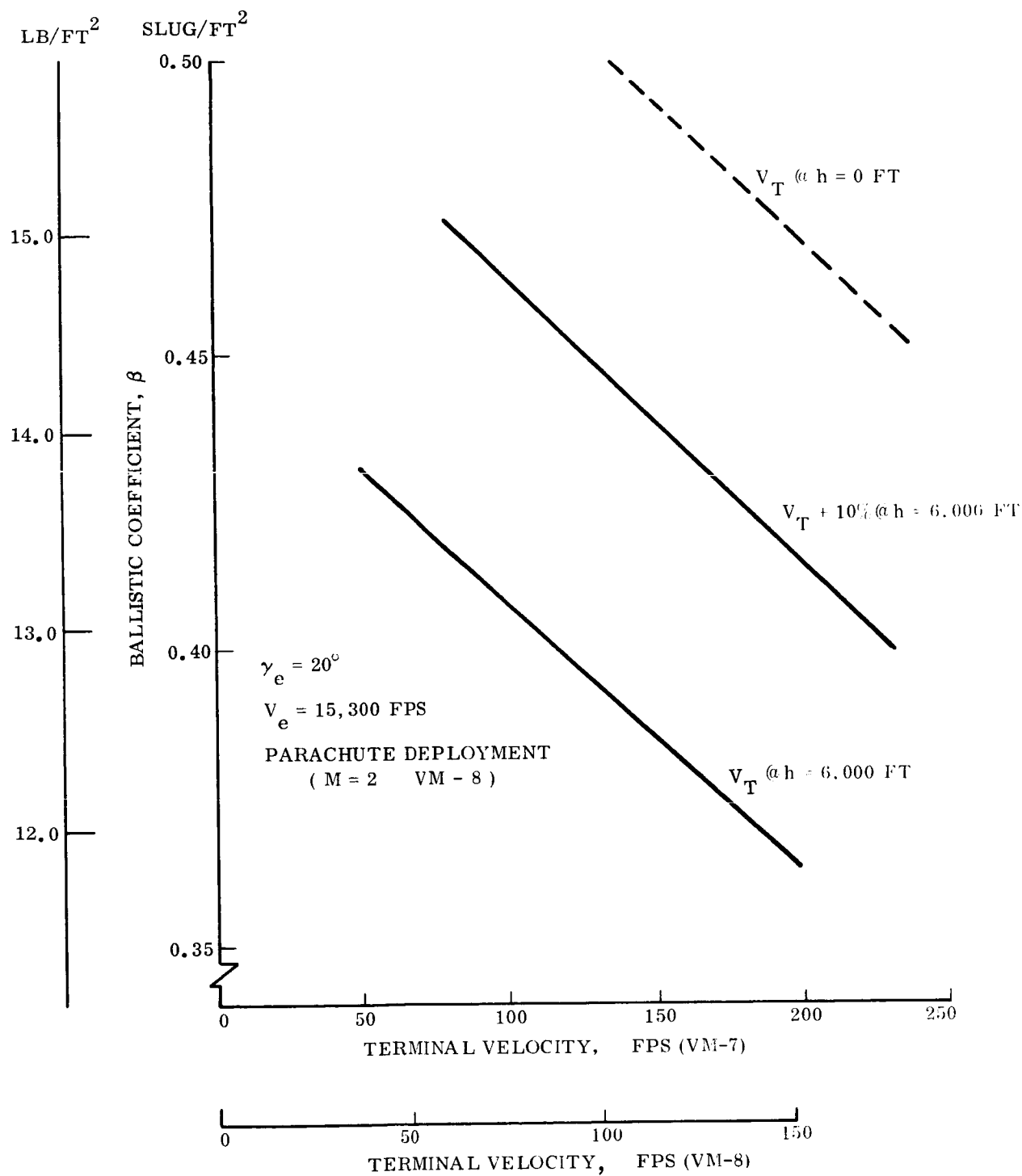


Figure 3.2.2-1. Maximum Ballistic Coefficient Variation with Terminal Velocity ($\gamma = 20^\circ$) Direct Entry

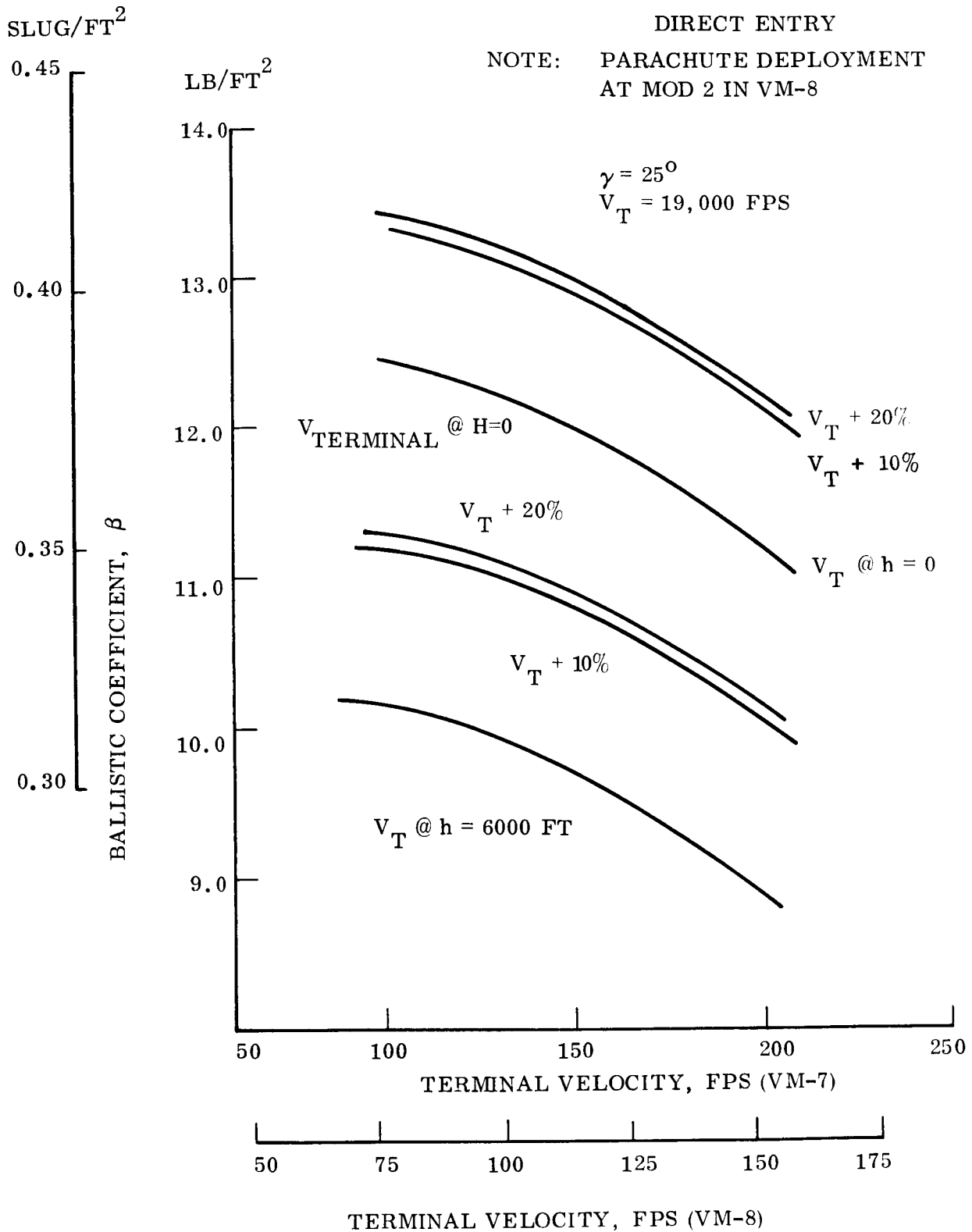


Figure 3.2.2-2. Maximum Ballistic Coefficient Variation with Terminal Velocity ($\gamma = 25^\circ$) Direct Entry

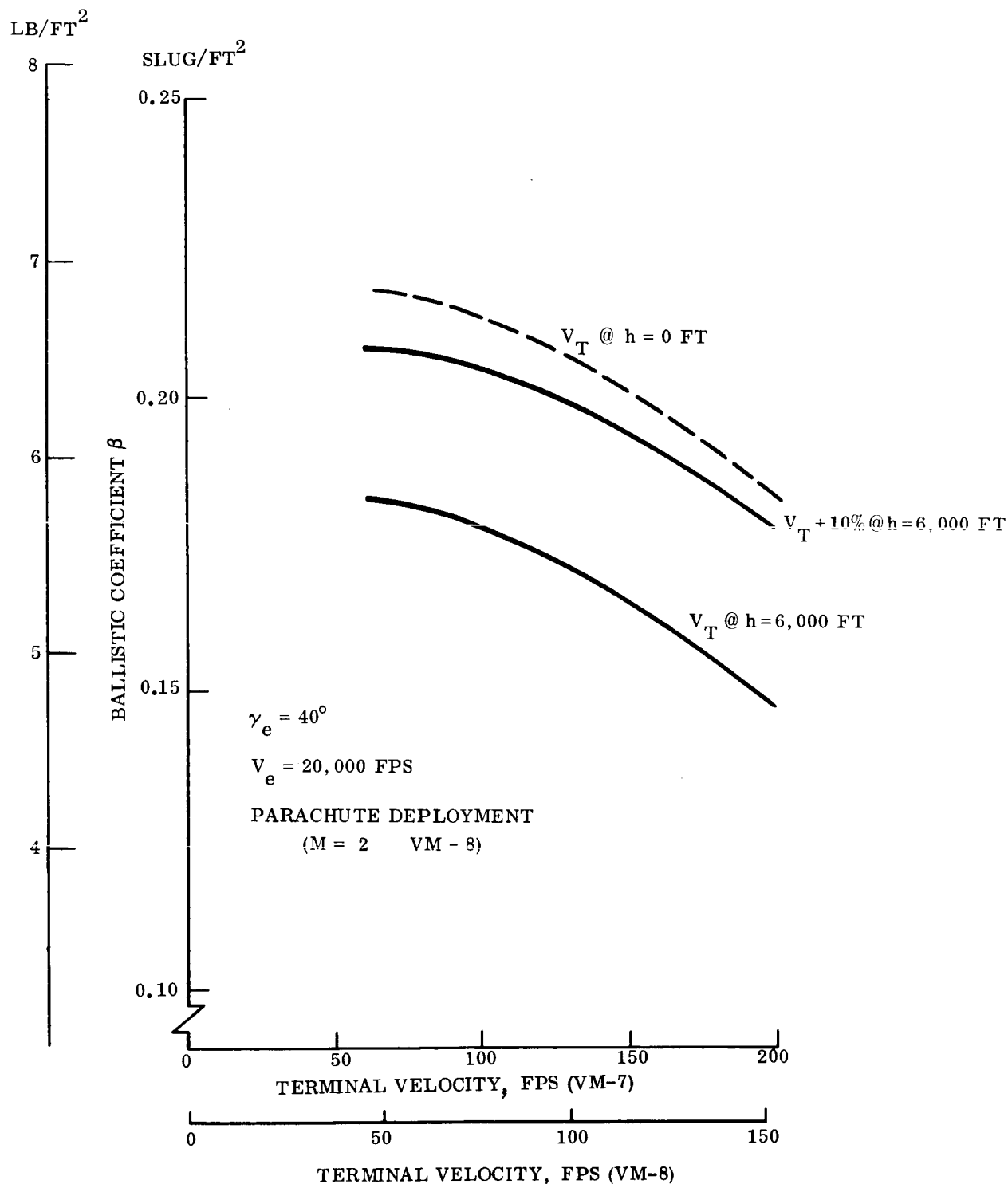


Figure 3.2.2-3. Maximum Ballistic Coefficient Variation with Terminal Velocity ($\gamma = 40^\circ$) Direct Entry

entry vehicle β as a function of the terminal vertical velocity of the Lander descending on the parachute for the out-of-orbit case. It can be seen that as the terminal velocity is decreased by enlarging the parachute, the allowable β increases. This is due to the fact that for the large parachutes associated with decreased terminal velocity, less altitude is lost during deceleration from Mach 2 to the terminal velocity at $h = 6,000$ ft. Therefore, since less altitude is lost, Mach 2 can occur at a lower altitude for a higher allowable β . Since the VM-8 atmosphere results in Mach 2 occurring at the lowest altitude, it has been used to determine the parachute drag staging by the RESSEP computer program. In addition to the terminal velocity at $h = 6,000$ ft of fig. 3.2.2-1, other curves are presented which indicate increases in β that can be gained by modification of deceleration requirements. First allowing the vertical velocity of the Lander to be 10 percent greater than terminal, can raise the allowable β significantly. Also, eliminating the requirement of impact at an altitude of 6,000 ft raises the allowable β .

The effects of terminal velocity on β shown in figs. 3.2.2-1, -2, and -3 were determined as follows. Each entry velocity and path angle has associated with it a maximum ballistic coefficient which will permit a particular terminal velocity on a parachute. For the current study, three entry modes were considered, two direct and one out-of-orbit along with a matrix of terminal velocities. Since for a given entry mode, the minimum altitude at the specified Mach 2 deployment occurs in VM-8, this model was used to determine altitude loss while on a chute.

A matrix of ballistic coefficients were used to generate vehicle trajectories for each of the $V_e - \gamma_e$ entry conditions. Using this matrix, plots of ballistic coefficient variation with altitude at which Mach 2 occurs were generated for each of the 3 $V_e - \gamma_e$ conditions. The RESSEP program was then used to determine what minimum altitude (or max β) would allow the chute to deploy and still permit it to reach the particular terminal velocity at 6,000 ft. A typical parachute trajectory is shown in fig. 3.2.2-4 for direct entry, $\gamma = 40^\circ$ using the maximum ballistic coefficient of 0.17 slug/ft² corresponding to a 115 ft/sec terminal velocity at 6,000 ft (ref. fig. 3.2.2-3). The figure shows a parachute deployment at Mach 2 at 14,500 ft in a reefed condition; disreef occurs at 11,200 ft. The majority of the velocity loss occurs before 10,000 ft or over an altitude range of 4,500 ft. Assuming the same velocity loss would occur at the slightly lower altitude would allow parachute deployment at 10,500 ft at Mach 2. Looking at a plot of altitude at Mach 2 vs β (fig. 3.2.1-3) shows that the corresponding β would be 6.2 lb/ft² (0.193 slugs/ft²). The previous case corresponds to a $V_{TERM} + 10$ percent (i. e., 126.5 ft/sec) as shown in fig. 3.2.2-3. Also shown on that figure is the allowable β resulting from a terminal velocity at impact which was derived by the altitude loss method described above.

The same analysis was applied to the other entry conditions permitting the determination of the corresponding allowable β 's. Fig. 3.2.2-5 shows the parachute trajectory corresponding to a 100 fps impact velocity in VM-7, and the altitude penalty paid in reaching terminal velocity of 76.5 fps at 6,000 ft in a VM-8 atmosphere.

Fig. 3.2.2-2 presents limiting β for direct entry at $\gamma_e = 25$ degrees ($22 + 3$ degree). As a result of the increased flight path angle and higher entry velocity, the limiting β is less than that of the out-of-orbit case. Again, the variations in β that correspond to alternate terminal conditions are presented.

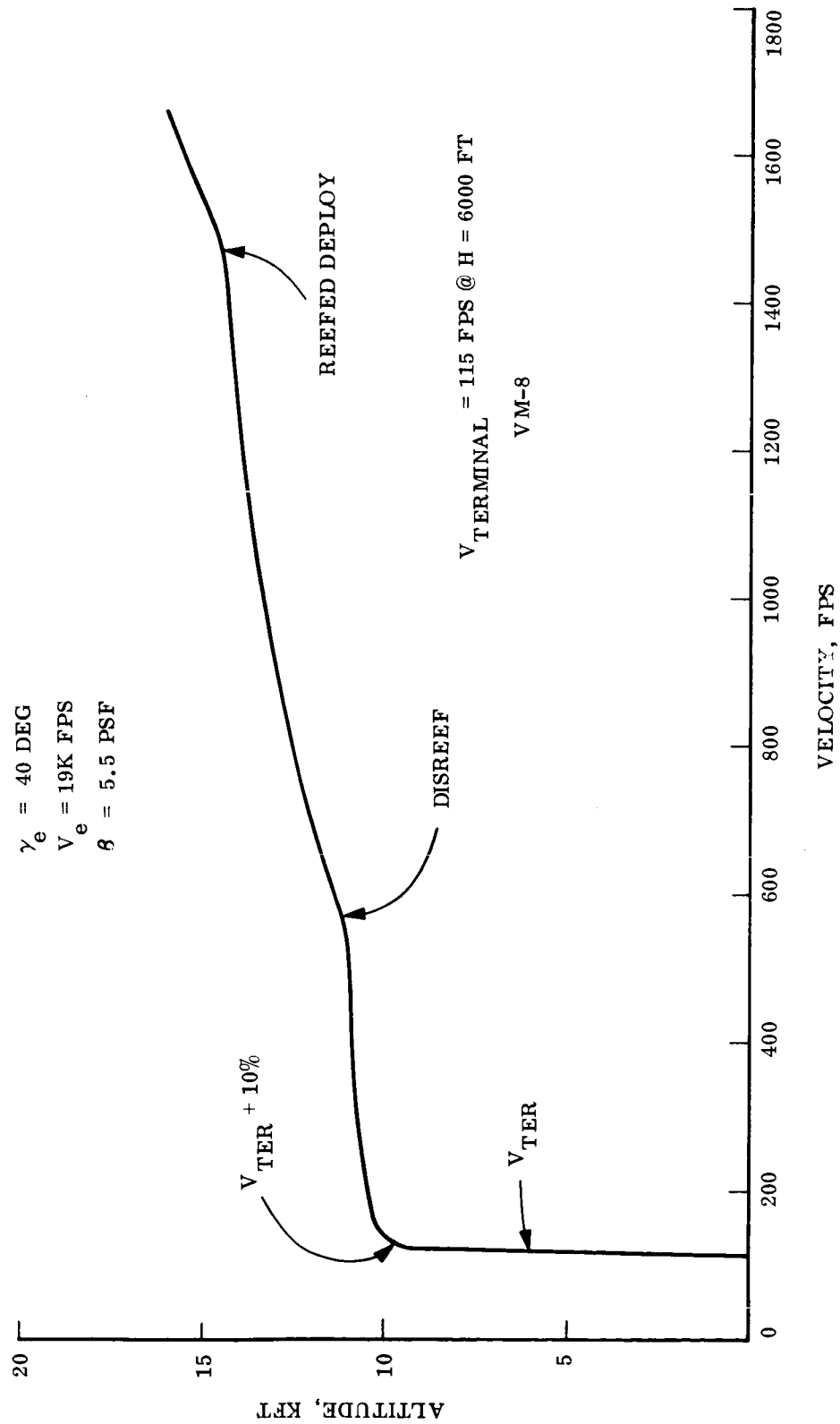


Figure 3.2.2-4. Altitude-Velocity History, 52.2 ft Parachute

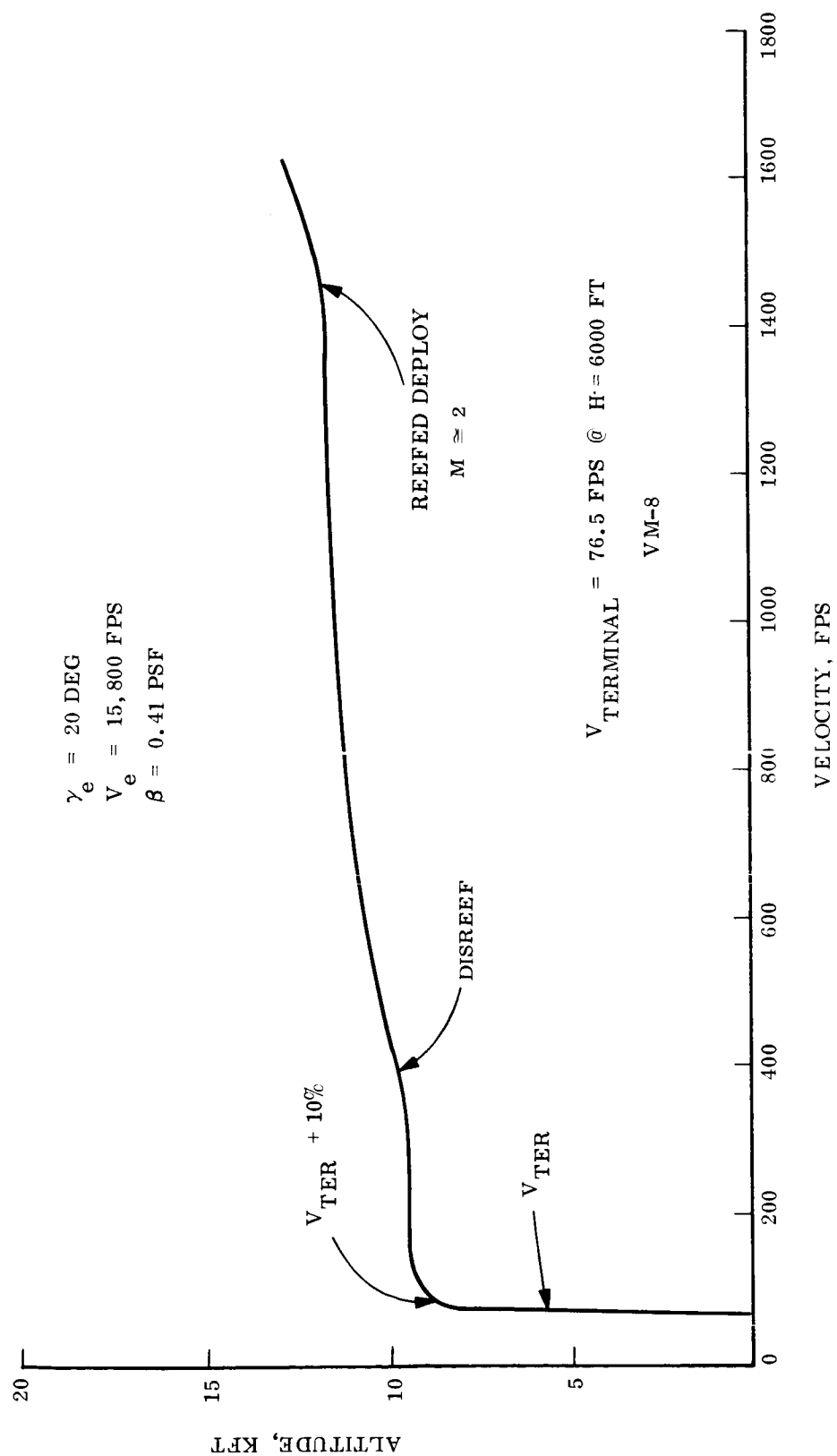


Figure 3.2.2-5. Altitude-Velocity History, 80 ft Parachute

Fig. 3.2.2-6 demonstrates the effect of the use of lift to increase the allowable β . Superimposed on this figure is the ballistic ($L/D = 0$) plot. For a 100 fps terminal velocity in the VM-7 atmosphere, the allowable β may be increased from less than 6 PSF (0.19 slug/ft²) at $L/D = 0$ to nearly 14 PSF (0.44 slug/ft²) at $L/D = 0.3$. A lower maximum path angle of 30° will also have the same effect as increasing allowable β .

3.2.3 SUBSYSTEM PERFORMANCE

3.2.3.1 RESSEP Studies and Trajectories

The parachute parametric study encompasses the complete range of decelerator loads specified for study for both direct entry and de-orbit modes. The significant parameter of interest is the weight of the parachute system as a function of the vertical descent velocity, Lander weight, entry vehicle ballistic coefficient, model atmosphere and entry condition.

Entry vehicle ballistic coefficient limits, discussed previously, reflect the Mach 2 parachute deployment limit. Typical of parachute descent trajectories, fig. 3.2.3-1 shows the variation of trajectory parameters for a Mach 2 deployment condition of a 90 ft nominal diameter modified Ringsail parachute. This parachute was designed by the RESSEP program to provide a vertical descent velocity of 150 fps in the VM-7 model atmosphere. The same parachute will provide a vertical velocity of 115 fps in the VM-8 model atmosphere at an altitude of 6,000 ft. The parachute is deployed reefed in order to provide an efficient parachute design. The feasibility of reefing a large parachute in a simulated Martian environment was demonstrated by the large parachute PEPP test documented in ref. 3-58. Although this test was unique in that test program, it is considered to be severe since the parachute material porosity has been indicated to have been marginally high. Such a characteristic, in general, discourages the opening of a parachute. The reduced energy of the wake behind bodies as blunt as the current aeroshell must be considered in the design of the drogue extraction parachute, as well as the prediction of the reefed drag area. In most instances the full open parachute will not be affected to a significant degree by the aeroshell, or by Lander forebody and wake. The drag deficiency in the wake of an extremely blunt vehicle (ref. 3-67) is shown to increase with Mach number in free flight ground tests of model drogues. For a trailing drogue, the wake-to-freestream drag coefficient ratio was shown to be about 0.35 at Mach 4 and about 0.6 at Mach 2 in the tests reported in ref. 3-67. Trailing decelerator tests reported in ref. 3-68, were shown to indicate that model Hyperflo drag coefficient reductions behind a blunt nosed cylinder at Mach 4 correlated with a wake centerline dynamic pressure to freestream dynamic pressure ratio of 0.31.

In the present study, reefed parachute diameters are generally of the order of the aeroshell base diameter. A resultant drag efficiency of 0.6 at Mach 2 for a desired 8 percent reefed drag area would reflect a reefed diameter increase of about 30 percent for the case shown in fig. 3.2.3-1. The reefed diameters considered here are compatible with the diameter ratios of trailing device to forebody base diameters used in obtaining the data of refs. 3-67 and -68. This parameter has been shown to be significant to trailing decelerator performance in numerous past tests.

DIRECT ENTRY

$h = 6,000 \text{ FT}$

$M = 2.0$

$\gamma_E = 40^\circ$

$V_E = 20,800 \text{ FPS}$

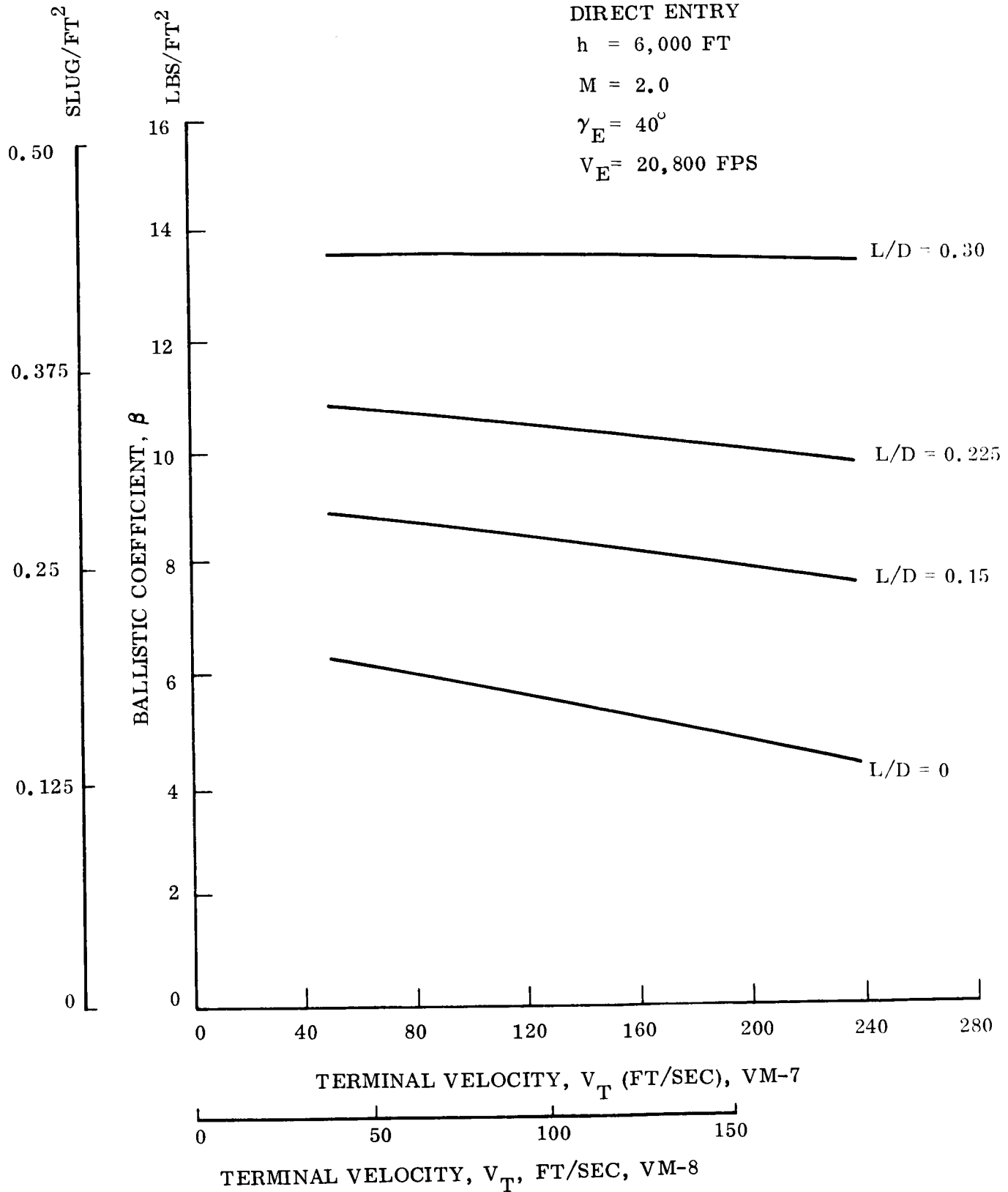


Figure 3.2.2-6. Variation of Limit Ballistic Coefficient with Terminal Velocity, Lifting Entry, Direct Entry

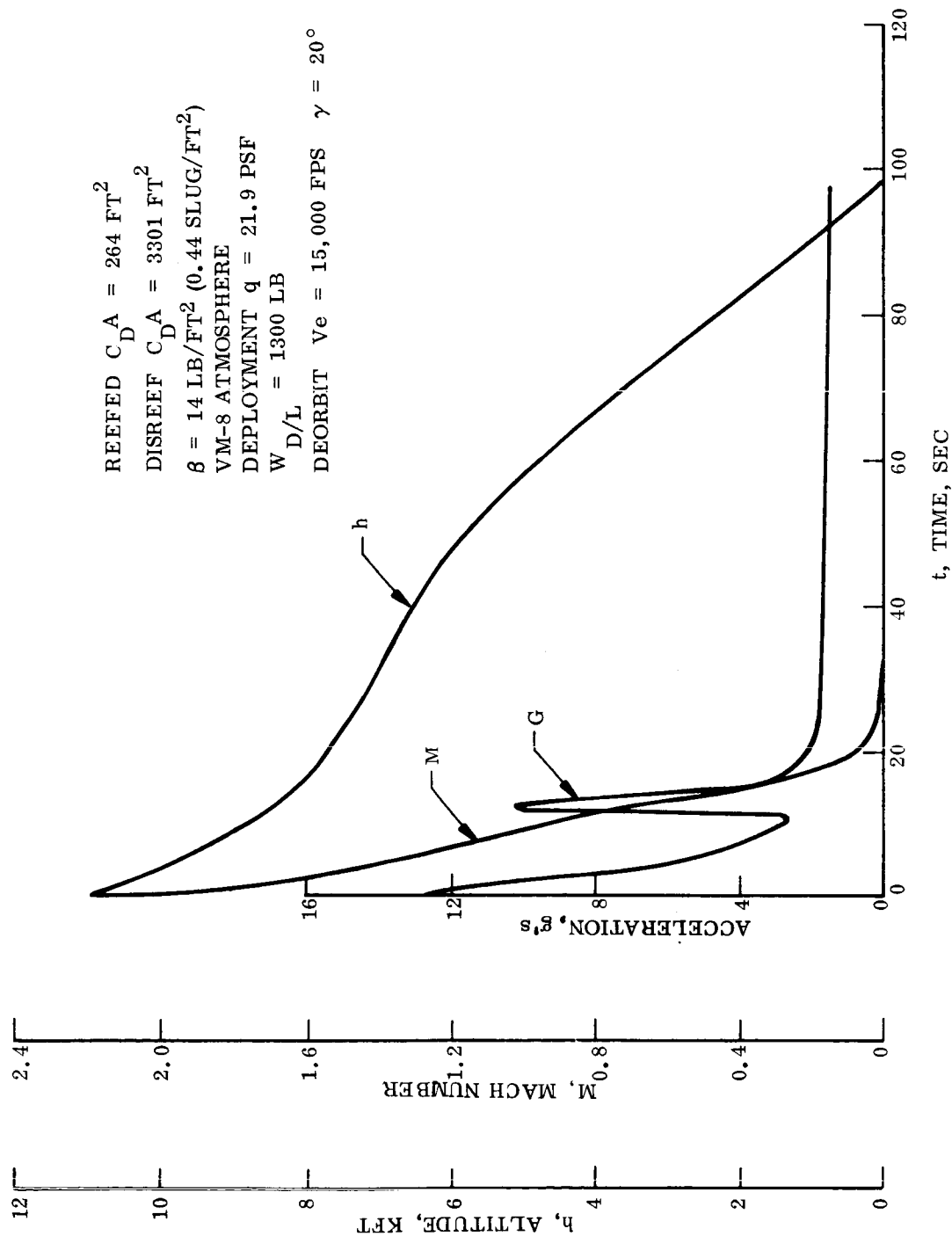


Figure 3.2.3-1. Parachute Descent Trajectory, Out-Of-Orbit Entry, $V_T = 115 \text{ fps}$

The variation of parachute size with terminal descent velocity for various suspended loads is shown in fig. 3.2.3-2. Parachute diameter is presented as a function of the terminal descent velocity in the VM-7 model atmosphere, and are based upon a typical modified Ringsail drag coefficient of 0.52. The strong effect of terminal velocity on parachute size for a given suspended weight is apparent.

3.2.3.2 Trajectories in Alternate Atmospheres

Parachute designs, and critical recovery system sequencing studies have been conducted on the basis of using VM-7 model atmosphere for parachute size determination (least dense model at low altitudes) and upon model VM-8 for sequencing since a given Mach number occurs at the lowest altitude in this atmosphere. Studies of Mars entry trajectories in all ten model atmospheres have shown that the dynamic pressure at Mach 2 is in the 18 to 20 psf range in all atmospheres.

Deployments which are limited to Mach 2, therefore would not exceed this loading condition. Since Mach 2 is a limitation in the present study the parachutes are designed for loading as well as system sequencing in the VM-8 atmosphere.

Deployments which are based upon Mach 2 alone will therefore be subjected to similar loading environment, regardless of the wide altitude differences in the various model atmospheres. In those cases where descent time is desired to be minimized and an altitude sensor is used, recovery in atmospheres other than VM-8 will occur at low - near terminal - dynamic pressures. These terminal values will not be much greater than the ballistic coefficient - 2 psf (0.062 slugs/ft^2) for a $\beta = 5 \text{ psf}$ (0.155 slug/ft^2) vehicle in the Mars atmosphere. The deployment of large parachutes at low velocity, and with a reefed stage designed for the high dynamic pressure (Mach 2 in VM-8) will require longer filling times hence the minimum altitude which is feasible for the design recovery system must be evaluated under these low velocity deployment conditions. Accordingly, studies have been performed in VM-7 and VM-9 atmospheres to evaluate the parachute system corresponding to that previously shown in fig. 3.2.3-1 for the VM-8 atmosphere. A heavy (1300 lbs) decelerator load parachute trajectory with an entry $\beta = 14 \text{ psf}$ (0.435 slug/ft^2) is represented by fig. 3.2.3-1 for a terminal velocity of 115 fps in VM-8 at 6000 ft altitude. In fig. 3.2.3-3, the parachute trajectory is shown for the high β configuration deployment at 15,000 ft altitude. It may be seen that the terminal velocity of 150 fps (VM-7) is reached below 6,000 ft altitude. A larger parachute to provide a terminal velocity of 100 fps in VM-7 provides the trajectory shown in fig. 3.2.3-4. The larger parachute in this case provides the desired terminal velocity at near 6,000 ft altitude.

Both small and large parachutes deployed at 15,000 ft altitude in model VM-9 atmosphere for the 1300 lbs decelerator load result in terminal velocity being reached above 6,000 ft altitude.

To evaluate a minimum terminal velocity condition, a direct entry $V_e = 20,000$ and $V_e = 40^\circ$, $\beta = 5 \text{ psf}$ (0.155 slug/ft^2) recovery is represented by the trajectory in fig. 3.2.3-5 for the same vehicle weight (1300 lb). It may be seen that the small parachute giving a terminal of 115 fps at 6,000 ft in VM-8 does not reach its terminal of 150 fps in VM-7 until near impact as in the high β case for the same terminal condition.

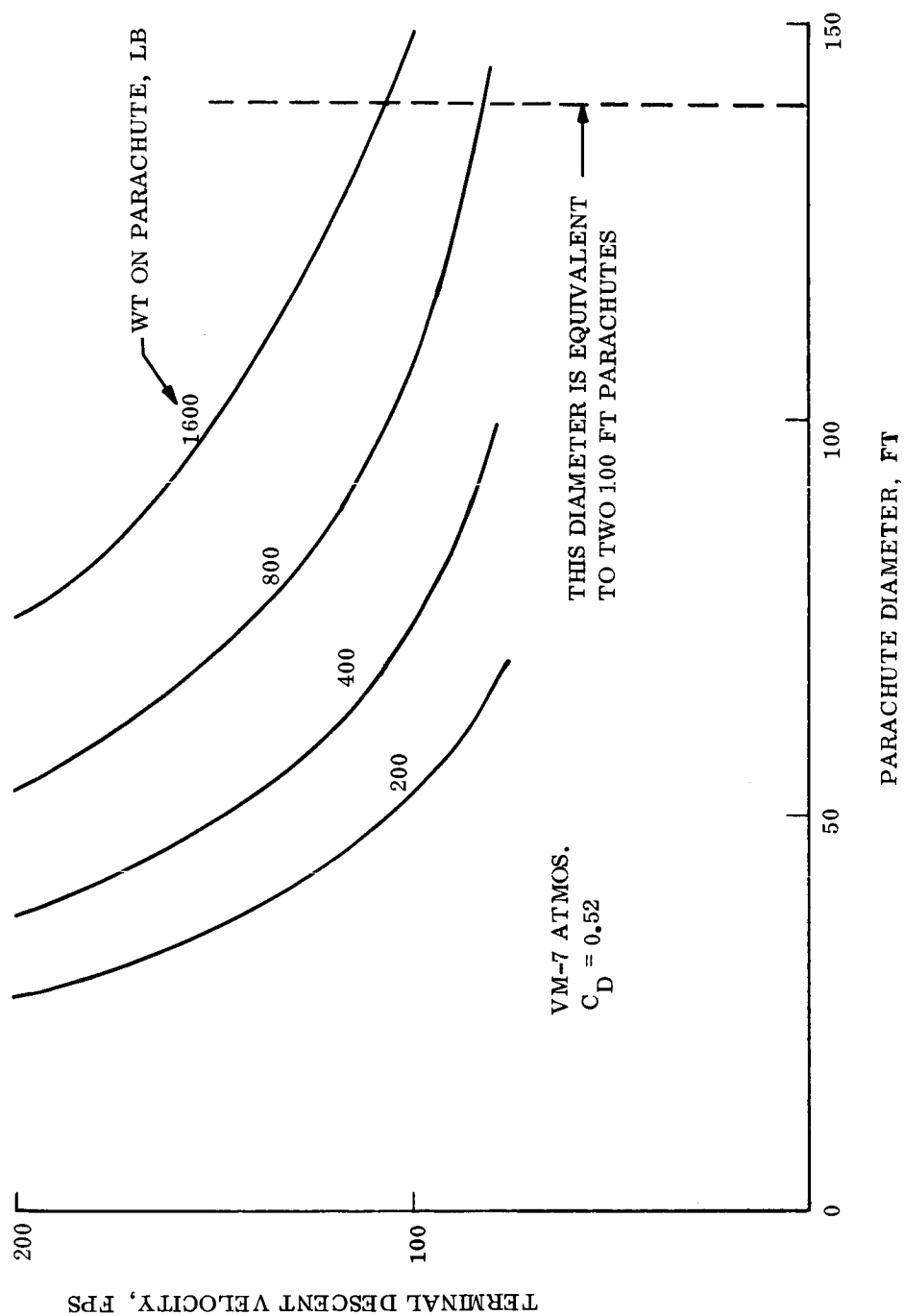


Figure 3.2.3-2. Variation of Descent Velocity with Parachute Size

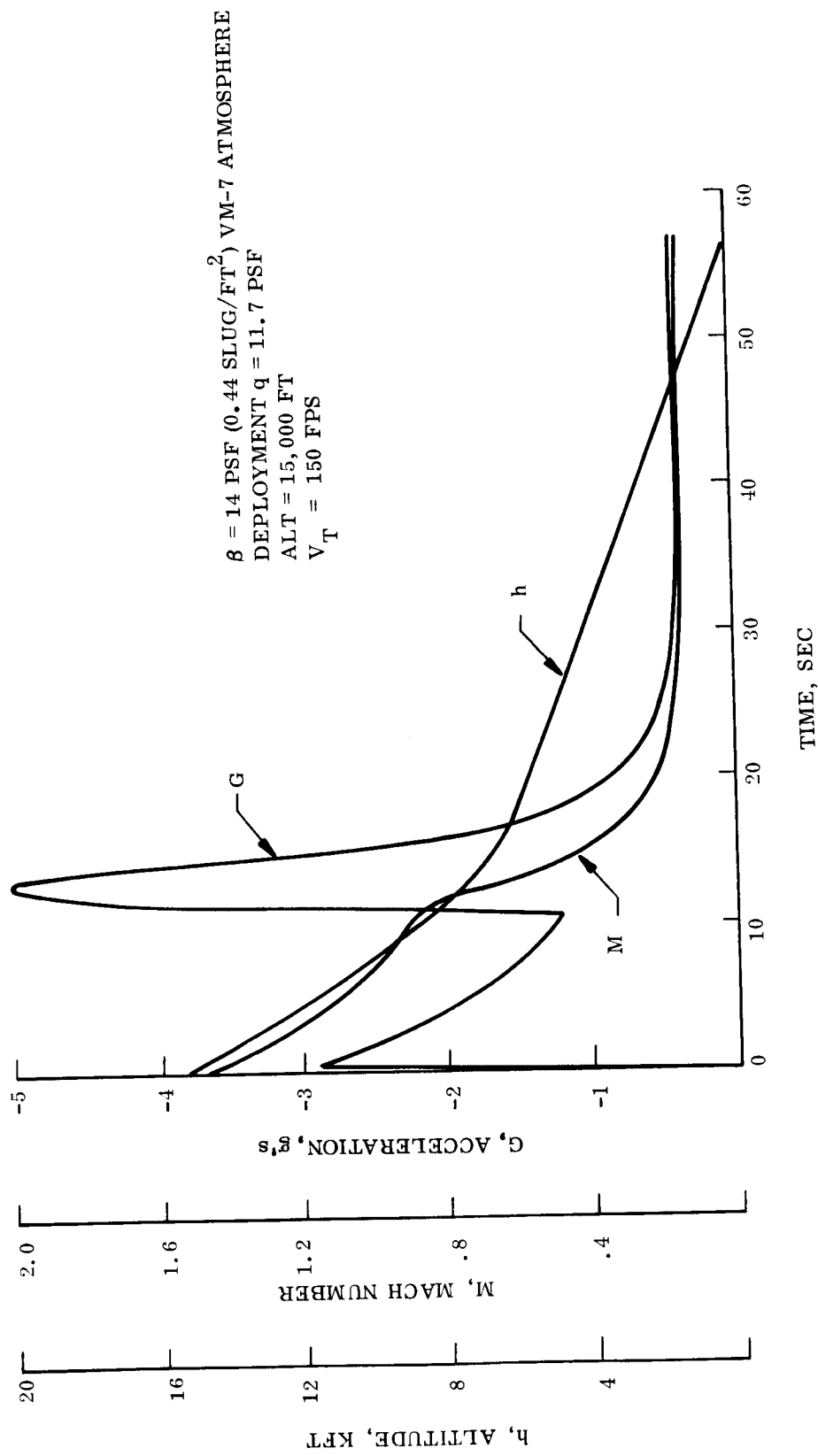


Figure 3.2.3-3. Parachute Descent Trajectory, Out-of-Orbit Entry, $V_T = 150 \text{ fps}$

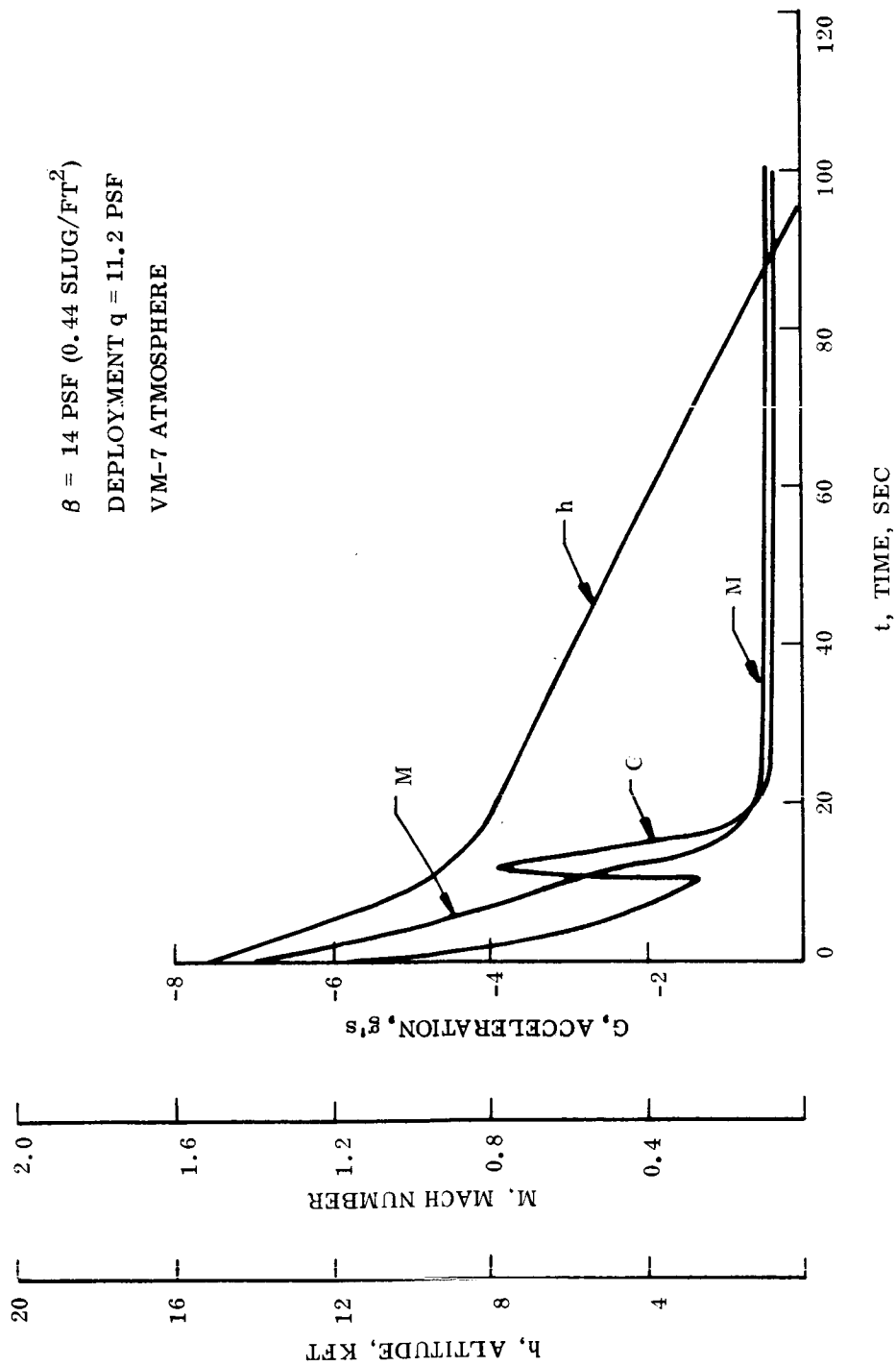


Figure 3.2.3-4. Parachute Descent Trajectory, Out-of-Orbit Entry, $V_T = 100 \text{ fps}$

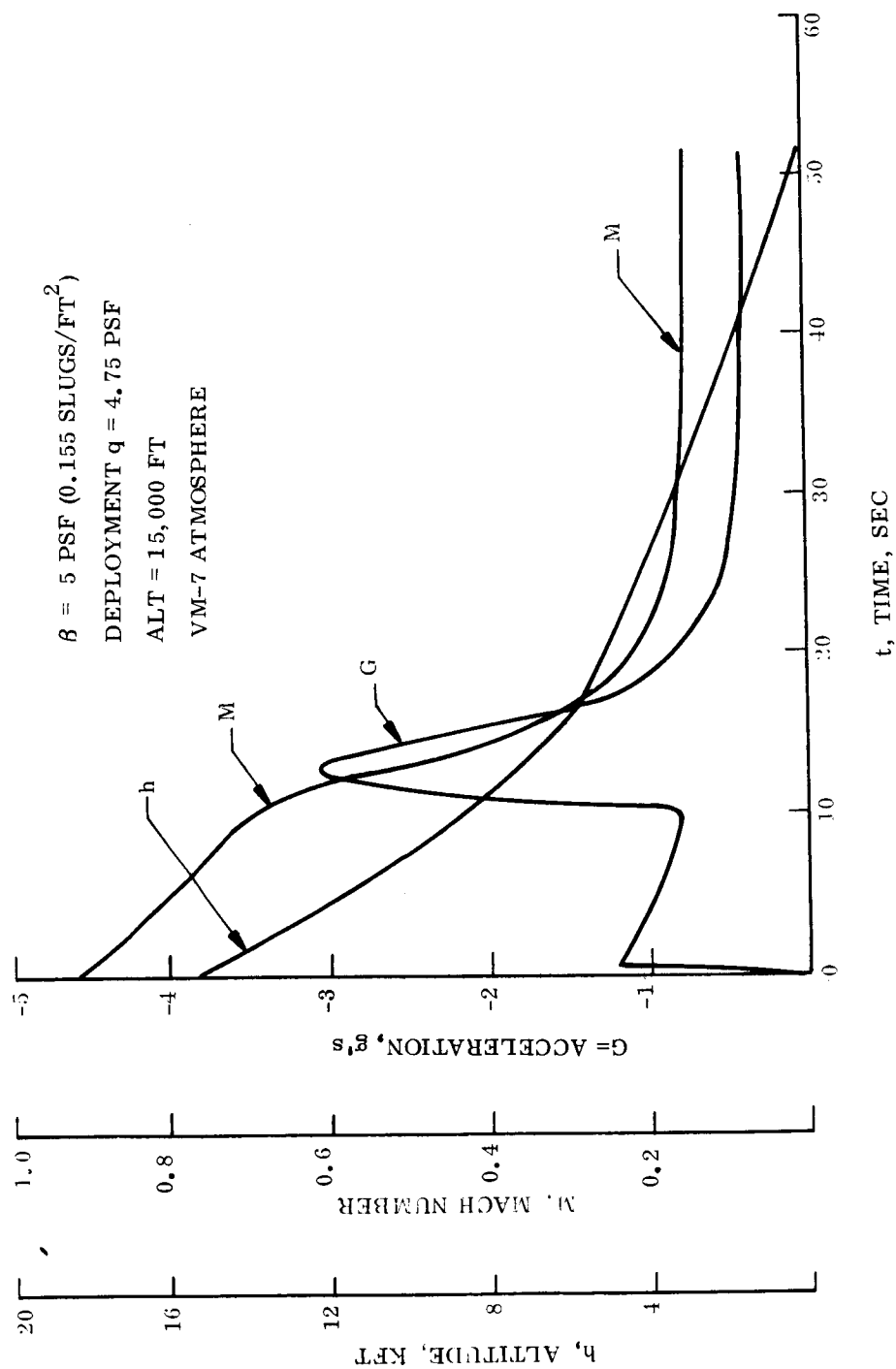


Figure 3.2.3-5. Paracnute Descent Trajectory, Direct Entry, $V_T = 150 \text{ fps}$

It is apparent that to provide terminal velocities higher than 100 fps in VM-7, the parachute deployment altitude must not be less than approximately 20,000 feet.

For these studies, the parachute reefing ratio and reefing time used for the alternate atmosphere evaluations were as selected in the parametric RESSEP parachute design program. Detail design optimization of both reefing ratio and time will better indicate minimum recovery altitudes. Results of these studies are discussed in the Retardation Subsystem sections of Volume IV for each point design.

3.2.3.3 Deployment Sensing

Because of the uncertainties in the Martian atmosphere, most of the sensing systems used for initiating parachute deployment in the Earth's atmosphere are unacceptable for initiating deployment in the Martian atmosphere.

Reference 3.69 presented an analysis of sensing systems to select a system for initiation of parachute deployment in the Martian environment. The criteria for determining whether a deployment was acceptable or not were that the deployment not occur at too low an altitude or at too high a Mach number. The same two criteria apply for this study but an additional constraint has been added. Contact between the Orbiter and the Lander is desired throughout the entry phase (including impact) and thus large tolerances on entry time are not acceptable. Reference 3-69 analyzes sixteen different sensing systems and presents an additional eleven systems which are not analyzed. The conclusion is drawn in the reference that the best sensor system is one which senses the ratio of base pressure to axial acceleration. Base pressure may be expressed as:

$$P_B/P_\infty = f(M, Re, \text{Body Shape})$$

and axial acceleration as:

$$a_x = (a C_D A/m) = (\gamma/2) P_\infty \frac{M_\infty^2}{m} C_D A$$

and

$$\begin{aligned} \frac{P_B}{a_x} &= \frac{(P_B/P_\infty) P_\infty}{(\gamma/2) P_\infty M_\infty^2} \frac{m}{C_D A} \\ &= \frac{(P_B/P_\infty)}{(\gamma/2) M_\infty^2} \frac{m}{C_D A} \\ &= \frac{2 K}{M_\infty^2 \gamma} \end{aligned}$$

It is apparent that this parameter is essentially a Mach number indicator for a given entry vehicle geometry and ballistic coefficient. Slight variations in the specific heat ratio for the various model atmospheres result in a small deviation in Mach number.

The use of this parameter to signal the parachute deployment will assure that a limit deployment Mach number will be maintained. If the value of β is chosen properly, deployment can be initiated at a high enough altitude. However, with the entry time constraint that is now imposed, this system by itself is not adequate. Table 3.2.3-1 shows the altitude at which deployment would occur if the sensor were set to deploy the chute at Mach 2 in VM-8. For the deployment at $h = 15,900$ ft in VM-8 total entry time would be on the order of 400 sec while deployment at $h = 96,600$ ft in VM-9 results in entry time of about 1200 sec. This 800 sec spread in entry time is unacceptable. Therefore all of the systems in ref. 3-58 were re-evaluated in an attempt to find a system which would reduce the spread in entry time. This re-evaluation was done using GE generated out-of-orbit trajectories. Since the atmospheric pressure and density are so different in VM-8 and VM-9, virtually all the sensors except those employing radar altimeters have this same deployment altitude problem. All of the systems which employ pressure sensors (either stagnation or base pressure) would show similar spreads in deployment altitudes between VM-8 and VM-9. In the VM-9 atmosphere the vehicles reach terminal conditions at very high altitudes ($h = 90$ kft) while in VM-8 they reach terminal at low altitude (12 kft) and thus the acceleration matrix, inertial path angle, towed body and any acceleration sensors will present large discrepancies in deployment altitudes. Time after maximum acceleration is ruled out also because in VM-8, about 56 sec lapsed between maximum acceleration and $M = 2$ and in VM-9 this amount of time will result in parachute deployment at about 100 kft. Some time function of maximum acceleration might work but the equation could become too complex. (A linear type equation was checked and showed large discrepancies in deployment altitude.) Insufficient data are available to fully evaluate an oscillation encountered, but due to uncertainties in aerodynamic moment coefficient and other parameters it does not appear as though it would constitute a very reliable sensor. Deployment at some fixed altitude in all atmospheres would greatly reduce the spread in entry time and thus systems using radar altimeters appear very attractive. With the radar altimeter, however, there is danger of deployment at too high a Mach number. This problem could be overcome by (1) coupling it with a P_B/A_x sensor or (2) using a sufficiently low β so that in all atmospheres the Mach number x would be sufficiently low at the chosen deployment altitude. The latter solution may not be acceptable because it may not be possible to package such a low β vehicle and thus it is recommended that a radar altimeter be coupled with a P_B/A_x sensor for initiating parachute deployment.

3.2.3.4 Design Impact Conditions

The attitude and velocity of the vehicle at impact must be known so that proper impact protection can be provided. The impact velocity can be easily obtained from the terminal velocity of the vehicle and the wind velocity. However, the impact attitude is not easily obtained in this case because of the high wind conditions which may prevail on Mars. Thus it becomes necessary to devise a method for simulating the motion of a parachute-payload system in the presence of wind shears and/or gusts.

In order to fully simulate the relative motion of a parachute-payload system requires a two-body six degree-of-freedom simulation with non-rigid attachment of the bodies, multi-attach points, suspension line wrap-up, and other factors. However, work done by many people has shown that good approximations of motion can be obtained with simplified simulations. Since rolling motion is not generally a major factor during

TABLE 3.2.3-1. DEPLOYMENT ALTITUDES FOR VARIOUS ATMOSPHERES,
MACH = 2.0

Deployment Altitude using P_B/A_x Sensor	
$V_e = 15,300$	
$\gamma_e = 16^\circ$	
$\beta = 16 \text{ PSF, (0.5 slug)}$	
Atmosphere	Altitude at $P_B/A_x = .0742 \text{ PSF/Ft/sec}^2$
VM 1	47,800
VM 2	24,700
VM 3	64,700
VM 4	33,400
VM 5	80,000
VM 6	47,400
VM 7	31,400
VM 8	15,900
VM 9	96,600
VM 10	64,400

parachute descent the first obvious simplification is to a two-body three degree-of-freedom simulation. This type of two-body program is now in general use. However, to obtain quick but reasonably good results even simpler programs can be used. The work done by Moog in ref. 3-70 using a two-body three degree-of-freedom program has shown that for relatively large chutes there is very little relative motion between the payload and the parachute. Therefore, it should be possible to obtain good results using a one body three (or six) degree-of-freedom program as long as mass properties are computed for the combined system and the apparent mass of the chute is handled properly. This type of approach has in essence been used by White and Wolf in ref. 3-71 with five degrees-of-freedom (no roll moment). This approach, simplified to three degrees-of-freedom, was used in the Mars Hard Lander study.

In order to further demonstrate the validity of this approach an attempt was made to simulate one of the parachute tests done in the NASA/Langely Planetary Entry Parachute Program (PEPP). At best it is difficult to simulate a parachute drop because of the uncertainties in the aerodynamic characteristics and in mass properties, but the intent was to show that the amount of oscillation predicted was of the same magnitude as that actually observed. Fig. 3.2.3-6 shows the attitude-time history obtained in the drop test discussed in ref. 3-58. This attitude-time history was obtained when the vehicle traversed the wind profile shown in fig. 3.2.3-7. Fig. 3.2.3-8 shows the altitude-time history so that a correlation can be made between figs. 3.2.3-6 and -7. Superimposed on fig. 3.2.3-6 is the attitude-time history from the simulation. The simulation was initiated a few seconds prior to the earliest time shown on fig. 3.2.3-6 but the oscillations were regular and the response of the simulation to perturbation was desired in this study. The correlation between the actual and predicted is extremely good between 80 and 105 sec as far as magnitude is concerned but the frequency of oscillation is somewhat different. By making adjustments to the mass and aerodynamic characteristics, the frequency could probably also have been matched. From 105 sec to the end of the run the correlation is not quite as good but is still acceptable and slightly conservative (i.e., predicts higher oscillation than actually occurs). It should also be noted that in the region around 150 sec, where the actual drop encountered a perturbation which altered the normal oscillatory motion, the simulation also encountered a rather similar change in oscillation amplitude. Further, at the end of the run the actual data shows a diverging oscillation which the simulation also reveals. Thus it can be concluded that the proposed method does in fact simulate the motion of a parachute-payload combination under the influence of a wind profile.

This simulation technique was then used to determine the motion of the Mars Hard Lander parachute-payload system in the postulated Martian environment. Since the most severe wind profiles are encountered in the VM-7 and VM-8 atmospheres, these atmospheres were chosen for use in this study. Most of the structural analysis on the Hard Lander has been done using results from the VM-8 atmosphere model and thus the impact study was initiated using the VM-8 atmosphere. Most of this study was done using a 391 lb Lander attached to a 79.8 ft Ringsail parachute but the effects of chute size were also investigated. Figs. 3.2.3-9 and -10 show the nominal winter wind profiles (fig. 3.2.3-9 is high altitude, fig. 3.2.3-10 is low altitude) used in this study with possible wind shear superposed. Wind gusts of 0.1 ft/sec per ft were also considered. The parachute trajectories were initiated at an altitude of 16,000 ft on an out-of-orbit no-wind trajectory for a vehicle with a ballistic coefficient of 16 psf (0.496 slug/ft²). The starting conditions are of little consequence to this study because with the large parachute the velocity and path angle are changed rather abruptly to the parachute terminal conditions.

Fig. 3.2.3-11 shows the response of the parachute system to the nominal wind profile. Shown on fig. 3.2.3-11 are the Capsule attitude (angle between Capsule center-line and the local vertical) and capsule angle-of-attack (angle between Capsule center-line and the velocity vector) as a function of altitude. The Capsule attitude is the angle which has direct bearing on the Capsule design. The Capsule angle-of-attack is also shown because parachute canopies have a tendency to collapse at large angles-of-attack (>40 degree). For the nominal wind, it can be seen that both the angle-of-

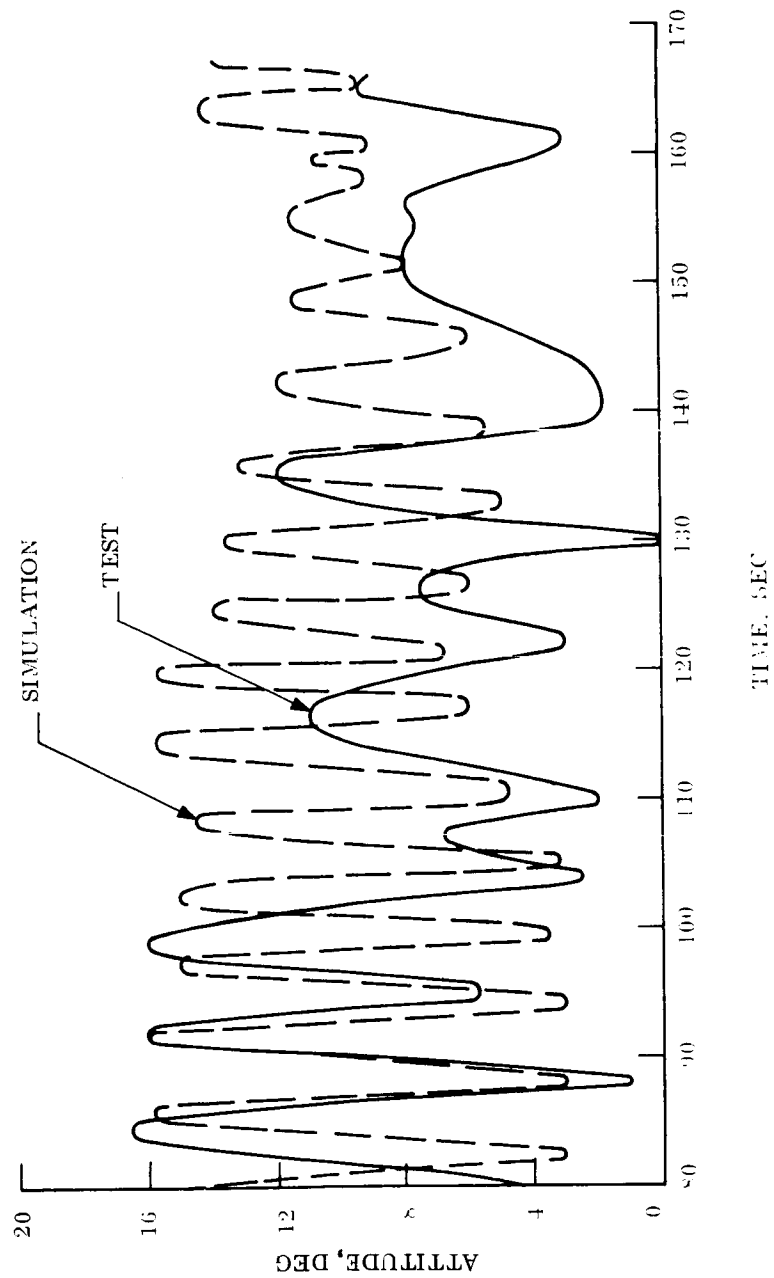


Figure 3.2.3-6. PEPP Test Simulation, Altitude Angle vs Time

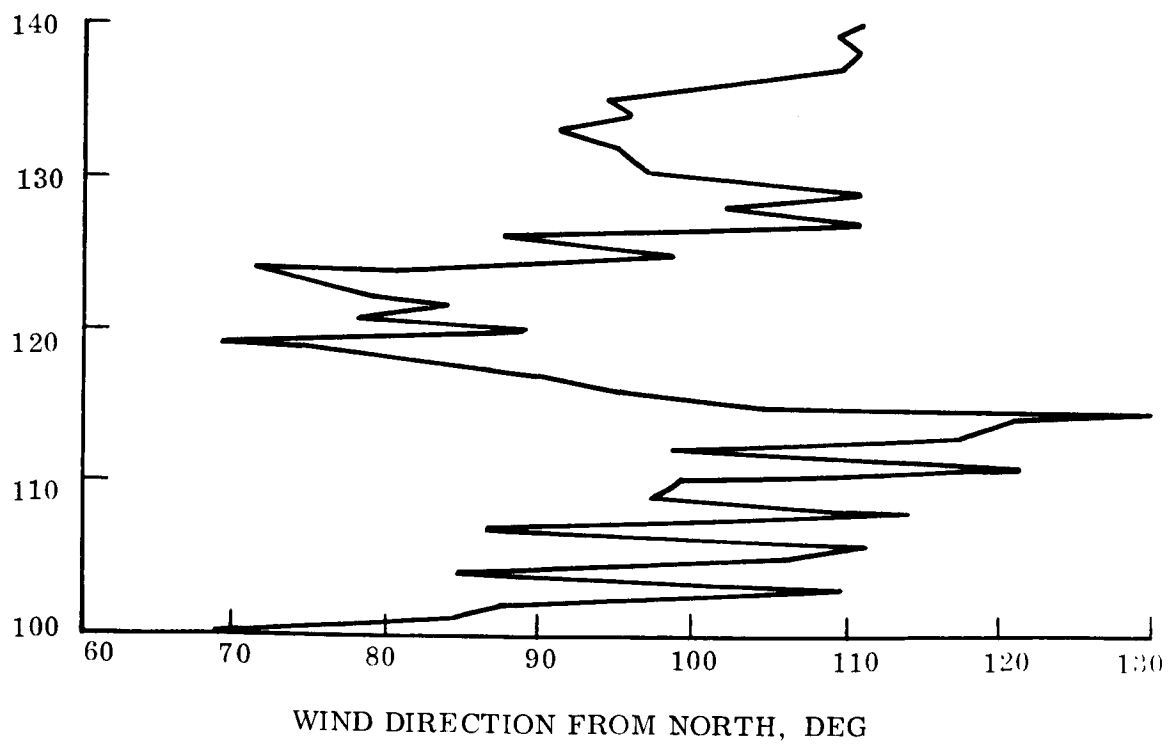
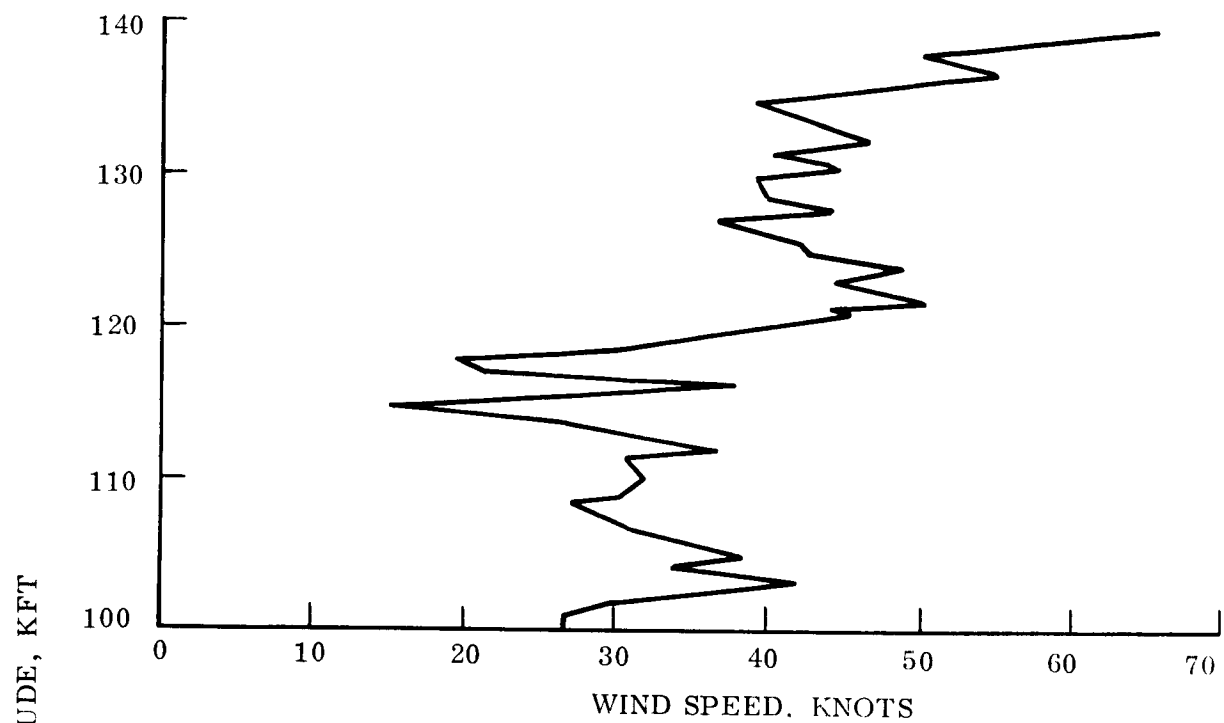


Figure 3.2.3-7. PEPP Test Wind Profile

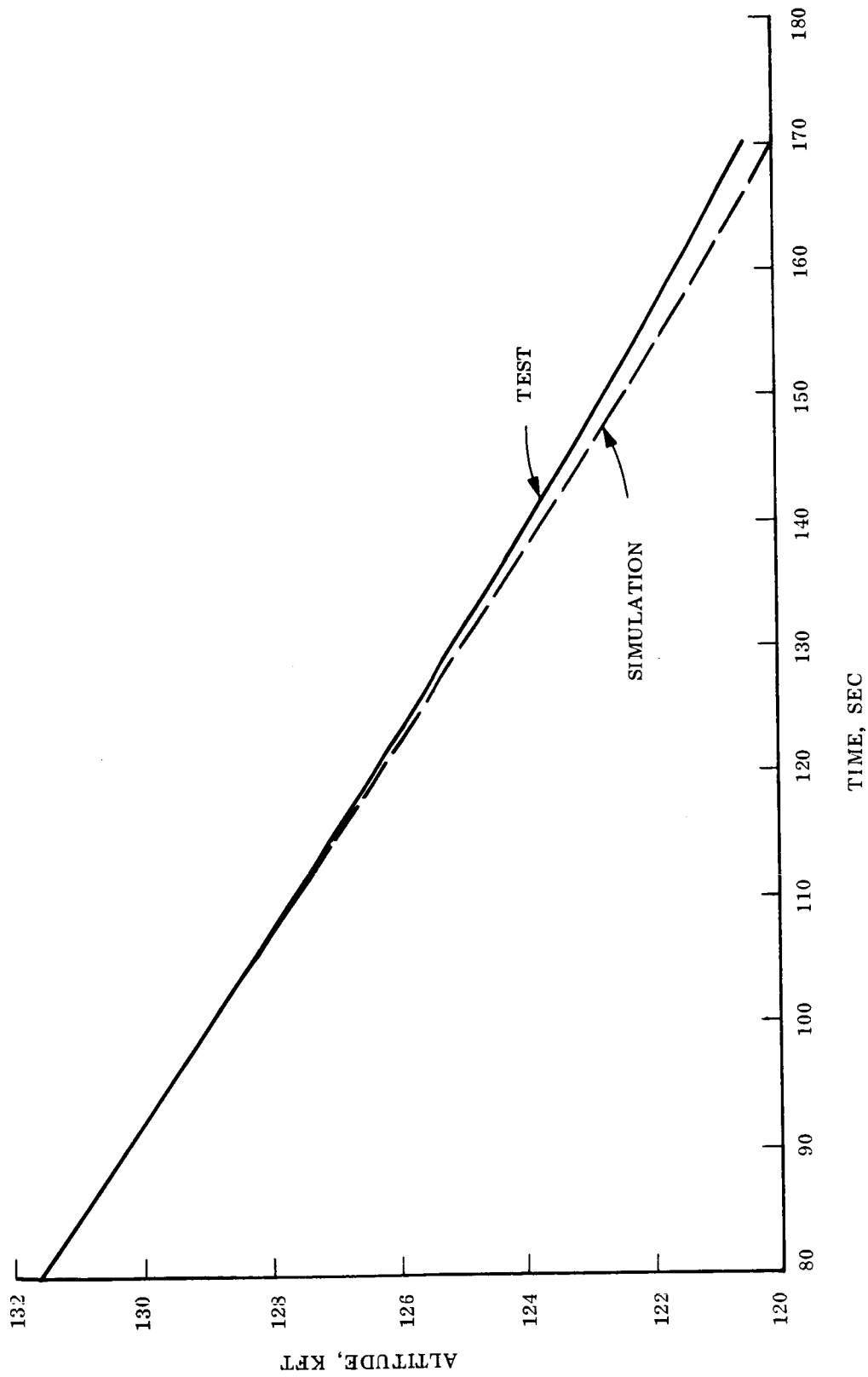


Figure 3.2.3-8. PEPP Test Simulation, Altitude vs Time

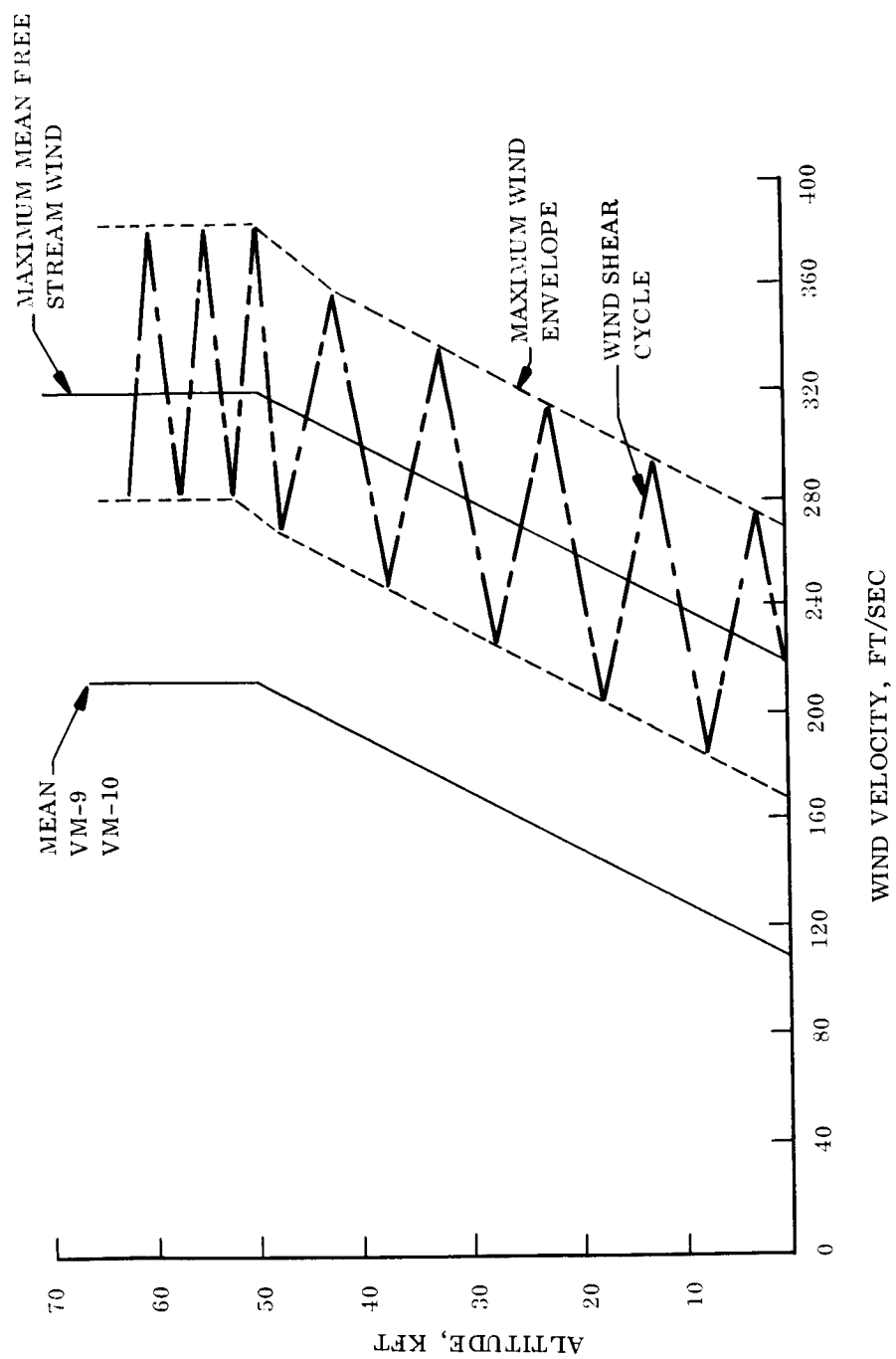


Figure 3.2.3-9. Entry Wind Profile for VM-7 and VM-8 Atmospheres

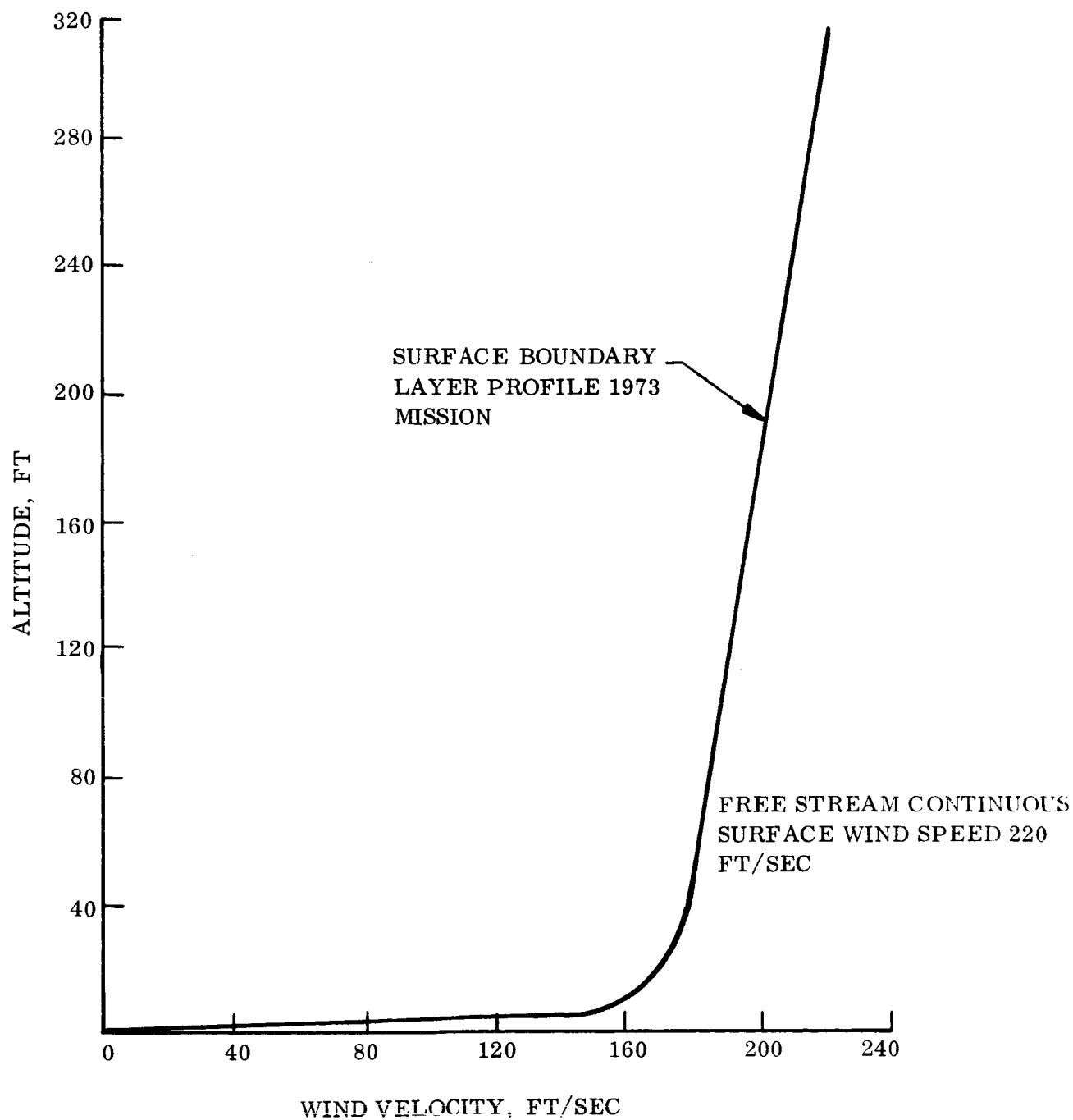


Figure 3.2.3-10. Surface Boundary Layer Profiles for VM-7 and VM-8 Atmospheres

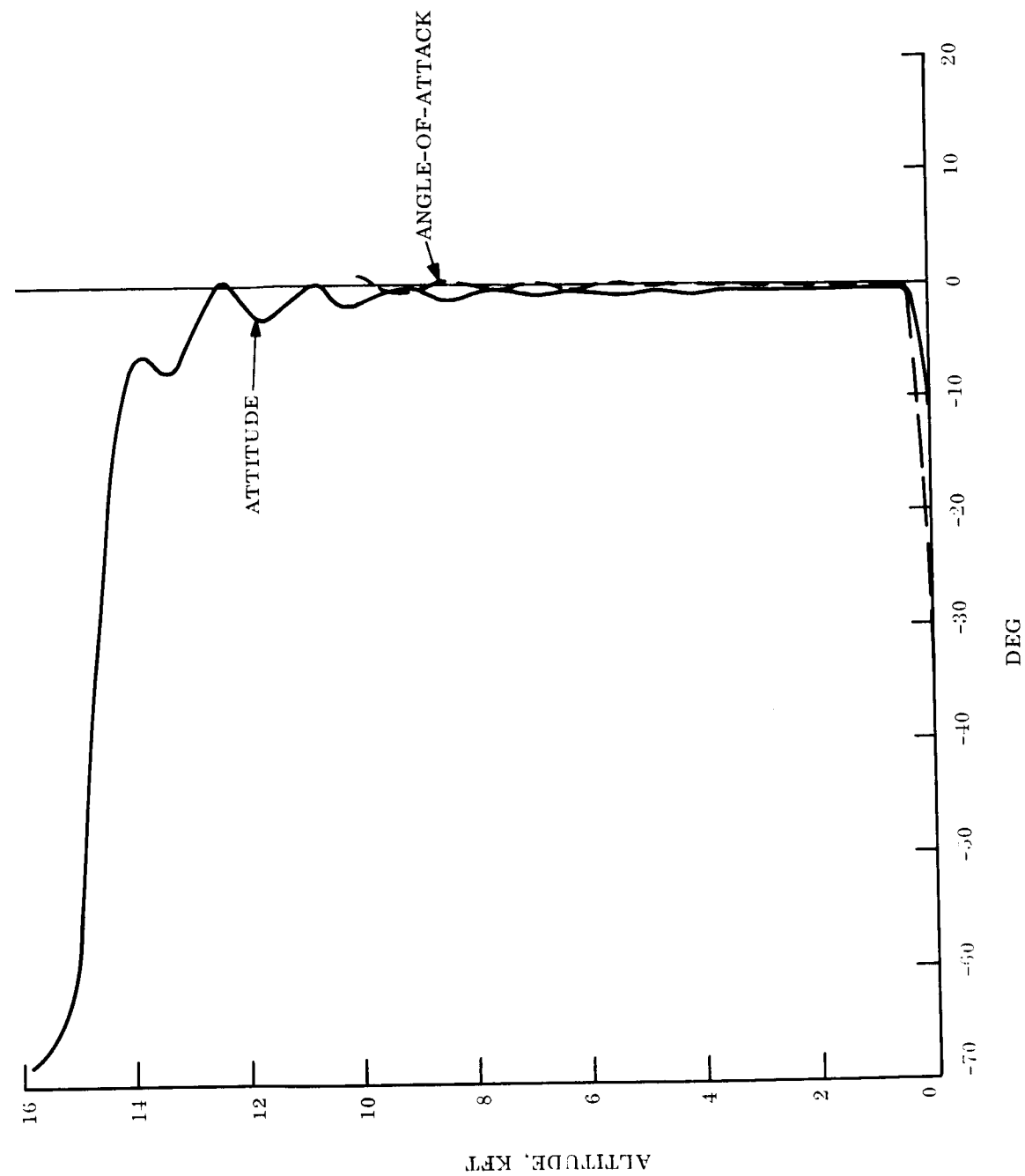


Figure 3.2.3-11. Parachute Response to Nominal Wind

attack and the attitude adjust quickly to the wind and remain at low values until the boundary layer near the surface is encountered. The parachute angle-of-attack goes to 40° at the surface but parachute collapse at this low altitude would not be a serious problem. There is not enough time for the capsule to respond to this wind shear (i. e., boundary layer) and thus the attitude angle remains low.

Fig. 3.2.3-12 shows the angle-of-attack and attitude that the parachute-lander system would traverse when encountering a wind shear at low altitudes. The vehicle traverses the nominal wind profile down to an altitude of 10,000 ft, where the wind velocity profile is made to follow the wind shear cycle shown in fig. 3.2.3-9 down to impact. Neither angle-of-attack nor attitude demonstrate large oscillations with the largest perturbations again encountered in the near surface boundary layer.

Fig. 3.2.3-13 shows the effects of wind gusts on the parachute-lander system. In this instance the nominal wind profile was used down to 2000 ft where a gust was applied with an onset rate of 0.1 ft/sec per ft. The vehicle was assumed to impact with the gust still increasing. The angle-of-attack does not have large oscillations ($>10^\circ$) until near the surface, but the attitude angle reaches almost 38° at about 1200 ft. Because of the high attitude angle acquired in this gust condition, this wind profile was used to study the effect of parachute size. Fig. 3.2.3-14 shows the attitude angle for three different parachute sizes. All reach about the same attitude angle but the large chute responds faster and reaches its peak attitude angle at a high altitude.

Because of the severity of the wind gust case it was decided to investigate the effect of wind gust from a zero wind (instead of from the nominal wind). In this case, nominally descending with no wind, the system encounters a gust at 4000 ft with an onset rate of 0.1 ft/sec per ft. At 2000 ft the wind gust starts to relax, decreasing at a rate equal to the onset rate. Both the initial onset and the subsequent relaxation resulted in peak attitudes of 35 to 40° as can be seen in fig. 3.2.3-15.

The VM-7 atmosphere is less dense at the low altitudes than VM-8 and thus a vehicle would have a higher terminal velocity in VM-7 for the same size chute than it would in VM-8. A trajectory was computed for the vehicle descending the VM-7 atmosphere to determine how this increased descent rate with a lower density atmosphere might affect the motion of the system. Only the nominal wind was investigated and it can be seen in fig. 3.2.3-16 that the motion showed the same trends as the motion in VM-8.

It now appears as though the Lander will reach Mars at a time near the summer solstice and the high winter winds considered thus far would not occur. The nominal summer wind is 118 ft/sec at 330 ft and increases the rate of 2 ft/sec per 1000 ft up to an altitude of 50 fkt. The wind gust has the same onset rate but has a 66 ft/sec limit instead of 200 ft/sec limit for the winter winds. A trajectory was computed for the vehicle traversing a summer wind profile in the VM-8 atmosphere.

The specific profile traversed was the nominal summer wind down to 2000 ft where a gust onset begins. The gust reaches its peak about 1350 ft and holds that

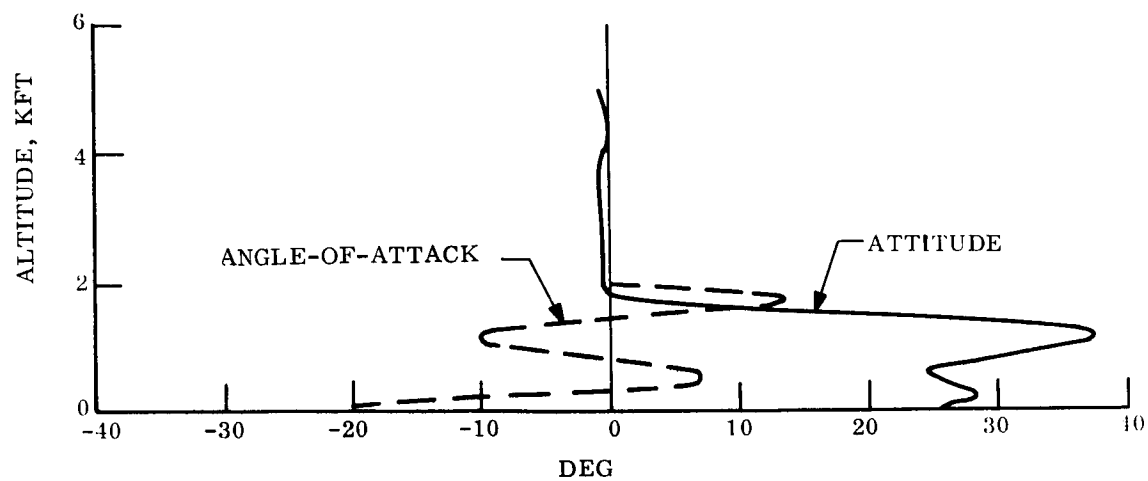


Figure 3.2.3-12. Parachute Response to Wind Shear

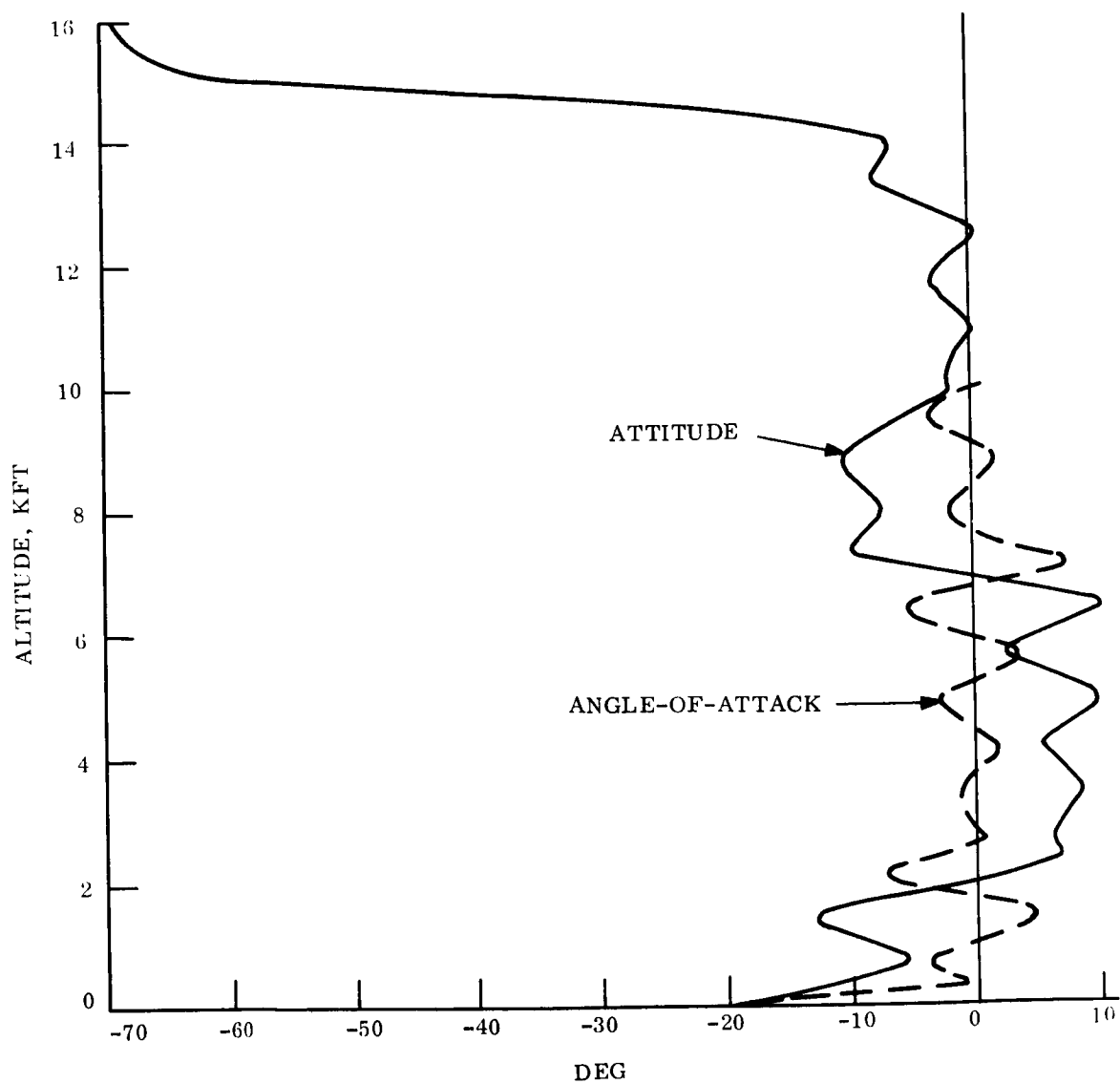


Figure 3.2.3-13. Parachute Response to Wind Gust

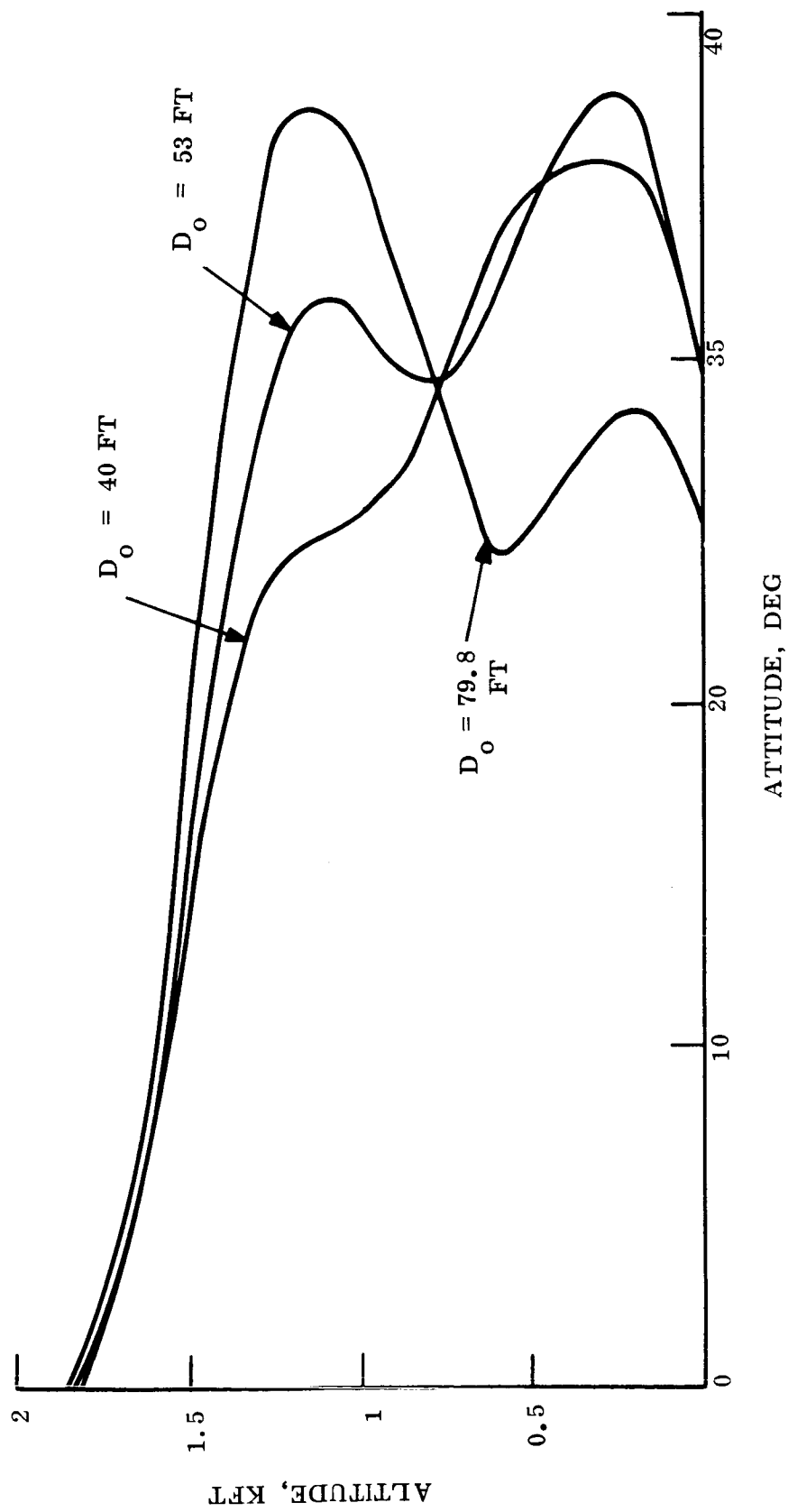


Figure 3.2.3-14. Effect of Parachute Size on Attitude During Wind Gust

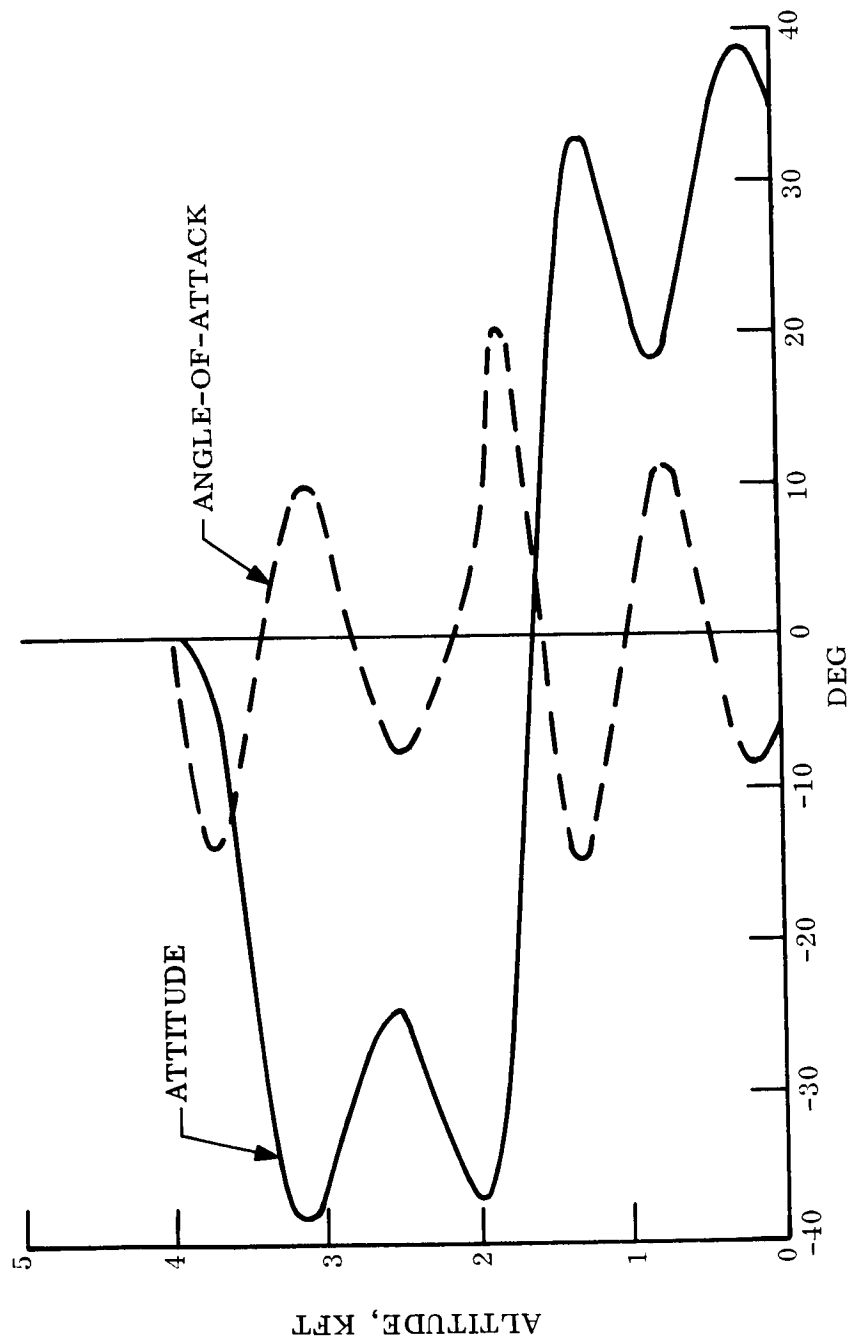


Figure 3.2.3-15. Parachute Response to a Wind Gust from Zero Initial Wind

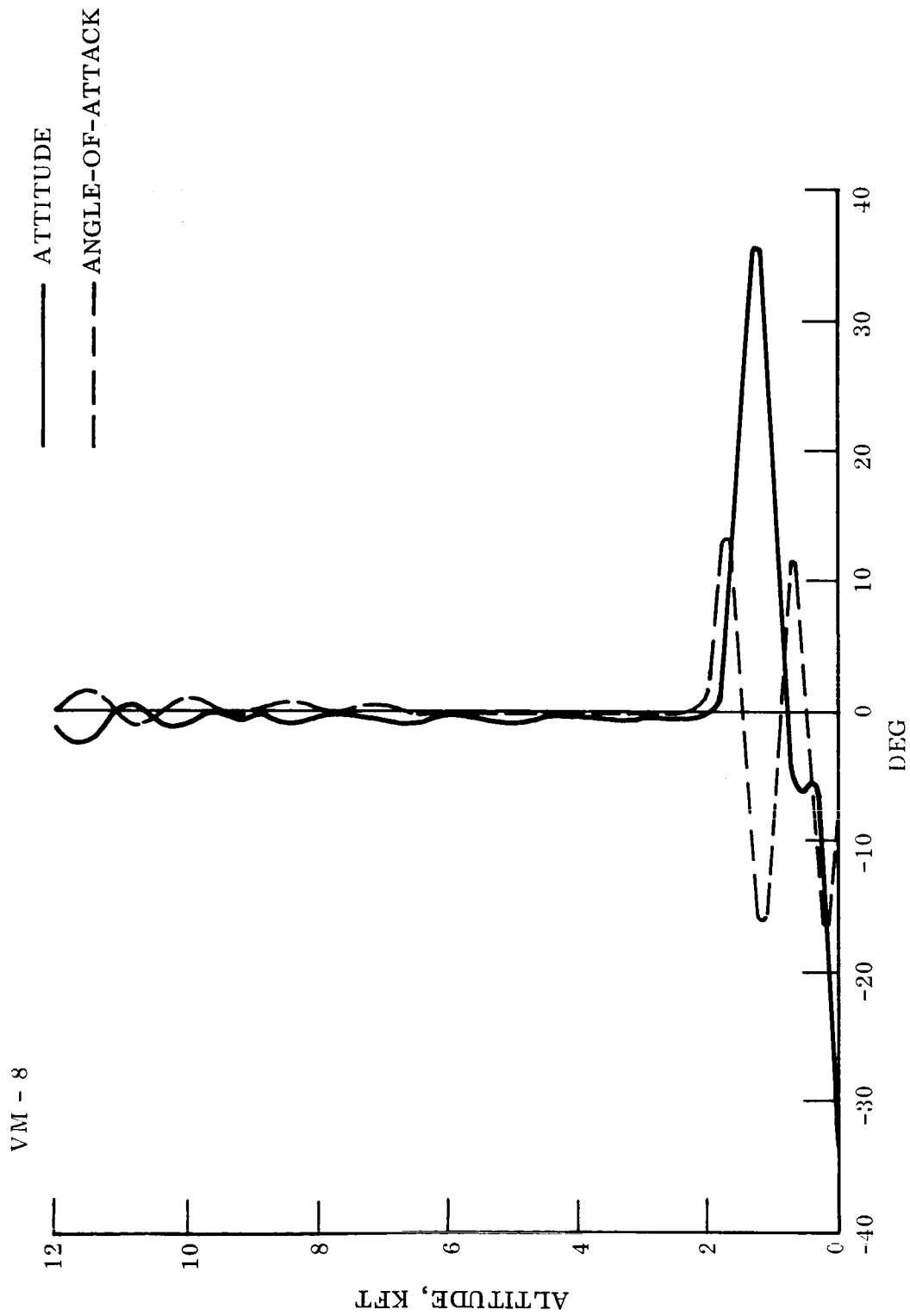


Figure 3.2.3-16. Parachute Response to Nominal Summer Wind

value to 650 ft where it begins to relax. It can be seen in fig. 3.2.3-17 that trans-
versal of this wind profile results in attitude angles of the same magnitude as those
encountered in the winter wind profiles.

From the results of this study it appears that the Lander must be designed to
survive impact with the Martian surface with attitude angles as high as 40 deg from
the vertical. The vertical impact velocity is a function of parachute size with the
nominal being 75 to 200 ft/sec. The maximum horizontal velocity would be on the
order of 420 ft/sec in the winter wind gust case (185 ft/sec for the summer wind gust
case) but would be slightly higher if impact occurred on a hill. This attitude angle
would also be increased if impact occurred on the side of a hill or mountain, where
the slope angle would be directly added to the descent attitude.

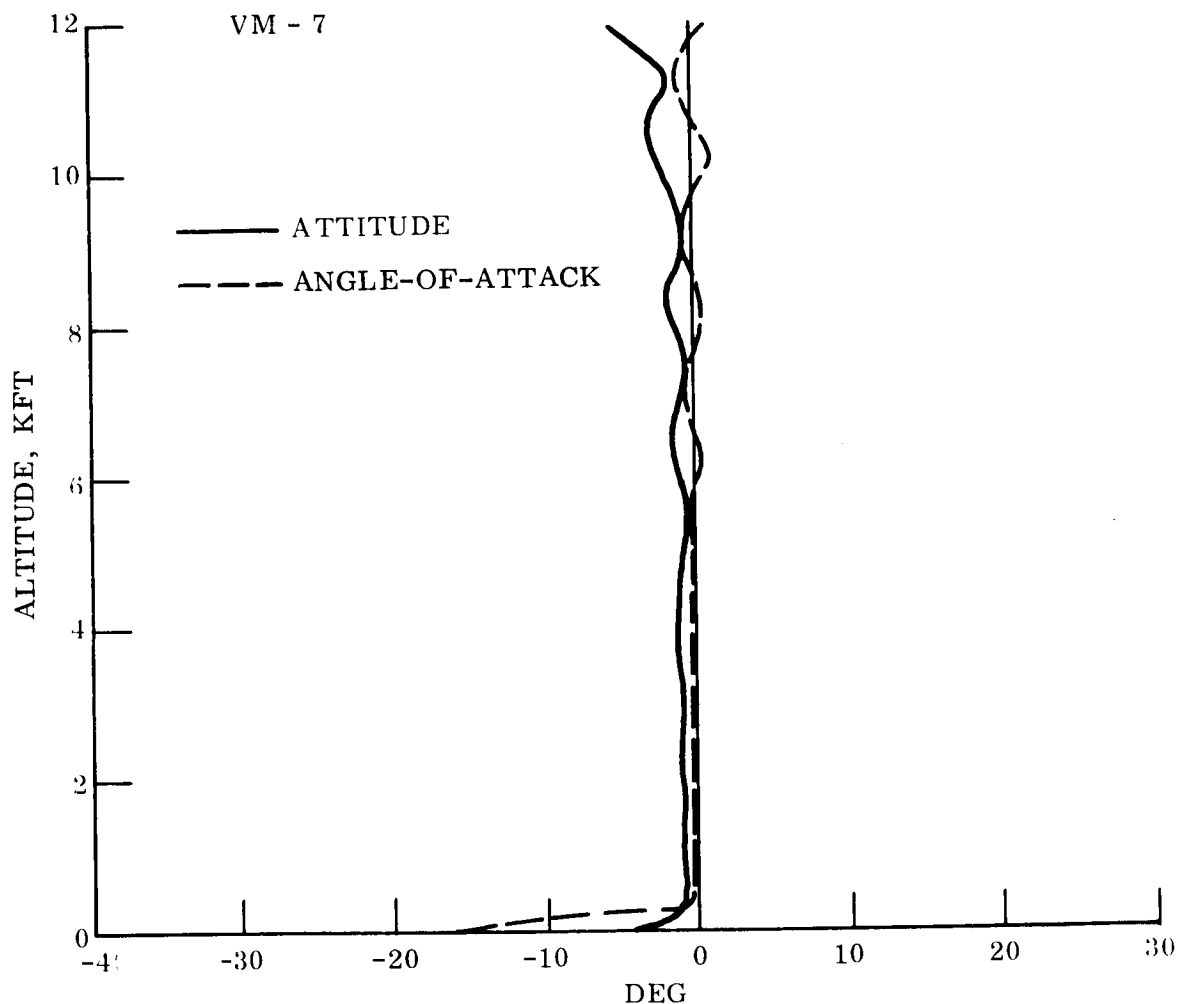


Figure 3.2.3-17. Parachute Response to Nominal Winter Wind

3.2.4 SUBSYSTEM TRADE-OFF STUDIES

3.2.4.1 Subsystem Description and Design Constraints

As discussed earlier in this report, the modified Ringsail parachute has been used as the main and pilot parachutes in the parametric studies and the point designs. The extensive usage and data accumulated with this parachute design in all manned NASA entry systems is considered to justify the choice. However, other candidate parachute types such as the Disc-Gap-Band and the Cross types may ultimately prove to be superior because of simpler design and fabrication characteristics. Tests to date have appeared to indicate favorable performance characteristics for these latter types.

Parachute materials assumed in the current designs were selected to be Dacron. Because of part sterilization environment tests, two fabric materials have demonstrated adequate resistance to thermal and vacuum environments. These are Dacron and Nomex. Since Dacron is readily available in a wide assortment of standard parachute materials, this material is recommended for selection to the present application, so that an optimization of parachute weight is possible.

By proper selection of weave type and yarn denier it is considered that material weights of from 1.1 to 1.25 oz/yd² will provide sufficiently low porosity and adequate strength to be used as the minimum weight material for the present designs.

In parametric studies, parachute sizes are limited to 100 ft diameter. This is based upon a long standing limitation determined by practical physical sizes and weights capable of ground handling, and has historically been of importance due to general re-usability of parachutes in many applications. In the point designs, however, the complexity and reliability of multiple parachute deployments and clustered performance prediction is weighed against fabrication and handling problems associated with a "one-shot" parachute application such as the current one. A single 127 ft diameter modified Ringsail parachute is specified for Point Designs 1 and 2A for example, rather than two smaller (90 ft) parachutes, each requiring a mortared pilot parachute. A single mortared pilot is required for the single large parachute extraction and deployment.

The deployment and opening of large parachutes having diameters up to 200 ft are not expected to introduce problem areas which are not normally encountered in the development of conventional Earth systems involving parachutes of the Apollo size or larger. Earth atmosphere tests have been conducted with a 189 ft diameter Ringsail parachute, and with a series of Ringsails having diameters in excess of 100 ft (Ref. 3-72). By the use of conventional pilot extraction and reefing techniques, orderly deployment and parachute opening is attained.

This was demonstrated in the tests reported in refs. 3-73 and -74 where near-minimum weight large Ringsail configurations were shown to withstand deployment dynamic pressures of 75 psf with little or no damage. Canopy specific drag area of from 47.3 to 52.2 ft²/lb were employed in these tests. Specific drag area is defined as:

$$\frac{\text{drag area}}{\text{wt of canopy and lines}} = \frac{C_{D_o} S_o}{W_p}$$

The specific drag area is a chute weight efficiency factor, which for the designs of refs. 3-73 and -74 applied to a dynamic pressure range of from 67 psf to 96 psf.

In the present application, parachute specific drag areas of the order of 35 ft²/lb are used for dynamic pressures with a maximum of 20 psf. Since, in general, textile material strengths vary directly with weight, the current hard Lander parachute designs appear to be conservative.

The main Ringsail parachutes in the recovery system designs are deployed by an extraction, or pilot parachute. The pilot, also a modified Ringsail, is ejected by means of a mortar which provides an ejection velocity adequate to overcome the vehicle deceleration and wake reverse flow regions. This is required to assure that the pilot chute will reach its line stretch and effective drag force to extract the main canopy. The determination of required ejection velocities have long been included as standard design practices by recovery system specialists.

Design ejection velocities required in the present study were determined from the GE cover or pack ejection computer program, described in Appendix A-5. A typical result of this program is shown in fig. 3.2.4-1 for a Mach 2 ejection of a one slug mass which would simulate a Mach 2 drogue ejection at recovery system initiation signal.

The recovery system initiation signal will result from a sensing system which consists of one or more of the following:

1. Base pressure to axial acceleration ratio.
2. Altitude sensor

As discussed previously, sensing parameter (1) will provide a Mach 2 limitation. In addition, an altimeter may be employed to equalize descent time regardless of the atmosphere encountered.

In order to provide a recovery system which will be compatible with either or both sensing capabilities, the drogue (or pilot) parachute is designed in the present system to be capable of extracting the main parachute under minimum dynamic pressure conditions (alternate atmosphere study) and taking into account the reduced wake energy resulting from the blunt forebody. Pilot parachutes are sized assuming a free-stream dynamic pressure of 20 psf, and a 1 sec time for main chute extraction. The time for main parachute extraction under the minimum dynamic pressure deployment condition of 2 psf is evaluated to determine the main parachute extraction time penalty. Parachute component snatch loads and opening loads are determined in accordance with standard design practices.

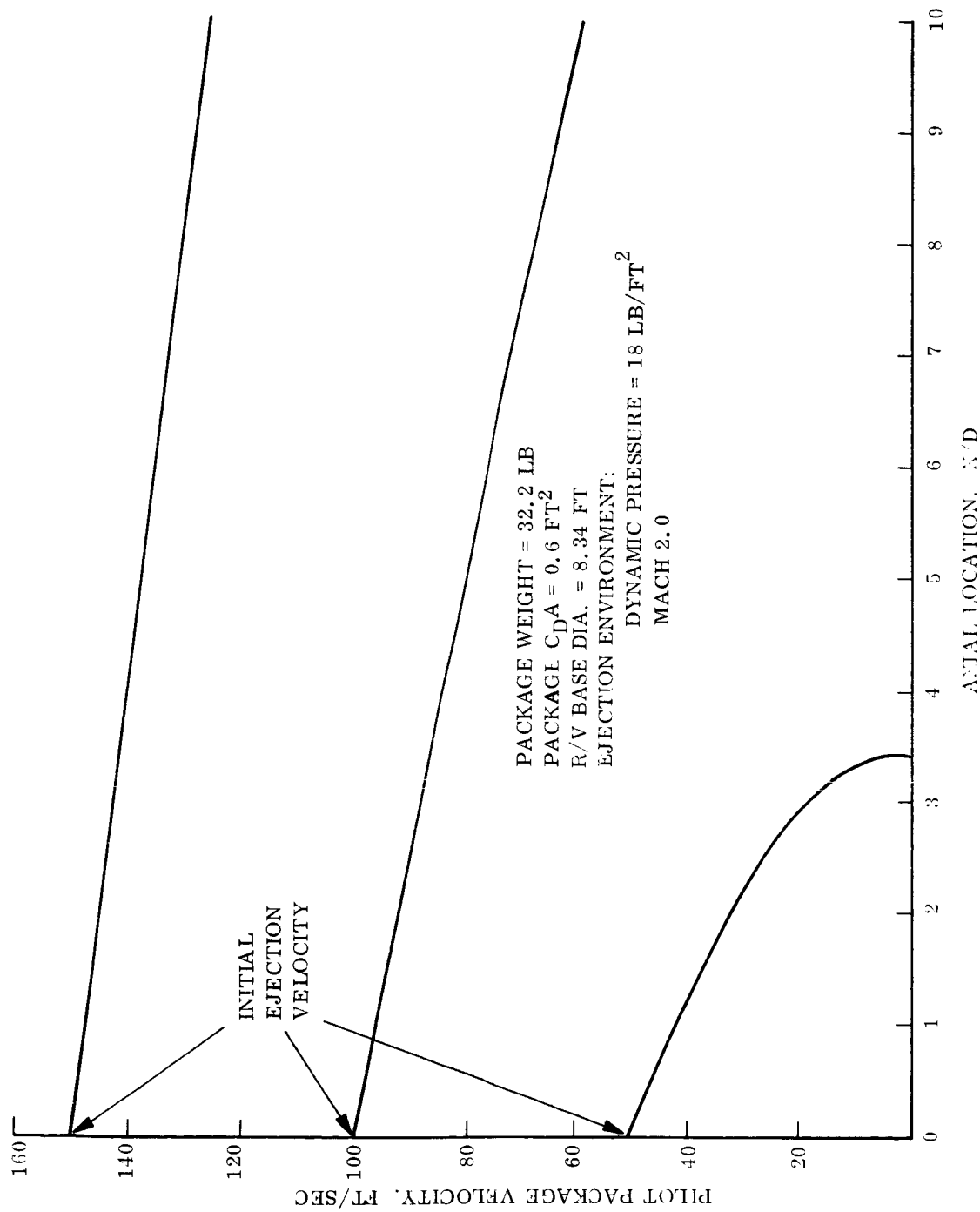


Figure 3.2.4-1. Variation of Package Velocity with Axial Wake Location and Initial Velocity

As an example of preliminary parachute component sizing and sequencing as described above, components and loads are shown for a system which is applicable to Point Designs 1 and 2A. For these configurations, a 15.7 ft pilot parachute extracts a 127 ft diameter main canopy for a desired terminal velocity of 100 fps in a model VM-7 atmosphere. Finalized recovery system designs are given in Volume IV.

3.2.4.1.1 Pilot Parachute

The pilot parachute is sized assuming a freestream dynamic pressure of 20 psf, a separation velocity of 100 fps, and a 1 sec time interval for main parachute extraction at Mach 2. A Ringsail extraction parachute with a drag coefficient of 0.52 is assumed. In addition, a 40 percent reduction in freestream dynamic pressure is used to represent the reduced energy wake of the blunt aeroshell at Mach 2 (refs. 3-67 and -68).

To evaluate the recovery system deployment time and loss of altitude during deployment under the low dynamic pressure conditions, calculations are performed in a manner similar to the high dynamic pressure case using the reduced freestream dynamic pressure and vehicle deceleration. In such cases the vehicle is at or near equilibrium velocities. Freestream dynamic pressure is reduced by 20 percent to reflect the subsonic energy loss in the vehicle wake (ref. 3-67).

To obtain the maximum force which will be seen by the parachute suspension lines and riser, two conditions are analyzed; the deployment snatch force and parachute opening load. Opening force calculations assume a maximum dynamic pressure of 20 psf, an opening shock factor of 1.05, and a load factor of 1.0. As shown in table 3.2.4-1 the opening force is largest, and is used for force calculations.

The number and length of suspension lines is roughly the diameter of the parachute considered and the force on them is (in the present example) the opening force divided by the number of lines; however, a factor of safety of 2.3 is included. These results are shown in table 3.2.4-1. Standard commercial lines of 375 lb strength (minimum used in study) are adequate. For all pilot chutes the riser length is between 5 and 6 vehicle base diameters. The force in the riser is determined as the opening force divided by 0.686, which is a strength reducing factor reflecting material stretching, abrasion, line angles, etc., loss factors. A safety factor of 1.5 is then applied to give the results shown in table 3.2.4-1.

3.2.4.1.2 Main Canopy

To minimize parachute system weight, main canopies are reefed. The reefed drag area ratio is considered to be a fraction of the full open canopy drag area, with the reefed drag coefficient (based upon the frontal reefed area) taken to be equal to the full open drag coefficient. For most Ringsail designs, the reefed drag coefficient (based upon the reefed projected area) is higher than the full open drag coefficient. The reefed area must be determined in development tests to provide the desired reefing ratio in a final system design. The use of past data in the preliminary design will result in a minimum adjustment of the reefing line length to provide the desired ratio.

TABLE 3.2.4-1. TYPICAL RECOVERY SYSTEM PRELIMINARY DESIGN SUMMARY. POINT DESIGN 1

LOADS								
Component	Do (ft)	Deployment Time (sec) q = 2 psf	Snatch Force (lb)	Opening Load (lb)	Suspension Line		Riser	
					Load (lb)	Length (ft)	Load (lb)	Length (ft)
Pilot Para-chute	15.7	3.56	910.	2180.	313.5	16	4770	38.6
Main Canopy	127.		8500	11,300	203.	127.	24,700	25.4

WEIGHTS

	Canopy and Suspension Lines (lbs)	Riser (lbs)	Total (lbs)
Pilot Parachute	3.1	2.1	5.2
Main Canopy	180.0	5.0	185.0
Bridles, Mortar, Attachment Hardware, Canisters, etc.			47.5
		Total	237.7

The reefing ratio in the present analyses is the ratio of the reefed area to the parachute nominal area and parachute filling times are based upon a linear area growth with time.

The general analysis for the main canopy loading is similar to that for the pilot. However, the main canopy size is determined on the basis of the decelerator load and desired terminal velocity. For Point Design 1 (and also 2A) a 127 ft diameter modified Ringsail is used. Since the main canopy is deployed reefed at a reefed area ration of 8 percent of full open drag area, opening loads are small, and minimum weight materials may be used. Results of the main parachute load calculations are shown in table 3.2.4-1.

3.2.4.2 Parachute Release Considerations

In order to insure that the canopy does not land on and cover the Lander after ground impact, various methods have been considered to prevent such an occurrence. The critical parameters involved in the problem solution are: (1) the short time between the Lander and parachute impacts, (2) the large main canopy diameters, and (3) the low dynamic pressures (low densities).

A number of solutions become apparent which will provide relative displacements of the Lander and parachute at ground impact. Some of these are as follows:

1. Release the parachute before impact ($h \approx 100$ to 200 ft)
2. Sense wind direction and accelerate payload upwind (e. g. , solid propellant rocket on Lander)
3. Sense wind direction and accelerate parachute downwind (e. g. , spilling the canopy by lengthening one of two risers, by rockets on canopy, or by the use of a lifting canopy, such as an all flexible parawing replacing the main canopy).

Although extensive studies of this problem are required to establish an optimum solution, the system designs of the current study assume that the parachute is released from the Lander before Lander impact. This is sensed either by an on-board altimeter, or by means of a drop line which would also serve to sense wind direction, as illustrated in fig. 3.2.4-2. Release of the parachute before landing is primarily based upon the non-wind, worst-case condition, but requires some form of altitude sensing, and a Lander impact attenuation design based upon a slightly higher impact velocity than would exist for a parachute release after ground impact.

Release of the parachute from the Lander at the instant that Lander is 100 ft above the ground, for example, will require slightly less than one sec for the Lander to impact. Neglecting drag on the Lander, its impact velocity would be about 112 fps assuming a 100 fps terminal at parachute release. Assuming that an 85 ft nominal diameter parachute is providing this terminal velocity and weighs approximately 85 lbs,

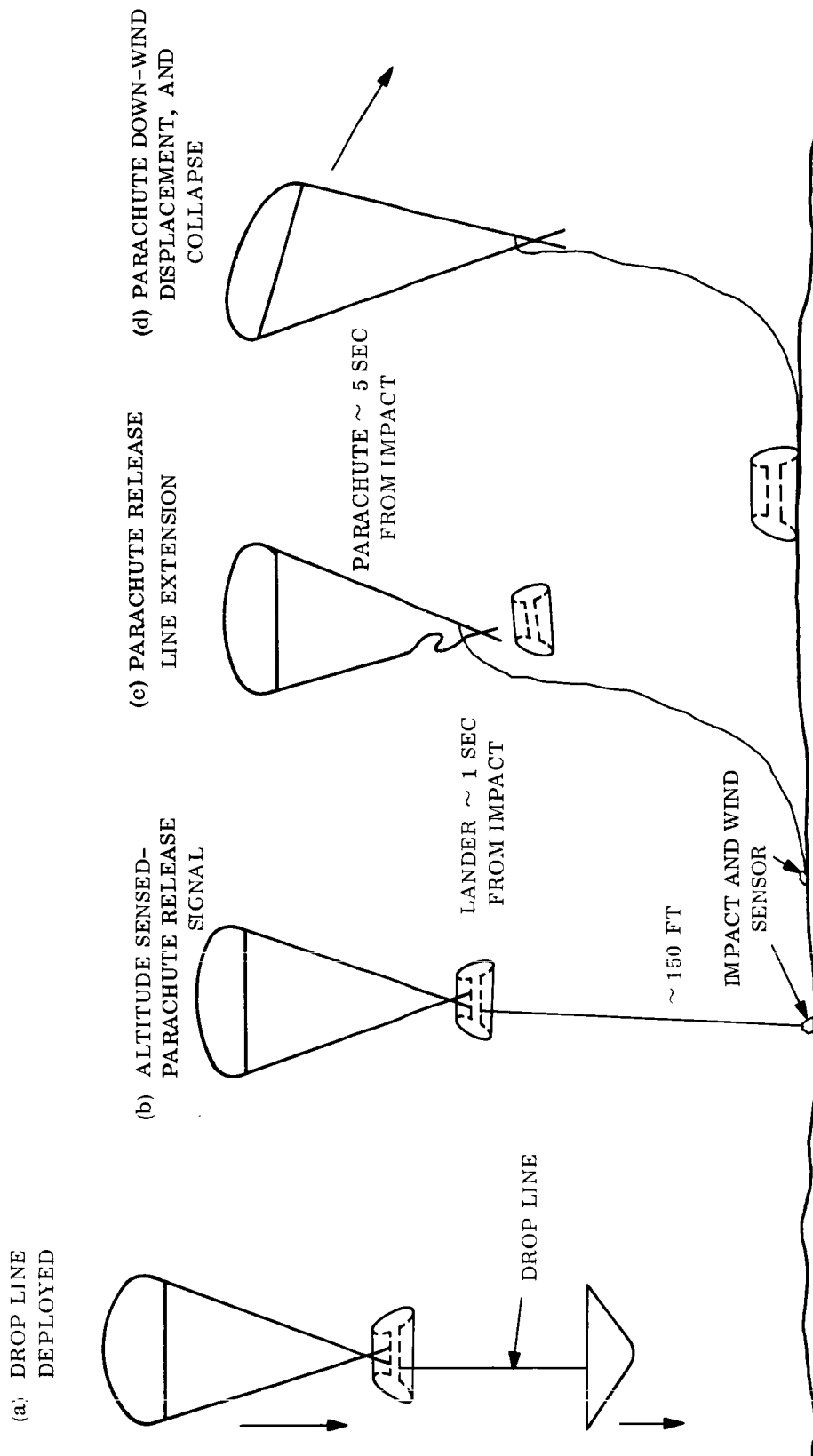


Figure 3.2.4-2. Parachute Release Concept

its $W/C_D A$ without the Lander is about 0.0113 psf (0.00035 slugs/ft²). With this as the terminal dynamic pressure, the parachute's terminal velocity in VM-7 is:

$$\begin{aligned} V_T &= \sqrt{\frac{2q_T}{\rho}} \\ &= \sqrt{\frac{2 \times 0.0113}{1.36 \times 10^{-5}}} \\ &= 41 \text{ fps} \end{aligned}$$

When the Lander is 100 ft above the ground the parachute center of gravity is about 185 ft above the ground and would require about 4.5 sec to impact. This would represent the maximum parachute alone descent time, assuming the full canopy $C_D A$ would remain effective to impact. This is probably unlikely. It is apparent that a significant time difference will exist in any case between the Lander and parachute impacts. A small wind or small sideslip of the parachute will result in significant horizontal range differences for the Lander and parachute impact points. Very small lateral impulses on the Lander and/or the parachute if directed properly relative to the prevailing wind will produce separation of the two objects at impact. In the case of no wind, the direction of impulse doesn't matter.

Studies of parachute-lander dynamics, discussed previously, have indicated that the parachute-lander attitude does not exceed 40 deg from the local vertical in response to wind profiles (shears) and/or wind gusts. Angular rotation rates are near zero at maximum attitude; hence, release of the parachute at this time would not result in a ground impact attitude significantly different from that at release. Proper design of release mechanisms and attachments will not result in a significant tip-off at release.

At near zero attitude angles, maximum vehicle angular rates were found to be less than 15 deg per second; hence, impact angles in this case would be less than in the maximum attitude release case.

As a result of the previous discussion, it is apparent that parachute release prior to ground impact will provide a means by which the parachute may be displaced from the Lander on impact, and further that a multi-directional impact attenuation is adequate. It is expected that the large weight penalty associated with the omni-directional impact attenuator designs would not be required with a parachute release prior-to-impact concept of the type herein described.

3.2.4.3. Parachute Weights

As a result of numerous parametric RESSEP parachute design studies, the ratio of the terminal parachute weight to the decelerator load is summarized in fig. 3.2.1-3 as a function of terminal velocity in VM-7. As shown by specific points labelled for

different entry conditions, the effects of entry angle, mode of entry and entry vehicle ballistic coefficient are minor. The predominant factors affecting the parachute weight are the suspended weight, the terminal velocity, and the model atmosphere.

Since all main parachutes are designed employing reefing, if required to optimize the design, the parachute weight does not reflect an effect of the deployment dynamic pressure range of the current study. This is shown in fig. 3.2.4-4 where the parachute weight is found to be essentially a function of canopy area, since minimum weight materials are used in most designs as a result of a reefing stage.

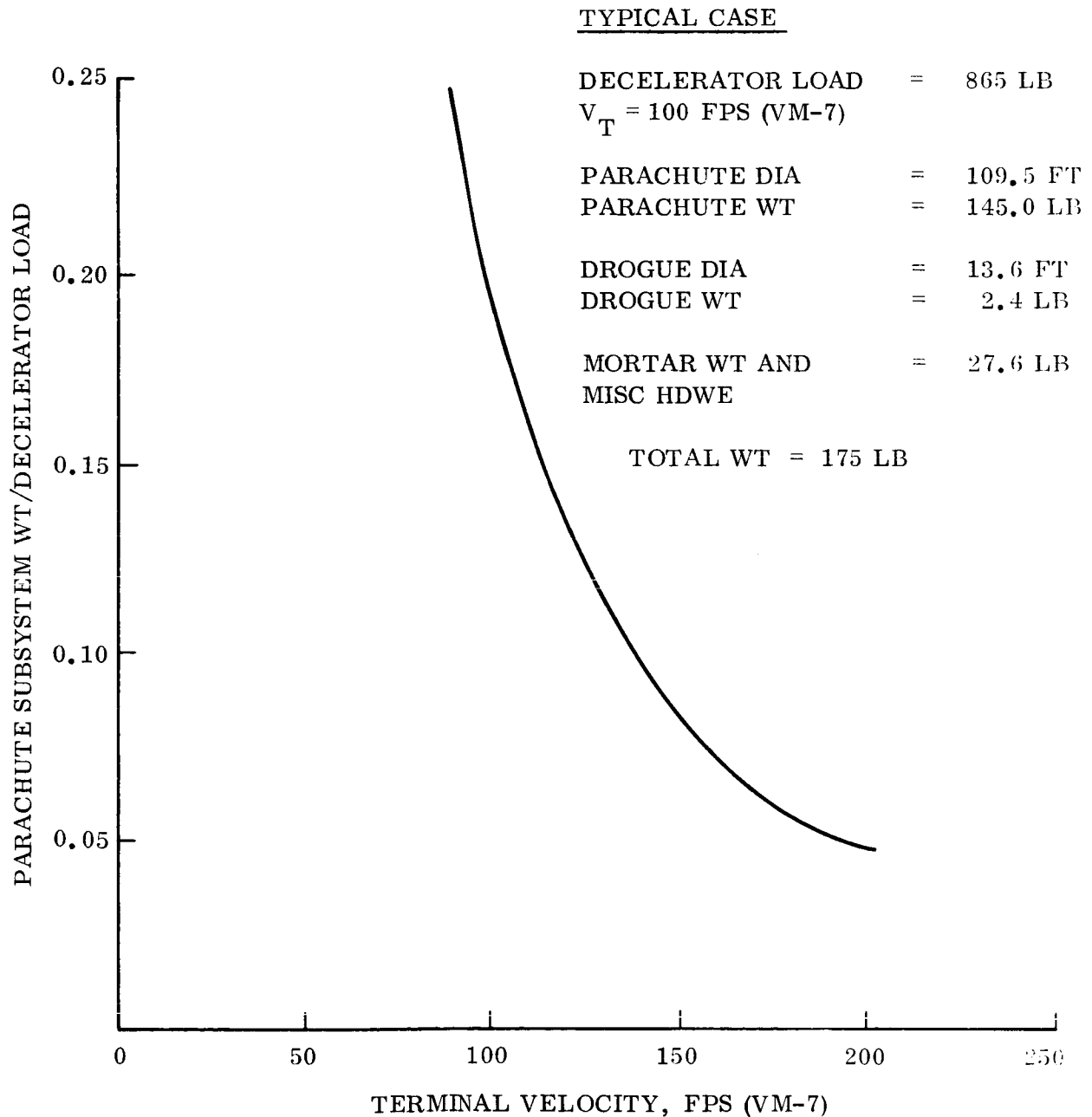


Figure 3.2.4-3. Variation of Parachute Subsystem Weight with Terminal Velocity

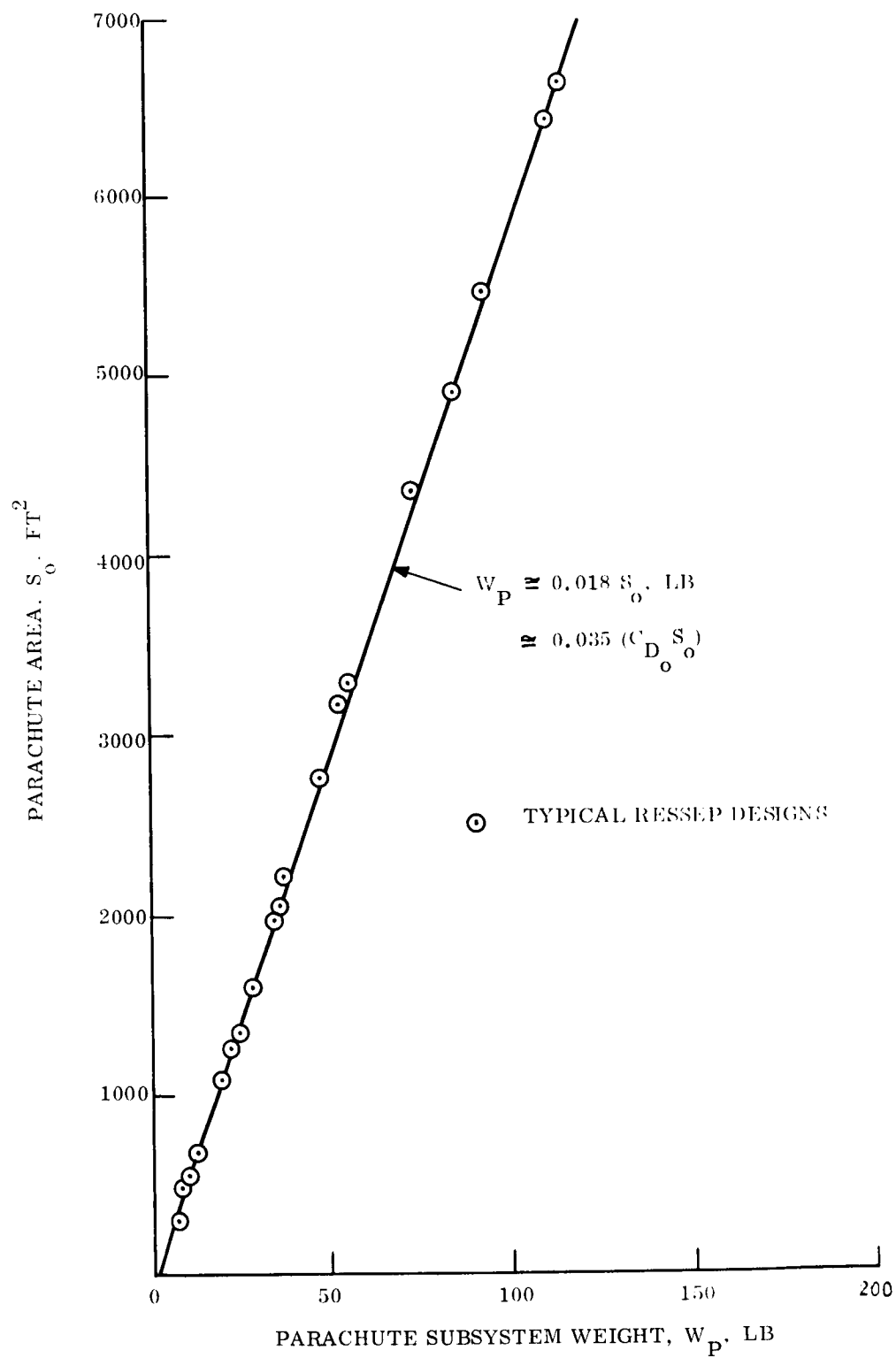


Figure 3.2.4-4. Variation of Parachute Subsystem Weight with Canopy Area

The parachute system weight may be alternately expressed as:

$$W_{PS} = \frac{0.028W_E}{\rho V_T^2} \quad \text{where}$$
$$W_E = \text{the decelerator load (Earth wt)}$$
$$\rho = \text{atmospheric density, slug/ft}^3$$
$$V_T = \text{terminal velocity, fps}$$

This weight includes a 25 percent factor over the basic parachute, lines and riser, to account for a deployment bag, harnesses, attachment hardware, deployment mortar, and pilot for main chute extraction.

As described in Volume III, Appendix A of this report, parachute weights are based upon parachute design and fabrication specialists data and information which are incorporated in the RESSEP program. The data of ref. 3-58 served as the basis for Ringsail type parachute designs and weights resulting in the present study. More recent data involving parachute sizes up to 129 ft nominal diameter (refs. 3-59 and -60) indicate parachute weights, and system weights which are closely predicted by the RESSEP program.

3.3 ENTRY AND RETARDATION TRADE-OFFS

For the baseline Hard Lander (i.e., entry aeroshell, parachute retardation, and crushable impact attenuation), deceleration of the landed subsystems occurs in three basic steps: (1) during entry into the atmosphere the aeroshell protects and decelerates the decelerator load to conditions compatible with the parachute; (2) the parachute then further decelerates the Lander to conditions selected for landing; and (3) the crushable impact attenuation subsystem performs the final deceleration in a rather abrupt manner. Altogether, the landed subsystems are decelerated from entry conditions (20,800 fps for direct and 15,300 fps for out of orbit) to zero on the surface of Mars.

In order to maximize the landed weight, it is necessary to optimize the combined deceleration subsystems. This is the entry and retardation trade-off. To perform the optimization, it is necessary to relate all three subsystems (aeroshell, parachute, and impact attenuation) to a common parameter. It is obvious that the parachute and impact attenuation subsystem are related by the terminal vertical velocity of the lander on the parachute. From Section 3.2 (Parachute Decelerator), we have the entry vehicle ballistic coefficient also related to terminal velocity. Therefore, the aeroshell subsystem can be related to terminal velocity. Fig. 3.3-1 presents an optimization curve where aeroshell (entry weight - decelerator load), parachute (decelerator load - Lander weight), and impact attenuation (Lander weight - landed weight) are related to one another through terminal velocity. The particular terminal velocity shown is for the VM-7 atmosphere; however, it can be related to any other atmosphere through the square root of the surface density ratio, atmosphere to atmosphere.

In fig. 3.3-1 for the out-of-orbit mode, the aeroshell diameter is selected as 8.34 ft and impact conditions are noted. From the variation of allowable β with terminal velocity (see Section 3.2) and the aeroshell diameter, the entry weight variation can be derived as shown in the figure. Subtracting the aeroshell weight from the entry weight yields the decelerator load shown. The parachute subsystem weight is then subtracted from the decelerator load resulting in the Lander weight. At this point, a preferable terminal velocity appears; however, the impact attenuation must still be accounted for. This results in the landed weight for the particular assumptions made for the design impact condition (i.e., g's, VH , θ slope, etc.). From fig. 3.3-1, it can be seen that the optimum Lander weight (790 lb) for the multi-directional Lander occurs at a terminal velocity of approximately 150 fps. The omnidirectional design results in less landed weight and an optimum velocity of approximately 120 fps.

Figure 3.3-2 presents a similar story for the direct entry $\gamma_e = 25^\circ$ case with landed weight being less than for the out-of-orbit case. Direct entry at $\gamma_e = 40^\circ$, fig. 3.3-3, results in a much lower landed weight.

If a 16 ft shroud is considered, figs. 3.3-4 and 3.3-5, the landed weight can be increased to the point where it is comparable to that for out-of-orbit with the 10 ft shroud.

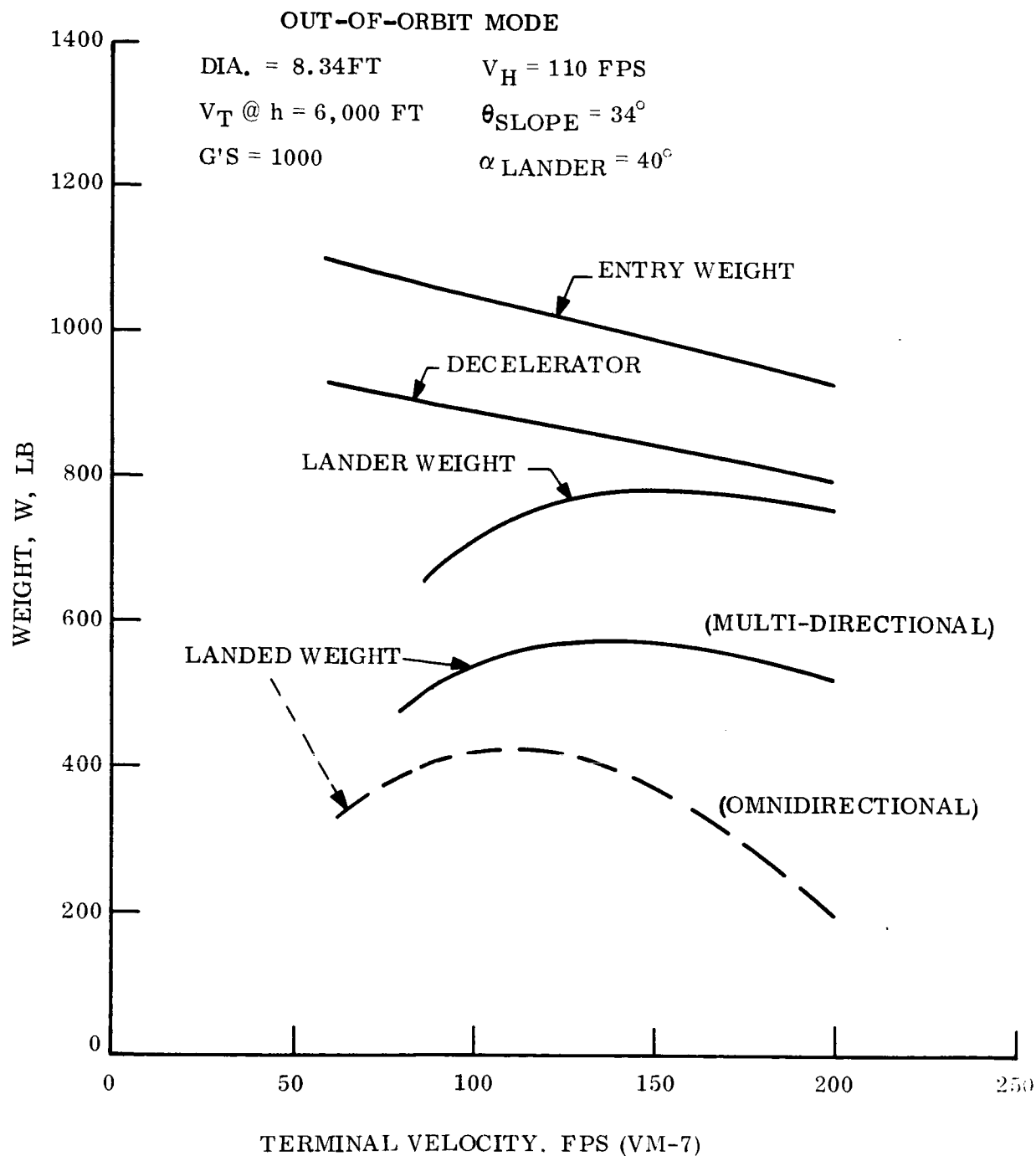


Figure 3.3-1. Impact Velocity Optimization

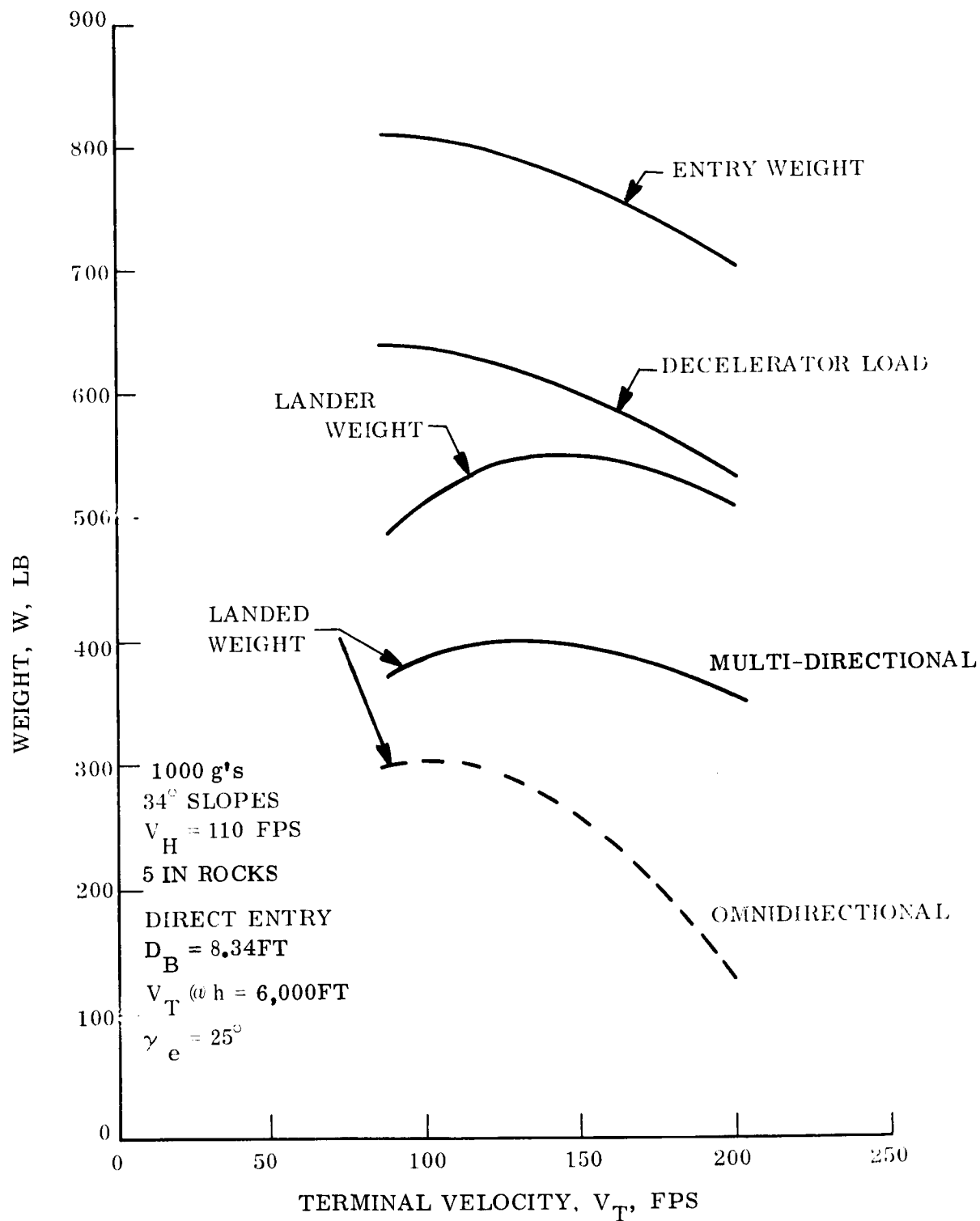


Figure 3.3-2. Impact Velocity Optimization, Diameter 8.34 ft, $\gamma = 25^\circ$

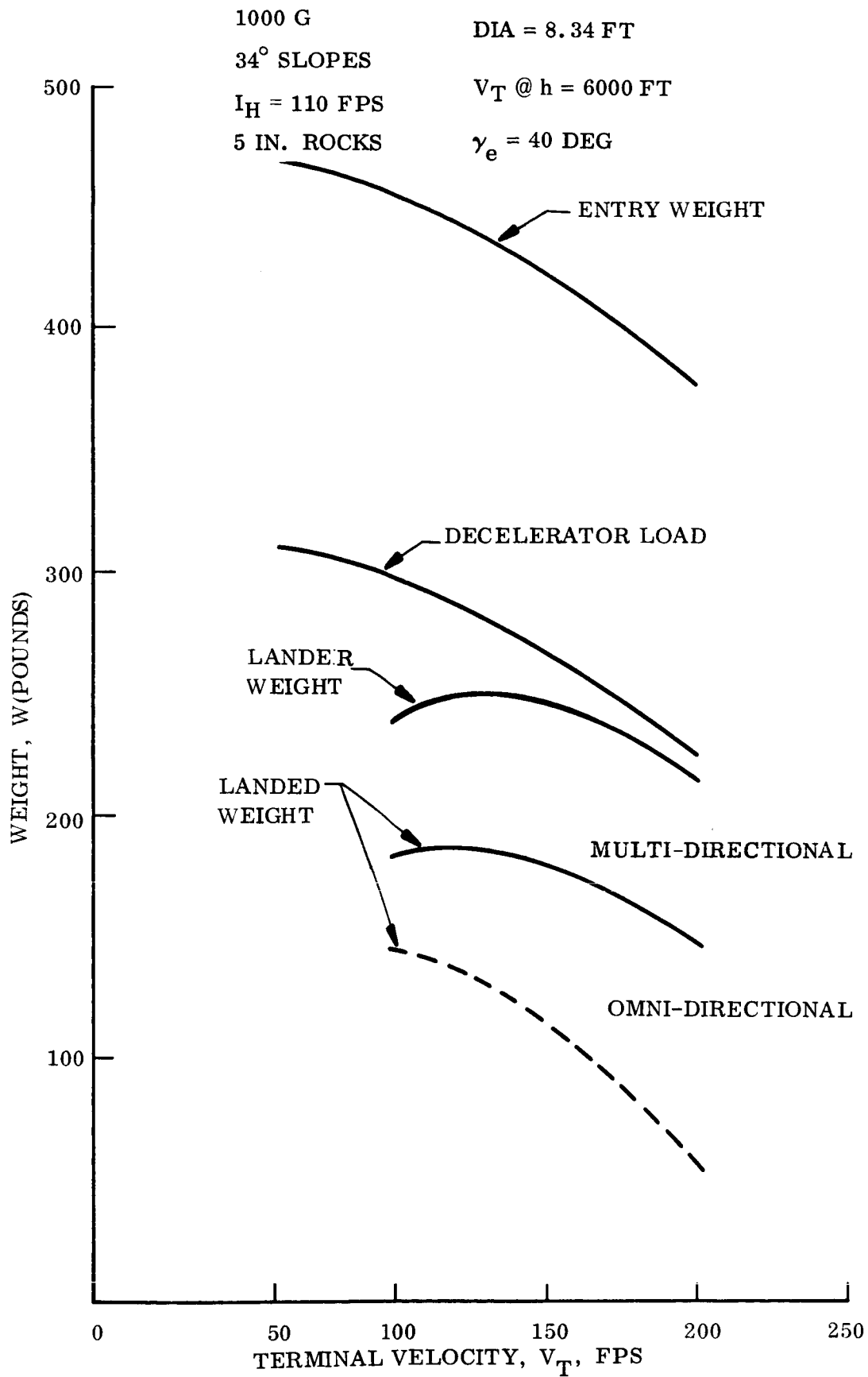


Figure 3.3-3. Impact Velocity Optimization, Diameter 8.34 ft, $\gamma = 40^\circ$

1000 g
 34° SLOPES
 $V_H = 110$ FPS
 5 IN ROCKS

DIRECT ENTRY
 DIA. = 14.3FT
 V_T @ h = 6,000FT
 $\gamma_e = 25$ DEG

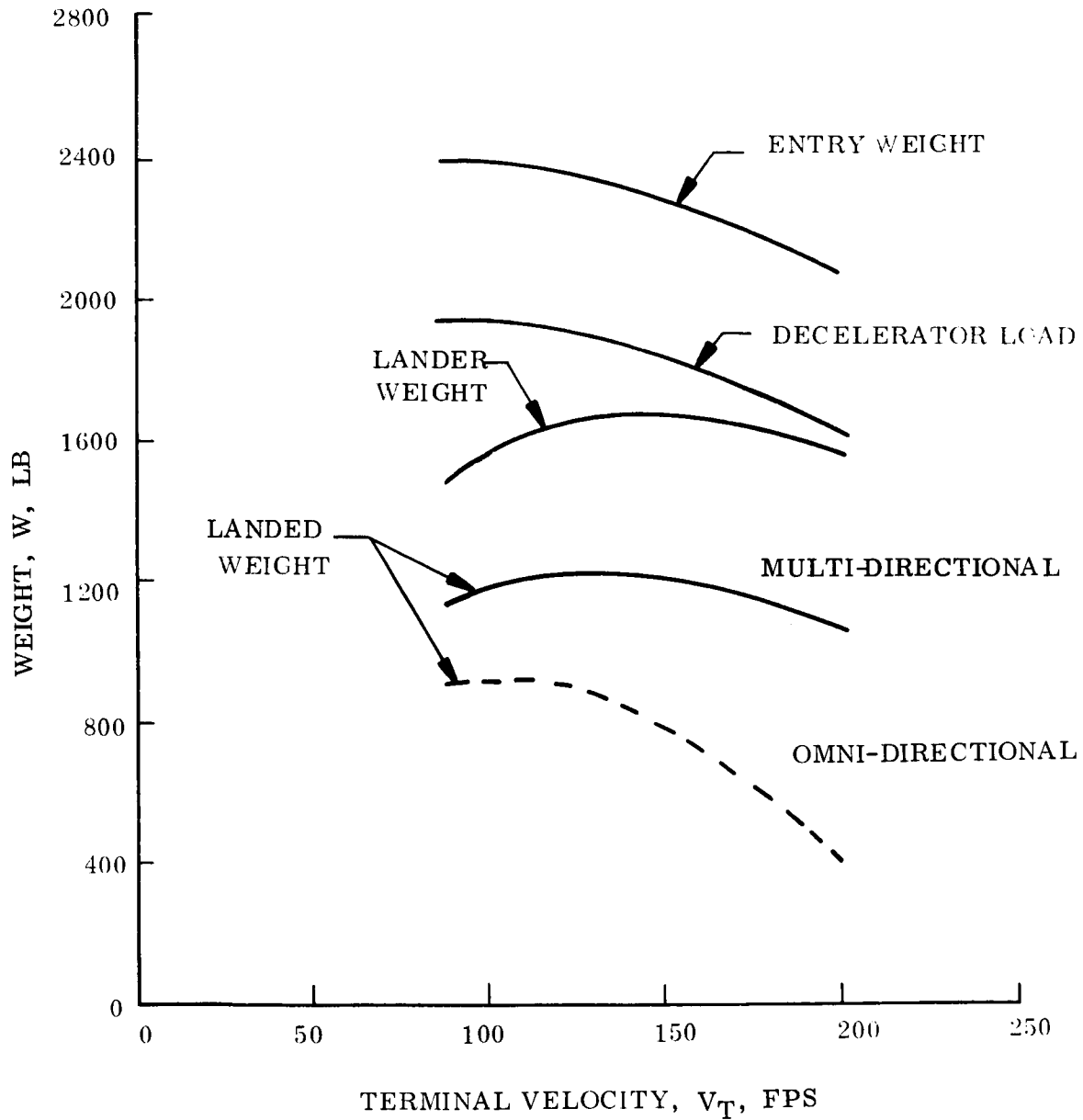


Figure 3.3-4. Impact Velocity Optimization, Diameter = 14.3 ft, $\gamma = 25^\circ$

IMPACT VELOCITY OPTIMIZATION

1000 G

DIRECT ENTRY

34° SLOPES

DIA = 14.3 FT

$V_A = 110$ FPS

$V_T @ h = 6,000$ FT

5 IN. ROCKS

$\gamma_e = 40$ DEG.

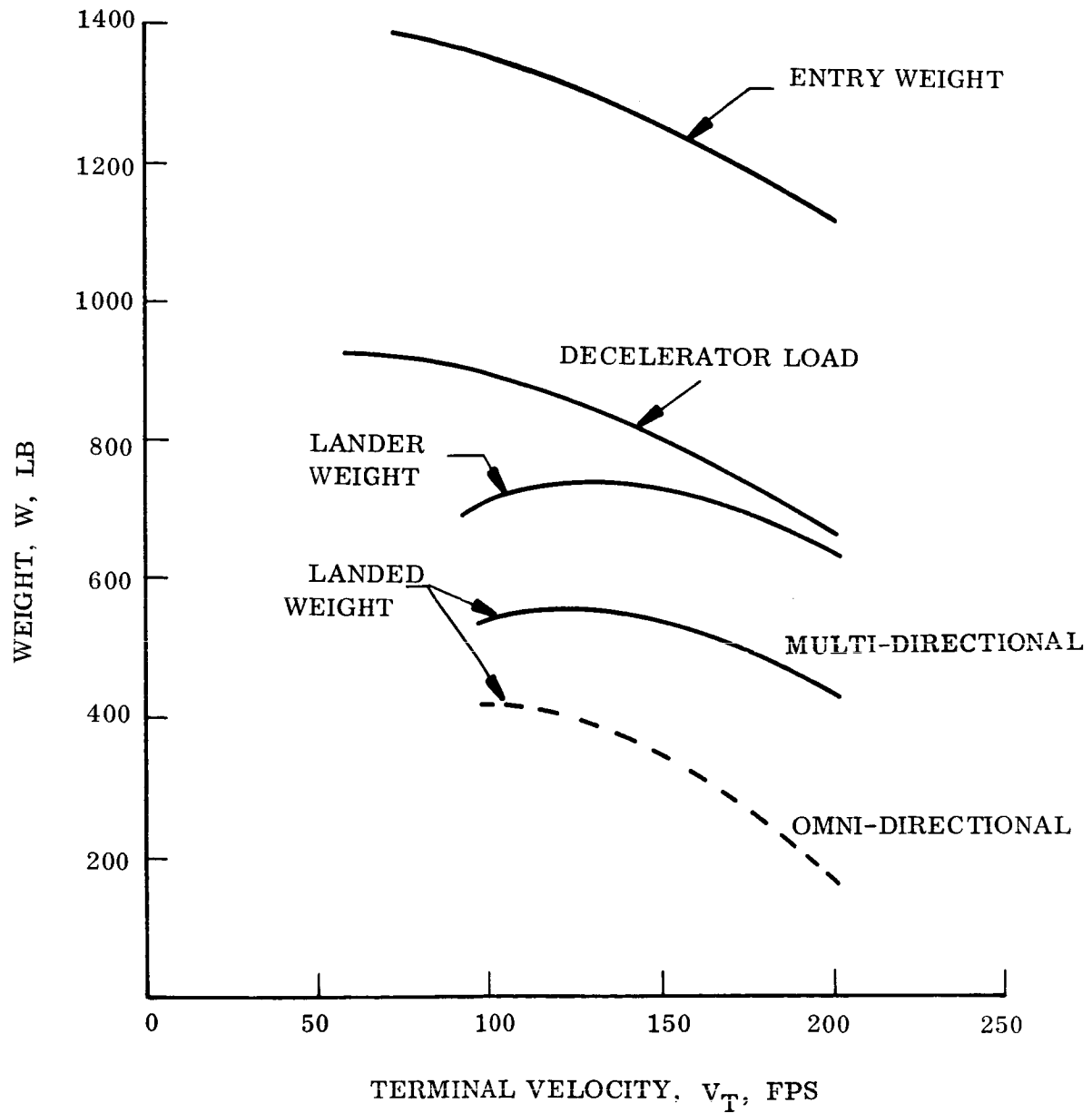


Figure 3.3-5. Impact Velocity Optimization, Diameter = 14.3 ft, $\gamma = 40^\circ$

The optimizations described above serve to point out the terminal velocity that should be selected for vehicle design to maximize landed weight. Since the curves are rather flat near the maximum, some leeway exists so that other considerations may influence the final design, such as Lander packaging for higher velocities where increased impact attenuation stroke is required to maintain a 'g' limit. From the optimization, a terminal velocity can be selected, however, so that the number of variables placed on the Lander maps can be reduced. Selection of a terminal velocity specifies an entry vehicle ballistic coefficient and thus a relationship between aeroshell diameter and decelerator load. These relationships are discussed in the following sections.

3.3.1 AEROSHELL AND PARACHUTES

Selection of terminal conditions on the parachute of $V_T = 120$ fps at an altitude of 5,000 ft results in an allowable β equal to 0.40 slugs/ft² for out-of-orbit and 0.174 slugs/ft² for direct entry at $\gamma_e = 40^\circ$. With these β 's, the decelerator load as a function of aeroshell diameter can be determined for a particular aeroshell configuration. Fig. 3.3.1-1 presents the decelerator load variation for an aeroshell cone angle of $\theta_c = 60^\circ$. Both the out-of-orbit and direct entry cases are shown. Also presented in the figure are the aeroshell diameter limits for the 10 and 16 ft shrouds. For the 10 ft shroud, the decelerator load is 870 lbs for out-of-orbit and 270 lbs for direct entry at $\gamma_e = 40^\circ$. The 870 lb value is compatible with hard Lander payload requirements; however, the 270 lb direct entry case is not. Extending the aeroshell to the 16 ft shroud represents a bulbous configuration for the launch vehicle.

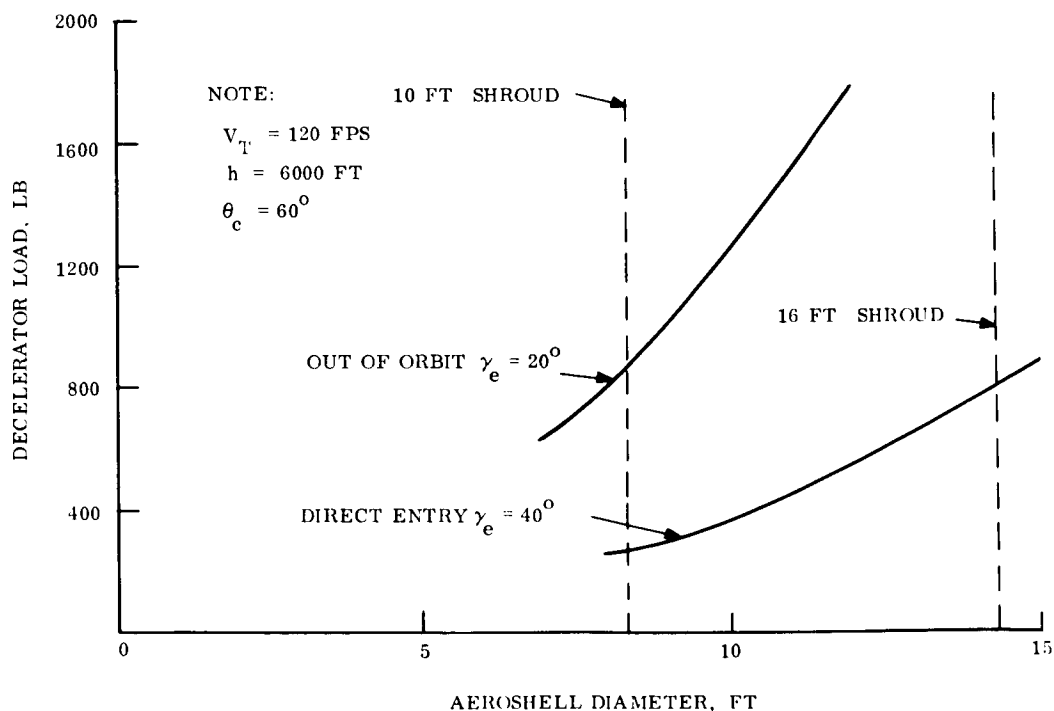


Figure 3.3.1-1. Aeroshell and Parachute

3.3.2 AEROSHELL, INFLATABLE AND PARACHUTE

The use of an inflatable afterbody can significantly increase the decelerator load capability for direct entry without exceeding the 10 ft shroud limit. An example of the decelerator load capability for a $\theta_c = 60^\circ$ aeroshell with a Goodyear Aerospace Corp. Airmat inflatable afterbody is presented in fig. 3.3.2-1. In this figure, the decelerator load is plotted as a function of the inflated diameter (D_{AC}) and the basic "hard" aeroshell diameter (D_O). The $D_O = 8.34$ ft corresponds to the 10 ft shroud limit. Two other aeroshell diameters are included to illustrate the small influence of D_O on decelerator load. The important point to be obtained from fig. 3.3.2-1 is that the 870 lb decelerator load associated with out of orbit entry and the 10 ft shroud limit can be matched for direct entry at $\gamma_e = 40^\circ$ by the use of a 12 ft inflated Airmat cone diameter. Other inflatable concepts discussed in paragraph 3.1.2.3 have the same or better capability, except for the post-entry heating deployment concept.

The pre-entry inflated afterbody performs quite well in maintaining the 10 ft shroud limit by increasing the decelerator load for direct entry to a value compatible with the desired hard Lander payload. If direct entry becomes the preferred mode, further effort on inflatable afterbodies is definitely recommended.

3.3.3 AEROSHELL, FLAPS AND PARACHUTES

The use of extendable flaps for adding drag to the aeroshell during entry has been discussed in para 3.1.2.4. Such a technique provides the required drag; however, packaging of the flaps in the stowed position presents problems which together with the flap weight results in very little gain in decelerator load. Since this concept appears marginal in comparison with the inflatable afterbody concepts, satisfactory engineering solutions to packaging and deployment should be developed before extendable flaps could appear worthy of further considerations.

3.3.4 AEROSHELL, POST-HEATING INFLATABLE AND PARACHUTE

Para 3.1.2.3 described the requirements associated with the post-heating inflatable (deployed at approximately Mach 5). For the Hard Lander where this concept would be used in conjunction with a parachute subsystem, no significant net gain in decelerator load was found.

3.3.5 LIFTING AEROSHELL AND PARACHUTE

The use of a lifting aeroshell for the direct entry mode is another way of increasing the decelerator load without exceeding the 10 ft shroud limit as shown in fig. 3.3.5-1. Note that for a decelerator load of 870 lb (the weight associated with out-of-orbit and 10 ft shroud limit), and $L/D \approx 0.28$ is required. Packaging studies currently in progress indicate that the hard Lander can be packaged in a 60° aeroshell with sufficient center of gravity offset to achieve this L/D . Thus, lifting entry offers another alternative for achieving desirable payloads for direct entry without exceeding the 10 ft shroud limit.

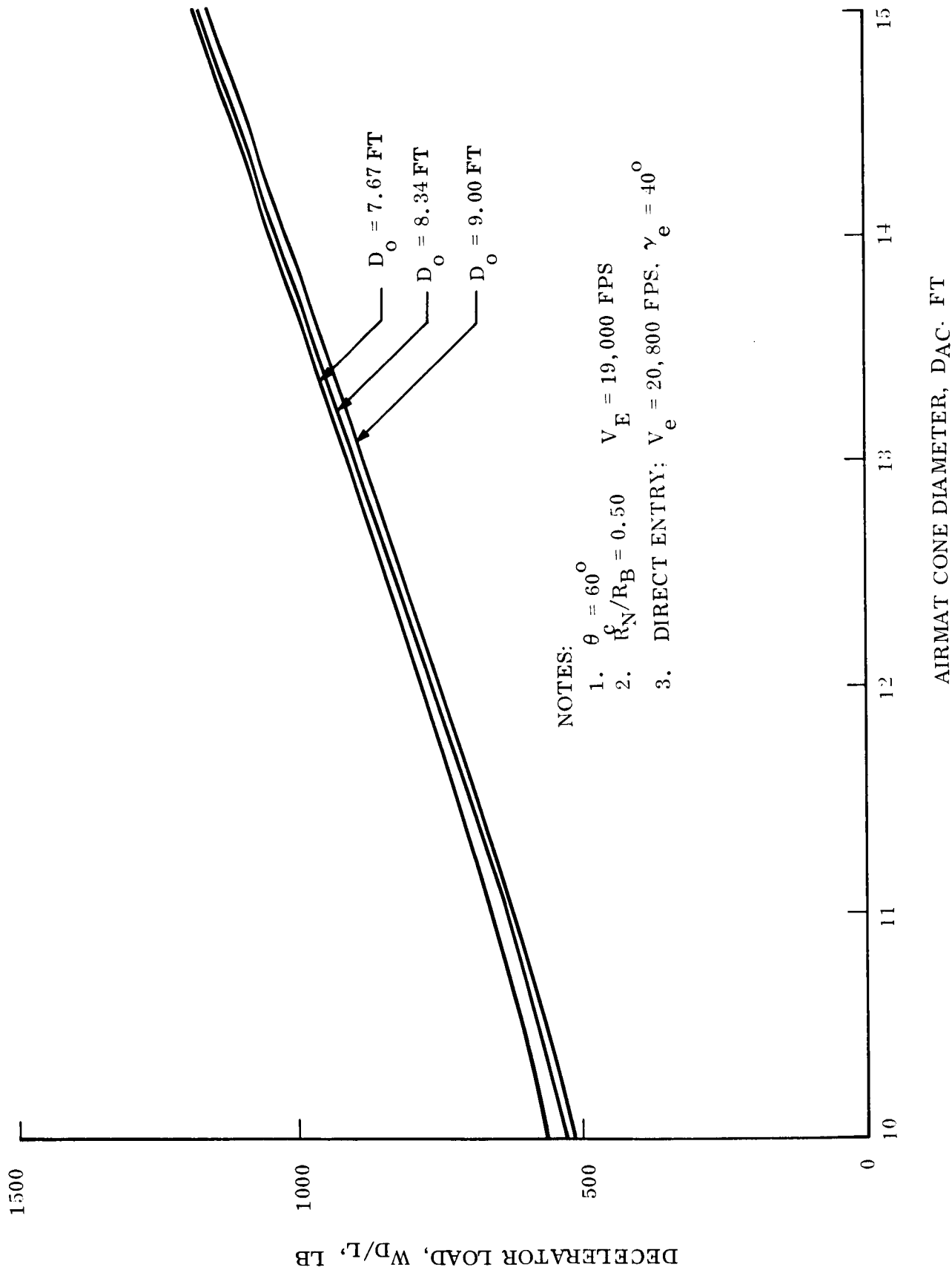


Figure 3.3.2-1. Decelerator Load vs Airmat Cone Diameter

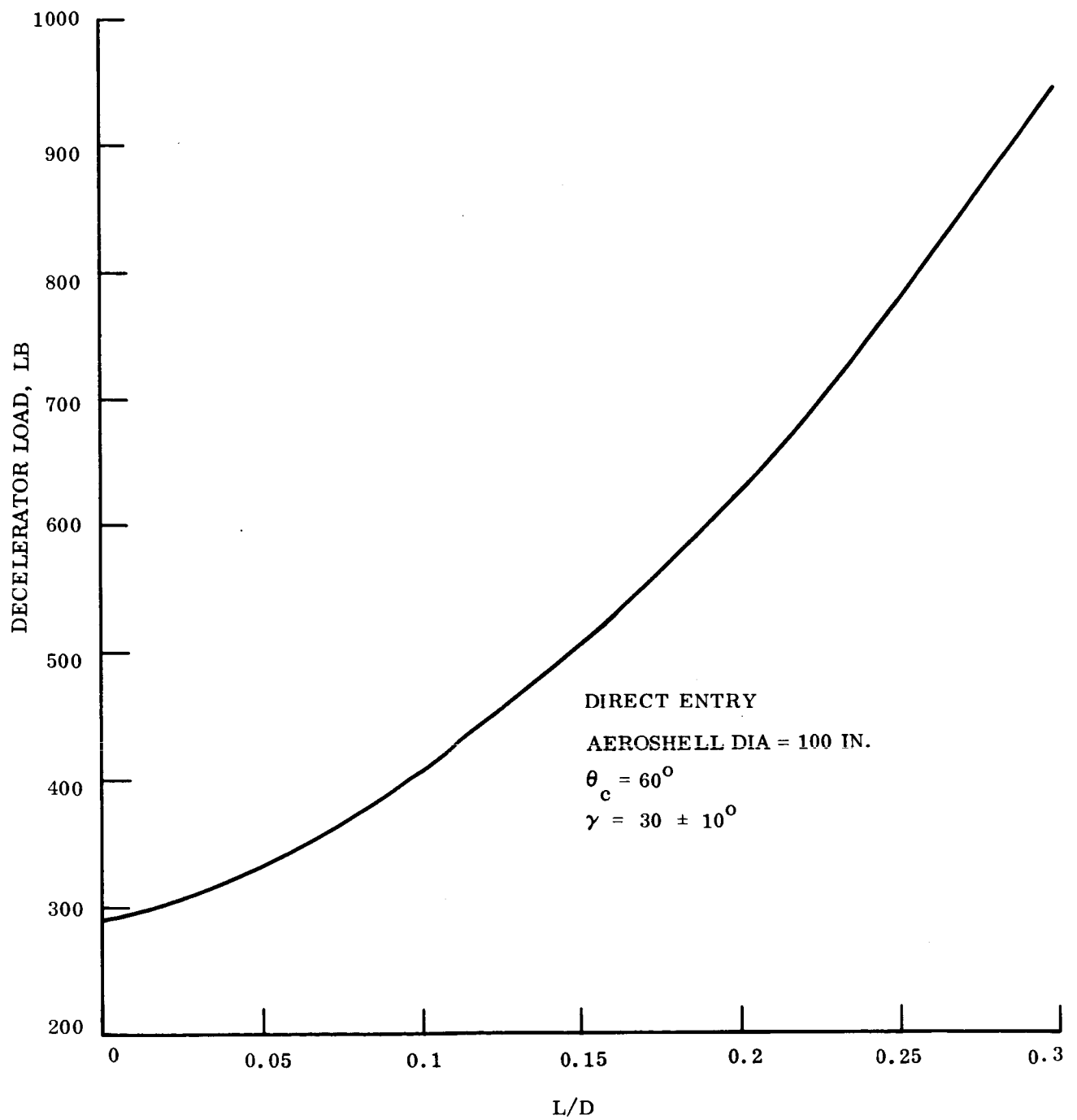


Figure 3.3.5-1. Variation of Decelerator Load with Vehicle L/D

3.4 ENTRY SCIENCE

3.4.1 GENERAL

The parametric studies on the entry science were conducted using the established Experiment Simulation Technique described in Volume II. Experiment Simulation is a concept by which the data from an experiment is mathematically simulated, analyzed, and interpreted before the experiment is actually conducted. By observing the effect on the simulated results of various changes in the type of data considered, the experiment design, instrumentation properties, and analysis techniques used, decisions on the optimization of these factors for the actual experiment can be made. The technique is used to parametrically evaluate the achievable experiment accuracy as a function of ballistic coefficient, sampling rate, sensor accuracy, atmospheric model, and subsystem errors. The results are given below. The study was limited primarily to the subsonic regime and clearly demonstrates that mean molecular weight and density can be obtained without a spectrometer and direct density measurements.

Characteristic of the parametric study results is the output of the simulation computer program in the form of a series of automatic plots for a given input. These plots succinctly summarize the important features of the simulated mission. An example of the runs used in this study is given figs. 3.4.1-1 through -11. The mission parameters characterizing the data are provided in table 3.4.1-1.

The general study approach was to use temperature, pressure, density, and altitude errors as a function of altitude (figs. 3.4.1-5, -7, -9, and -11) and take the mean values with standard deviations. These data form the basis for evaluating changes in measurement accuracy as functions of these cited error parameters.

TABLE 3.4.1-1. MISSION PARAMETERS

Case	Mars Parachute (90° Entry)
Trajectory	Direct
Atmosphere	VM-8
β =	0.393 lb/ft ² (0.0121 slug/ft ²)
Sampling Time =	0.5 sec
Random Errors	σ = 3 percent of Full Scale Pressure 1 percent of Full Scale Temperature

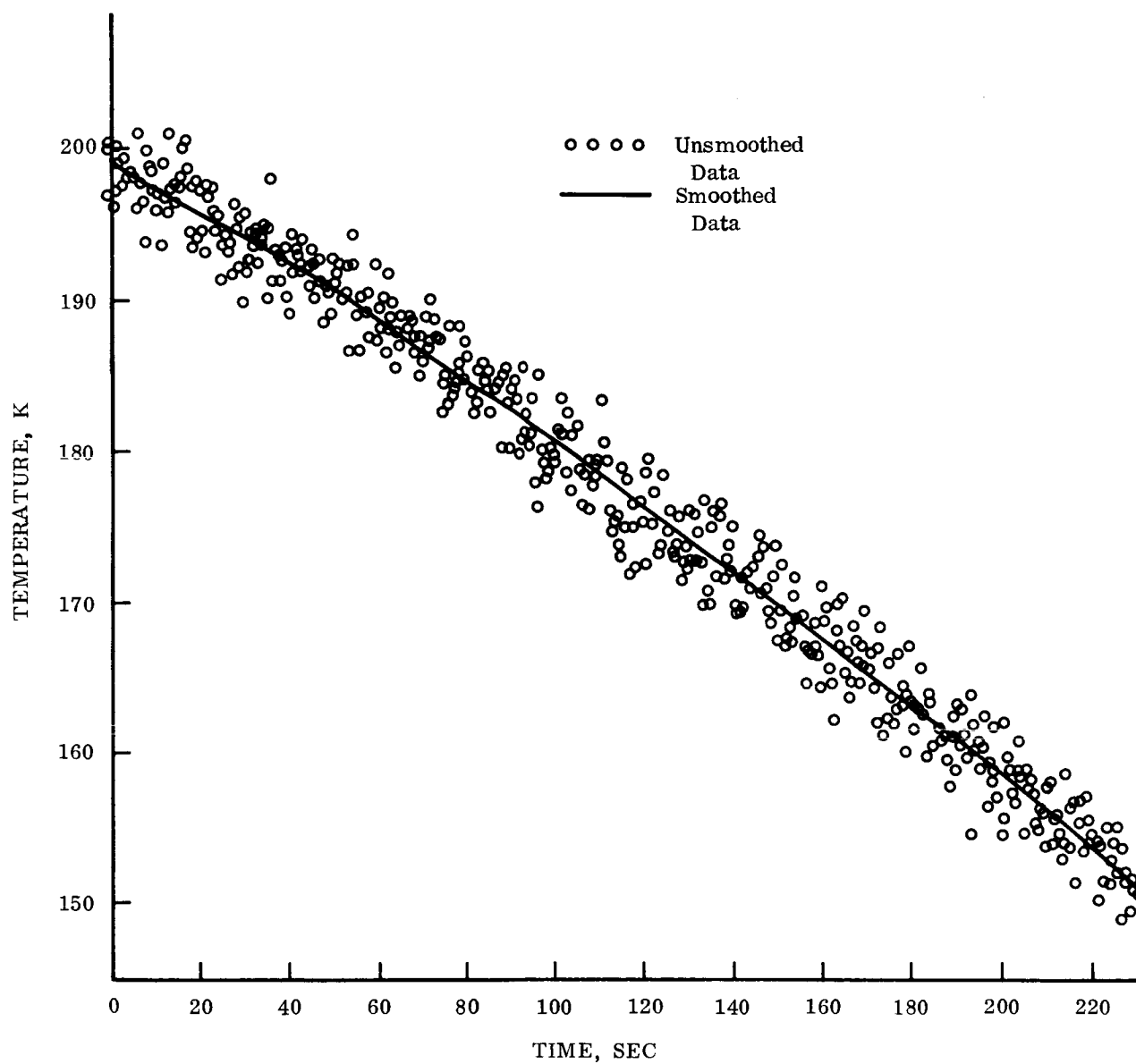


Figure 3.4.1-1. Unsmoothed and Smoothed Temperature Data

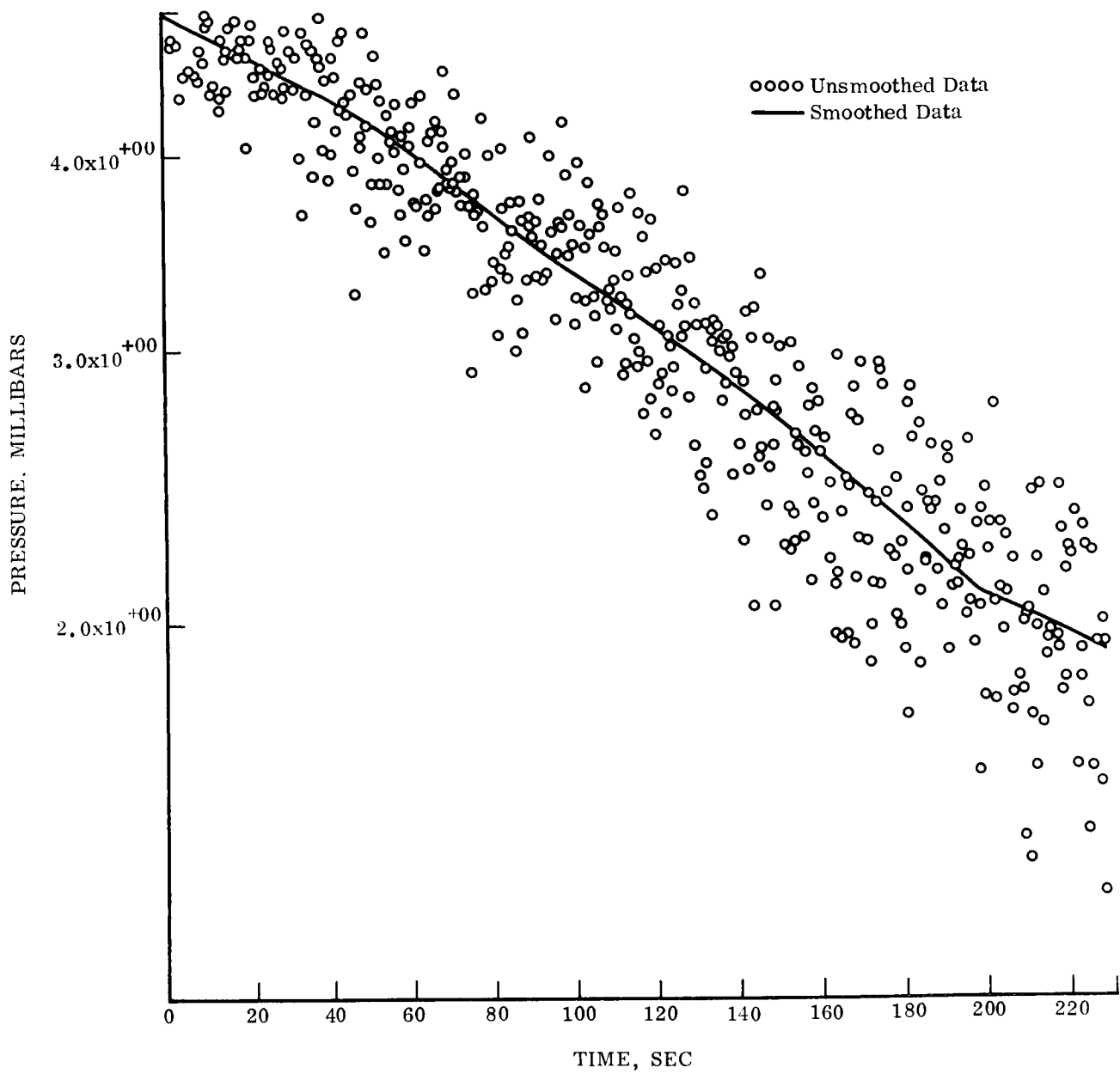


Figure 3.4.1-2. Unsmoothed and Smoothed Pressure Data

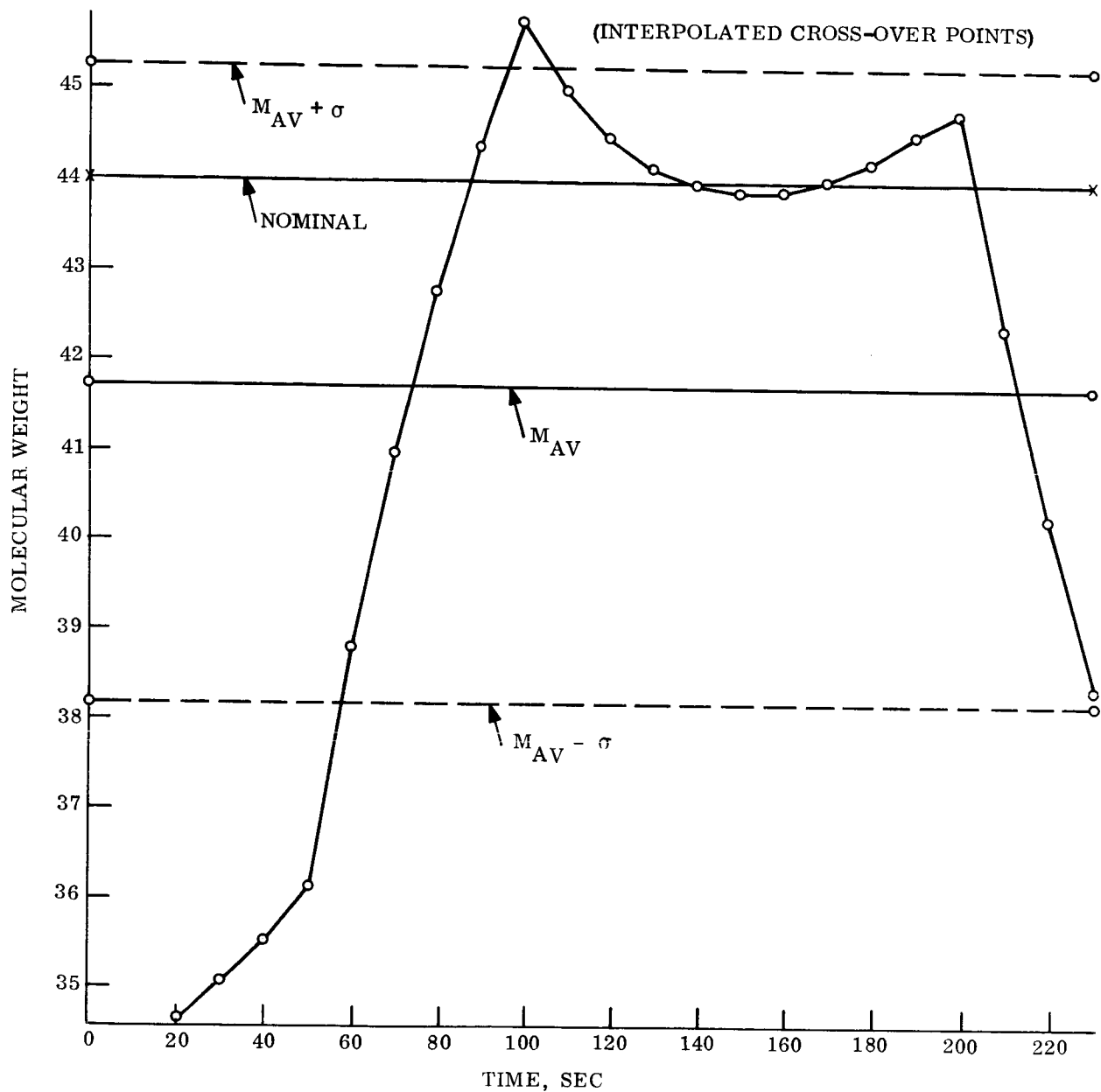


Figure 3.4.1-3. Calculated Values of Molecular Weight

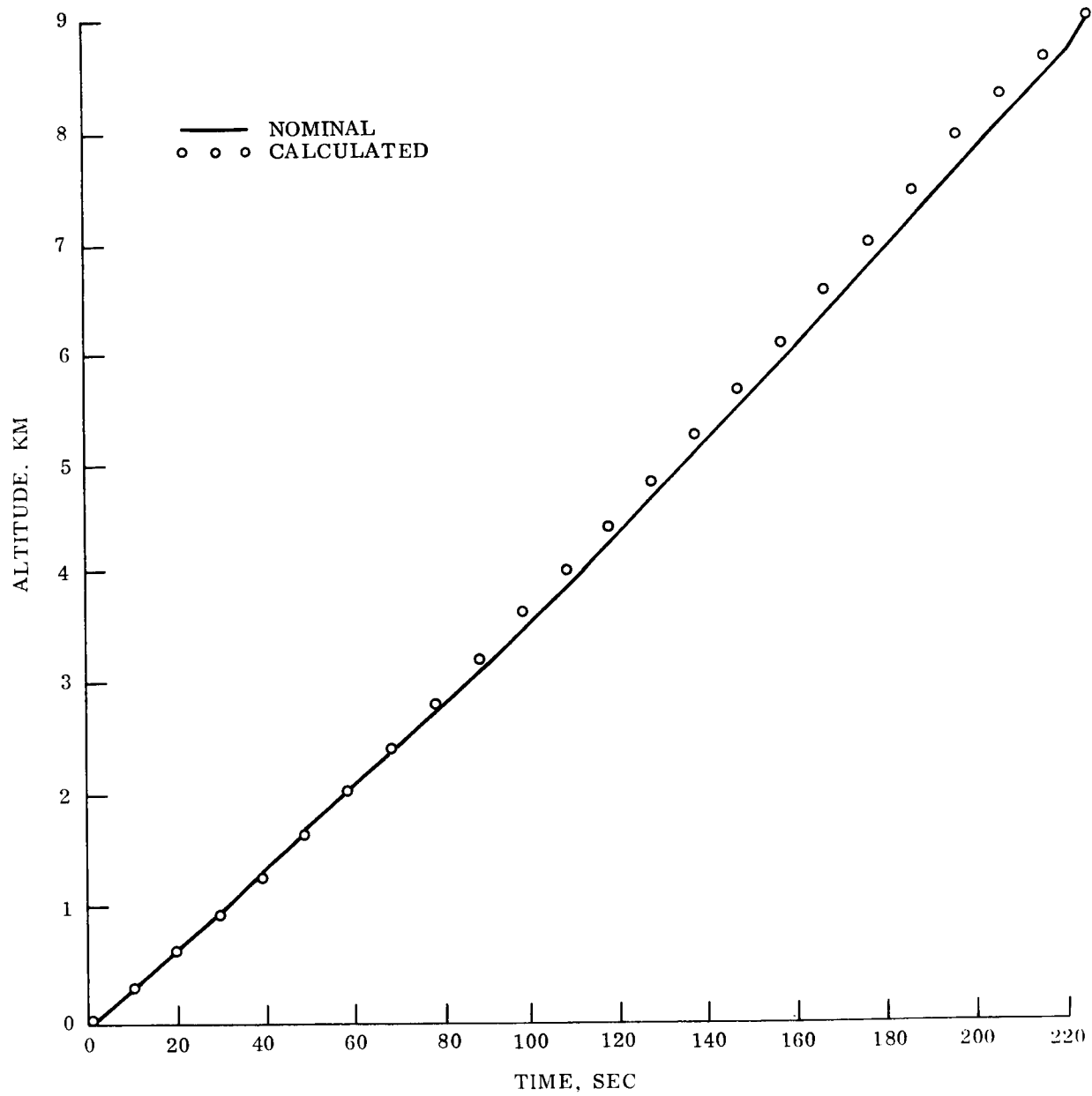


Figure 3.4.1-4. Altitude vs Time (Calculated and Nominal Values)

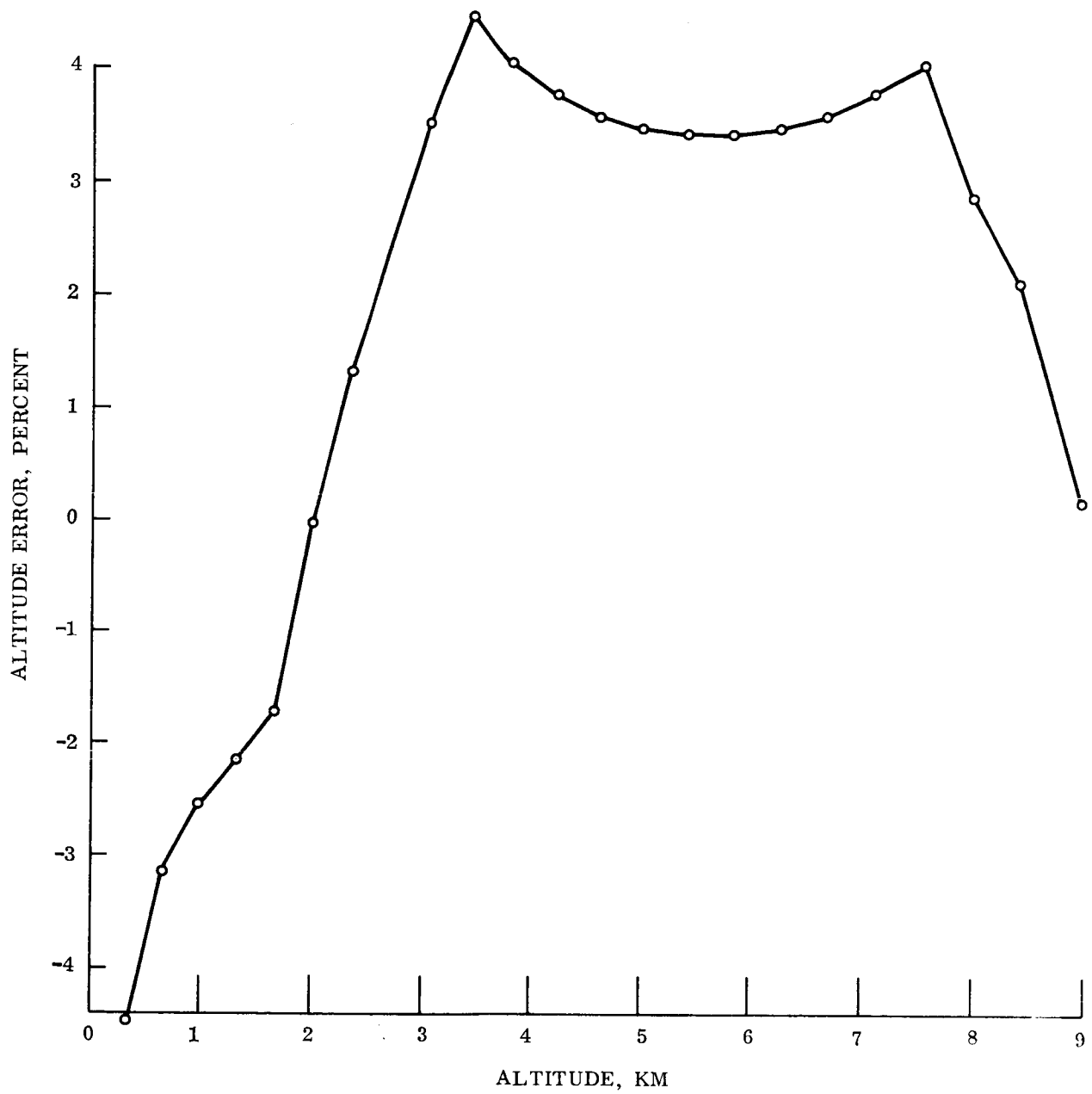


Figure 3.4.1-5. Altitude Error vs Altitude

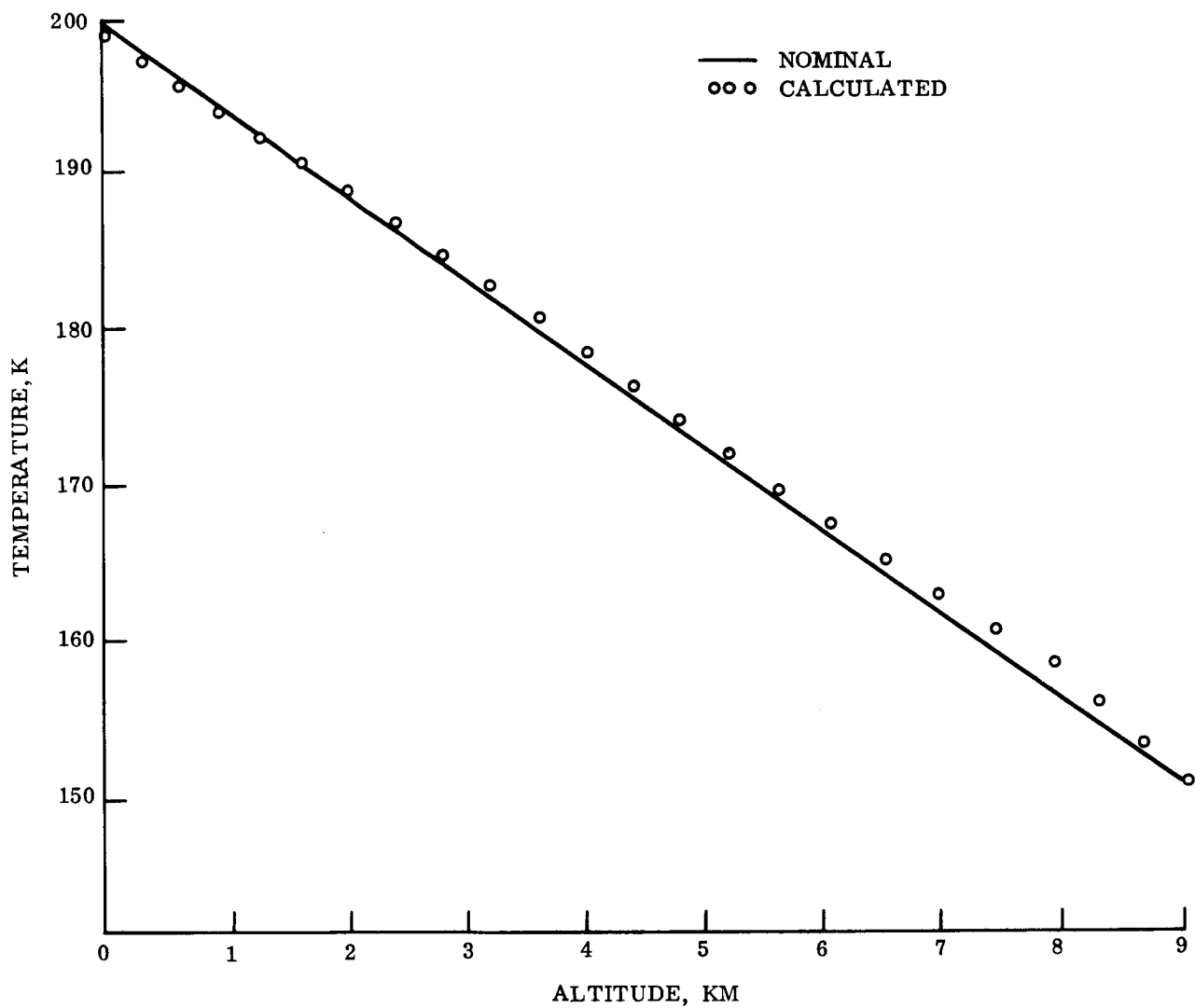


Figure 3.4.1-6. Temperature vs Altitude (Calculated and Nominal Values)

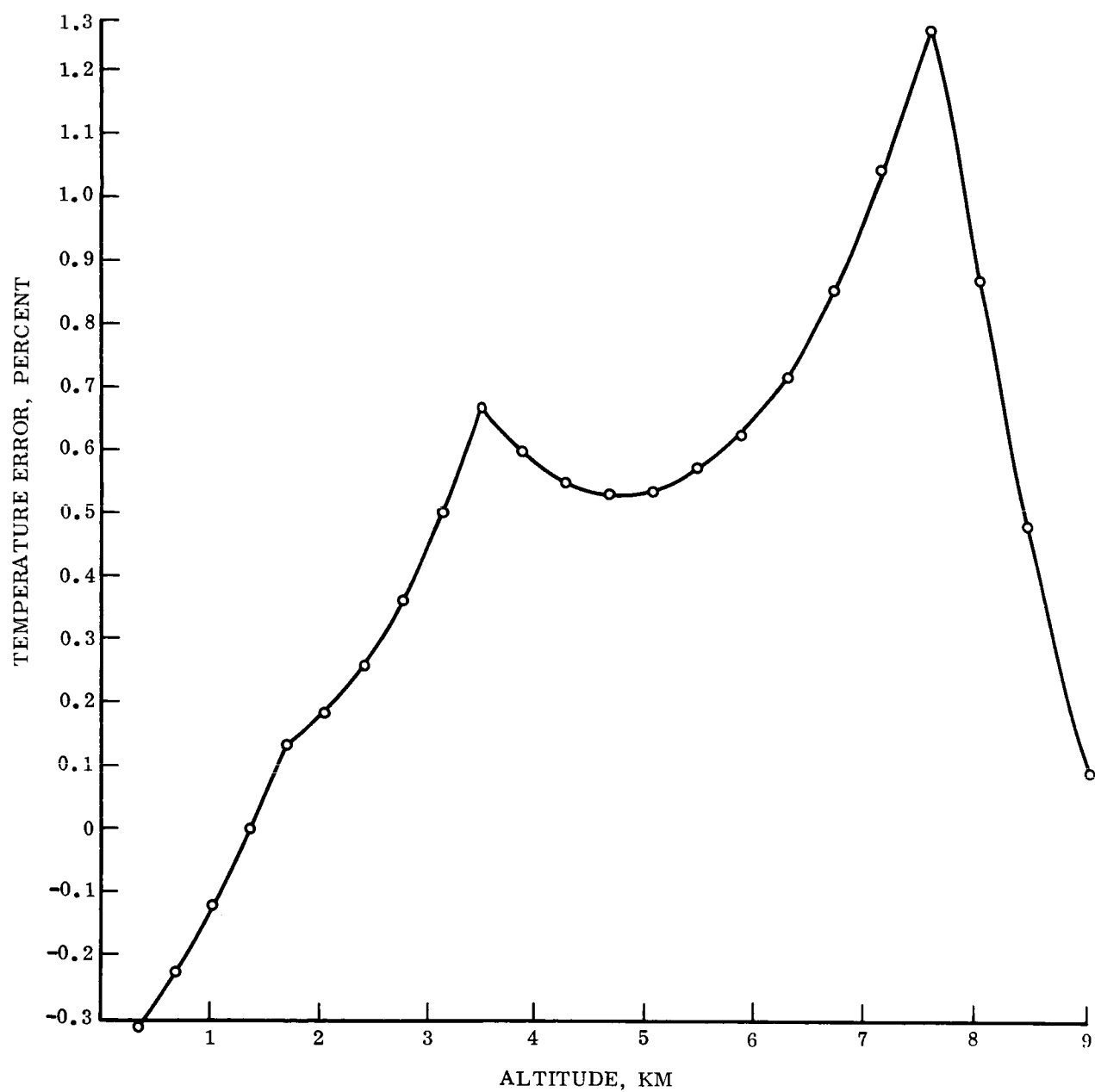


Figure 3.4.1-7. Temperature Error vs Altitude

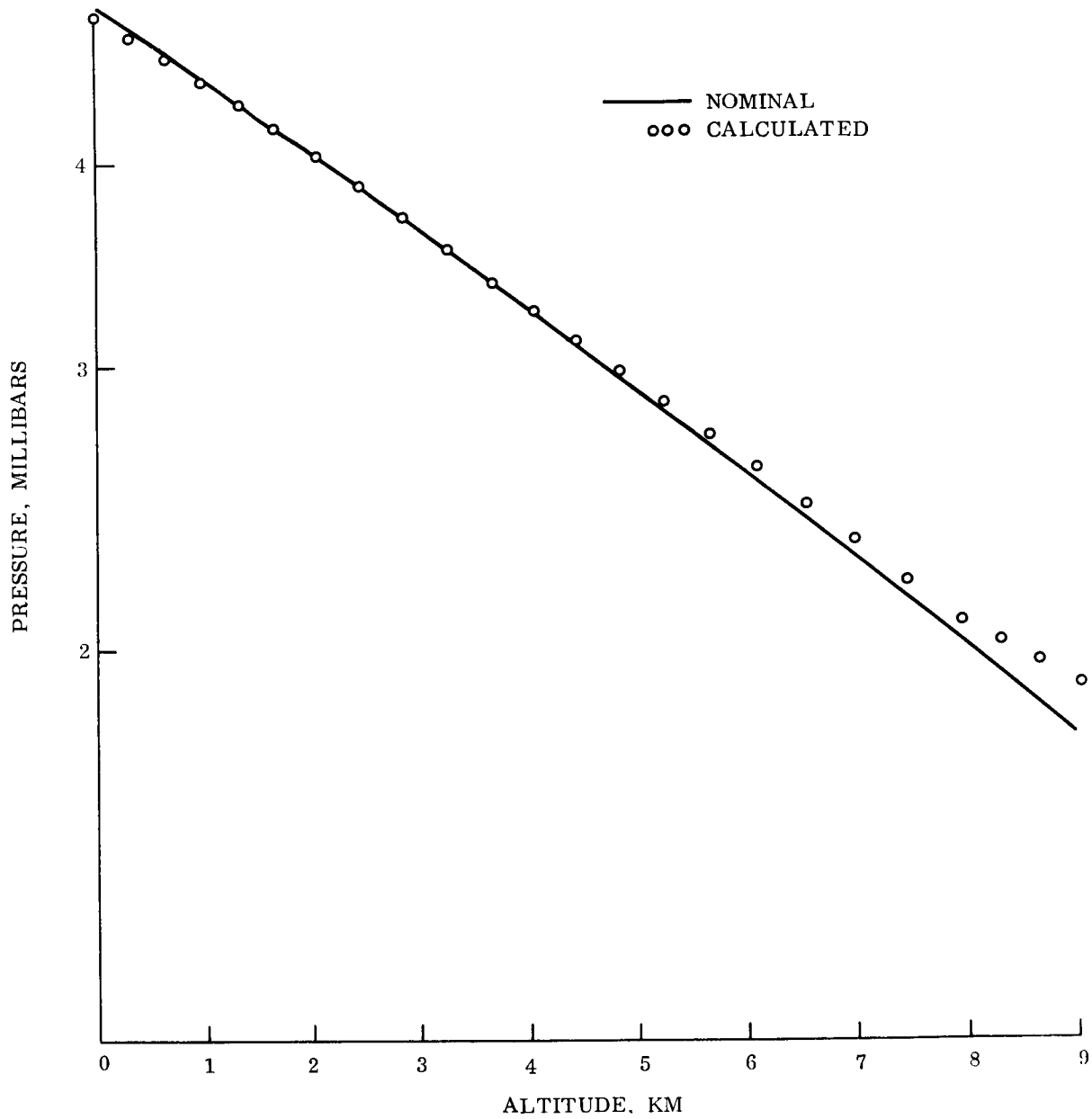


Figure 3.4.1-8. Pressure vs Altitude (Calculated and Nominal Values)

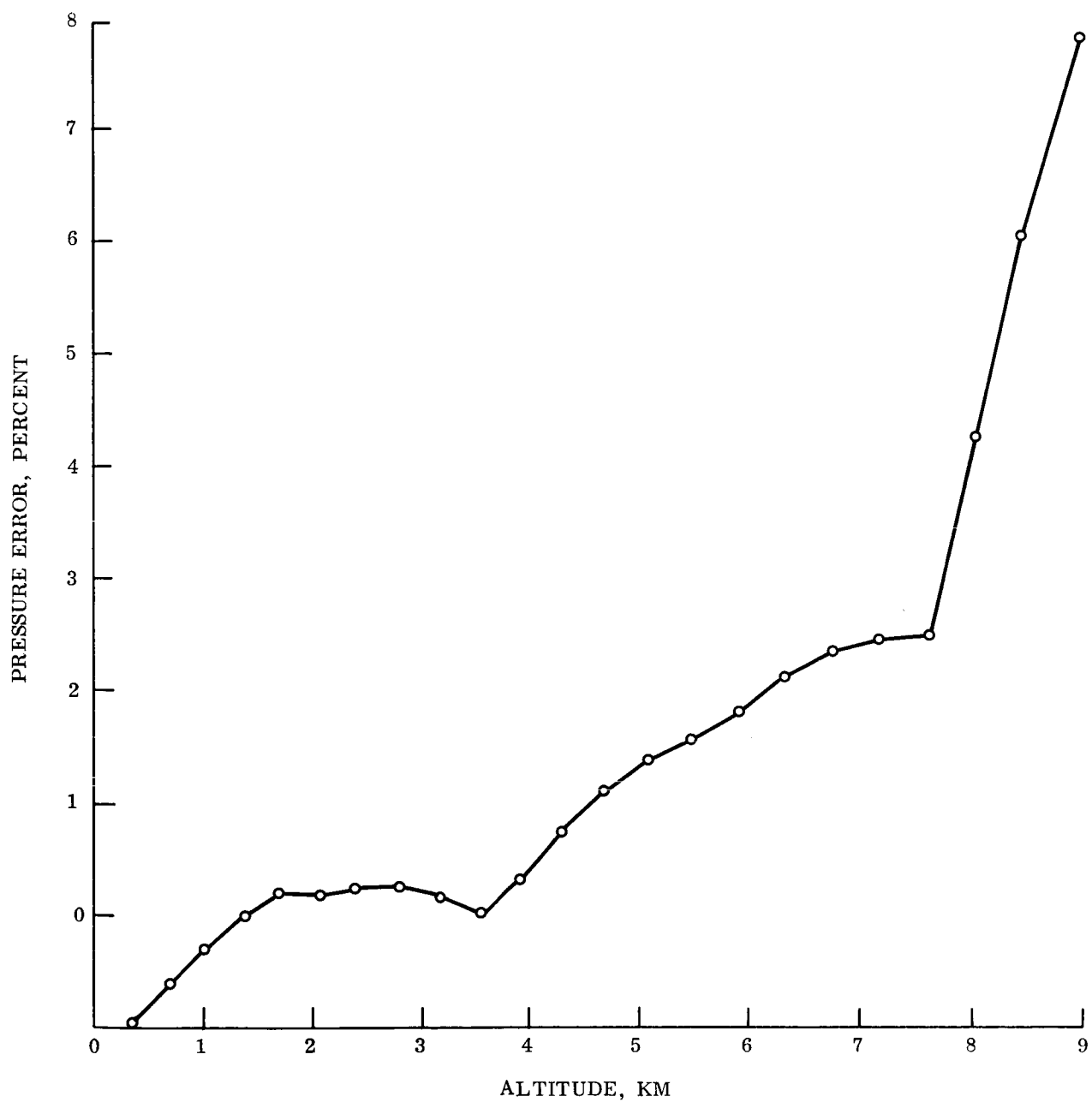


Figure 3.4.1-9. Pressure Error vs Altitude

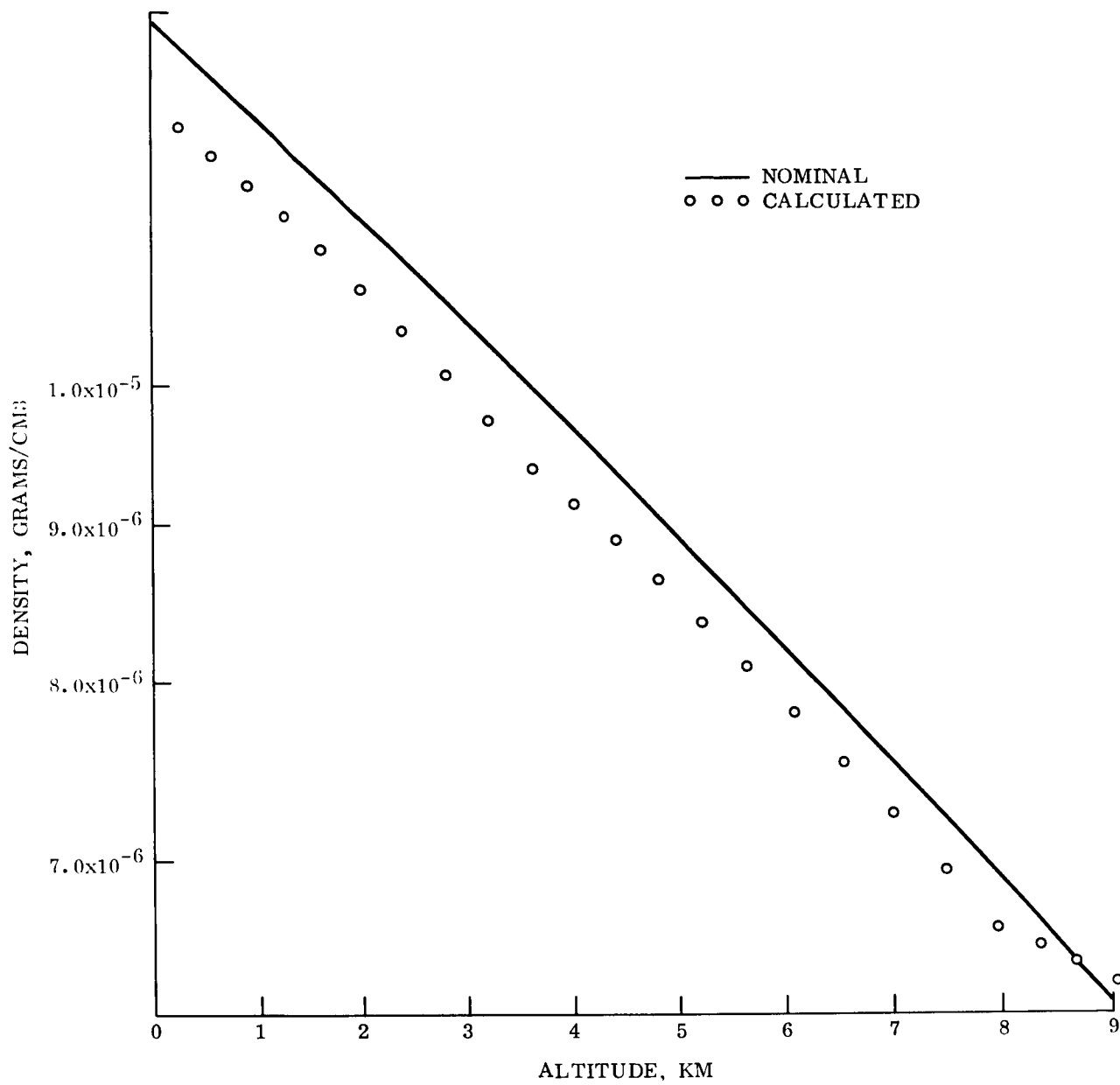


Figure 3.4.1-10. Density vs Altitude (Calculated and Nominal Values)

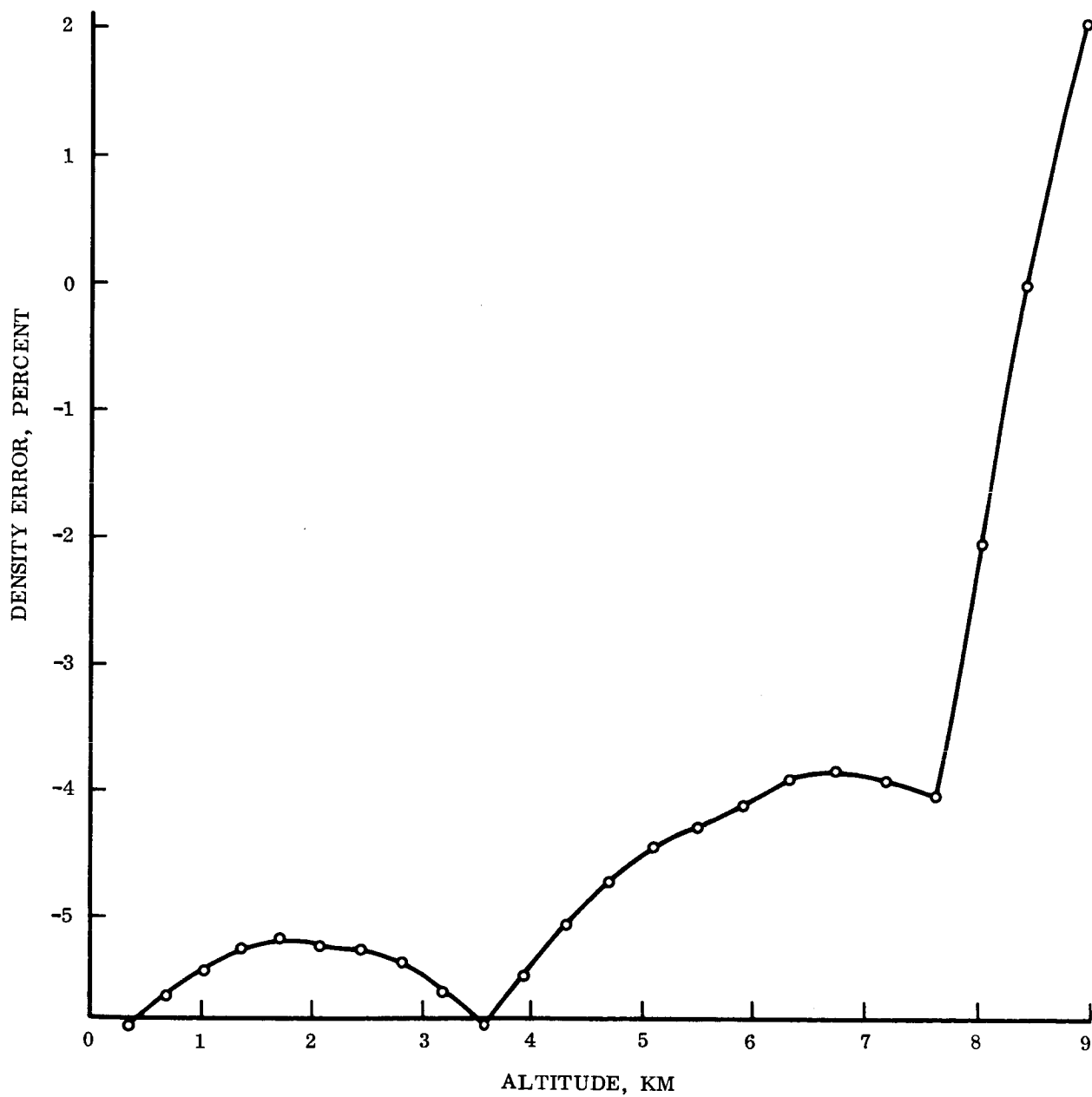


Figure 3.4.1-11. Density Error vs Altitude

3.4.2 PARAMETRIC ANALYSIS RESULTS

Assuming the minimum science payload, the experiment simulation method was applied to the temperature and pressure measurements in the subsonic (post-chute) regime. The procedure resulted in calculated mean molecular weights, trajectories, and density vs. altitude profiles. The quantities actually studied were the average accuracies of the various atmospheric parameters; that is, their percent deviations from the true values. The accuracies were averaged over the altitude intervals in question to obtain more statistically reliable results. Applying the data analysis technique to the minimum science payload enables evaluation of the necessity or advantage of adding additional instrumentation such as a mass spectrometer, altimeter, speed-of-sound sensor, etc. It should be noted that the data analysis technique assumes that the correlation between measured and ambient values of temperature and pressure will be reliably known from pre-flight test data. At present these correction curves are not yet available.

The experiment simulation method may also be applied to the accelerometer data, used to determine density in the supersonic (pre-chute regime). This was done in previous studies (ref. 3-75) but the present cases have not yet been studied in this manner.

The parametric study conducted here consisted of collecting and analyzing the data from a considerable number of runs of the Experiment Simulation computer program. The cases considered are enumerated in table 3.4.2-1. It should be borne in mind that each of the runs listed ($12 \times 2 \times 3 = 72$ in all) really consisted of three "sub-runs" since two complete iterations are involved in each.

Due to the random nature of the errors introduced into the raw data, the data after smoothing continues to have a randomness associated with it, particularly with regard to its time slope. Since the entire atmospheric profile depends very strongly on the calculated mean molecular weight, which in turn is proportional to the square of the pressure time slope ($\bar{M} \propto \dot{p}^2$) it is obvious that any randomness in this slope can cause a considerable variation in the resultant data. In the present study, the data was smoothed in connected linear segments, the length of the segments being chosen so that about four covered the region of interest. In previous studies (ref. 3-76) some other methods were examined, including logarithmic smoothing but the type used here seemed to yield the best results. Further investigation in this area is required, however, to assure that the optimum technique is used for a given type of data.

The uncertainty in the slope of the smoothed pressure data results in an unpredictable variation from run to run which makes it difficult to draw precise quantitative

conclusions. Using an averaging technique described below, however, some definite trends and results may be derived. Taking, as an example, the data for low random errors, where the spread in data is smallest, figs. 3.4.2-1 and 3.4.2-2 show that the resulting altitude and density accuracies are relatively uninfluenced by the type of atmosphere and trajectory considered. That is, the significant fact about figs. 3.4.2-1 and -2 is that the accuracy is greater (smaller percent), and the spread smaller, for the low β curves than for the nominal β curves, rather than that one specific curve shows a larger error than another. For this reason, it was deemed appropriate to look at the averages of the results denoting variations in atmosphere and entry conditions, rather than at the individual data points themselves. In this way, the accuracies used are desensitized to the atmosphere and entry conditions, and depend only on ballistic coefficient, total random data errors, and sampling interval.

Results of this form, for atmospheric density, are given in figs. 3.4.2-3 and -4. There is still considerable scatter in the data due to the factors mentioned above. This indicates that the data population is not large enough to assure similar smoothed data when every second or fourth point is taken (sample interval doubled and quadrupled). Although there are some anomalies in the unreliable high error data, several trends are immediately obvious:

1. Accuracy decreases, and its spread increases, as the random data errors increase.
2. Accuracy decreases as sampling interval increases.
3. The low β results show considerably higher accuracy and less spread than the nominal β results.

Taking the mean results from figs. 3.4.2-3 and -4 (extrapolating if necessary), the sampling intervals necessary to produce average accuracies in density of five percent (the design goal) can be estimated. These results are shown in fig. 3.4.2-5. It can be seen that for medium random errors in the data, a sampling interval of one sec or less is required when a nominal β vehicle is used (the value of one sec is that recommended in this study). When the errors decrease to the low values, however, the sampling interval is permitted to double. The use of lower β vehicles would also allow longer sampling intervals. Because of its extremely erratic nature, the high error data was not included in fig. 3.4.2-5.

When a procedure similar to that used with the density data was applied to the pressure and temperature data the results shown in figs. 3.4.2-6 and -7 were obtained. The accuracy goal for pressure was five percent and that for temperature was two percent. Both results indicate higher allowable sampling intervals than for the density case. This is to be expected since the calculation of density involves more complex data analysis than do temperature and pressure. Thus the density requirement is the driving function for the choice of sampling interval.

TABLE 3.4.2-1. CASES CONSIDERED IN PARAMETRIC STUDY

Case	Atmosphere	Trajectory	Total Random Errors
1	VM-8	Direct	Low
2	VM-8	Direct	Medium
3	VM-8	Direct	High
4	VM-8	Out-of-Orbit	Low
5	VM-8	Out-of-Orbit	Medium
6	VM-8	Out-of-Orbit	High
7	VM-9	Direct	Low
8	VM-9	Direct	Medium
9	VM-9	Direct	High
10	VM-9	Out-of Orbit	Low
11	VM-9	Out-of-Orbit	Medium
12	VM-9	Out-of-Orbit	High

Note:

1. All cases were run for two values of post-chute ballistic coefficient β_0 , and three sampling intervals.

	β_0 (lb/ft ²)	Sampling Intervals (sec)		
		Nominal β	Low β	High β
Nominal	0.393	0.5	1	1
Low	0.0982	1	2	2
		2	3	4

2. The trajectory parameters are as follows:

	Entry Velocity (km/sec)	Entry Angle
Direct	6.34	25°
Out of Orbit	4.66	16°

3. The total random errors are identified as follows (in percent of full scale, one sigma value):

Errors (%)	Pressure	Temperature
Low	1	0.5
Medium	3	1
High	5	2

4. The nominal β out-of-orbit case corresponds to Point Design 1, while the nominal β direct entry case corresponds to Point Design 2.

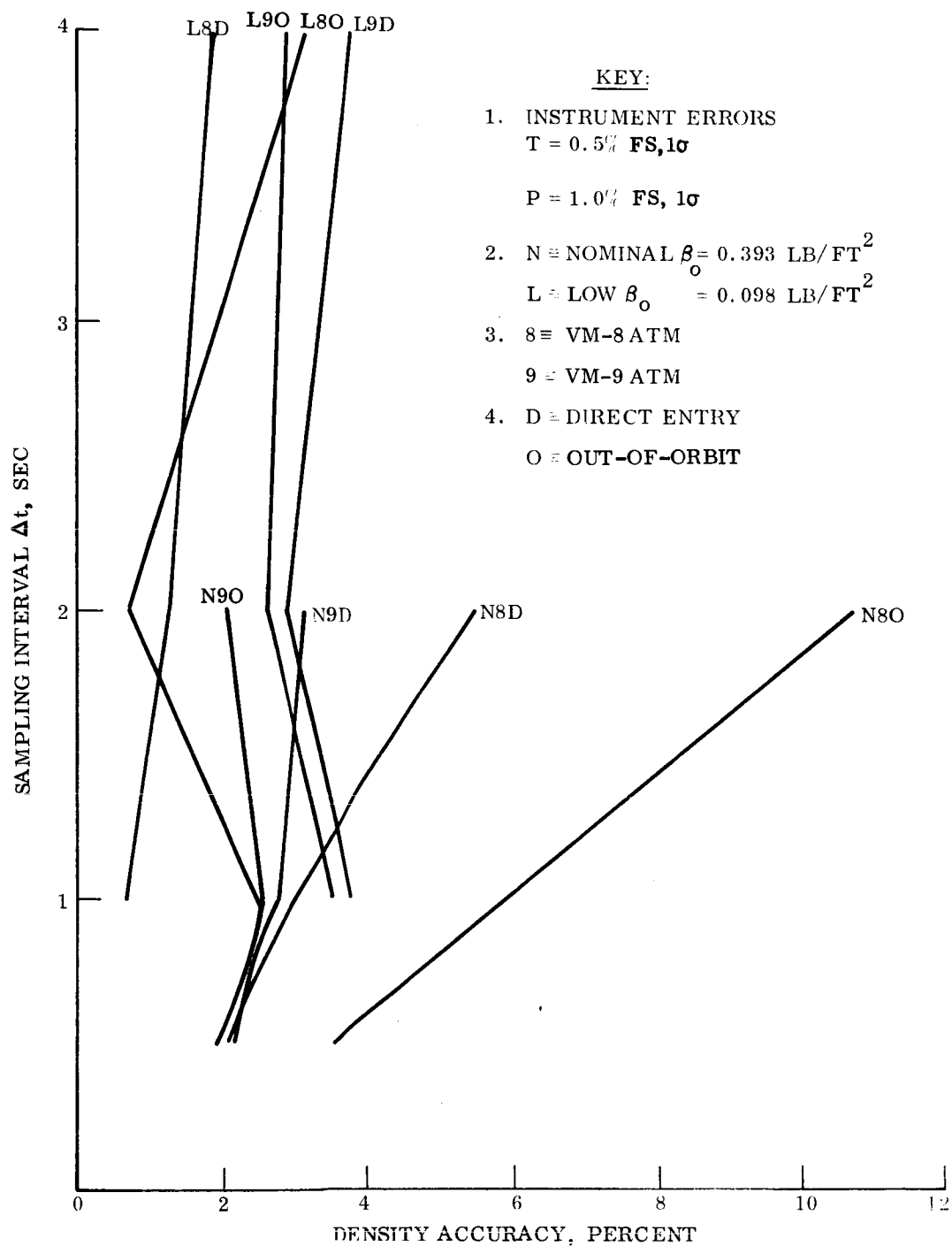


Figure 3.4.2-1. Average Altitude Accuracy vs Sampling Interval
(Low Random Errors)

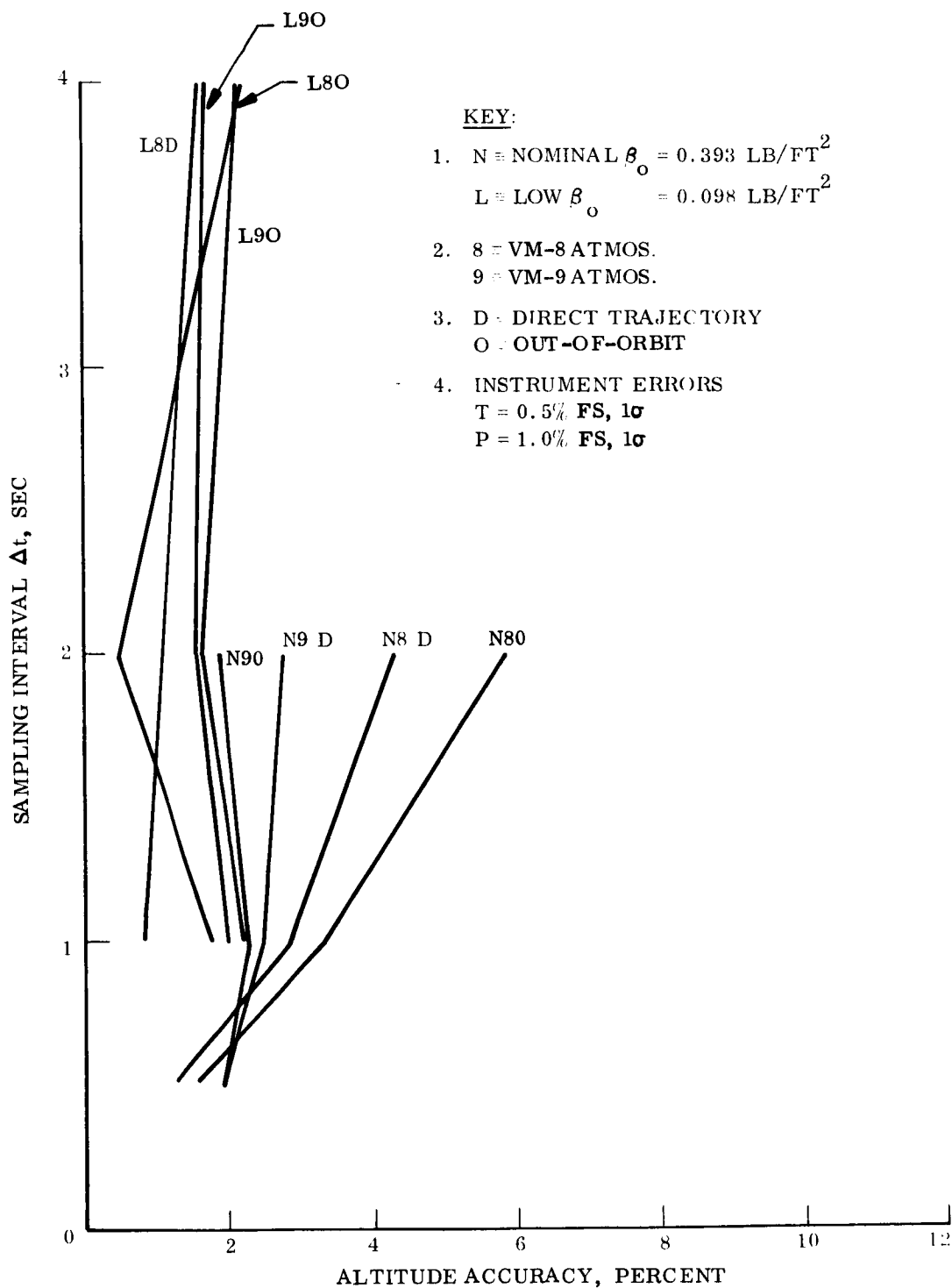


Figure 3.4.2-2. Average Density Accuracy vs Sampling Interval (Low Random Errors)

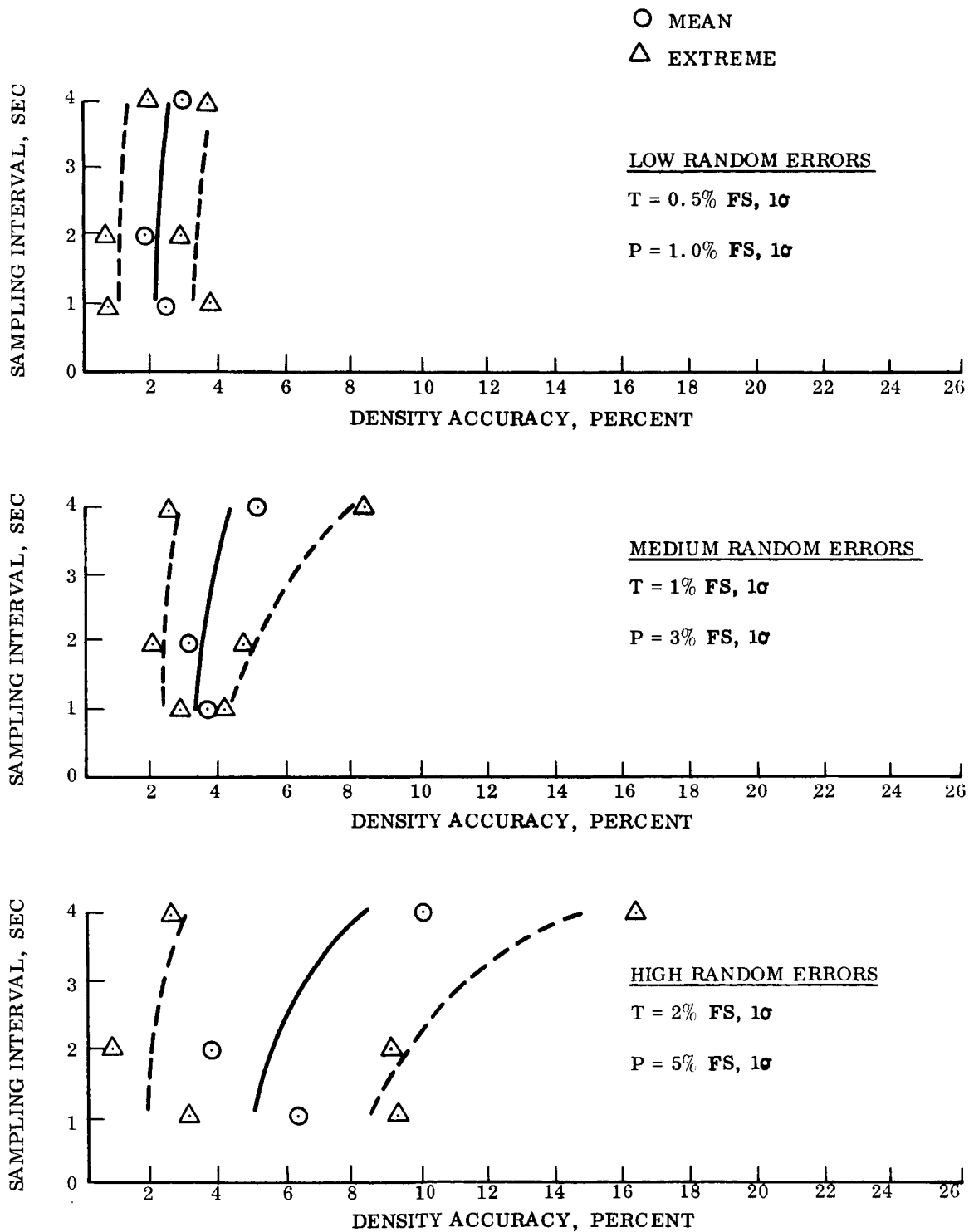


Figure 3.4.2-3. Comparison of Uncertainty in Density Accuracy vs Data Errors for Low $\beta = 0.0982 \text{ lb/ft}^2$

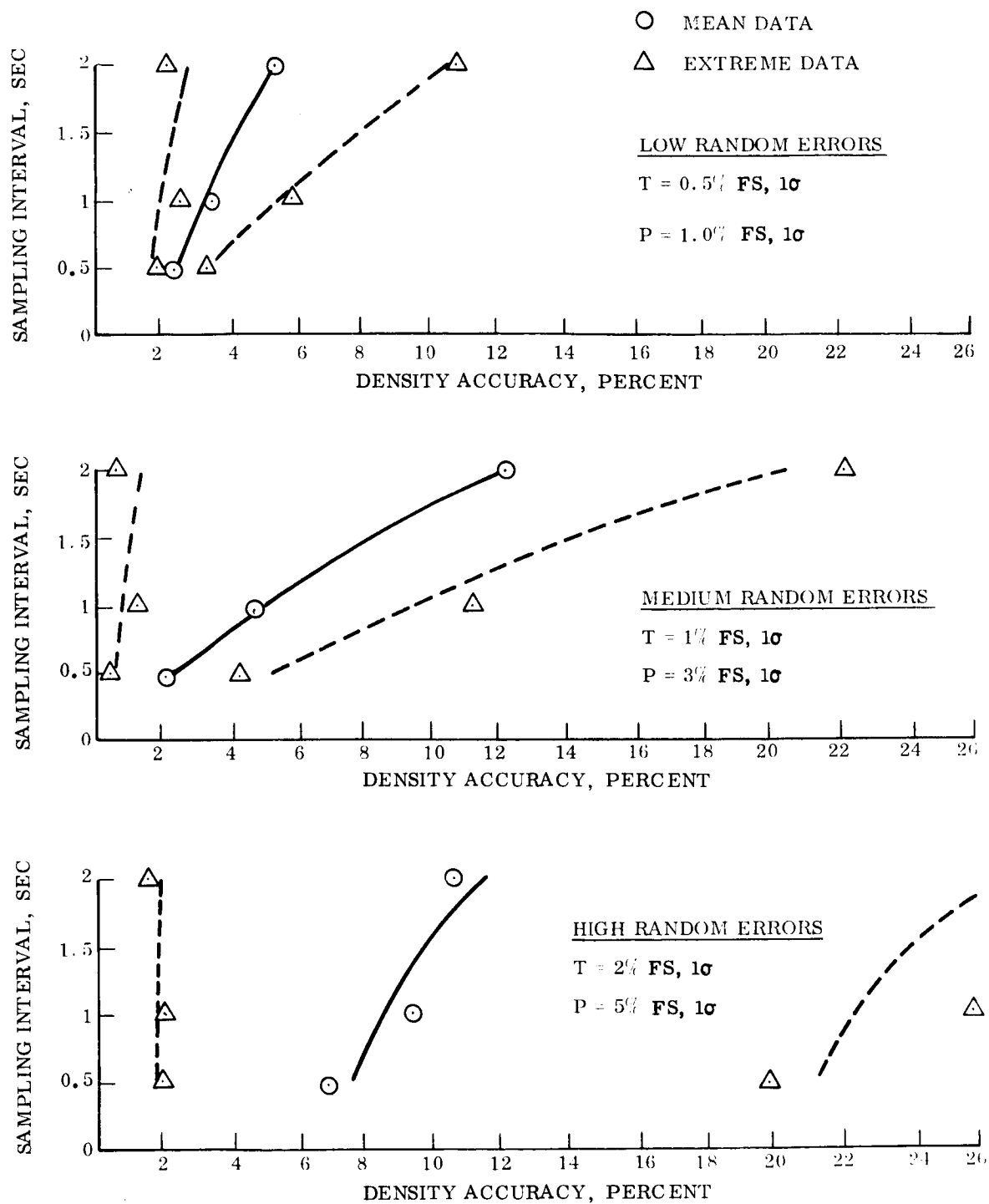


Figure 3.4.2-4. Comparison of Uncertainty in Density Accuracy vs Data Errors for Nominal $\beta = 0.393 \text{ lb/ft}^2$

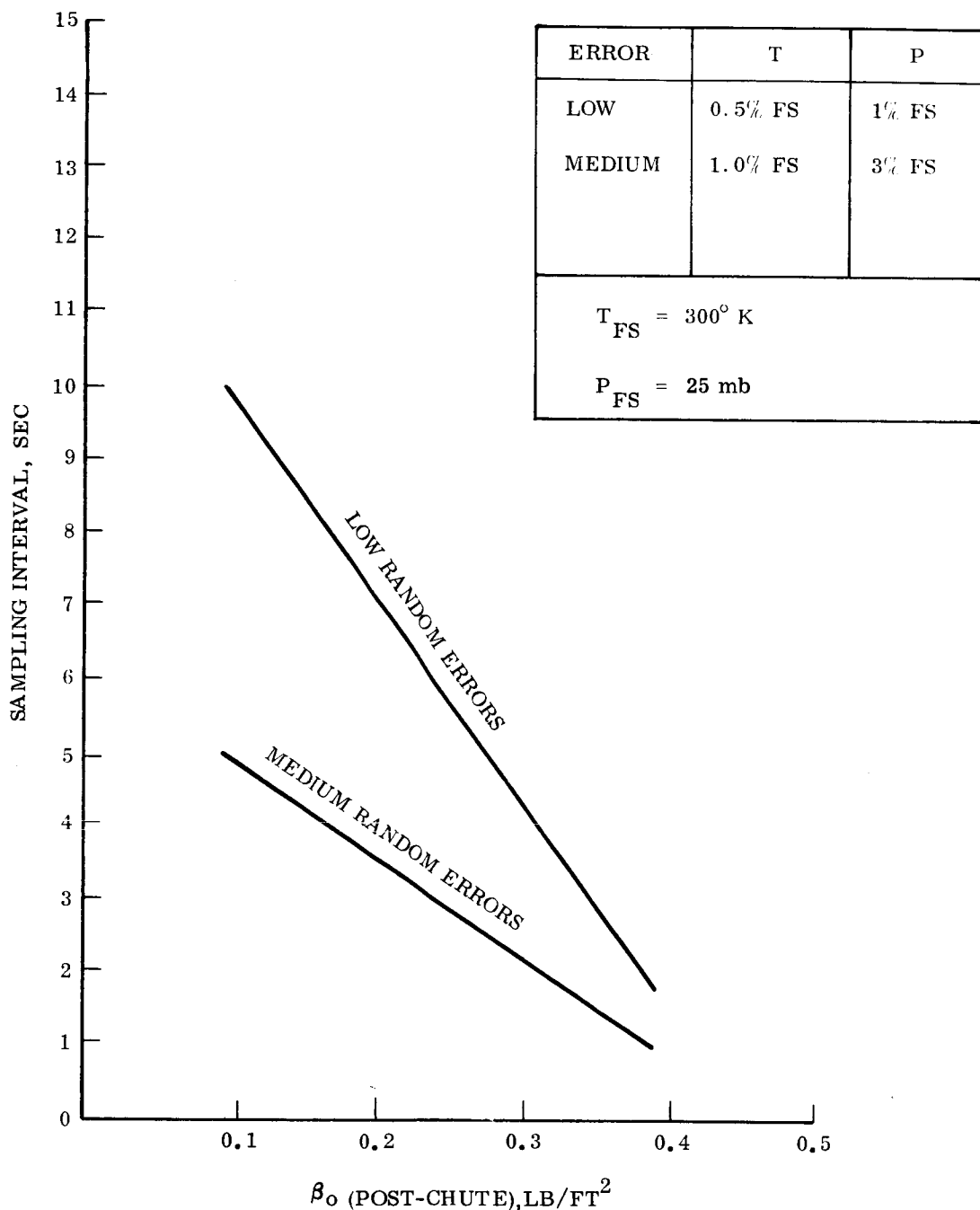


Figure 3.4.2-5. Sampling Interval - Ballistic Coefficient Relations for Achieving 5 Percent Measurement Accuracy of Density-Altitude Profile

As an additional result, fig. 3.4.2-8 shows the consequence of applying the above procedure to the calculation of mean molecular weight. The percent accuracies refer to deviations from the nominal values of 44.0 for the VM-8 atmosphere and 31.2 for VM-9. Choosing an accuracy goal of five percent it can be seen that a sampling interval of one second would suffice. The analysis technique used, therefore, could serve as a valuable back-up to the mass spectrometer in obtaining composition information.

It is clear that the results shown, although based on considerable data, still constitute a far from adequate statistical sample. In order to obtain truly reliable results many more cases should be run, and each case should be repeated several times to average out the random input data errors. In this study each case evaluated involved only a "single entry" into the atmosphere, that is, one set of simulated raw data, containing one set of random errors. Also, as pointed out earlier, more attention must be given to the choice of appropriate smoothing techniques. When these steps are taken much more reliable prediction of experiment accuracies will be attained.

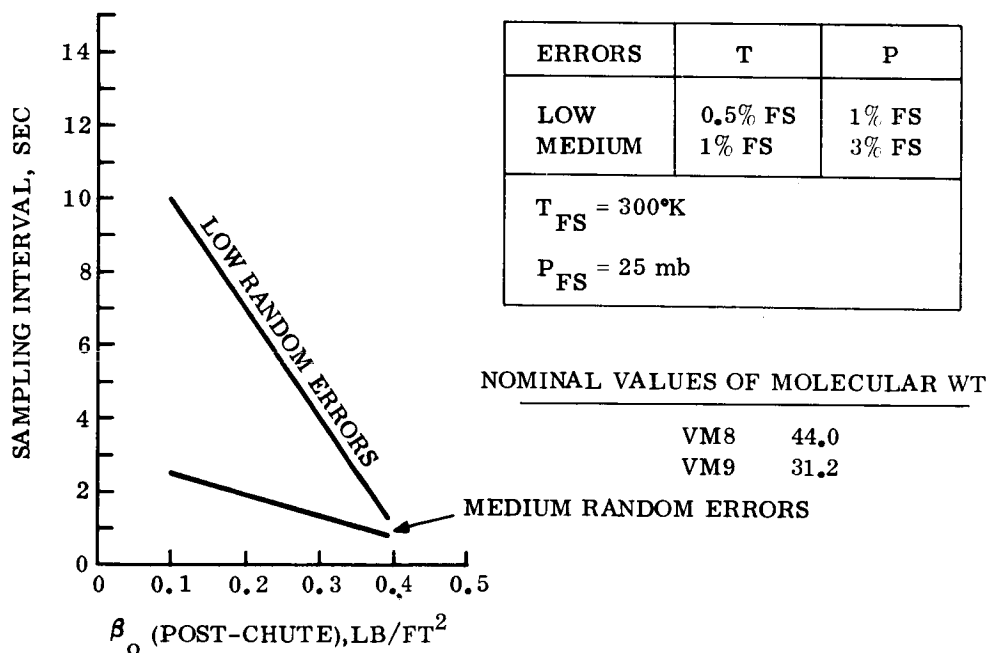


Figure 3.4.2-8. Sampling Interval - Ballistic Coefficient Relative for Achieving 5 Percent Accuracy of of Molecular Weight

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