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SAIC Science Applications International Corporation
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With addendum of new results

December 6, 1989

MEMO TO: Roger Bourke, EIS Team
FROM: Alan Friedlander, SAIC
SUBJECT: Analysis of Success Probability/Cost Trades for Small Landers in a Mars Network

The premise to be tested in this analysis is whether cost economies may accrue by delivering more landers designed to lower reliability of operation (compared to fewer landers of higher reliability) to obtain a desired probability of achieving a given number of lander successes. Generally, the application in mind is a network of penetrators, although the analysis may apply as well to other small lander concepts or even to simple rovers. In previous MRSR studies, the approach taken to raise the probability of a successful mission (e.g. a rover or sample return objective) was to invoke a dual launch policy utilizing identical flight systems. With this approach we found that a substantial improvement in achieving at least one total mission success was gained for realistic values of system element reliability, albeit at the expense of higher program cost and more complex operations. However, in the case of a large number of small landers whose recurring cost of production might be small compared to the development cost, a single spacecraft carrier may be sufficient to deliver these landers to Mars within acceptable limitations of spacecraft injected mass and launch vehicle performance capability. It seems reasonable to at least explore the question of potential economies if such landers were purposely designed to lower values of reliability. What is specifically meant by lower reliability in this context is that, while fewer lander emplacements will succeed, those that do succeed will accomplish the desired mission objectives. The underlying assumption here is a certain degree of independence of lander system failure modes such that objective-specific elements (science instruments and data communications) are highly reliable while delivery-specific elements (e.g. deorbit propulsion and aeroshell) are less reliable and developed at lower cost with attendant higher risk. This analysis leaves open the important question as to whether such an approach is at all realistic in terms of engineering design, but focuses instead on the first question of potential cost advantage.

The method of analysis is based on a probabilistic model of lander success and a related probabilistic model of project cost including the lander, spacecraft carrier, and integration, but not launch or operations costs. Quantitative results are obtained in a normalized and parametric fashion. Sensitivity to the assumed model parameters is also examined.

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Mission Success Model

Consider (n) landers each of which have the same level of reliability (p) for achieving individual mission success. Assuming that the actual failure events of different landers are statistically independent (even though the underlying failure modes for contributing components may be related), then the probability that exactly (m) of these landers are successful is given by the binomial distribution

$$P(\text{m successes}) = \left[\, n! \,/\, (m! * (n - m)!) \right] * p^{m} * (1 - p)^{n-m} \tag{1}$$

where ! denotes the factorial operator and * denotes multiplication. Mission success also depends on the reliability of the launch vehicle and the spacecraft carrier that delivers the landers to Mars. To take these factors into account, we define P_l as the probability of a successful launch event and P_c as the probability of a successful delivery event. Then, the overall probability P that at least m (i. e. m or more) landers will be successful (for a single launch) is calculated by the expression

$$P = P_l\, P_c \sum_{i = m \text{ to } n} \left[\, n! \,/\, (i! * (n - i)!) \right] * p^{i} * (1 - p)^{n-i} \tag{2}$$

The relationships of Equation (2) are illustrated in Figure 1 for $P_l = 0.94$, $P_c = 0.98$, and $p = 0.8$.

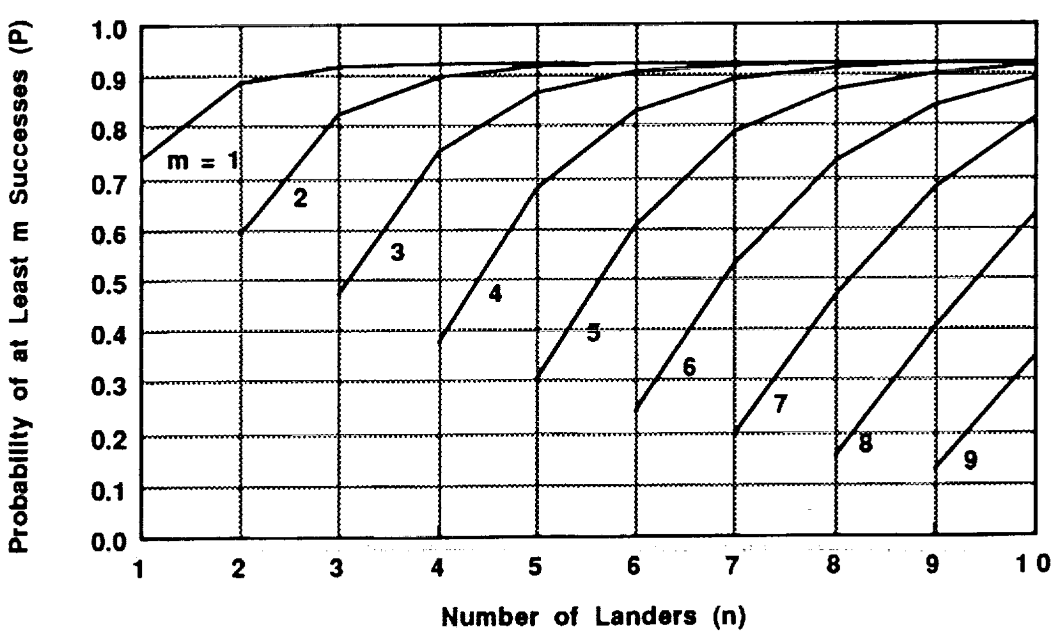


Figure 1 Lander Success Probability P(n, m) for Lander Reliability p = 0.8
Pl = 0.94, Pc=0.98

Mission Cost Model

The lander system development cost is modeled in terms of design reliability by the relationship

$$C_d = C_{do} / (1 - p)^a \tag{3}$$

where C_{do} is a "reference" development cost at p = 0, and the exponent (a) is a model parameter. This equation is graphed in Figure 2 for values of a = 0.1, 0.2, and 0.301.

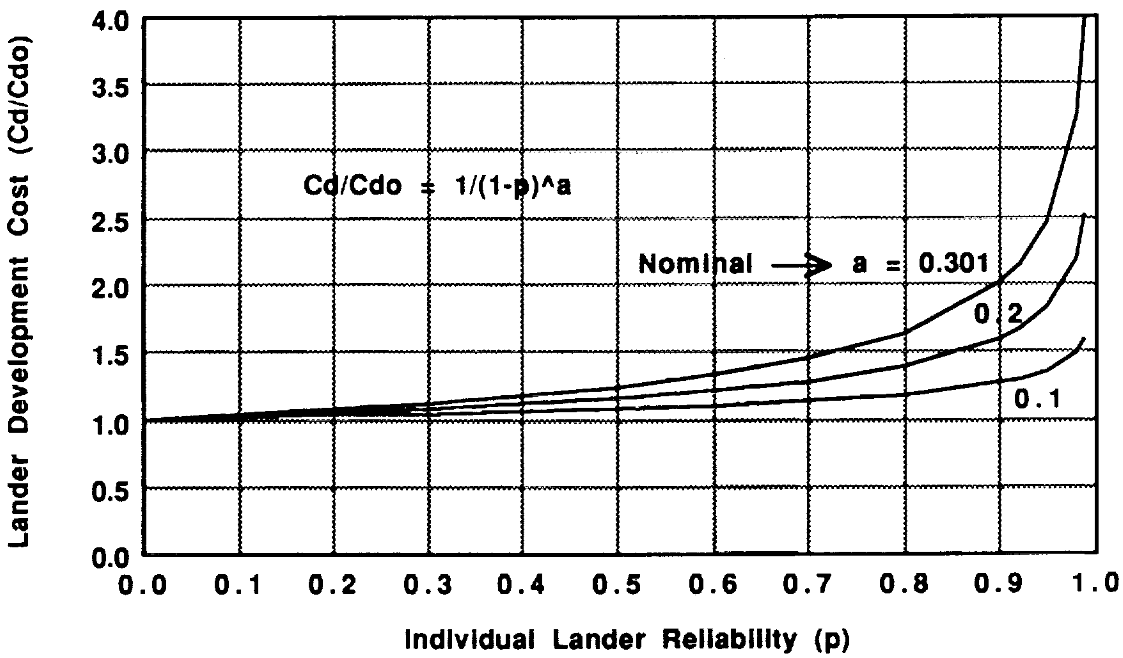


Figure 2 Lander Development Cost Model

The nominally selected value of the development cost parameter is a = 0.301 = $\log_{10} 2$, which gives a doubling of cost from p = 0 to p = 0.9 and doubling again for p = 0.99, etc. For a = 0.1 the increase in cost is only 25% for each additional 9 in reliability. The sensitivity to this parameter will be tested later. Recurring cost for each additional lander is assumed to be a constant fraction of the development cost. Hence, the lander system cost model is represented by

$$LC = C_d\,(1+k_1{*}n) = C_{do}\,(1 + k_1{*}n) / (1 - p)^a \tag{4}$$

where the nominal value of the constant is selected as $k_1 = 0.2$. Total project cost includes the lander, carrier, and a cost element associated with hardware integration, management, and contingency. The carrier spacecraft cost is taken to be proportional to the reference lander cost (development + recurring) at $p = 0$. Integration, management, and contingency is taken to be proportional to the sum of the carrier cost and the reference lander cost. Hence, the total project cost model is represented by

$$\begin{aligned} PC &= LC + k_2{*}C_{do}\,(1+k_1{*}n) + k_3{*}[LC + k_2{*}C_{do}\,(1+k1{*}n)] \\ &= C_{do}\,(1+k_1{*}n)\,[1/(1-p)^a + k_2\,(1+k_3) + k_3] \end{aligned} \tag{5}$$

where the nominal parameters are $a = 0.301$, $k_1 = 0.2$, $k_2 = 0.667$, and $k_3 = 0.4$. The final step in the cost model is to normalize LC and PC to their respective values LC^* and PC^* corresponding to one lander ($n = 1$) and reliability $p = 0.8684$ evaluated at the nominal values of the cost model parameters. Hence, $LC^* = 2.209\ C_{do}$ and $PC^* = 3.809\ C_{do}$. Lander system relative cost and total project relative cost are graphed in Figures 3 and 4 as a function of the number of landers and the individual lander reliability.

Results

Solution of the mission success model (Equation 2) was obtained for a constant probability $P = 0.8$ that at least (m) landers will be successful. These calculations assume the nominal values of 0.94 for launch success and 0.98 for carrier success; these values yield the reference lander reliability of $p = 0.8684$ for a single lander. Results are shown in Figure 5 which plots the required lander reliability as a function of the number of landers (n) and the minimum number of lander successes (m). The solution values for p are then used to evaluate the normalized total project cost which is graphed in Figure 6. Note that for each value of (m) there is a number of landers ($n > m$) that yields the lowest cost. Generally, (n) is greater than (m) by one or two lander units. This result substantiates the initial contention that more landers of lower reliability may provide cost economy. The intersection points along the minimum cost locus can be mapped into Figure 5 to determine the lander reliability values; the range is $p = \{0.64, 0.87\}$ as m varies from 1 to 8. For example, to obtain at least six lander successess ($m = 6$) at a probability of 80%, the minimum relative cost is $PC/PC^* = 2.084$ (i.e. twice the single lander project cost) with $n = 8$ and $p = 0.835$. Note also that the cost curve is fairly flat for $n > 8$, so that if $n = 10$ the project cost increases to only 2.195 but the required lander reliability decreases to $p = 0.711$. By comparison, if $n = m = 6$, then the required reliability is quite high at $p = 0.977$ and the project relative cost increases to 2.563. One could also interpret the results for a constant cost as (m) varies. For example, if $PC/PC^* = 2.0$ or less, then for values of $m = \{1, 2, 3, 4, 5\}$ the minimum necessary lander reliabilities are $\{0.18, 0.31, 0.42, 0.57, 0.74\}$ at corresponding values of $n = \{10, 10, 10, 9, 8\}$.

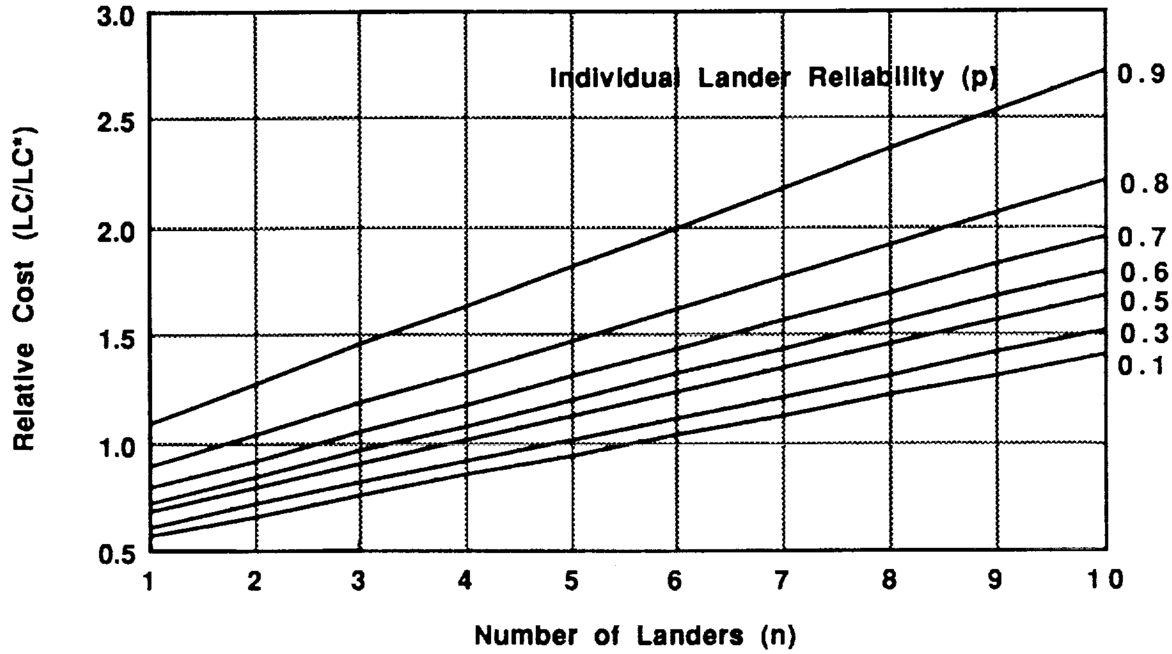


Figure 3 Lander System Relative Cost
LC* = 2.209 x Cdo (for n = 1, p = 0.8684)

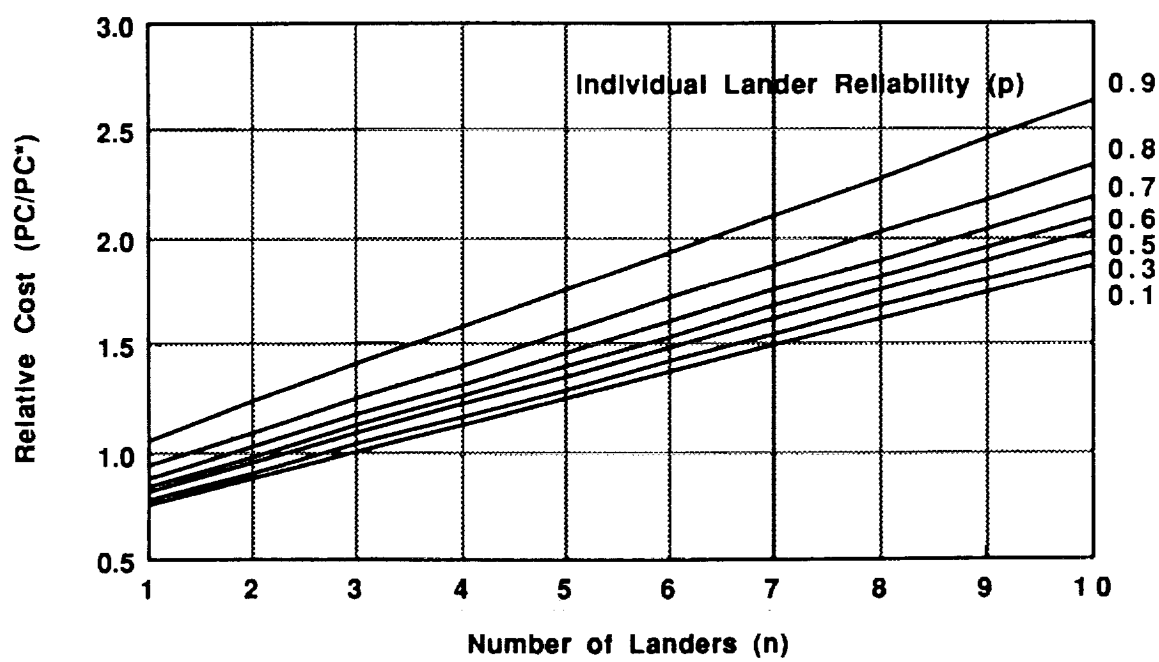


Figure 4 Total Project Relative Cost
PC* = 3.809 x Cdo (for n = 1, p = 0.8684)

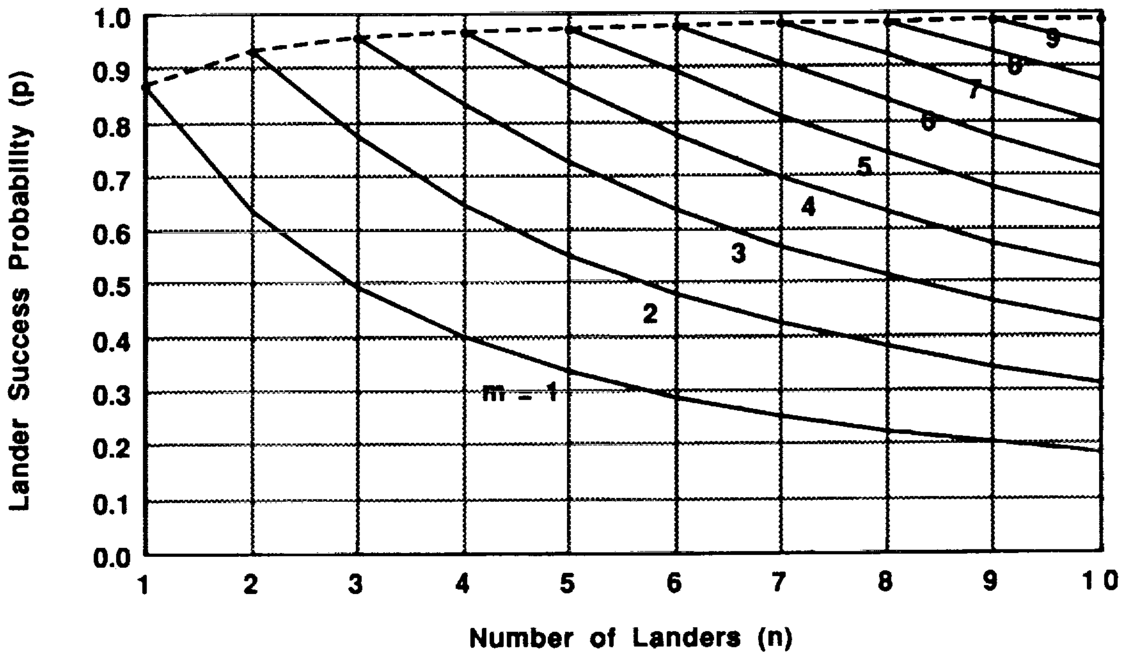


Figure 5 Required Lander Reliability for at Least m Lander Successes with Probability P = 0.8 (Pl = 0.94, Pc = 0.98)

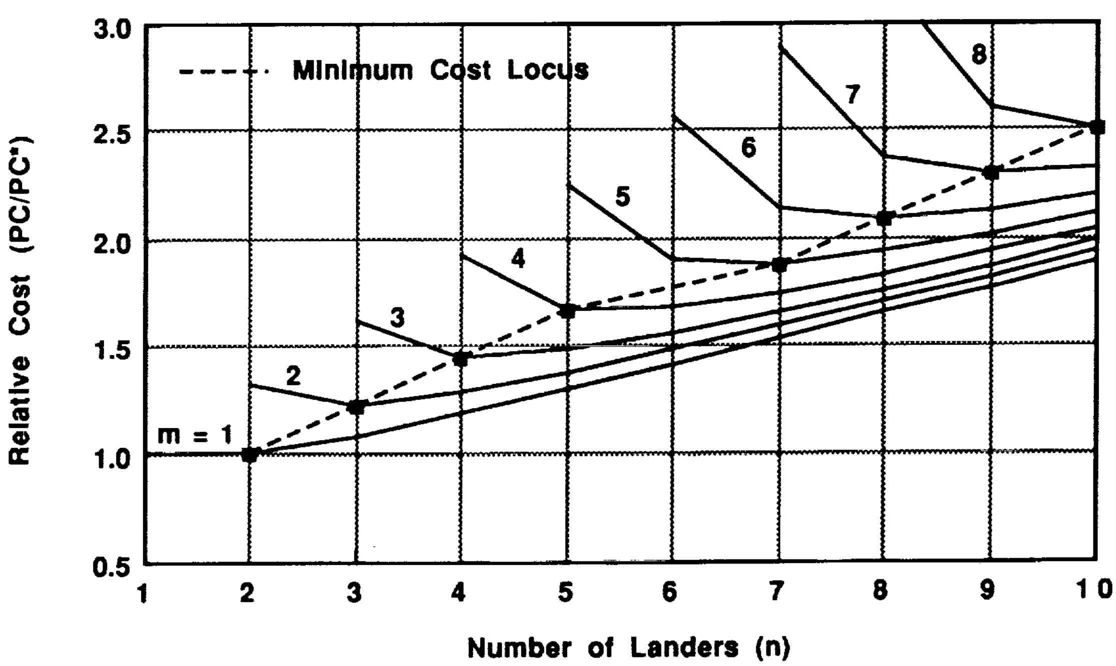


Figure 6 Project Cost for at Least m Lander Successes with Probability P = 0.8 (Pl =0.94, Pc = 0.98)

Similar types of solution data are presented in Figures 7 and 8 for a constant success probability of P = 0.9. In this case, of course, the level of both cost and required reliability is raised to satisfy the more demanding 90% success capability. For example, at m = 6, the minimum relative cost is $PC/PC^* = 2.327$ obtained for n = 9 and p = 0.866. If $PC/PC^* = 2.0$ or less, then for values of m = {1, 2, 3, 4} the minimum necessary lander reliabilities are {0.31, 0.45, 0.61, 0.76} at corresponding values of n = {10, 10, 9, 8}.

Sensitivity to Model Parameters

Model parameters were varied, generally one at a time, to determine the sensitivity of the minimum PC/PC^* solution. These calculations were made for the case of m = 6 and P = 0.8 with PC^* held constant at its reference value 3.809 C_{do}. Results are listed in Table 1.

Table 1
Sensitivity to Model Parameters for m = 6 and P = 0.8
(Parameters at Nominal Values Unless Otherwise Noted)

Varied Parameter		n	p	PC/PC^*
a =	0.1	6	0.9768	1.611
	0.2	7	0.9074	1.854
	0.301	8	0.8351	2.084
	0.4	9	0.7694	2.302
k_1 =	0.1	9	0.7694	1.441
	0.2	8	0.8351	2.084
	0.3	8	0.8351	2.725
k_2 =	0.5	8	0.8351	1.925
	0.667	8	0.8351	2.084
	1.0	8	0.8351	2.402
k_3 =	0.2	8	0.8351	1.857
	0.4	8	0.8351	2.084
	0.6	8	0.8351	2.312
P_l*P_c =	0.84	8	0.8910	2.240
	0.9212	8	0.8351	2.084
	1.00	8	0.8014	2.020
$P_l*P_c = 0.84$, $a = 0.4$, $k_1 = 0.3$, $k_2 = 1.0$, $k_3 = 0.6$		9	0.8338	4.127
$P_l*P_c = 1.0$, $k_1 = k_2 = 0$		9	0.7324	1.093

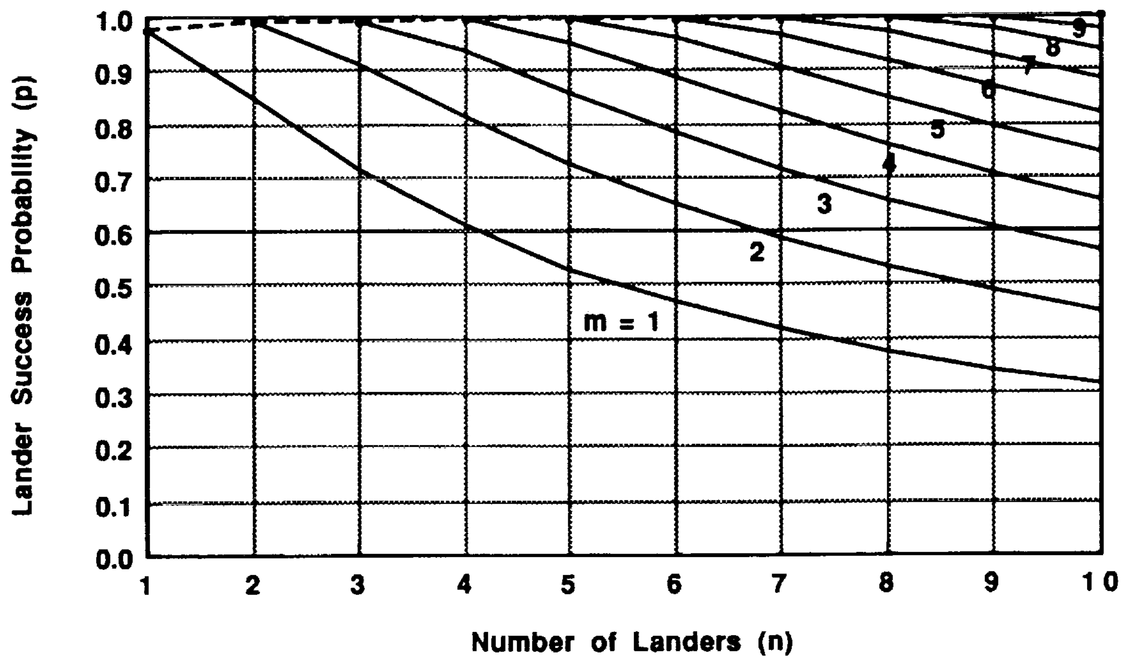


Figure 7 Required Lander Reliability for at Least m Lander Successes with Probability P = 0.9 (Pl = 0.94, Pc = 0.98)

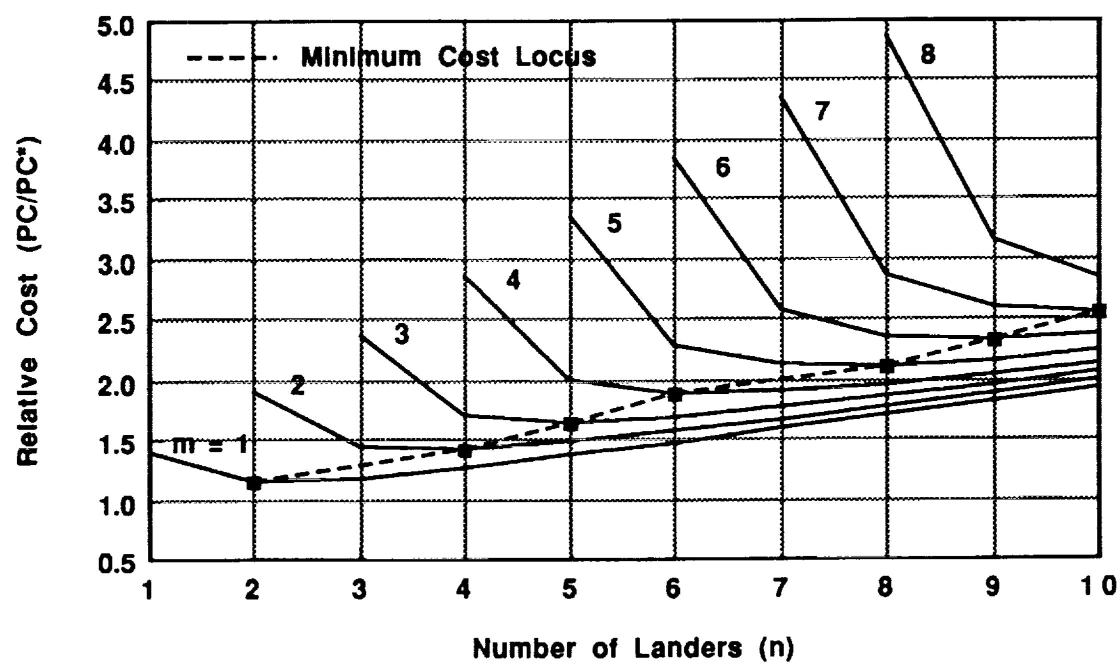


Figure 8 Project Cost for at Least m Lander Successes with Probability = 0.9 (Pl = 0.94, Pc = 0.98)

Comparison of One and Two Launch Scenarios

Results presented so far have been for a single launch of (n) landers. Additional calculations were made for two launches, but this required modification of the mission success and cost models. To calculate the probability P for at least (m) lander successes with two launches, it is necessary to use Equation (1) as the basic model for lander success, multiply each term by the product $P_l * P_c$ except for the m = 0 term which is adjusted to $[(1 - P_l * P_c) + P_l * P_c * P_{m=o}]$, and then obtain the various combinations for exactly (m) successess with two launches. The probability for at least (m) lander successes can then be calculated by summation of terms as in Equation (2). The project cost model for two launches is taken as a modification of Equation (5)

$$PC(2) = C_{do} \left\{ (1+ k_1 * n) \left[1/ (1 - p)^a + k_3 \right] + 2 * k_2 (1 + k_3) * (1+ k_1 * n/2) \right\} \tag{6}$$

where (n) is the total number of landers for two launches.

Employing the nominal values of model parameters, the first comparison case examined is n = 4 and a constant probability P = 0.8 that at least (m) landers will be successful. The single launch carries 4 landers while the dual launch system carries 2 landers each. Results for this case are listed in Table 2.

Table 2
Comparison of One and Two Launches for n = 4 and P = 0.8

	One Launch (n=4)		Two Launches (n = 2+2)	
At Least m Successes	p	PC/PC*	p	PC/PC*
1	0.3977	1.180	0.3675	1.417
2	0.6447	1.275	0.7747	1.615
3	0.8336	1.441	0.8951	1.806
4	0.9653	1.929	0.9854	2.561

Although the relative cost for two launches is always higher, if the criterion of comparison is the minimum value of lander reliability (p), then the results indicate that two launches is better only for the condition m = 1. If more than 2 lander successes is desired, a higher reliability is required because of the influence of possible launch and carrier failures.

The second comparison case examined is n = 8 and a constant value of p = 0.8 for the reliability of each lander. In this case we compare the mission success probability P(m) for m = 1 to 8. The relative project costs are $PC(1)/PC^* = 2.018$ and $PC(2)/PC^* = 2.263$ for all values of (m). Results are listed in Table 3.

Table 3
Comparison of One and Two Launches for n = 8 and p = 0.8

At Least m Successes	One Launch (n = 8) P	Two Launches (n = 4+4) P
1	0.9212	0.9936
2	0.9211	0.9898
3	0.9201	0.9665
4	0.9116	0.8992
5	0.8694	0.8008
6	0.7341	0.6763
7	0.4637	0.4271
8	0.1546	0.1424

These results indicate a "success performance" crossover point between one and two launches at the value m = {3, 4}. That is, two launches are better as measured by probability of success only for the condition m = 1, 2, or 3. If 4 or more lander successes is desired, then the single launch policy yields a somewhat higher probability of that occurrence.

Addendum to December 6, 1989, Memorandum

I did some more sensitivity studies relative to the cost model assumption. The results still confirm the conjecture (generally) that more landers at lower reliability yield lower project cost.

Basic Cost Model No.1 (as per memos)

$$C_d/C_{d_o} = \frac{1}{(1-p)^a}$$

Basic Cost Model No.2 (modified "Bourke")

$$C_d/C_{d_o} = \left(\frac{1}{1-p}\right)^a$$

Results for $m = 6$ and $P(m \geq 6) = 0.8$

			Minimum Cost Solution	
Cost Model	a	n	p	PC/C_{d_o}
No. 1	0.100	6	0.9768	6.136
	0.301	8	0.8351	7.938
	0.500	9	0.7693	9.561
	1.000	13	0.5767	10.334
No. 2	0.100	6	0.9768	6.130
	0.301	8	0.8351	7.703
	0.500	10	0.7113	8.710
	1.000	15	0.5110	9.513

Note: The greater the sensitivity (a), the more landers (n) desired.

Combining Independent and "Common Cause" Failure Events

Consider (n) landers on a single launch. Each lander has an independent reliability = p_i and a common cause (or bias) reliability component = p_d. Then, if S_m represents the event of exactly m successes, the total conditional probability formula is

$$P(S_m) = P(S_m/\bar{D})P(\bar{D}) + P(S_m/D)P(D)$$

where

$\bar{D}$ = event that common cause failure *does not* occur
D = event that common cause failure *does* occur
$P(S_m/\bar{D})$ obtained from binomial distribution, as before

$$P(S_m/D) = \begin{cases} 1.0 \text{ for } m=0 \\ 0 \text{ for } m>0 \end{cases}$$

$$P(\bar{D}) = p_d\,;\ P(D) = 1 - p_d$$

Distribution between failure event types

$$p = p_i\, p_d = (1 - f_i)\,(1 - f_d) = 1 - f$$

Let

$$k_d = f_d/f = \frac{1 - p_d}{1 - p_i\ p_d}\,;\ 0 \le k_d \le 1$$

or

$$p_d = \frac{1 - k_d}{1 - k_d\ p_i} = 1\ - k_d\ (1 - p)$$

Special case:

$$p_i\ = p_d = \sqrt{p}$$

$$k_d = (1 - \sqrt{p})/(1 - p)$$

p	$p_i = p_d$	k_d
0.5	0.7071	0.5858
0.6	0.7746	0.5635
0.7	0.8367	0.5445
0.8	0.8944	0.5279
0.9	0.9487	0.5132

Parametric Results For $m = 6$, $p_i = 0.94$, $p_c = 0.98$, $P = 0.8$

k_d	n	p_i	p_d	$\frac{p_c}{p_{c^*}}$ $\left(\text{using } \frac{p_i + p_d}{2} \text{ for cost}\right)$
0.2	6	0.9777	0.9945	2.861
	7	0.9153	0.9793	2.368
	$\boxed{8}$	0.8533	0.9646	$\boxed{2.314}$
	9	0.7984	0.9520	2.355
	10	0.7506	0.9413	2.433
0.5	6	0.9800	0.9804	2.650
	$\boxed{7}$	0.9336	0.9377	$\boxed{2.279}$
	8	0.8965	0.9062	2.281
	9	0.8716	0.8862	2.368
	10	0.8576	0.8754	2.494
$p_i = p_d$				
0.5051	6	0.9800	0.9800	2.645
0.5167	$\boxed{7}$	0.9350	0.9350	$\boxed{2.274}$
0.5259	8	0.9014	0.9014	2.281
0.5314	9	0.8817	0.8817	2.377
0.5340	10	0.8727	0.8727	2.515
0.8	6	0.9858	0.9463	2.368
	$\boxed{7}$	0.9678	0.8859	$\boxed{2.224}$
	8	0.9628	0.8705	2.352
	9	0.9622	0.8687	2.526
	10	0.9621	0.8684	2.705
0.9	6	0.9904	0.9205	2.243
	$\boxed{7}$	0.9838	0.8728	$\boxed{2.233}$
	8	0.9832	0.8687	2.404
	9	0.9832	0.8687	2.589
	10	0.9832	0.8687	2.774

Session B, Submittal No. 6

Manuel I. Cruz
TRW

EARTH PENETRATION CONCEPTS

AND

MARS PENETRATOR ENTRY TRADES

MANNY CRUZ
TRW

Earth Penetration Concepts

TABLE I

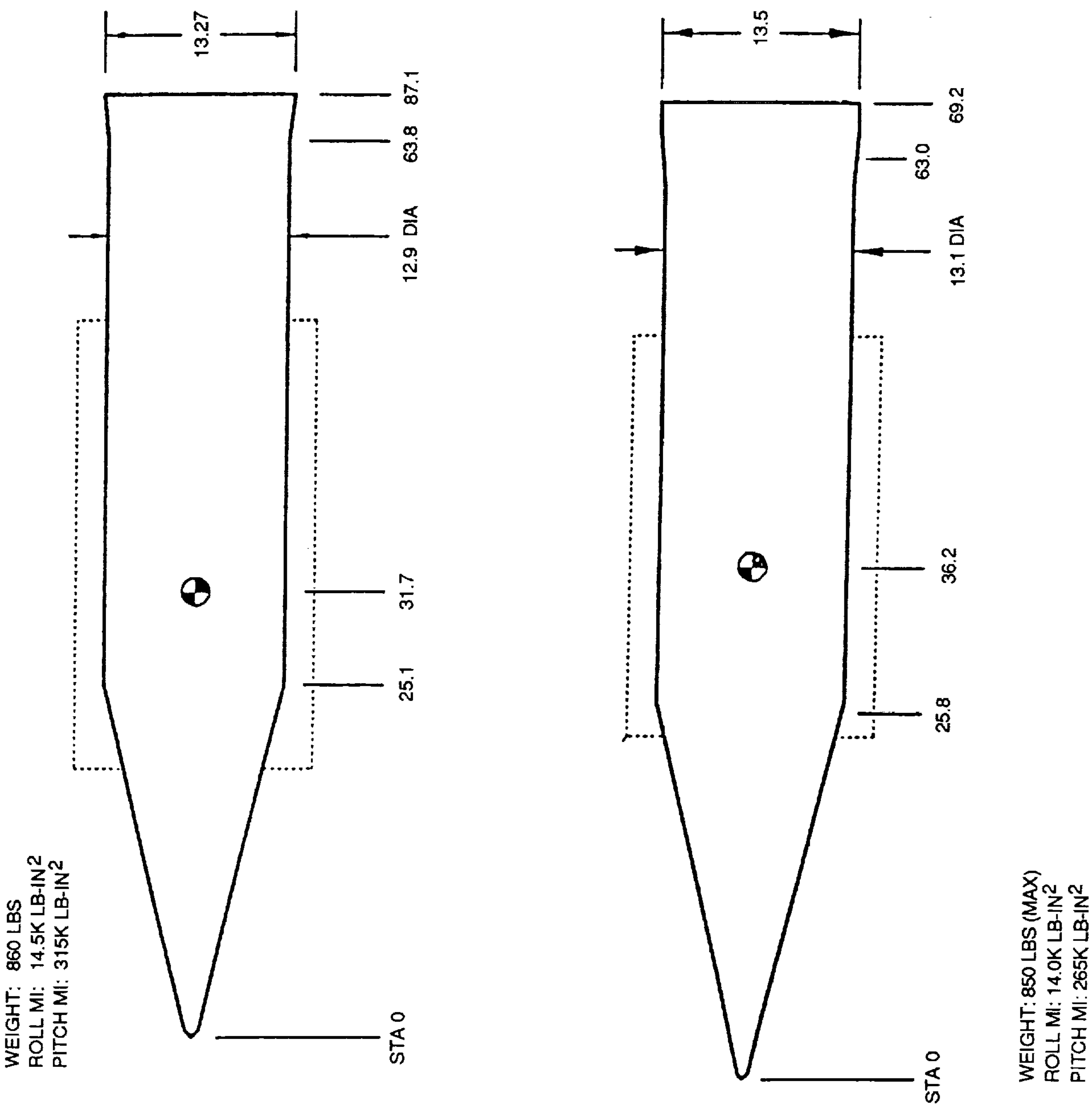
EARTH PENETRATOR RE-ENTRY TO IMPACT CONCEPTS

Concept	Control Mechanism	Ballistic Coefficient (kg/m2)	Staging Conditions			Impact Conditions			Comments
			V (mps)	Gamma (degree)	h (km)	V (mps)	Gamma (degree)	1σ Impact Error (meters/driver)	
Two Stage Ballistic Entry and High Altitude Drag Modulation	Drag modulation and staging from low B to high B	$10 \le B \le 15000$	180 to 250	90	30	300 to 900	90	~300/driven by low altitude winds ≤ 9 mps	Very good terminal velocity and flight path control; large CEP considerations
Two Stage Ballistic Entry	Staging from low B to high B	$10 \le B \le 15000$	180 to 250	90	30	300 to 900	90	~600 to 900/high altitude density variation ≤ 20 L/D low altitude winds ≤ 9 mps	Very good terminal velocity and flight path control, large CEP considerations
Two Stage Ballistic Entry	Staging from moderate B to high B	$730 \le B \le 15000$	300 to 900	50	18	450 to 900	50	~300 to 600/high altitude density variation ≤ 20 L/D low altitude winds ≤ 9 mps	Plagued by lateral load carrying capability of impact system medium strength rock
Ballistic Entry Moderate B	None	2500	N/A	N/A	N/A	215	60	~300	Plagued by rebound and/or ricochet
Ballistic Entry High B	None	$9800 \le B \le 15000$	N/A	N/A	N/A	2100 to 3400	30	Driven by delivery system	Exceeds load carrying capability of impact system
Maneuvering Re-Entry Pitch/Bank	Lift Modulation	$4900 \le B \le 15000$	N/A	N/A	N/A	300 to 900	90	Driven by delivery system	Very good velocity and flight path control throughout impact error exceeds requirements

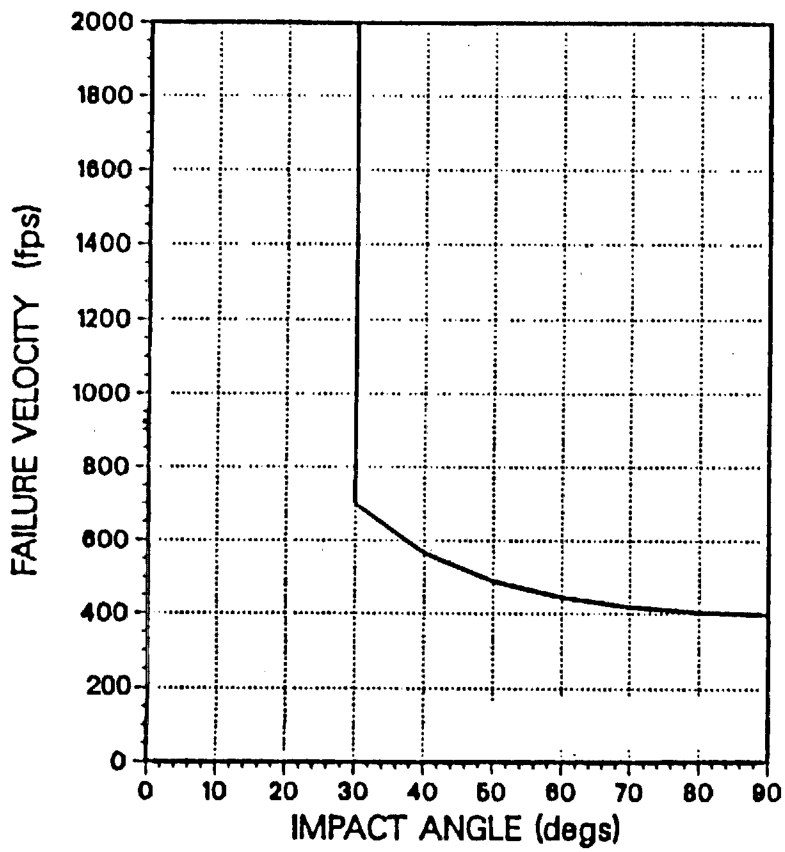
PENETRATION PREDICTION

$$D = 0.0031\ (S)\ (N)\ \sqrt{W/A}\quad (V-100), \quad V \geq 200\ \text{FT/SEC}$$

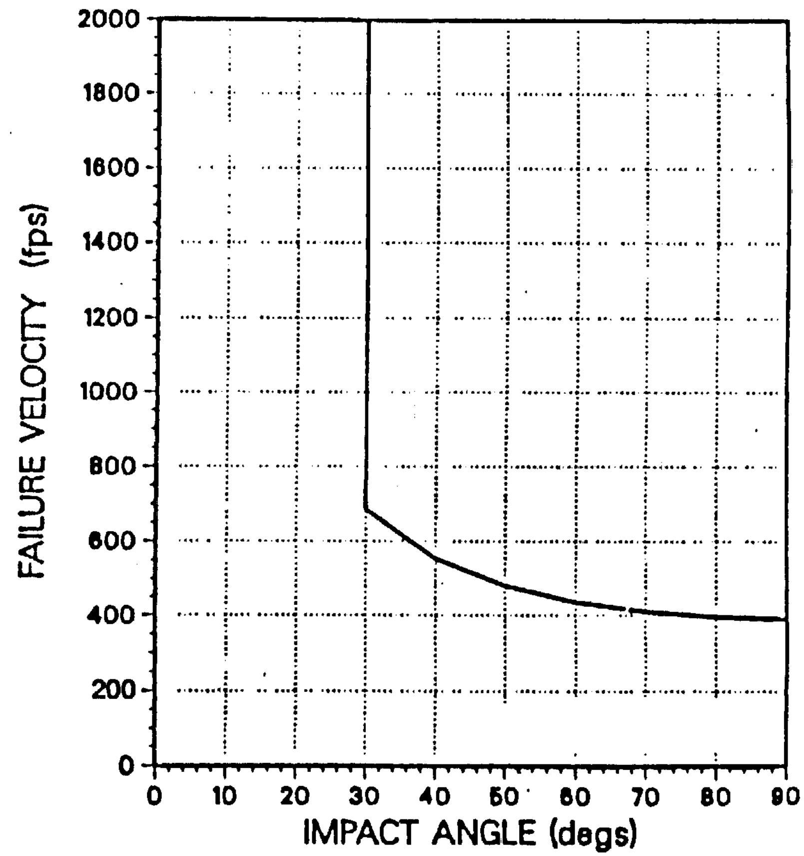
D = PENETRATION DEPTH (FT)
S = SOIL CONSTANT
N = NOSE PERFORMANCE COEFFICIENT
W = WEIGHT (LB)
A = CROSS-SECTIONAL AREA (FT^2)
V = VELOCITY AT IMPACT (FT/SEC)



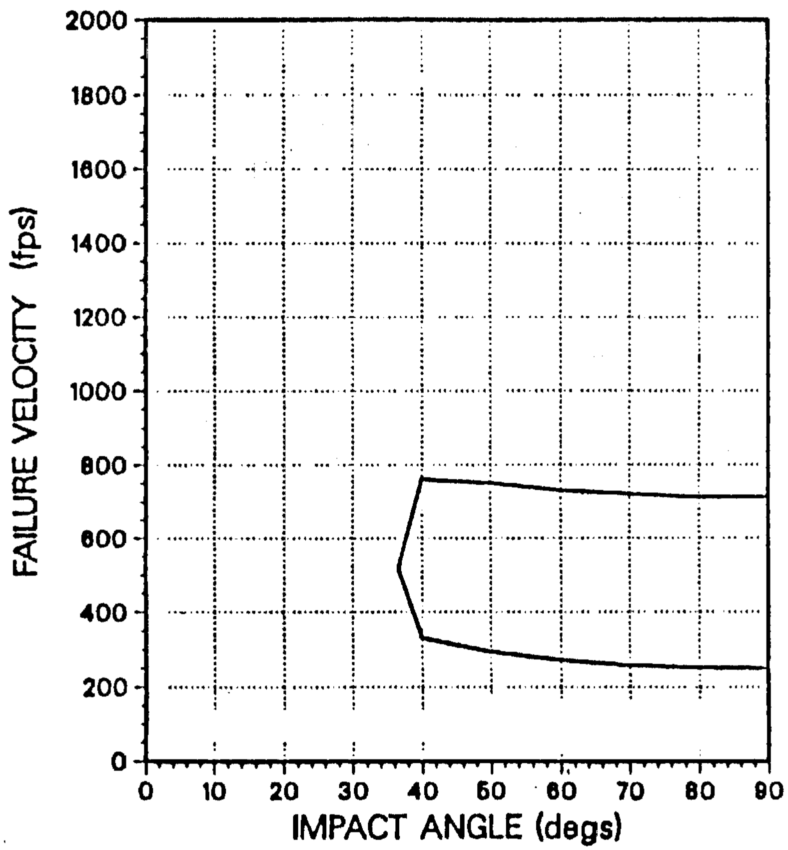
WEIGHT: 860 LBS
ROLL MI: 14.5K LB-IN^2
PITCH MI: 315K LB-IN^2
13.27
87.1
63.8
12.9 DIA
31.7
25.1
STA 0
13.5
69.2
63.0
13.1 DIA
36.2
25.8
STA 0
WEIGHT: 850 LBS (MAX)
ROLL MI: 14.0K LB-IN^2
PITCH MI: 265K LB-IN^2



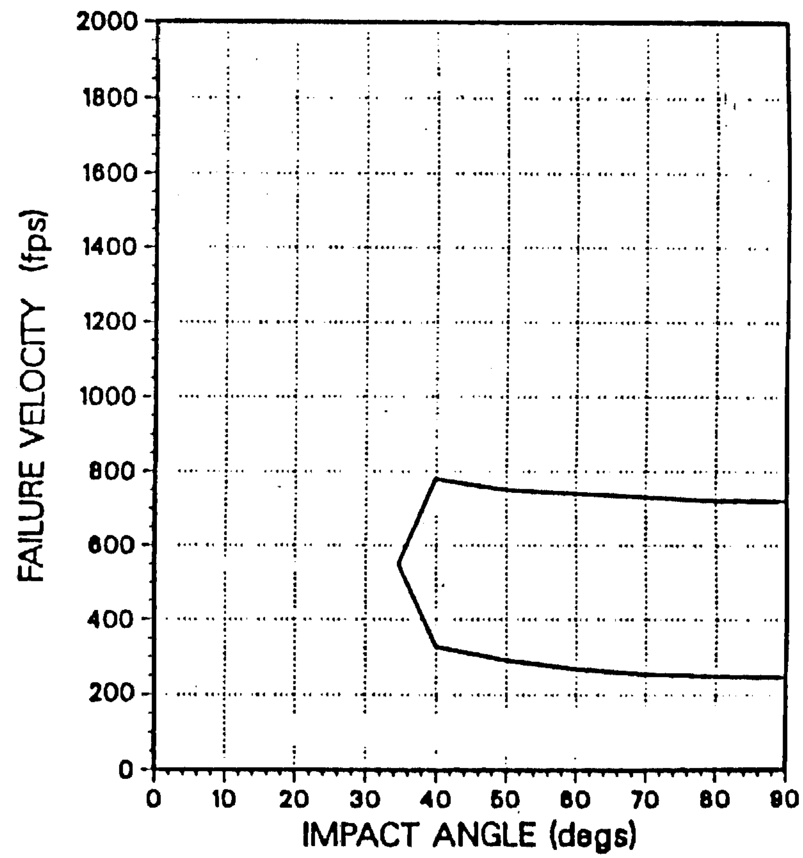
Baseline Design - Target Type 3 (Soil)
-2.0 Degrees Angle of Attack



860 lb Design Excursion 1 - Target Type 3 (Soil)
-2.0 Degrees Angle of Attack

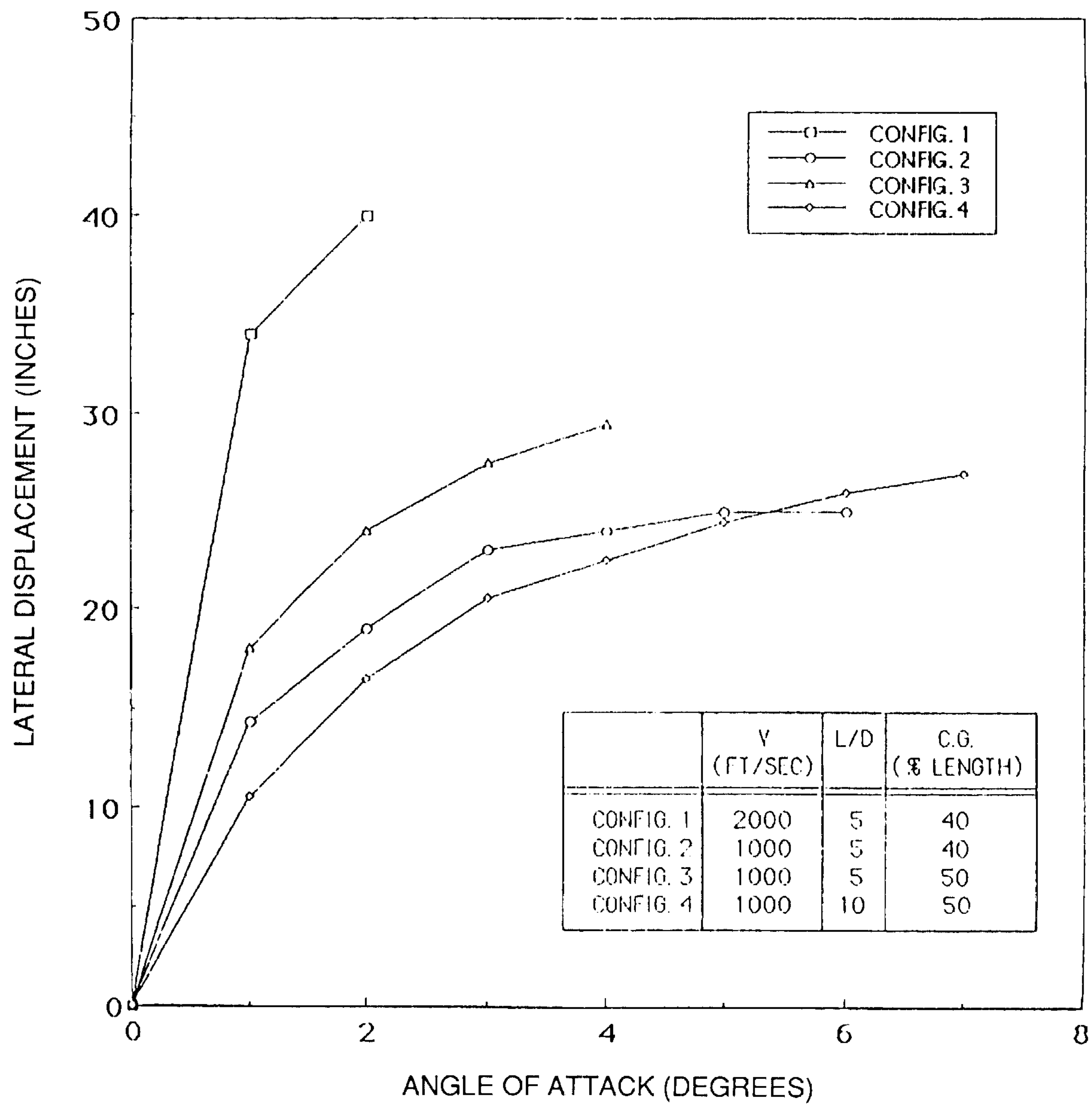


Baseline Design - Target Type 4 (Soil/Rock)
-2.0 Degrees Angle of Attack



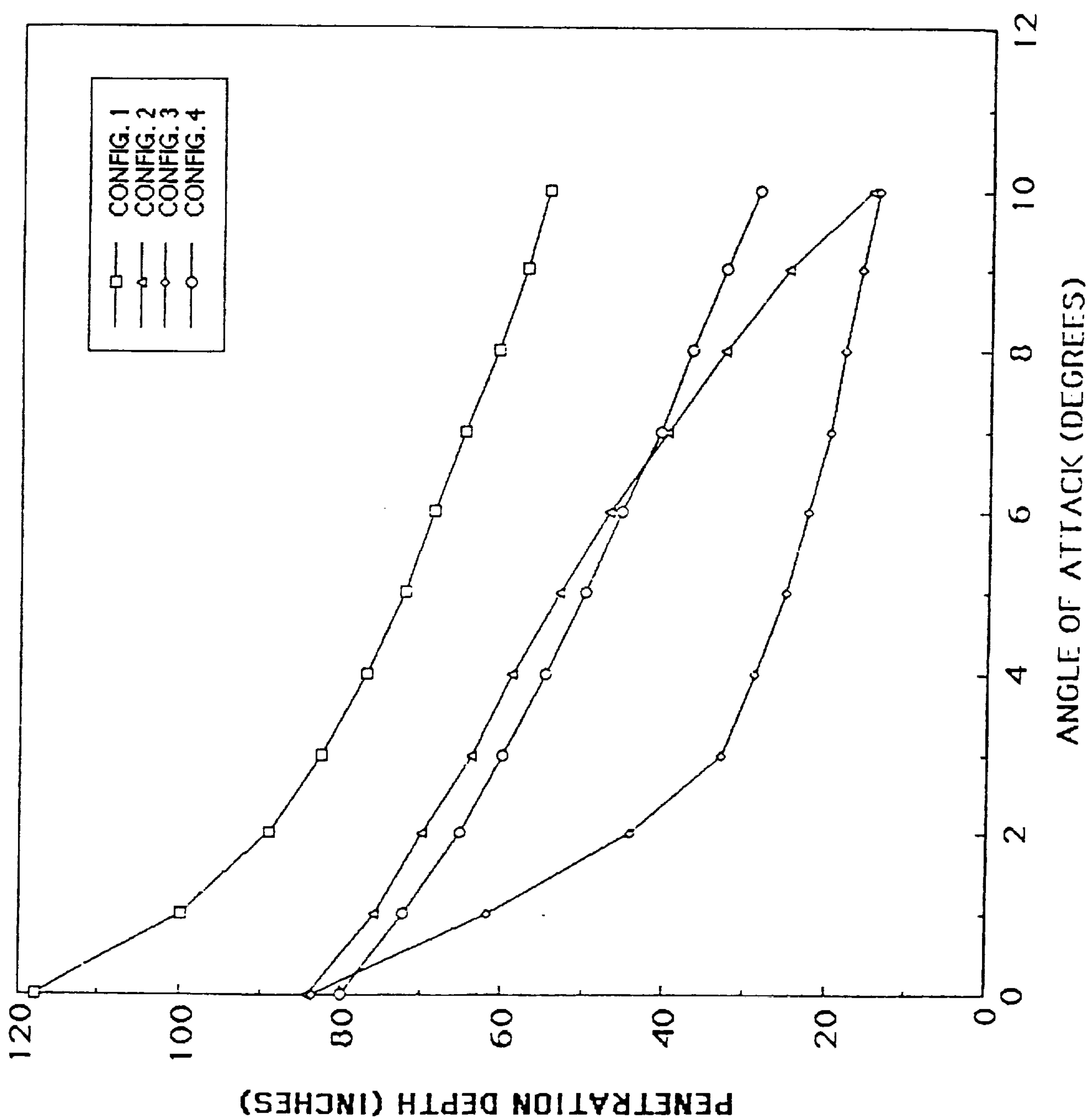
860 lb Design Excursion 1 - Target Type 4 (Soil/Rock)
-2.0 Degrees Angle of Attack

LATERAL DISPLACEMENT VS ANGLE OF ATTACK FOR 4 PENETRATOR CONFIGURATIONS
(MEDIA - SAND)



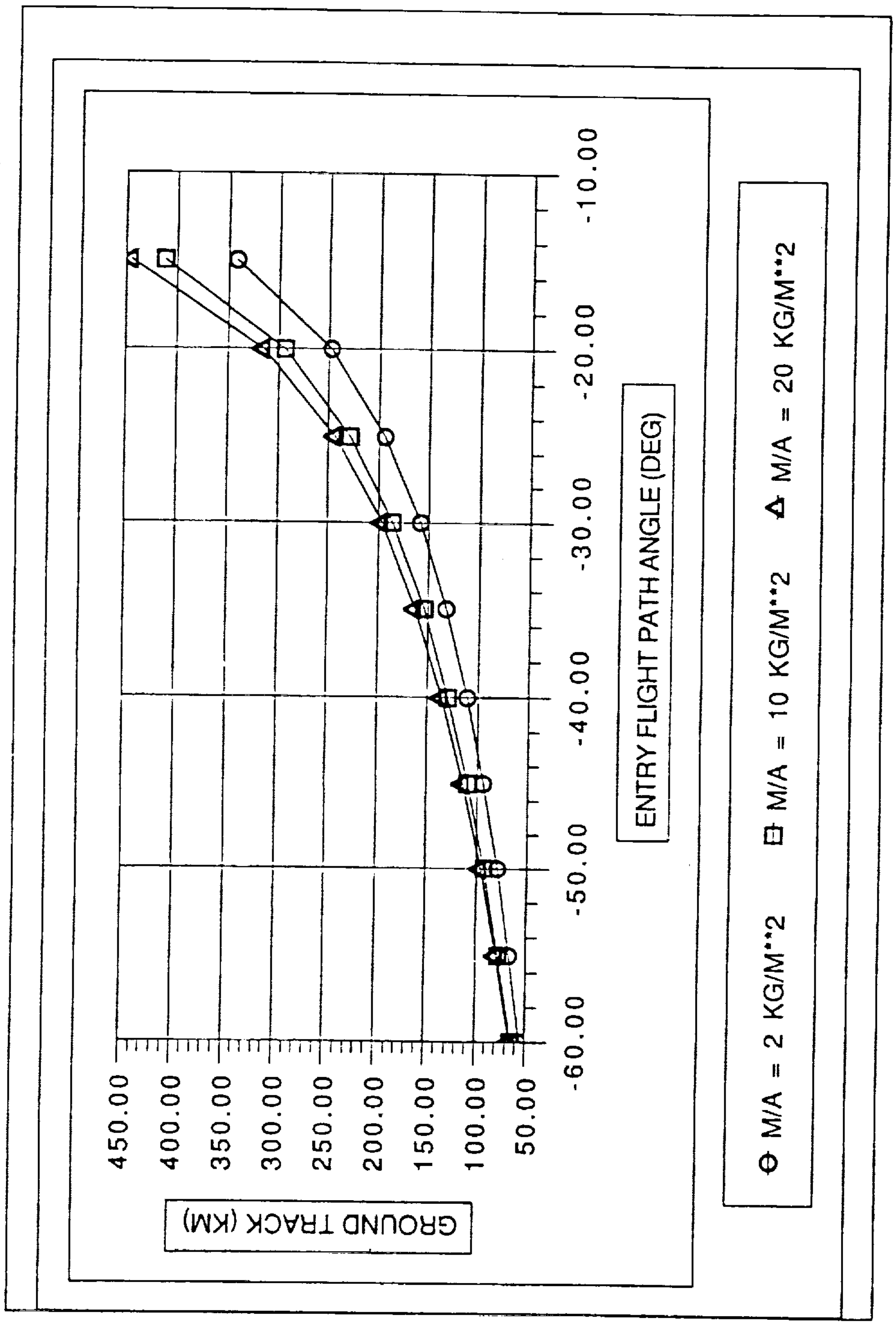
	V (FT/SEC)	L/D	C.G. (% LENGTH)
CONFIG. 1	2000	5	40
CONFIG. 2	1000	5	40
CONFIG. 3	1000	5	50
CONFIG. 4	1000	10	50

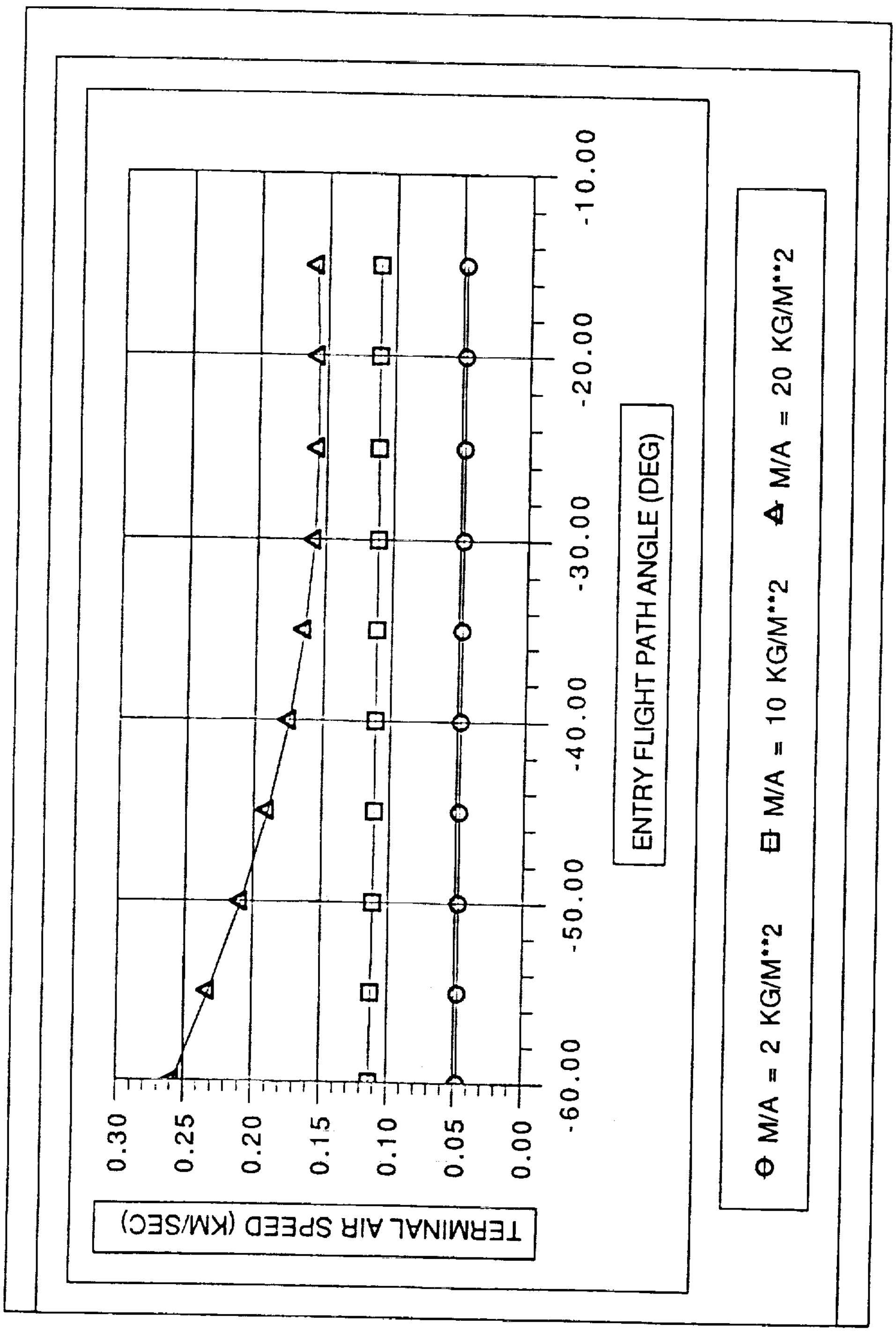
PENETRATION VS. ANGLE OF ATTACK FOR 4 PENETRATOR CONFIGURATIONS
(MEDIA - SAND)

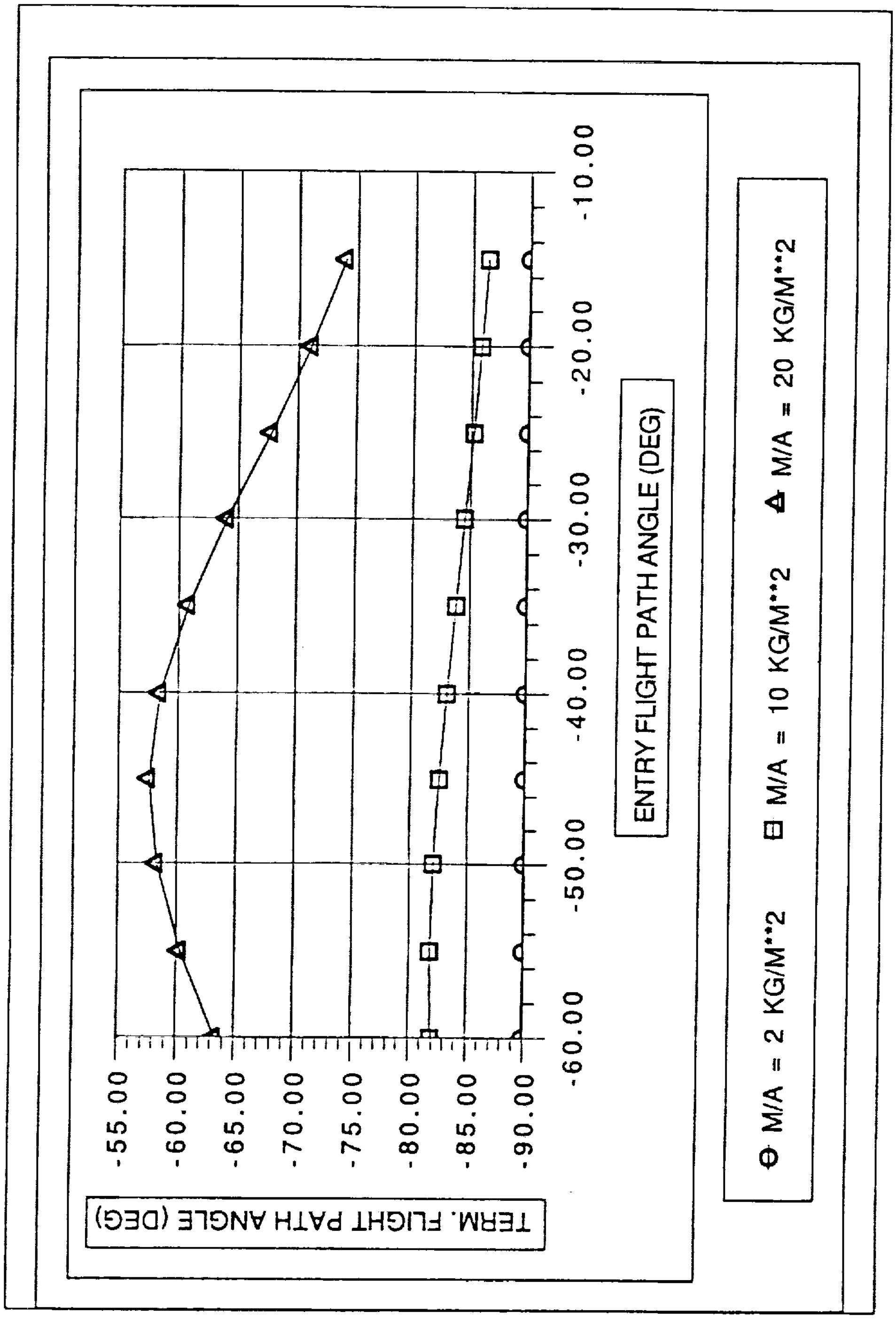


TRW

Mars Penetrator Entry Trades


GROUND TRACK (KM)
450.00
400.00
350.00
300.00
250.00
200.00
150.00
100.00
50.00
-60.00
-50.00
-40.00
-30.00
-20.00
-10.00
ENTRY FLIGHT PATH ANGLE (DEG)
M/A = 2 KG/M**2
M/A = 10 KG/M**2
M/A = 20 KG/M**2

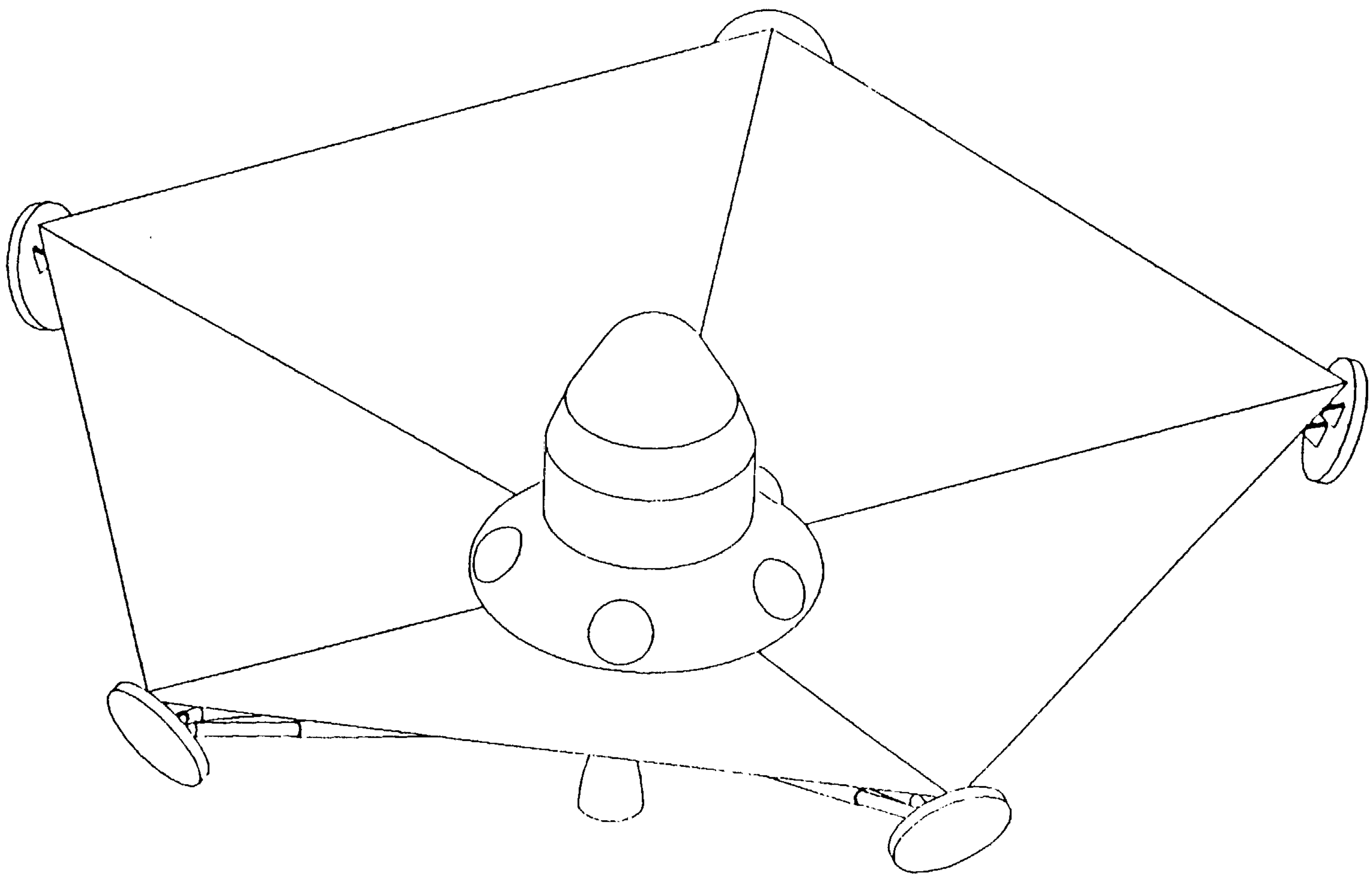

TERMINAL AIR SPEED (KM/SEC)
0.30
0.25
0.20
0.15
0.10
0.05
0.00
-60.00
-50.00
-40.00
-30.00
-20.00
-10.00
ENTRY FLIGHT PATH ANGLE (DEG)
M/A = 2 KG/M**2
M/A = 10 KG/M**2
M/A = 20 KG/M**2


TERM. FLIGHT PATH ANGLE (DEG)
-55.00
-60.00
-65.00
-70.00
-75.00
-80.00
-85.00
-90.00
-60.00
-50.00
-40.00
-30.00
-20.00
-10.00
ENTRY FLIGHT PATH ANGLE (DEG)
M/A = 2 KG/M**2
M/A = 10 KG/M**2
M/A = 20 KG/M**2

MARS
ROVER
SAMPLE
RETURN

AEROBRAKE DESIGN

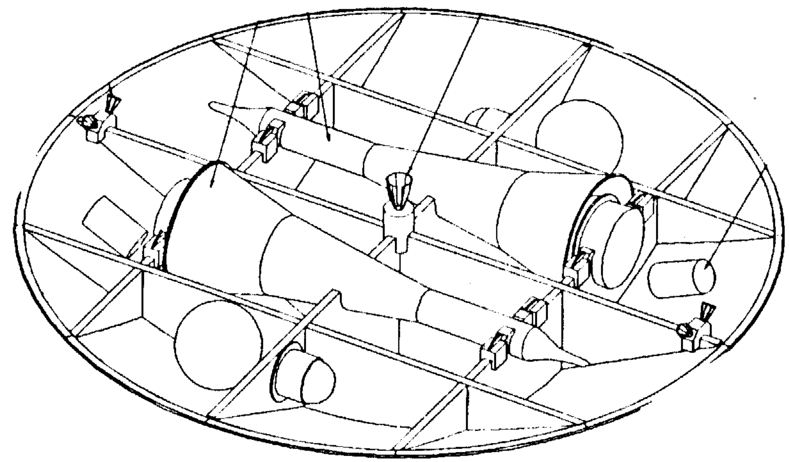
TRW



Session B, Submittal No. 7

Richard P. Reinert
Ball Space Systems Division

CRAF PENETRATOR-LANDER FOR MARS



PREPARED FOR MGNM WORKSHOP AT JPL

02/06/90

W. Boynton
University of Arizona

R. Reinert
Ball Space Systems Division

Ball Space Systems Division

BRIEFING OUTLINE

PRELIMINARY MISSION REQUIREMENTS

POLAR PENETRATOR DERIVED FROM CRAF CONFIGURATION

PV FUNCTIONAL SHOWS CENTRAL ROLE OF C&DH

SCIENCE OBJECTIVES SIMILAR TO CRAF

BASIC INSTRUMENT COMPLEMENT IS IDENTICAL EXCEPT FOR GRS

POLAR PENETRATOR MASS PROPERTIES MATCH BASELINE

POLAR PENETRATOR CONFIGURATION COMPATIBLE WITH BASELINE AEROSHELL

POLAR PENETRATORS OCCUPY ONE AEROSHELL

MPP AEROSHELL/PENETRATOR SYSTEM

PENETRATOR EXTRACTION FORM AEROSHELL

PENETRATOR DEPLOYMENT

PRELIMINARY MISSION REQUIREMENTS

REQUIREMENT	MPP	CRAF
SCIENCE PAYLOAD	MPP INSTRUMENT COMPLEMENT AS DESCRIBED BELOW	CRAF INSTRUMENT COMPLEMENT
MISSION DURATION: (APPROX)	CRUISE 330 D MARS ORBIT 30-60 D OPERATIONAL 7-14 D	6 YR 1 YR (NUCLEUS) 6-7 D
LAUNCH DATE	1998	1995
LAUNCH/TRANSPORT IV/CENTAUR/	TITAN IV/CENTAUR/	TITAN
	MODIFIED M/O BUS	MM/II BUS
DEPLOYMENT	**DEORBIT/MARS ATMOSPHERE ENTRY DUAL MANIFEST 120 DEG AEROSHELL 30 KG/M**2 M = 2 PARACHUTE EXTRACTION PARACHUTE PROVIDES IMPACT VELOCITY**	**FREE-FLIGHT TO NUCLEUS ONBOARD PRO-PULSION MODULE PROVIDES IMPACT VELOCITY**
IMPACT CONDITIONS		
S-RANGE	3 - 50	3-200
VELOCITY	50 M/S	20-50 M/S
MIN DEPTH	1 M	1 M
COMM RELAY		
BITS/DAY (AVG)	256 K	256 K
BER	1X10-5	1X10-5
RANGE (MAX)	600 KM	100 KM
PASSES/DAY	2 M	2 M
PASS DURATION	5 M MIN	10 SEC MIN

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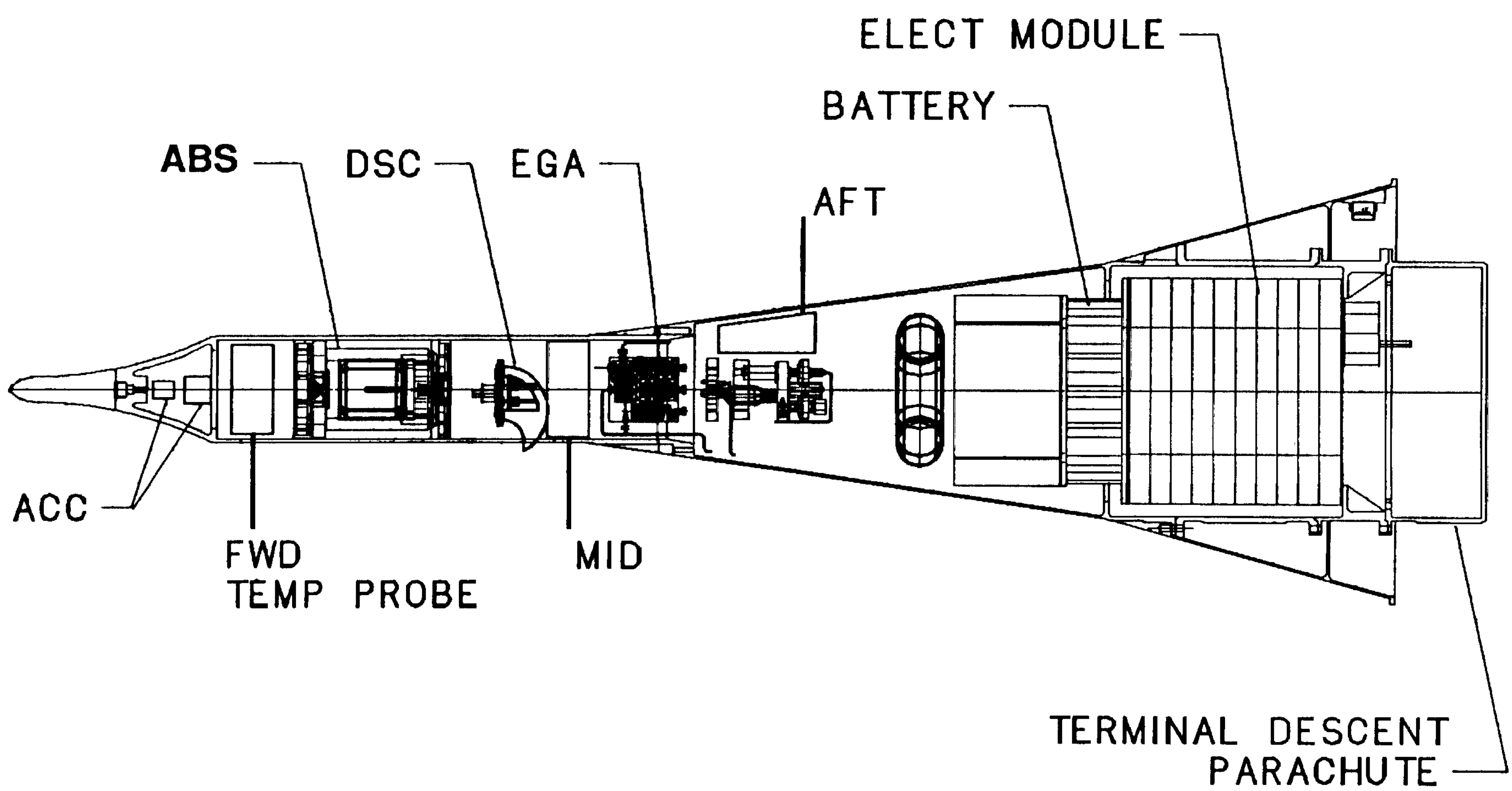
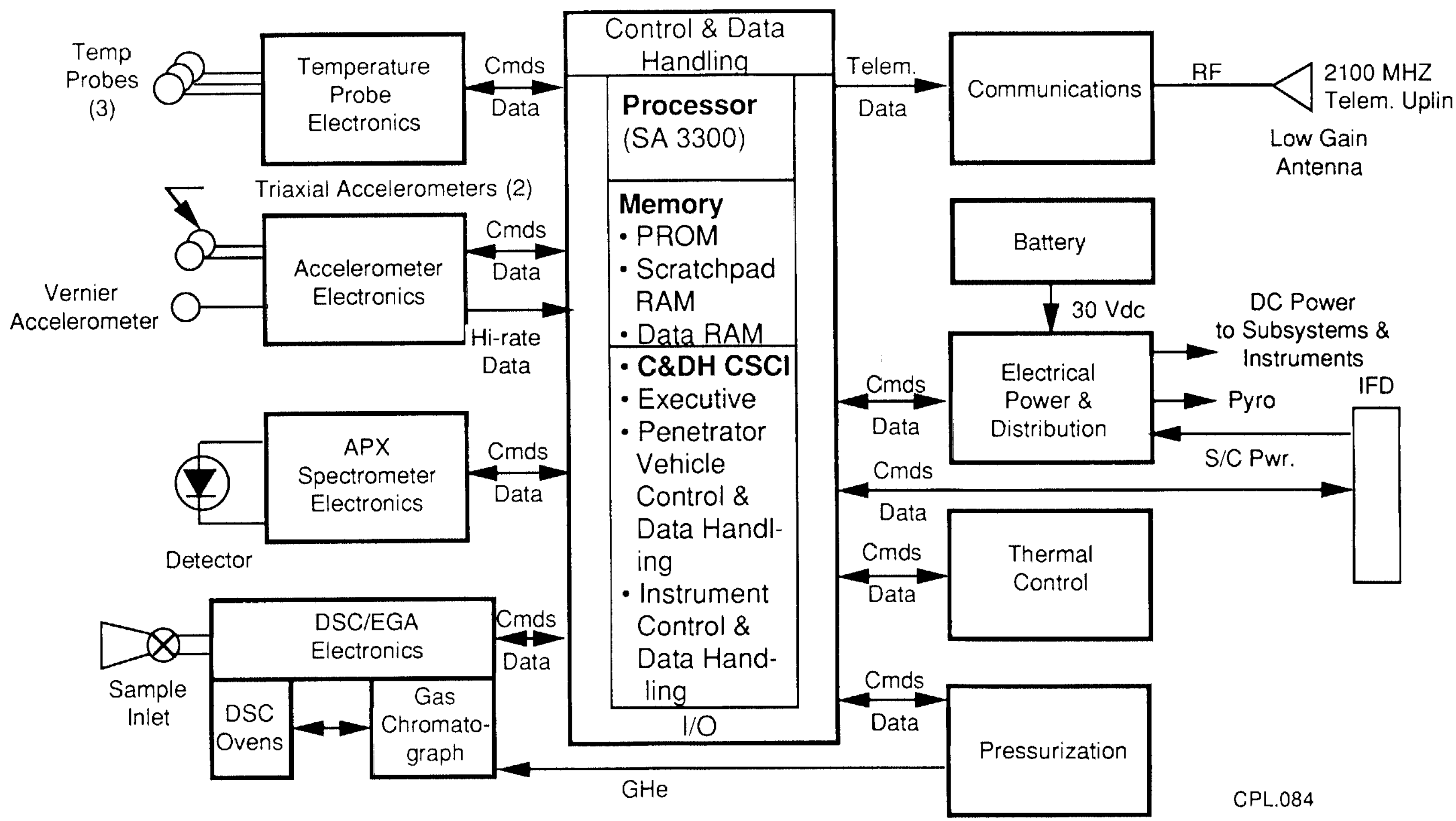
POLAR PENETRATORS DERIVED FROM CRAF CONFIGURATION



MPP FUNCTIONAL ARRANGEMENT IS SIMPLER



• Approx 90% commonality with CRAF PENL

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SCIENCE OBJECTIVES SIMILAR TO CRAF

- Measure strength, depth of penetration
- Detect stratigraphic layers

- Measure initial temperature profile
- Measure thermal diffusivity

- Determine elemental abundances of most major and minor and a few trace elements
- Determine the temperature and enthalpy of phase changes as a diagnostic of the mineralogical/molecular form of the major components
- Determine the identity and vapor pressure of major and trace volatile compounds as a function of temperature

BASIC INSTRUMENT COMPLEMENT IS IDENTICAL EXCEPT FOR GRS

ELEMENT	BASIC FUNCTION
SEVEN ACCELEROMETERS (ACCs)	MEASURE THE DECELERATION PROFILE ON IMPACT
THREE THERMAL PROBES (TPs)	MEASURE TEMPERATURES AND THERMAL DIFFUSIVITY AS A FUNCTION OF DEPTH
ALPHA BACKSCATTER SPECTROMETER (ABS)	MEASURE IN-SITU ELEMENTAL COMPOSITION
DIFFERENTIAL SCANNING CALORIMETER (DSC)	DETERMINE THE PHASE COMPOSITION OF THE ICES AND MINERALS PRESENT
EVOLVED GAS ANALYZER (EGA)	DETERMINE THE MOLECULAR COMPOSITION OF ENTRAPPED GASES EVOLVED FROM DSC HEATING

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POLAR PENETRATOR MASS PROPERTIES MATCH BASELINE

ITEM	MASS, KG (1)		RATIONALE
	CRAF	MPP	
MASS SPEC	3.7	0.5	REPLACE GRS WITH ABS
TEMP PROBE GROUP	1.5	1.5	IDENTICAL
ACCELEROMETER GROUP	0.5	0.5	"
DSC ANALYZER	1.9	1.9	"
EGA ANALYZER	5.0	5.0	"
INSTRUMENT ELECTRONICS	2.8	2.3	DELETE GRS HVPS
STRUCTURE	19.5	19.5	IDENTICAL
ELECTRICAL POWER	9.2	9.3	648 W-HR
C&DH	2.1	2.1	IDENTICAL
COMMUNICATIONS	0.6	0.6	16 KBPS/100 KM
THERMAL CONTROL	0.6	0.6	IDENTICAL
PROPULSION	6.9	-	NOT NEEDED
TOTAL	54.3	44.2	

NOTES:
(1) CONTINGENCY INCLUDED AT SUBSYSTEM LEVEL

(2) BASELINE MASS = 46.3 KG

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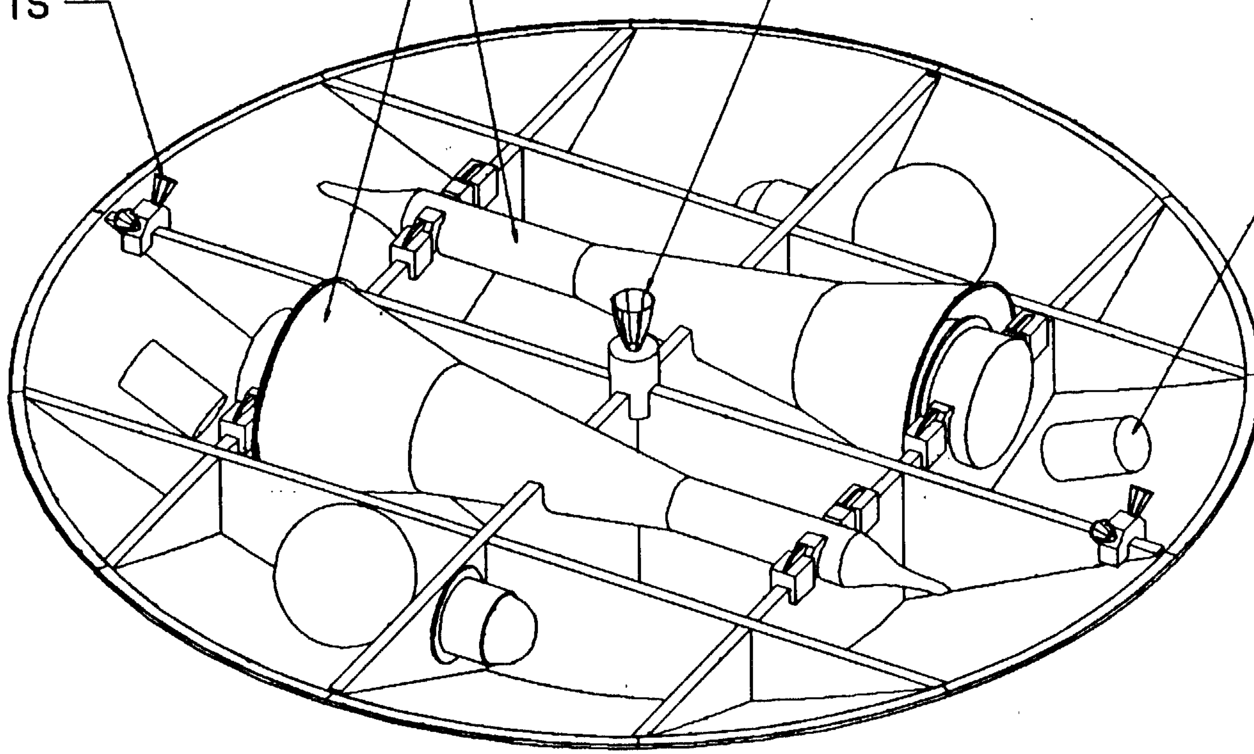
POLAR PENETRATOR CONFIGURATION COMPATIBLE WITH BASELINE AEROSHELL



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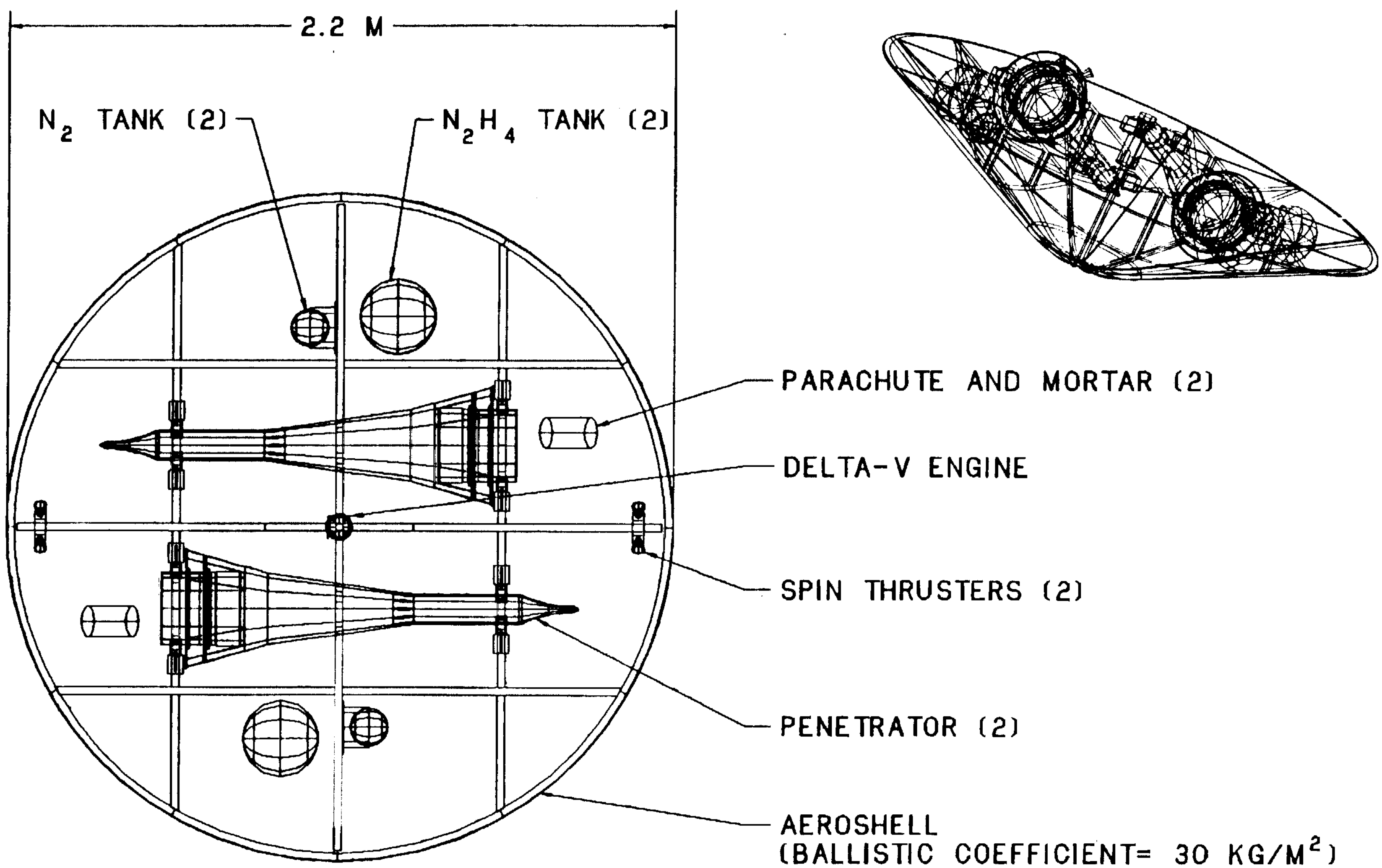
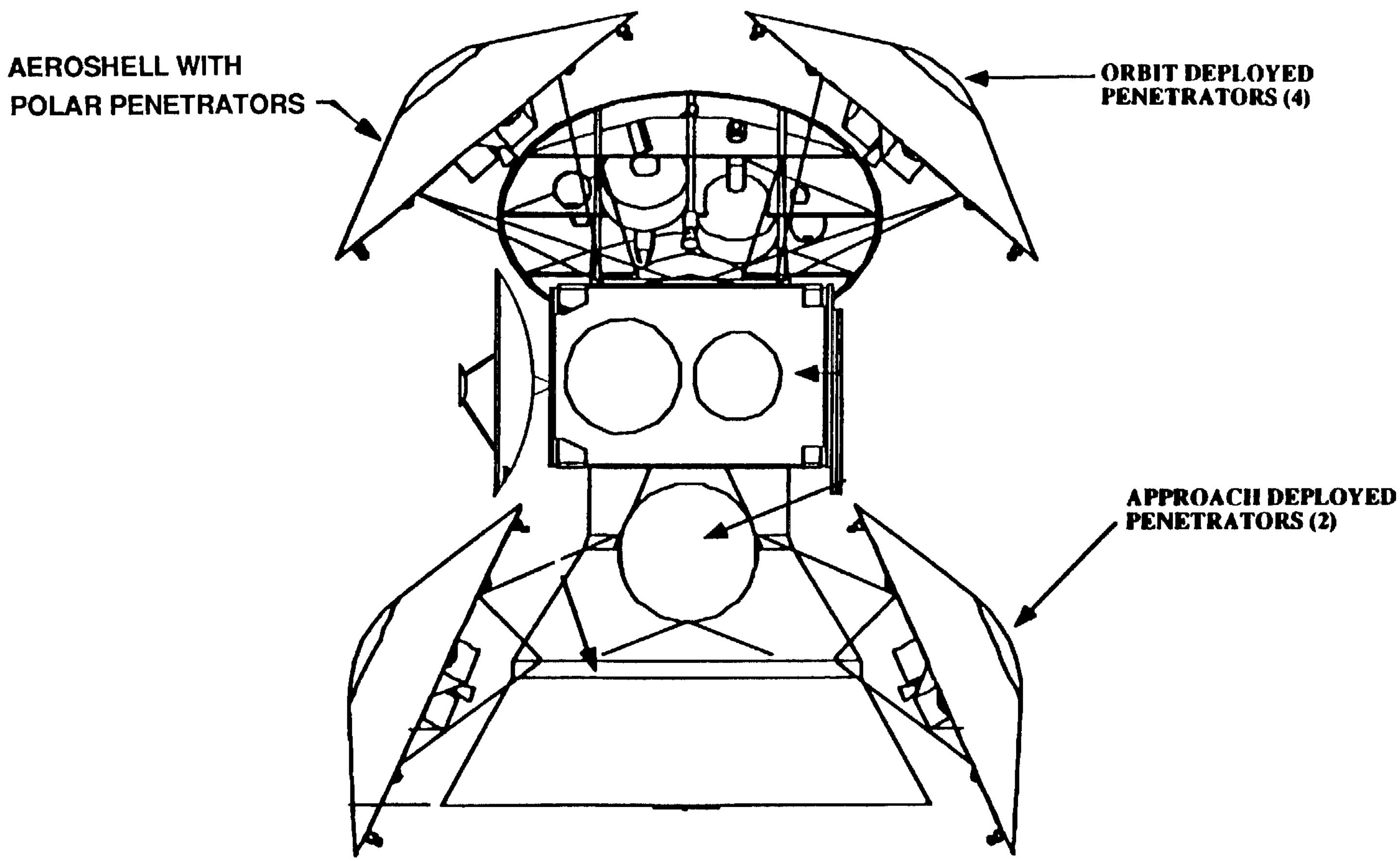
MPP AEROSHELL/PENETRATOR SYSTEM



POLAR PENETRATORS OCCUPY ONE AEROSHELL



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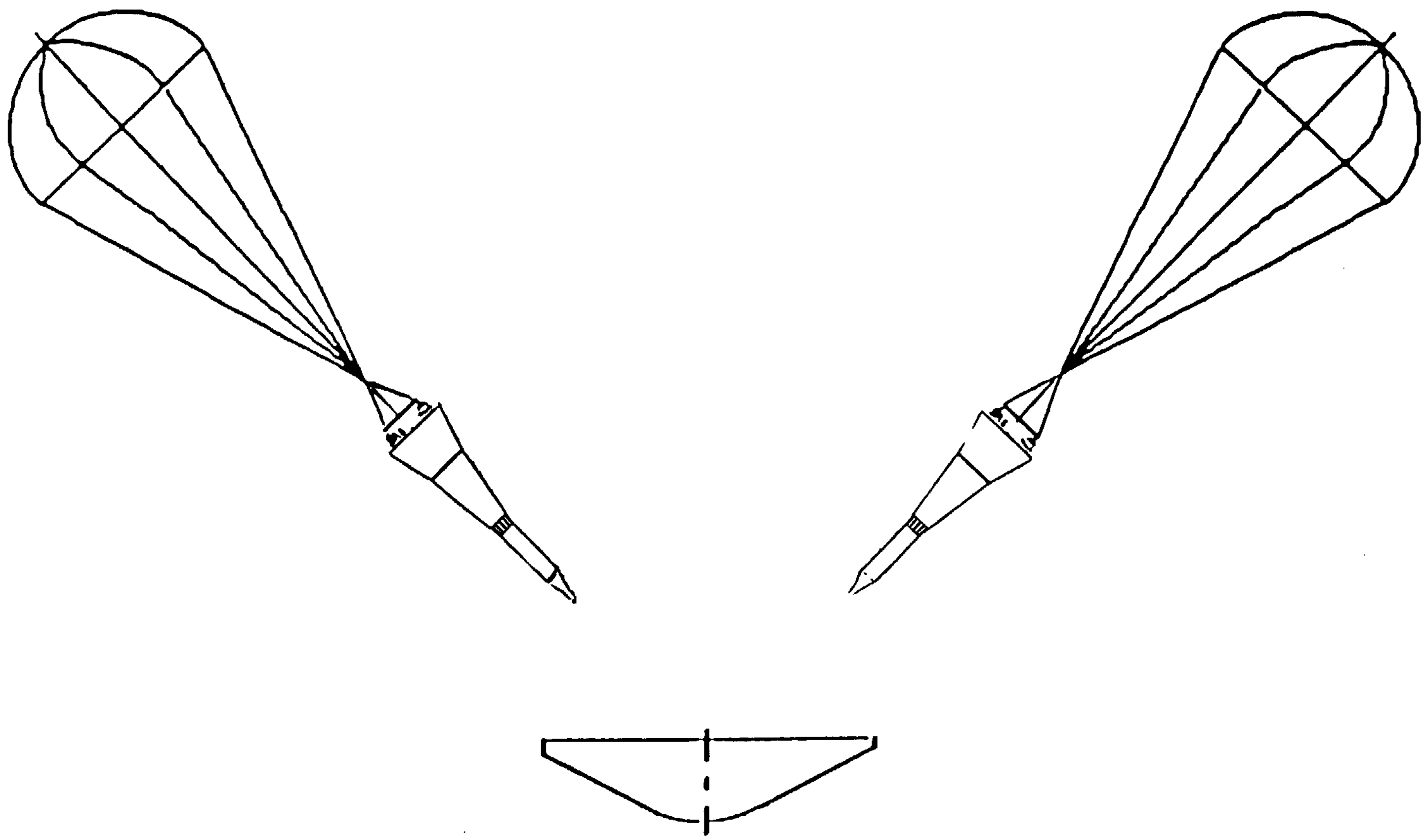
PENETRATOR EXTRACTION FROM AEROSHELL



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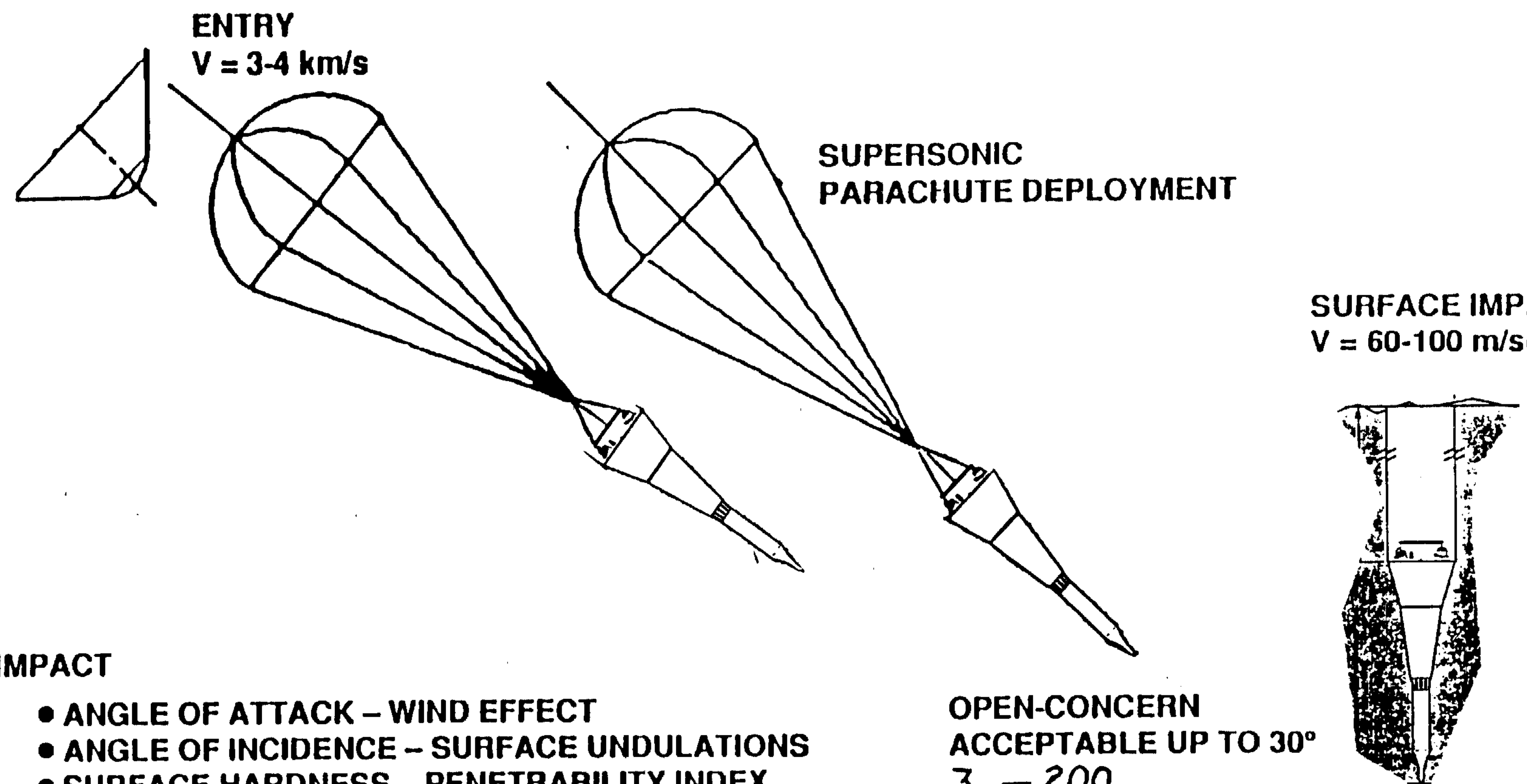
PENETRATOR DEPLOYMENT



IMPACT

- ANGLE OF ATTACK – WIND EFFECT
- ANGLE OF INCIDENCE – SURFACE UNDULATIONS
- SURFACE HARDNESS – PENETRABILITY INDEX

OPEN-CONCERN
ACCEPTABLE UP TO 30°
3 – 200

COMMUNICATIONS

- ACCESSIBILITY TIME
- BIT RATE, LINK BUDGET

ASSUME 2 TO 4 MINUTES PER PASS
12.5 Kbit/sec

Session B, Submittal No. 8

Joe D. Gamble
Johnson Space Center

MARS GLOBAL NETWORK MISSION WORKSHOP

ENTRY SYSTEM DESIGN CONSIDERATIONS

J. Gamble - NASA/Johnson Space Center

Introduction

This section addresses some of the design issues concerned with the specific workshop question, "What is the best entry system - fixed or deployed aeroshells; parachutes or direct impact?" To address these questions some information about the entry conditions in the Mars environment is required. Results from the 90 day human exploration initiative study were used as a reference point. The MRSR pre-phase A study results were also considered. Finally some parametric data was generated to specifically address the GNM entry design question.

Reference Mission

The 90 day study considered two flight systems each consisting of an orbiter/carrier vehicle with six aeroshells as shown in Figure 1. Each aeroshell contains two penetrator landers as shown in Figure 2 that use parachutes to extract them from the aeroshell just prior to landing. The rigid aeroshells are deployed from the carrier vehicle and spin stabilized at 60 rpm. Small propulsion systems provide the delta V required for the desired atmospheric entry conditions. The aeroshells do not have an active guidance and control system.

The aeroshell design incorporates a rigid conical aeroshell with a spherical nose cap. The aeroshell diameter is 2.2 m and has an entry mass of approximately 110 kg, yielding a ballistic coefficient of 30 kg/m2. The aeroshell uses an ablative heat shield.

Two of the six aeroshells are deployed 2-10 days prior to Mars arrival in order to land at polar sites. The other four aeroshells are deployed after capture into a 1/5 Sol Mars orbit.

Mars Approach Deployed Aeroshells

One of the primary concerns in the MRSR study was the ability to achieve the proper entry conditions during the Mars approach. The entry corridor is bounded by the skip out and maximum allowable g load boundaries as shown in figure 3. Figure 4 shows the entry corridor limits versus L/D for an entry velocity of 6 km/sec. The total corridor width is summarized in figure 5 and shows that the corridor is nearly independent of the ballistic coefficient. The ballistic coefficient determines whether the vehicle flies higher or lower in the atmosphere during the early portion of the entry. While the MRSR was concerned with aerocapture during the approach phase, the results are also applicable to the entry case. The estimated corridor width for ballistic vehicles are shown on figure 5. For a maximum entry load of 5 g's, the total corridor width is less than one degree. The corridor width increases to 3 and 5.5 for 10 and 15 g limits respectively. The MRSR study concluded that a minimum corridor width of approximately 3 degrees was required in order to accommodate navigation and atmosphere uncertainties. In order to achieve this accuracy, optical navigation was baselined for the study and trajectory corrections were considered within a few hours of entry.

GNM aeroshells deployed several days prior to entry and not having an active guidance and control system will almost certainly require much larger entry corridors than are necessary for the MRSR. It is very possible that a minimum corridor width of at least 5-10 degrees will be required. Figure 6 shows some preliminary results for the aeroshell defined by the 90 day study at an entry velocity of 6 km/sec at 125 km altitude. The figure indicates that g loads in excess of 20 g's will be required to provide a corridor width of 10 degrees. Figure 7 shows that for a 10 degree corridor width, downrange dispersions of +/- 2-5 degrees will occur for nominal entry angles of 15-20 degrees. These results were obtained from three degree-of-freedom simulations entering in a polar plane.

One proposal for decreasing the landing footprint dispersions is to enter at a much steeper entry angle. The results of entering at -35 and -45 degrees are shown as a function of ballistic coefficients in figure 8. The downrange dispersion for 10 degrees change in entry angle is less than one degree although it does increase as the ballistic coefficient increases. One of the primary problems with the steep entry angle is the large load factors that result. Figure 9 shows the maximum g loads (Earth g's) resulting from entry at -35 and -45 degrees. Load factors on the order of 40 - 60 g's result from these steep angles.

<u>Deployable Aeroshell Considerations</u>

Use of deployable aeroshell configurations will in general preclude the use of ablator heat shields and the ballistic coefficient will have to be small enough to limit the aeroheating during entry. To achieve a ballistic coefficient of 10 kg/m2 using the 90 day study mass of 110 kg would require an aeroshell diameter of approximately 3.8 m while a diameter of 8.5 m would be required to achieve a ballistic coefficient of 2 kg/m2. It would appear that use of deployed aeroshells of this size would have significant problems operating at 40-60 g's during entry. For this reason it is questionable whether use of deployable aeroshells for entry during Mars approach is a viable concept.

Mars Orbit Deployed Aeroshells

The lower entry velocity for aeroshells deployed from Mars orbit present much less of a problem than for those deployed during approach. Figure 10 shows that entry corridors of 15 degrees are possible at less than 10 g maximum load. Because the navigation is much better defined for the orbit deployed aeroshell than for the approach deployed case, the entry angle dispersions will be much less. Figure 11 indicates that for entry angle dispersions of +/- 1 degree, the dispersion in the downrange landing site will be well within +/- 1 degree. Aerodynamic heating for the orbit entry cases will also be much lower than for the approach deployed aeroshells. It would appear that these advantages definitely outweigh the delta V penalty associated with capturing the aeroshells into Mars orbit.

Parachute Considerations

One of the primary concerns with use of parachutes for the final surface delivery of the instrument packages is whether acceptable deployment conditions can be achieved during the aeroshell entry. The Viking program used supersonic deployed parachutes which were required because of the uncertainty in the Mars atmosphere. In general deployment of parachutes up to around Mach 2 (approximately 500 m/sec at Mars) is considered well within the state of the art. Figure 12 shows the aeroshell velocity at 5 and 10 km altitude as a function of entry angle for the 30 kg/m2 configuration with an entry velocity of 3.6 km/sec at 125 km altitude. The aeroshell is seen to be subsonic at both altitudes for the range of entry angles shown. Figure 13 shows the variation of the aeroshell velocity at 10 km altitude for various dispersions in the atmosphere. The low density cool COSPAR atmosphere results in barely supersonic conditions for the 30 kg/m2 configuration and even a severe 50% decrease in atmospheric density only produces a Mach 2 case. Therefore use of parachutes for landing of the payload should not present any significant deployment problems.

The bibliography lists several references with some applications to the Mars entry problem. A number of these also have extensive bibliographies.

MSNM STOWED CONFIGURATION
(ONE AEROSHELL REMOVED FOR CLARITY)

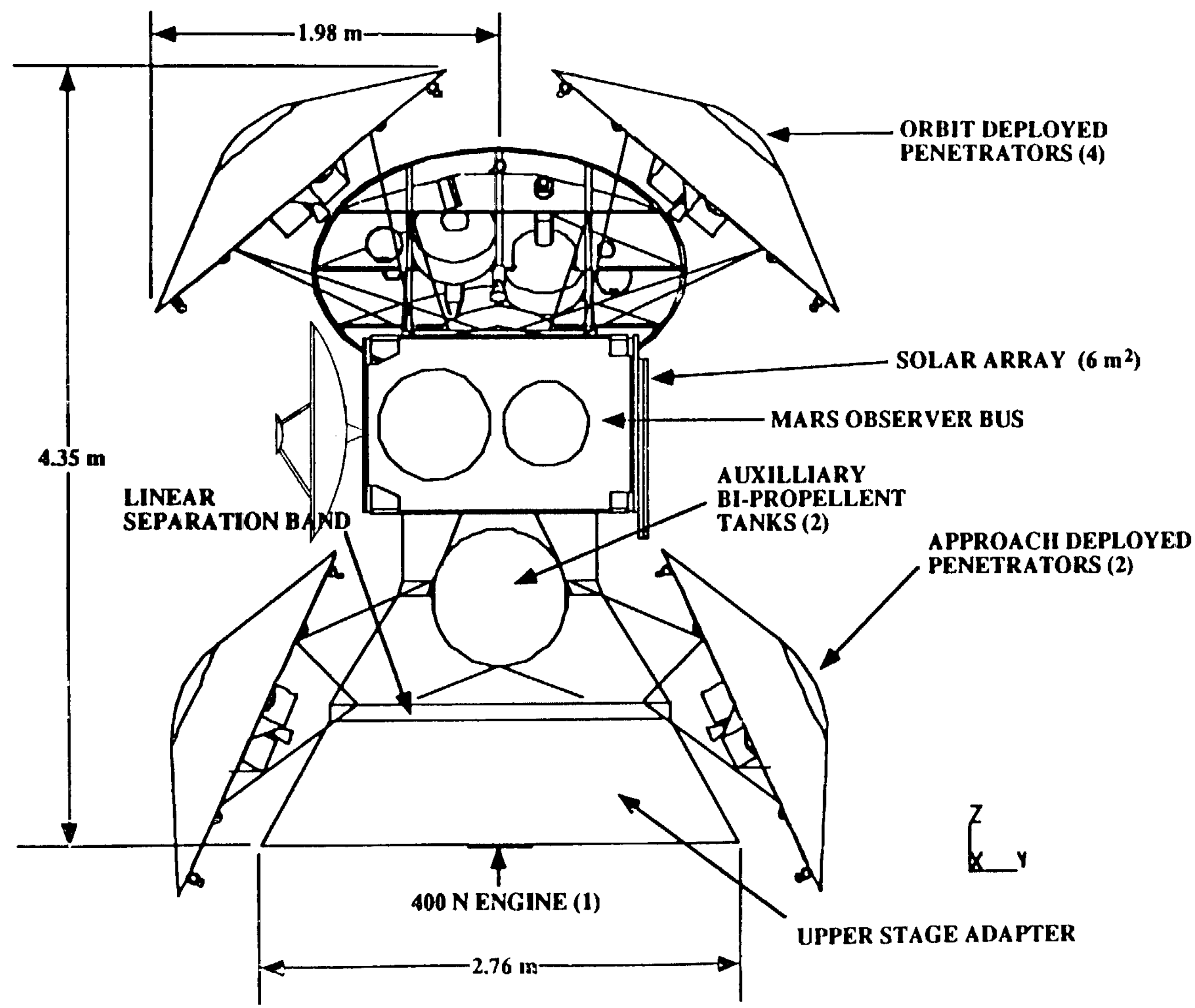
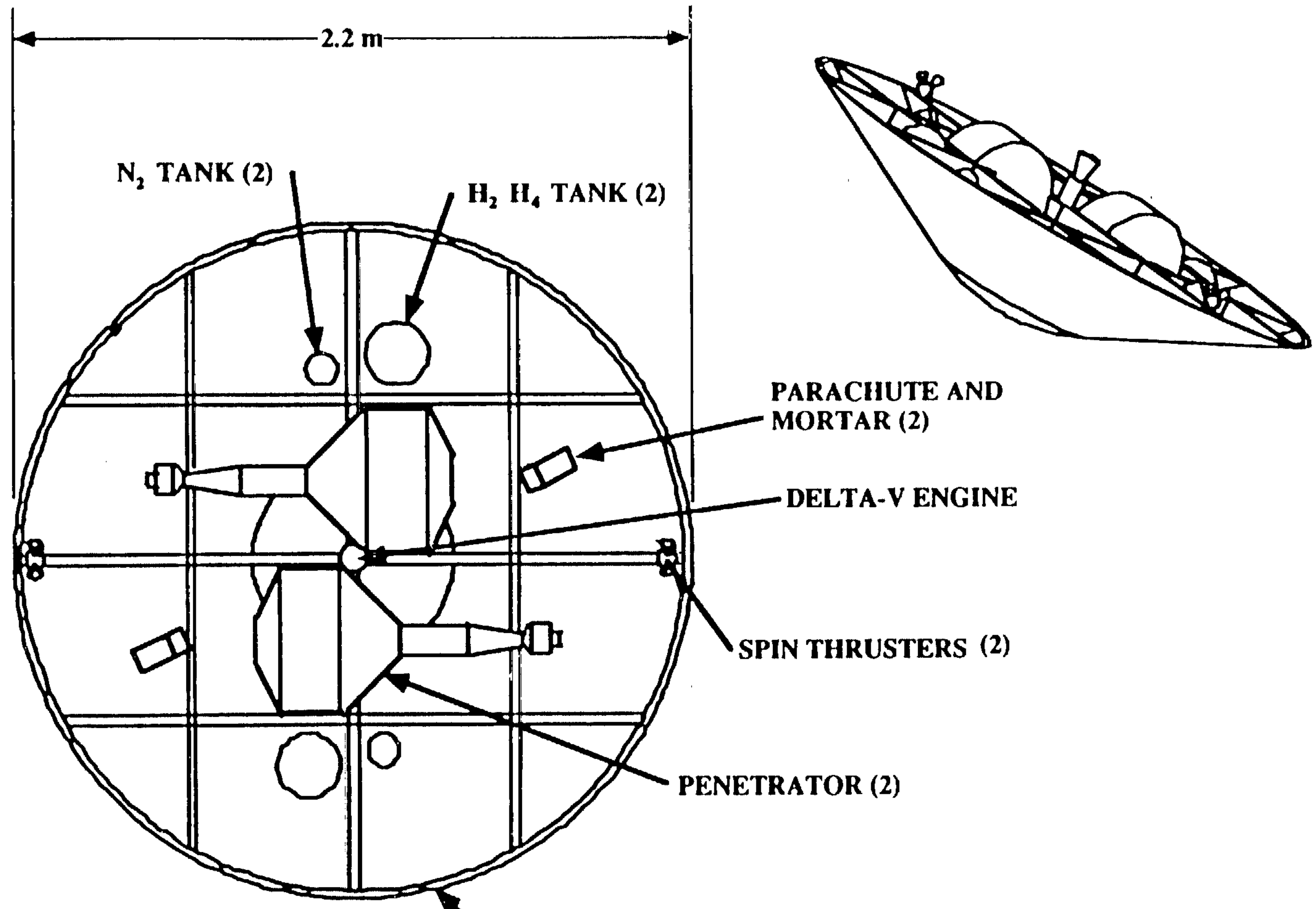


Figure 1.

MSNM AEROSHELL/PENETRATOR SYSTEM



$I_{XX} = I_{ZZ} = 20$ kg-m^2

$I_{YY} = 36$ kg-m^2

Figure 2.

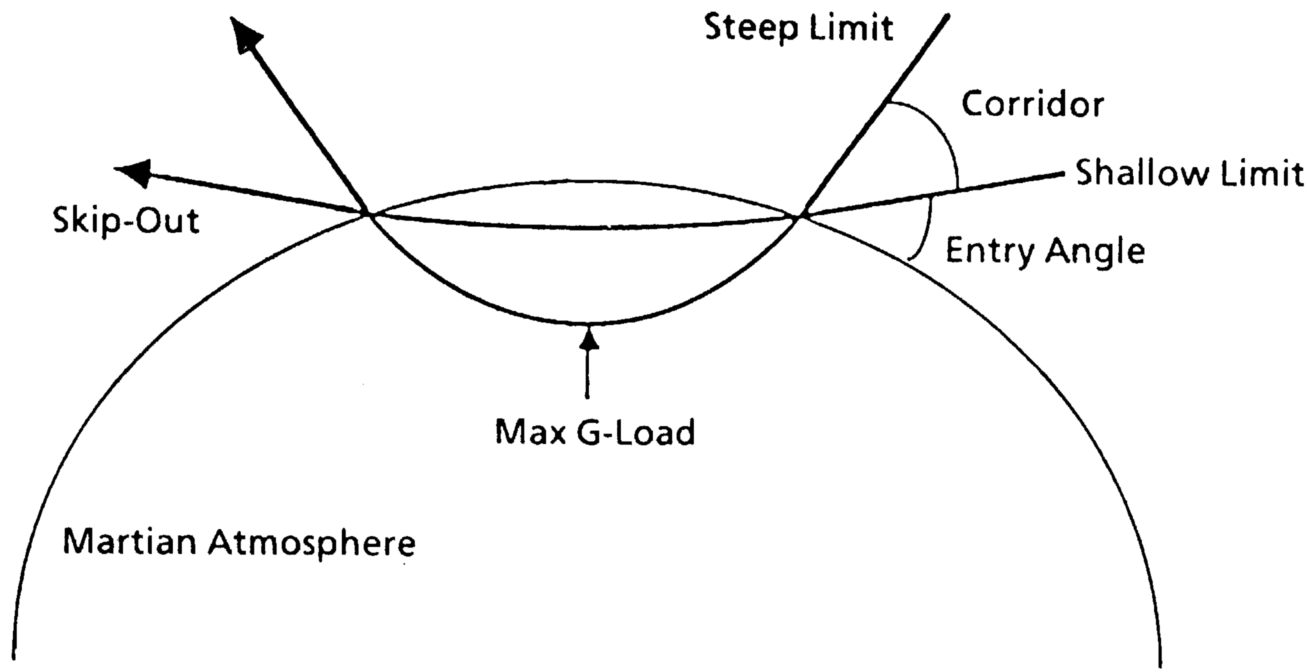

Steep Limit
Corridor
Shallow Limit
Entry Angle
Skip-Out
Max G-Load
Martian Atmosphere

Figure 3.

ENTRY CORRIDOR LIMITS

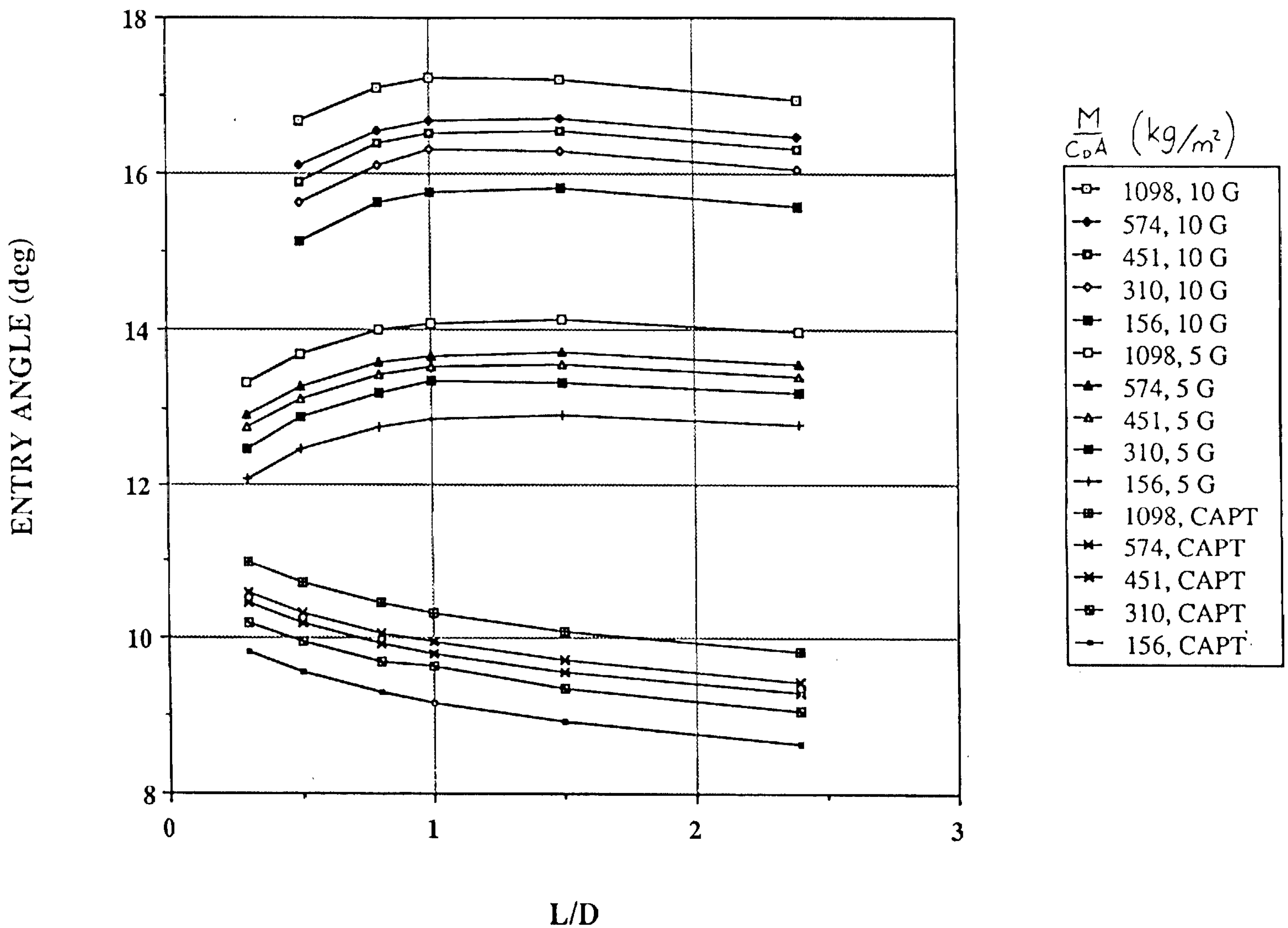


Figure 4.

EFFECTS ON CORRIDOR WIDTH

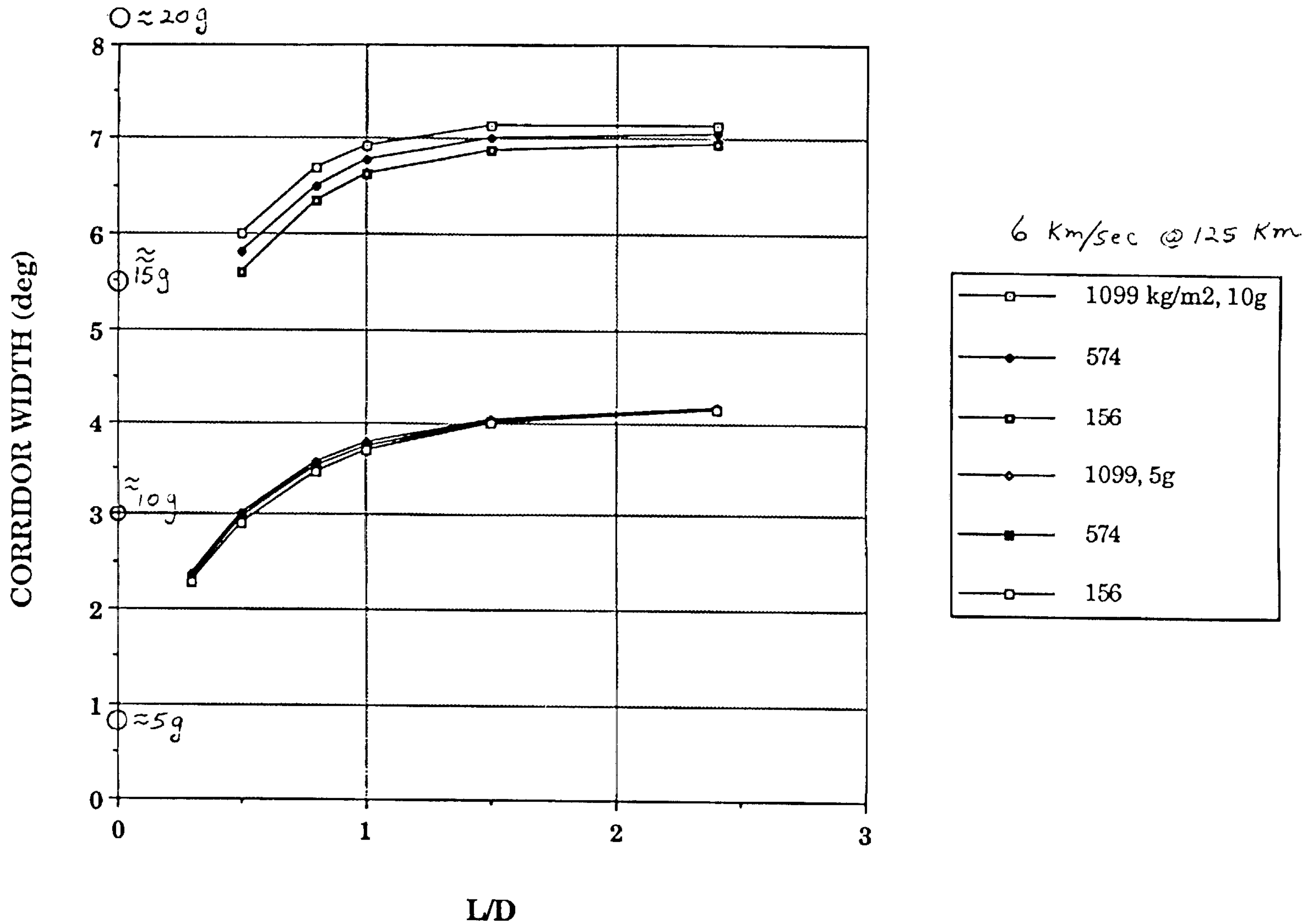


Figure 5.

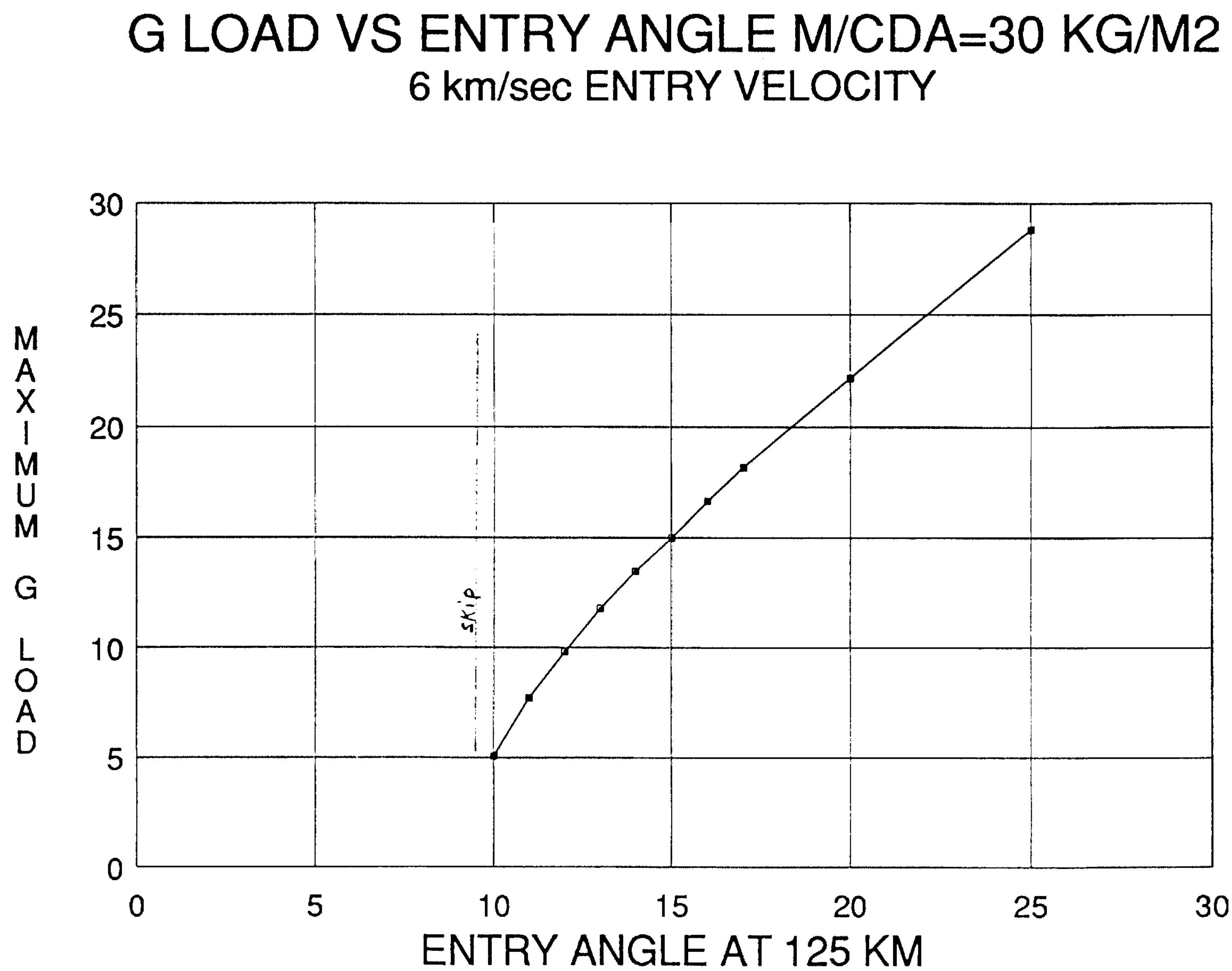

G LOAD VS ENTRY ANGLE M/CDA=30 KG/M2
6 km/sec ENTRY VELOCITY
MAXIMUM G LOAD
0
5
10
15
20
25
30
skip
0
5
10
15
20
25
30
ENTRY ANGLE AT 125 KM

Figure 6.

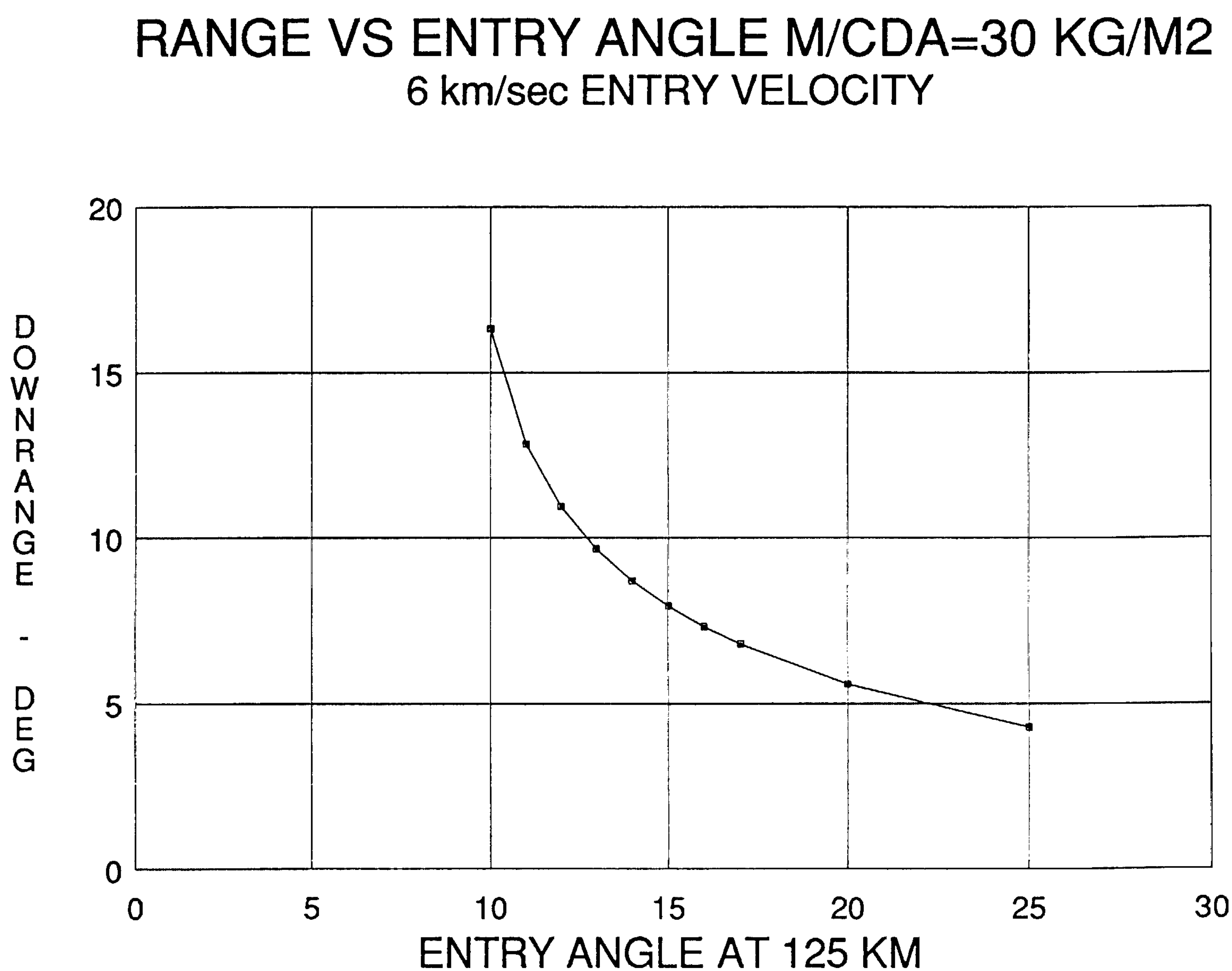

RANGE VS ENTRY ANGLE M/CDA=30 KG/M2
6 km/sec ENTRY VELOCITY
DOWNRANGE - DEG
ENTRY ANGLE AT 125 KM
0
5
10
15
20
25
30

Figure 7.

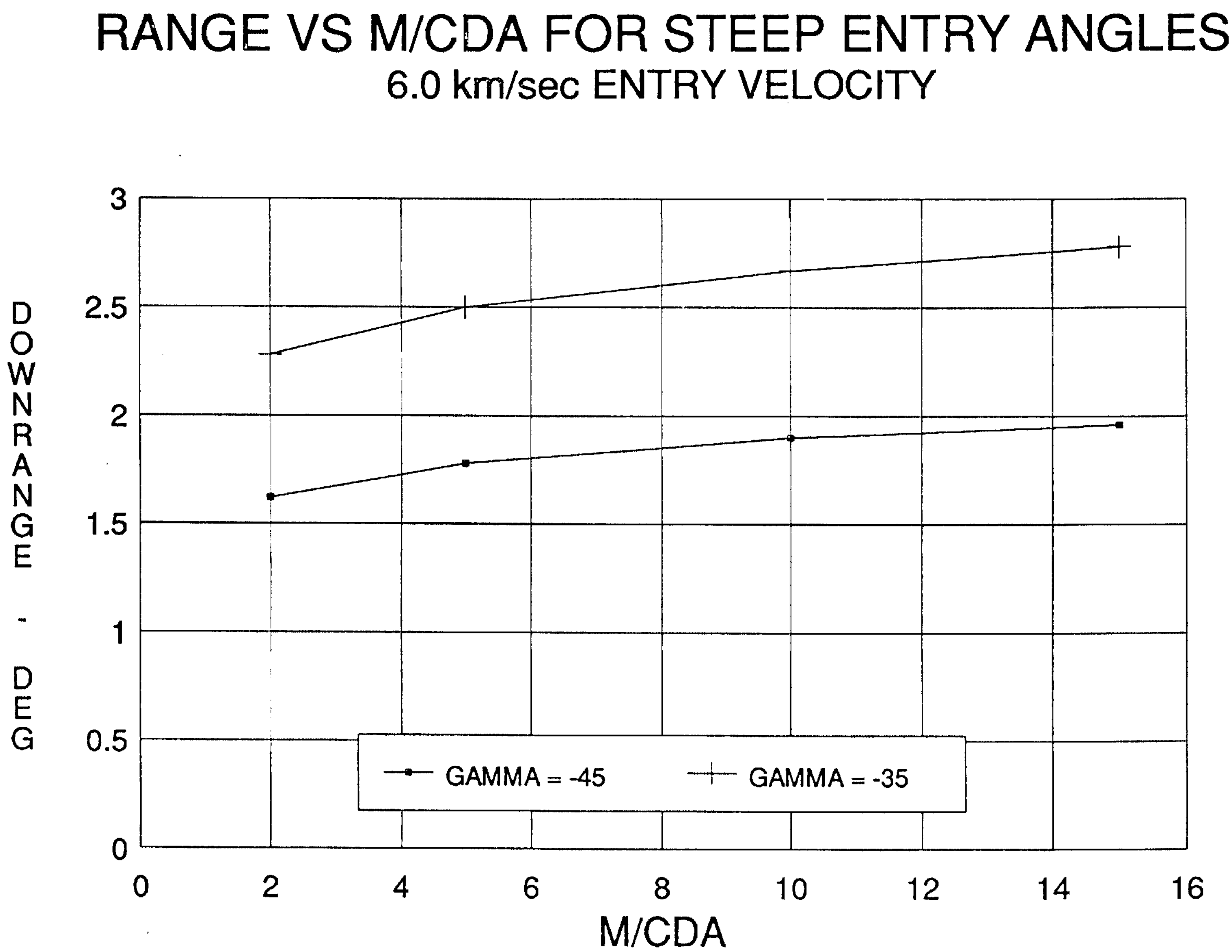

RANGE VS M/CDA FOR STEEP ENTRY ANGLES
6.0 km/sec ENTRY VELOCITY
DOWNRANGE - DEG
3
2.5
2
1.5
1
0.5
0
0
2
4
6
8
10
12
14
16
M/CDA
GAMMA = -45
GAMMA = -35

Figure 8.

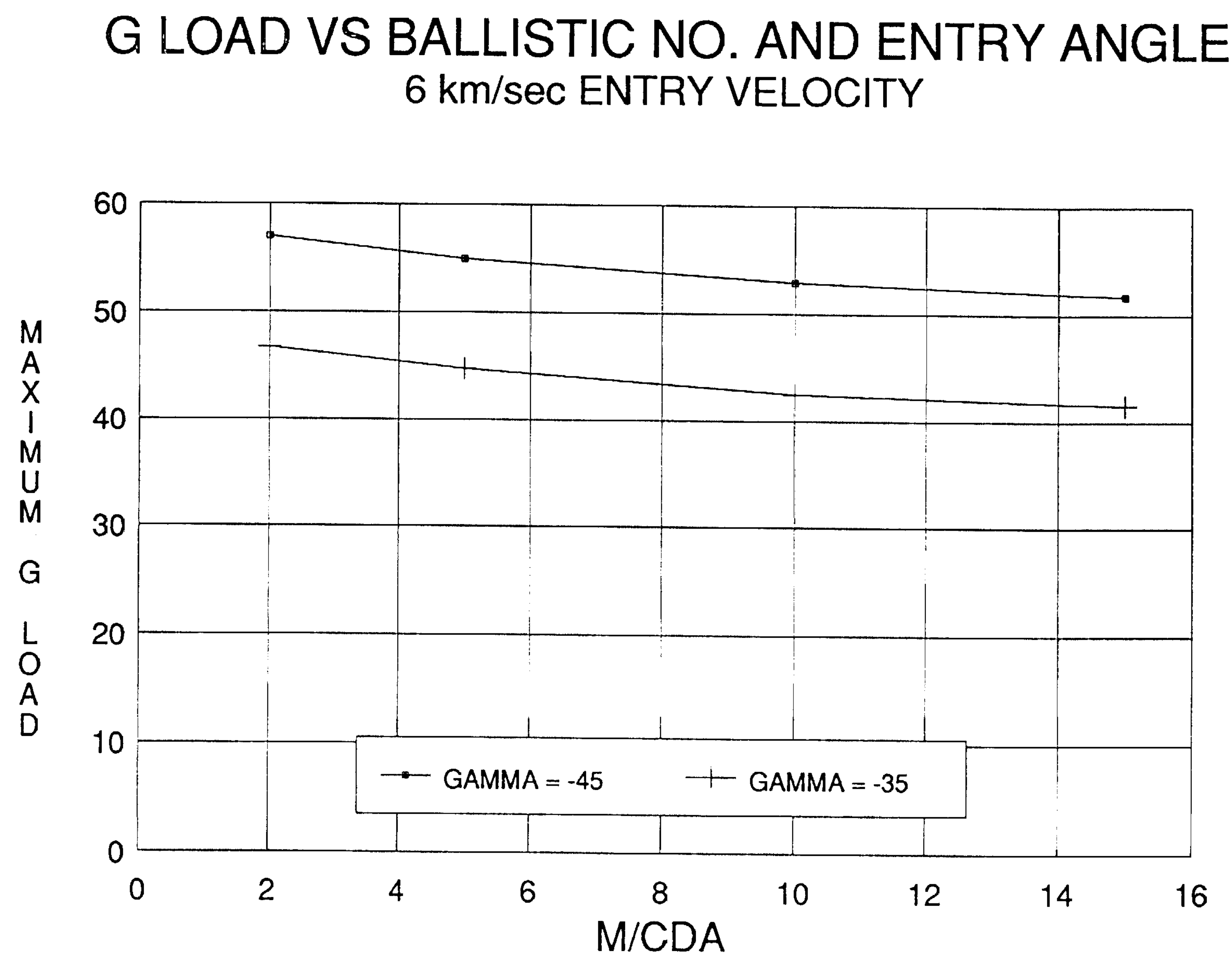

G LOAD VS BALLISTIC NO. AND ENTRY ANGLE
6 km/sec ENTRY VELOCITY
MAXIMUM G LOAD
60
50
40
30
20
10
0
0
2
4
6
8
10
12
14
16
M/CDA
GAMMA = -45
GAMMA = -35

Figure 9.

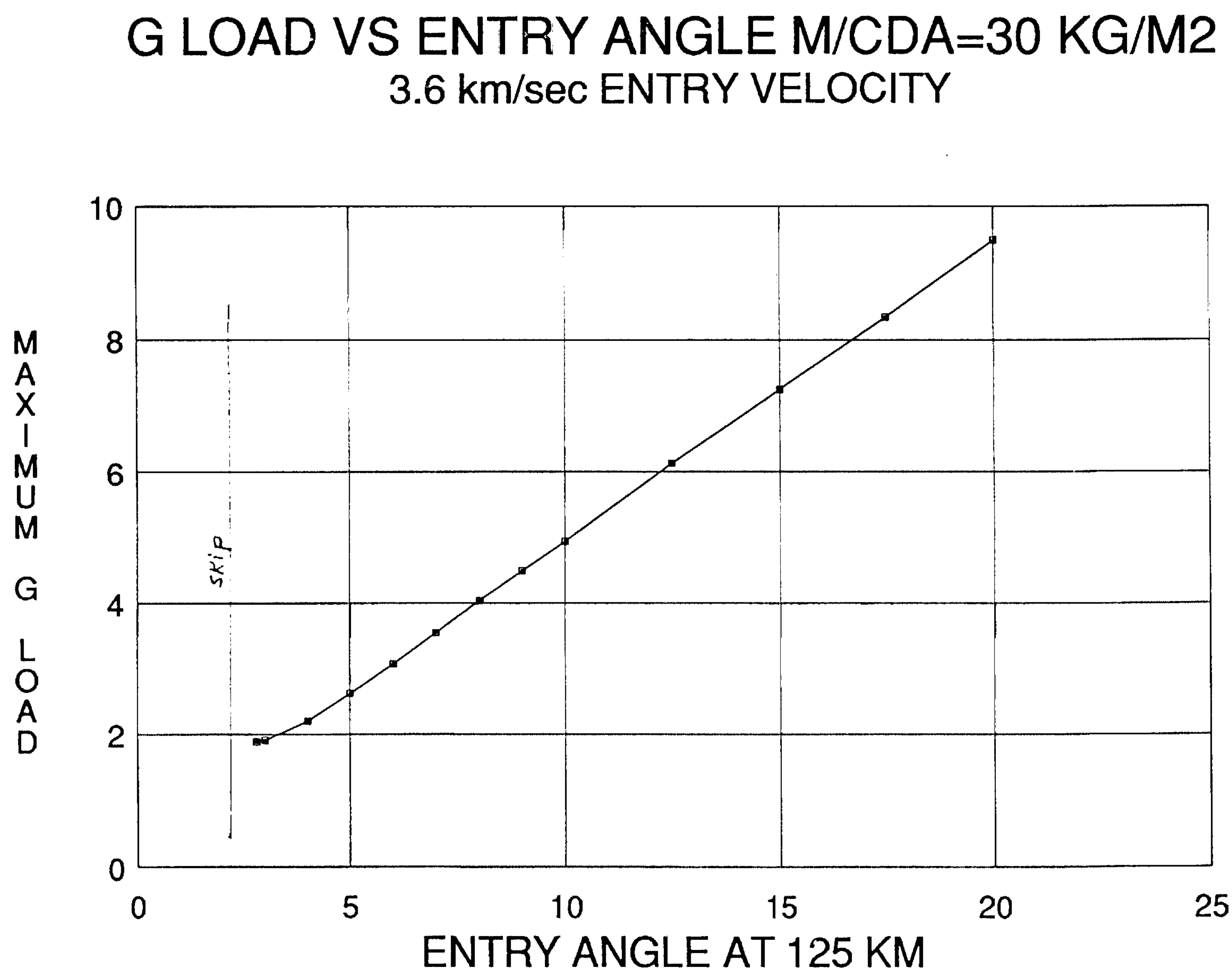

G LOAD VS ENTRY ANGLE M/CDA=30 KG/M2
3.6 km/sec ENTRY VELOCITY
MAXIMUM G LOAD
ENTRY ANGLE AT 125 KM
skip
0
2
4
6
8
10
0
5
10
15
20
25

Figure 10.

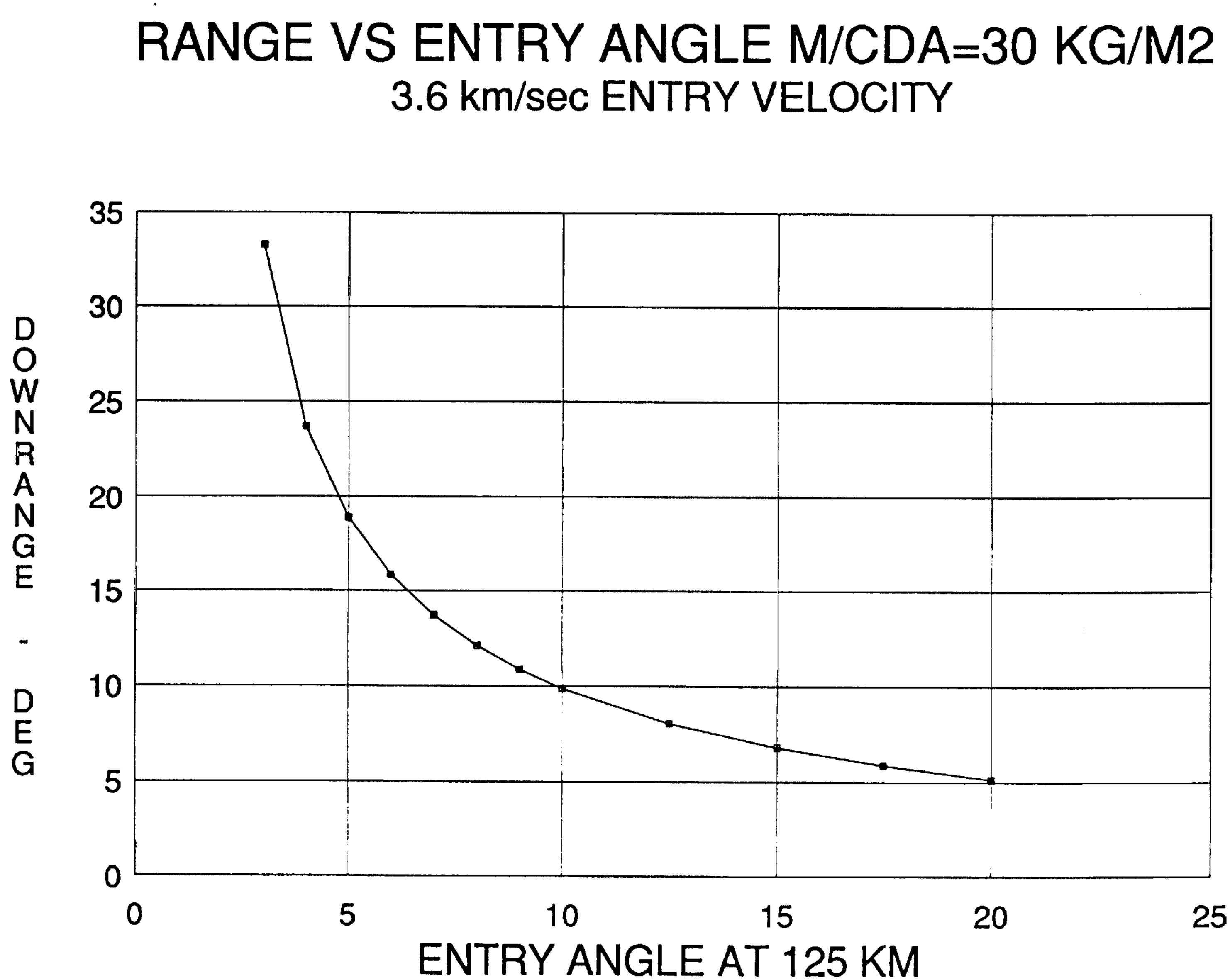

RANGE VS ENTRY ANGLE M/CDA=30 KG/M2
3.6 km/sec ENTRY VELOCITY
DOWNRANGE - DEG
35
30
25
20
15
10
5
0
0
5
10
15
20
25
ENTRY ANGLE AT 125 KM

Figure 11.

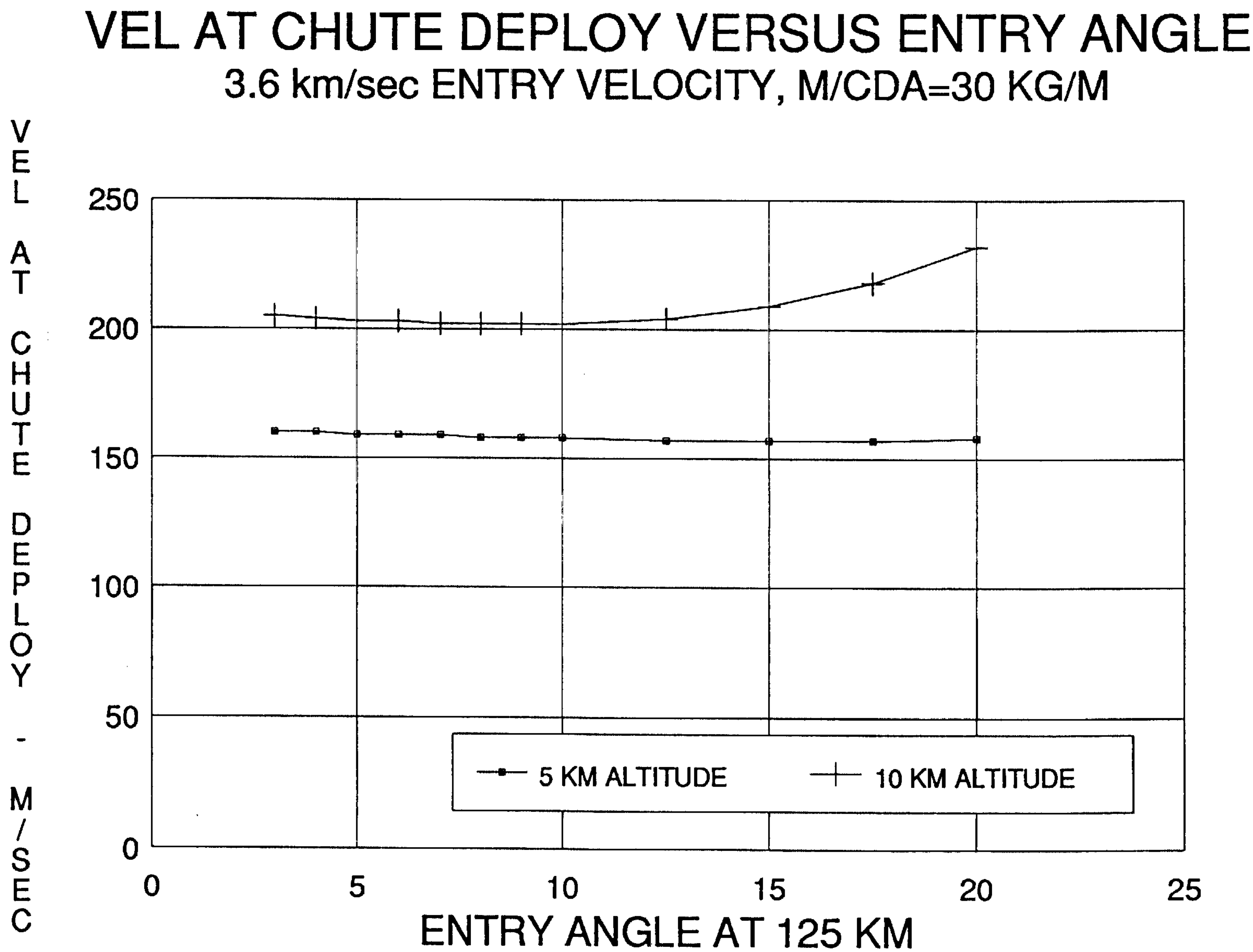
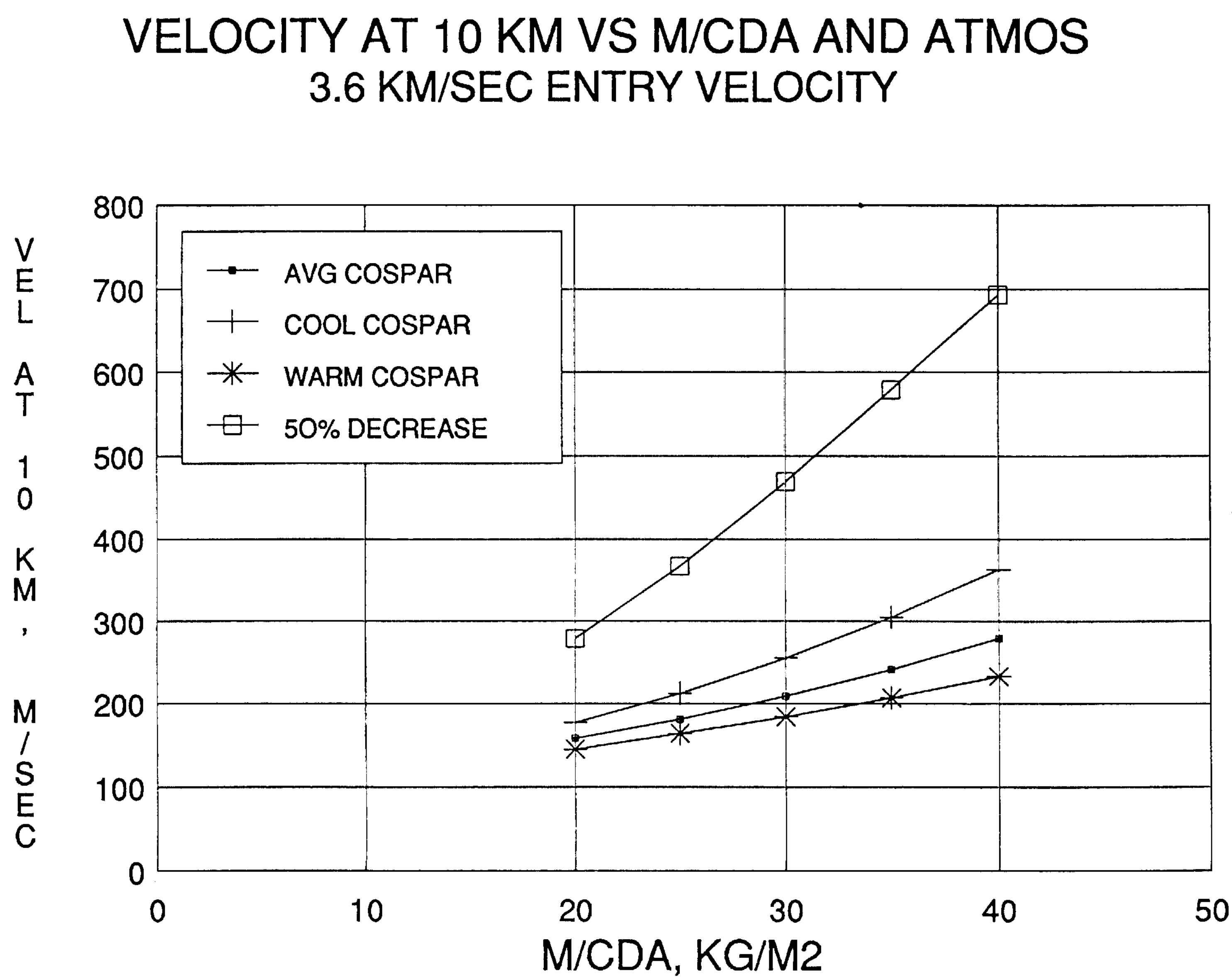

VEL AT CHUTE DEPLOY VERSUS ENTRY ANGLE
3.6 km/sec ENTRY VELOCITY, M/CDA=30 KG/M
VEL AT CHUTE DEPLOY - M/SEC
250
200
150
100
50
0
0
5
10
15
20
25
ENTRY ANGLE AT 125 KM
5 KM ALTITUDE
10 KM ALTITUDE

Figure 12.



VELOCITY AT 10 KM VS M/CDA AND ATMOS
3.6 KM/SEC ENTRY VELOCITY
VEL AT 10 KM, M/SEC
0
100
200
300
400
500
600
700
800
0
10
20
30
40
50
M/CDA, KG/M2
AVG COSPAR
COOL COSPAR
WARM COSPAR
50% DECREASE

Figure 13.

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Session B, Submittal No. 9

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Science Applications International Corporation

MARS GLOBAL NETWORK MISSION

ENTRY AND TERMINAL DESCENT SYSTEM DESIGN CONSIDERATIONS & ISSUES

PRESENTATION TO THE
MARS GLOBAL NETWORK MISSION WORKSHOP
FEBRUARY 6-7, 1990

BY

BYRON L. SWENSON

SAIC®

LANDER ENTRY SYSTEM DESIGN

REQUIREMENTS

- TO PROVIDE STABLE & CONTROLLED ATMOSPHERIC DECELERATION FROM HYPERBOLIC OR ORBITAL ENTRIES
- TO PROVIDE PROTECTION FROM AEROTHERMODYNAMIC HEATING
- TO DECELERATE LANDER TO TERMINAL DESCENT SYSTEM DEPLOYMENT CONDITIONS

DESIGN CONSIDERATIONS & ISSUES

- ENTRY CONDITIONS - SPEED, FLIGHT-PATH ANGLE, ANGLE OF ATTACK
- MARTIAN ATMOSPHERE - LOW DENSITY, VARIATIONS, SITE ELEVATIONS
- FREE-SPACE & ATMOSPHERIC DYNAMICS
- REQUIRED DELIVERY ACCURACIES
- BALLISTIC OR LIFTING TRAJECTORIES
- AERODYNAMIC & AEROTHERMODYNAMIC DATA BASE
- LOW BALLISTIC COEFFICIENT - TERMINAL SYSTEM DEPLOYMENT
- SPACECRAFT INTERFACE - TARGETING, SEPARATION, ENVELOPE
- AEROSHELL JETTISON

SAIC

ENTRY SYSTEM OPTIONS

CONFIGURATIONS

- BALLISTIC

  BLUNTED CONES - VIKING; PV, GALILEO; DISCOVERER

- LIFTING

  SYMMETRICAL (C.G. OFFSET) - BICONICS
  NON-SYMMETRIC - RAKED CONES (AFE); HALF-CONES (M-1)

AEROSHELL DESIGN

- RIGID

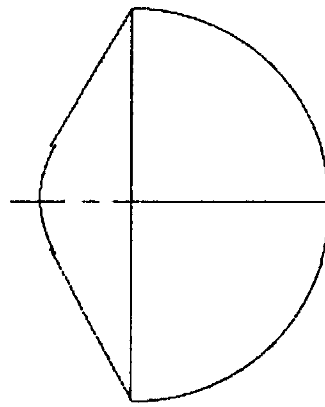
  SKIN/STRINGER; ABLATOR

- DEPLOYABLE

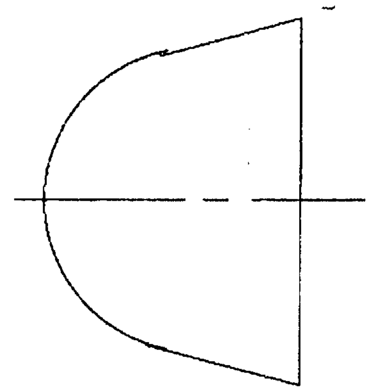
  INFLATABLE BALLUTES - FABRIC; RERADIATOR OR ABLATOR
  COLLAPSABLE - FABRIC/RIB & STRUT; RERADIATOR OR ABLATOR
  FOLDABLE - RIGID HINGED PANELS; RERADIATOR OR ABLATOR

CONFIGURATION OPTIONS

BALLISTIC

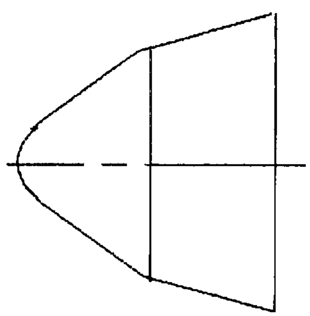


VIKING

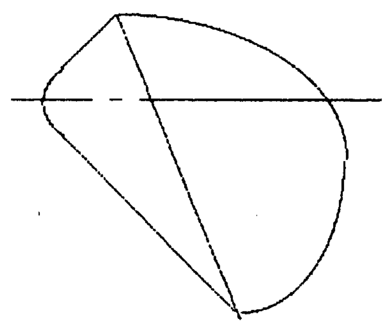


DISCOVERER

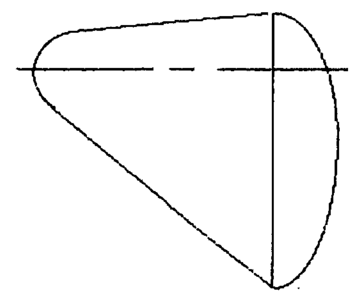
LIFTING



BICONIC

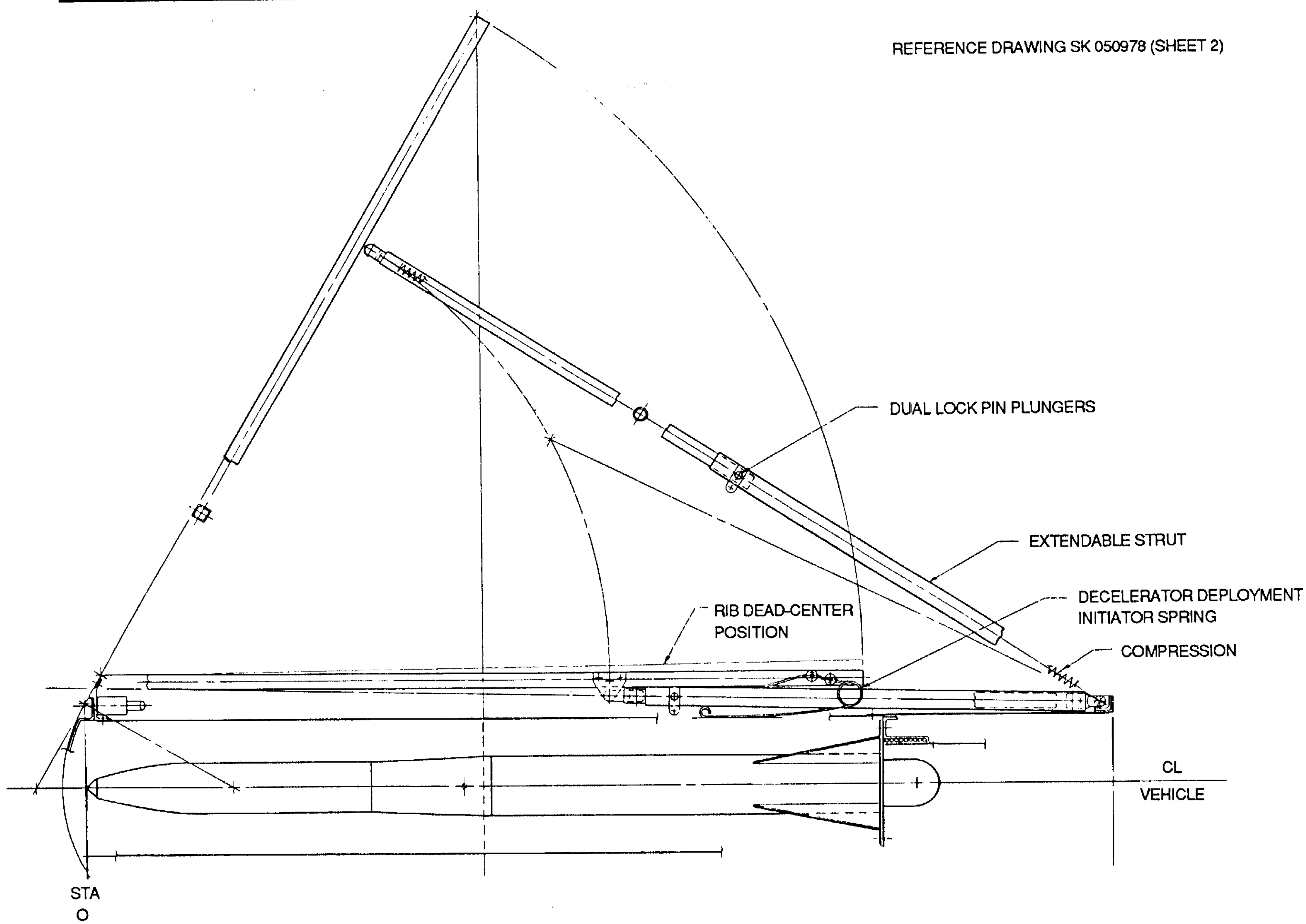


AFE



M-1

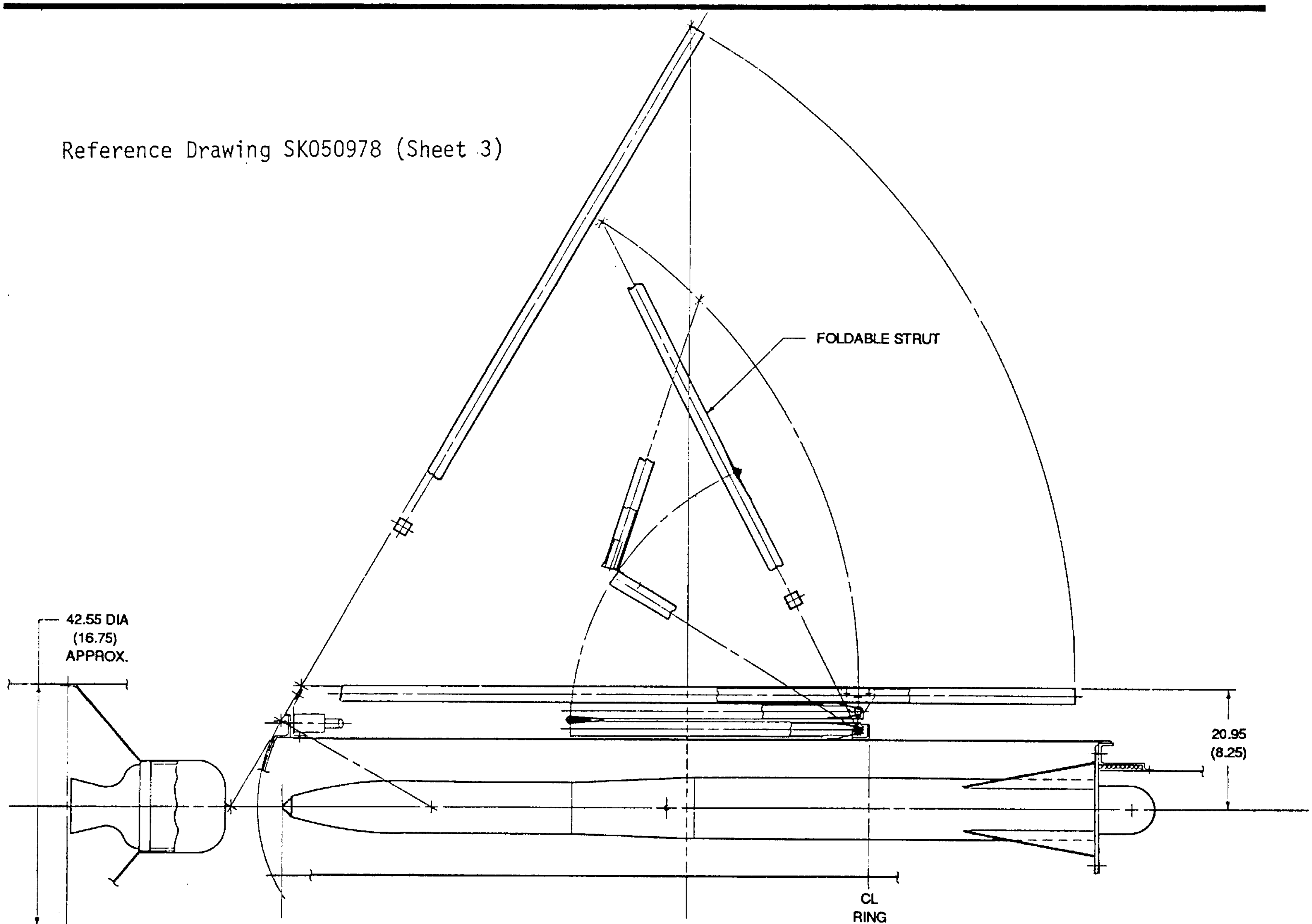
MARS PENETRATOR - EXTENDIBLE STRUT DECELERATOR



Mars Penetrator - Extendible Strut Decelerator

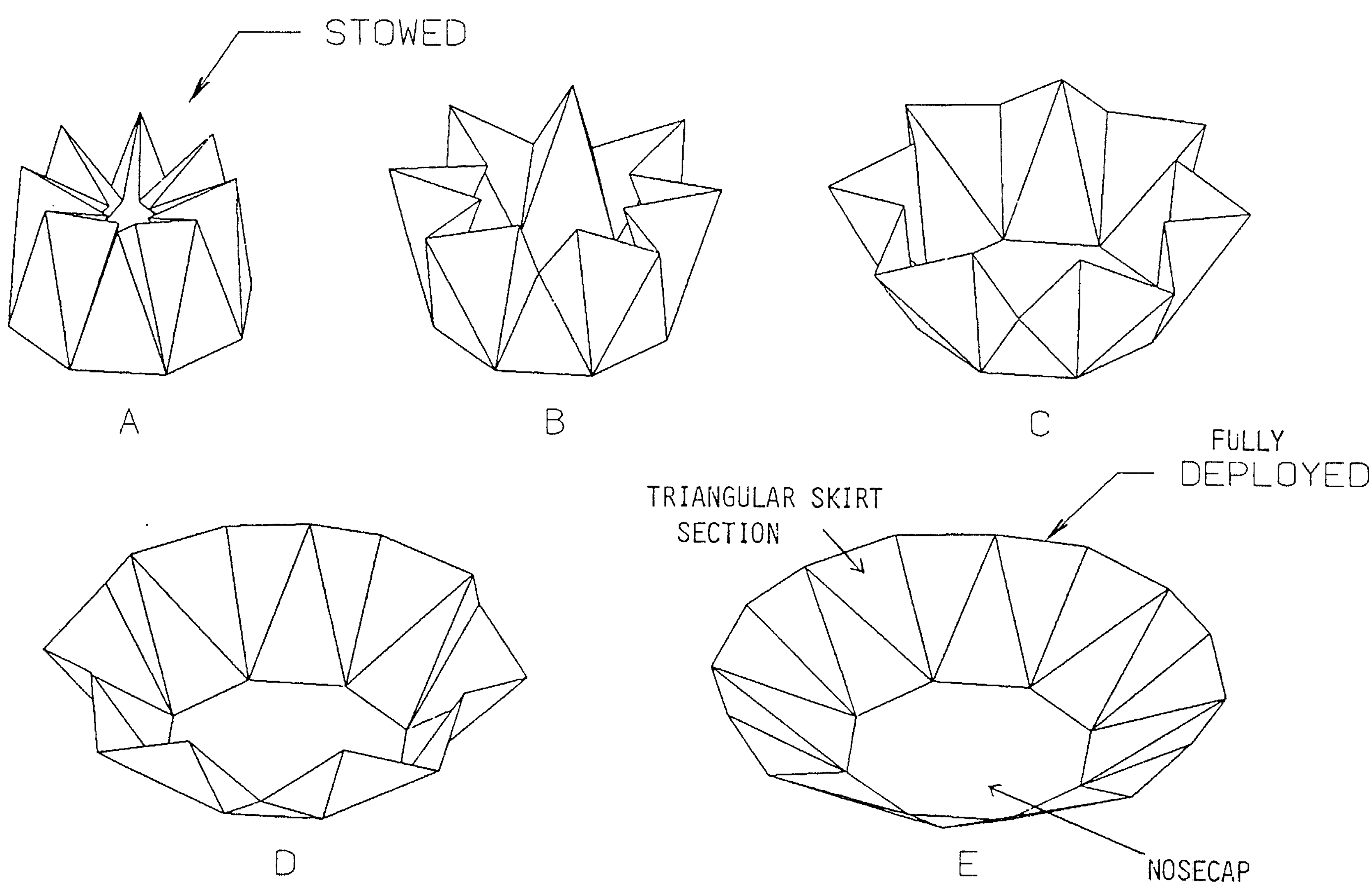
MARS PENETRATOR - FOLDABLE STRUT DECELERATOR

Reference Drawing SK050978 (Sheet 3)



Mars Penetrator - Foldable Strut Decelerator

HINGED PANEL CONCEPT FOR DEPLOYABLE DECELERATOR



AEROSHELL JETTISON OPTIONS

- PULL OUT OF THE REAR
  - REQUIRES LARGE PARACHUTE SYSTEM
- FALL-THRU THE NOSE
  - CONSTRAINS LANDER SHAPE
  - NOSE CAP JETTISON
- SEPARATE IN PIECES
  - HAZARDOUS
  - REQUIRES EXTENSIVE PYROTECHNICS AND VERY ACCURATE TIMING
- RETAIN TO LANDING
  - LANDING SYSTEM DESIGN IMPACT

SAIC®

TERMINAL DESCENT SYSTEM DESIGN

REQUIREMENTS

- TO PROVIDE A STABLE & CONTROLLED PLATFORM FOR DESCENT IMAGERY
- TO PROVIDE STABLE & CONTROLLED SURFACE APPROACH CONDITIONS FOR LANDING

DESIGN CONSIDERATIONS & ISSUES

- DEPLOYMENT CONDITIONS
- TECHNOLOGY BASE
- DESCENT TIME, RATE, & STABILITY FOR IMAGERY
- WINDS ALOFT & SHEARS
- ELEVATION DIFFERENCES OF SITES
- REQUIRED LANDING CONDITIONS
  - VERTICAL & HORIZONTAL VELOCITY
  - STABILITY & DYNAMICS
- SURFACE WINDS & SHEARS

Session B, Submittal No. 10

Arden Albee
California Institute of Technology

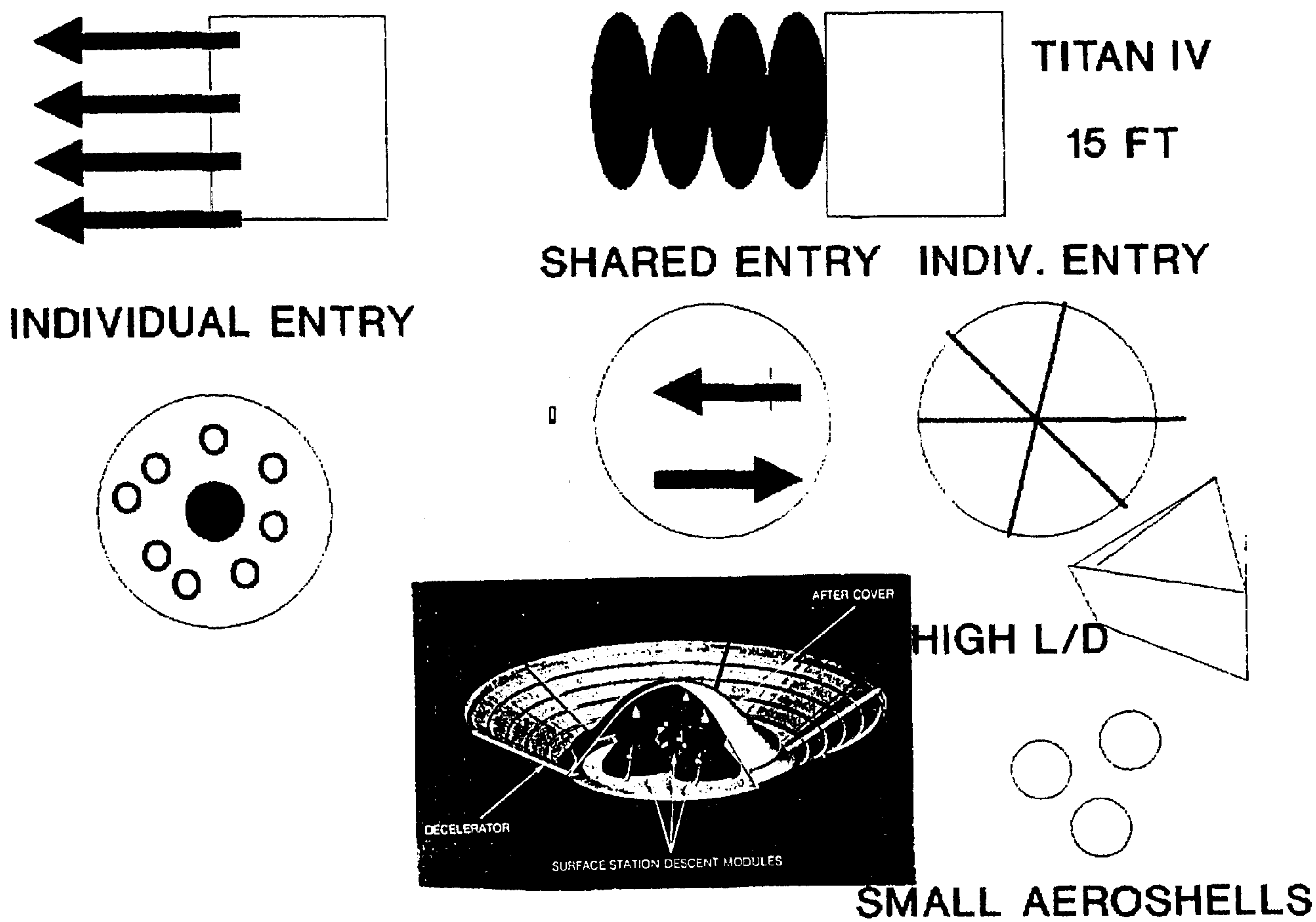
COMPARISON OF ALTERNATIVE LANDER STRATEGIES

ARDEN ALBEE

CALTECH

"MANY,GLOBALLY-DISPERSED" LANDERS DRIVE DESIGN

PACKING AND DEPLOYMENT VS STRUCTURE/PROP MASS



ALA 2-90

MARS ENTRY & LANDER SYSTEMS
DESIGN TREE

1) LANDER SYSTEM RELEASE--	ON APPROACH	FROM ORBIT		
2) INITIAL RETRO & ORIENT.--	BUS DEFLECT.	LANDER DEFLECT.		
3) ATM. ENTRY ANGLE--	CONSTRAINED	UNCONSTRAINED		
4) HIGH/HOT DECELERATION--	SHARED TPS	OWN TPS	GUIDED ENTRY	
5) LOW ATM. DECELERATION-- (Supersonic to subsonic)	BALLUTE	PARACHUTES	L/D SHAPES	
6) TERMINAL DECELERATION--	DESCENT RETRO.	PROX. RETRO.	IMPACT ABSORB.	SURFACE PENET
7) LANDER TYPE--	SOFT	ROUGH	HARD	PENETRATOR
8) TELEMETRY--	DIRECT	RELAY	W/ WO RECEIVER	

ALA-2-90

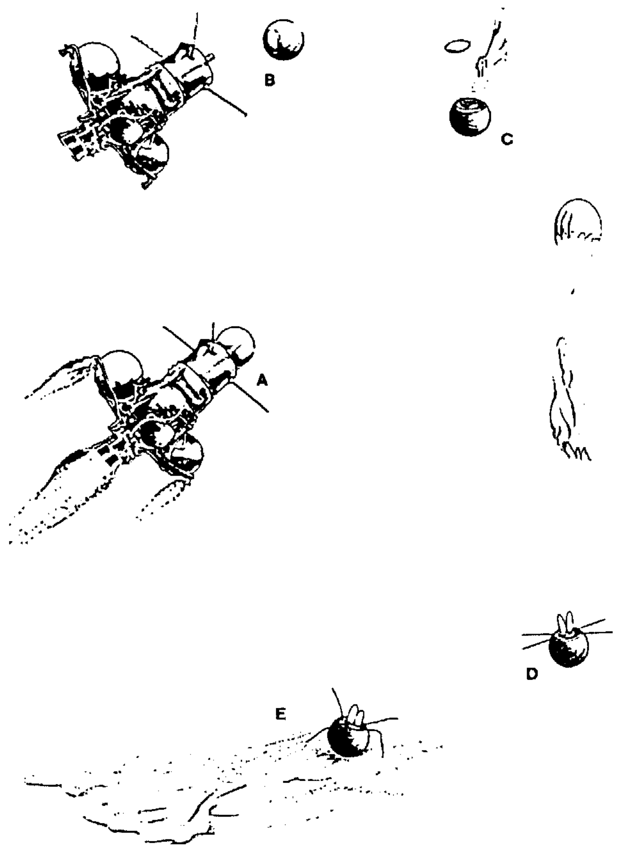
MARS ENTRY & LANDER SYSTEMS

DESIGN TREE

1) LANDER SYSTEM RELEASE--	ON APPROACH M, A, V8	FROM ORBIT VK		
2) INITIAL RETRO & ORIENT.--	BUS DEFLECT. A, V8	LANDER DEFLECT. M, VK		
3) ATM. ENTRY ANGLE--	CONSTRAINED M, VK, A	UNCONSTRAINED V8		
4) HIGH/HOT DECELERATION--	SHARED TPS	OWN TPS M, VK, V8, A	GUIDED ENTRY	
5) LOW ATM. DECELERATION-- (Supersonic to subsonic)	BALLUTE M, A	PARACHUTES M, VK, V8	L/D SHAPES	
6) TERMINAL DECELERATION--	DESCENT RETRO. VK	PROX. RETRO. M	IMPACT ABSORB. V8	SURFACE PENET A
7) LANDER TYPE--	SOFT VK	ROUGH M	HARD V8	PENETRATOR A
8) TELEMETRY--	DIRECT	RELAY M, VK, V8, A	W/ WO RECEIVER VK/M, A, V8	

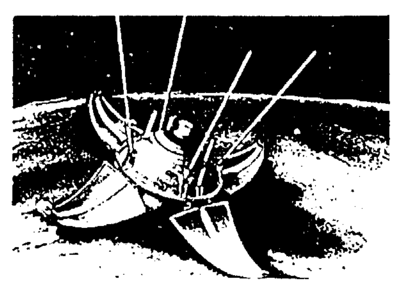
Key: M = Mars 3; VK = Viking; V8 = Venera 8; A = Ames 77

SPHERICAL ENTRY SYSTEMS

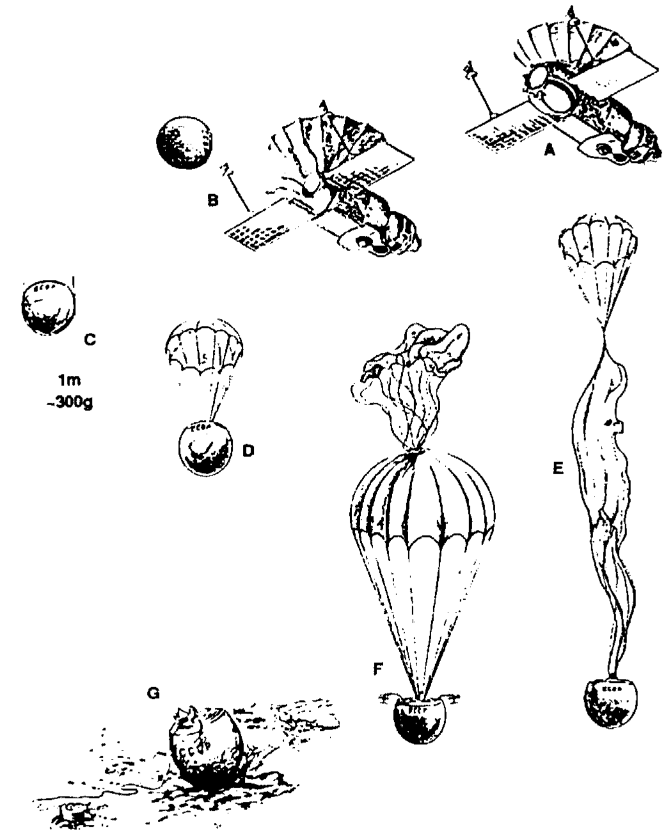


A. Blast-off from the moon
B. Ejection of the return capsule near earth
C. Drogue parachute deployment after reentry into earth's atmosphere; 20-g deceleration
D. Deployment of the main parachute
E. Touchdown of the return capsule on Soviet territory

The return of Luna 16 to earth (R. Escarcega)



Sketch of Luna 9 on the lunar surface



A. Venera 8 as it approaches Venus
B. Separation of descent capsule from carrier bus
C. Entry into Venerian atmosphere
D. Deployment of drogue parachute
E. Deployment of main parachute
F. Activation of scientific instruments
G. Landing on the Venerian surface

Descent sequence of Venera 8 (R. Escarcega)

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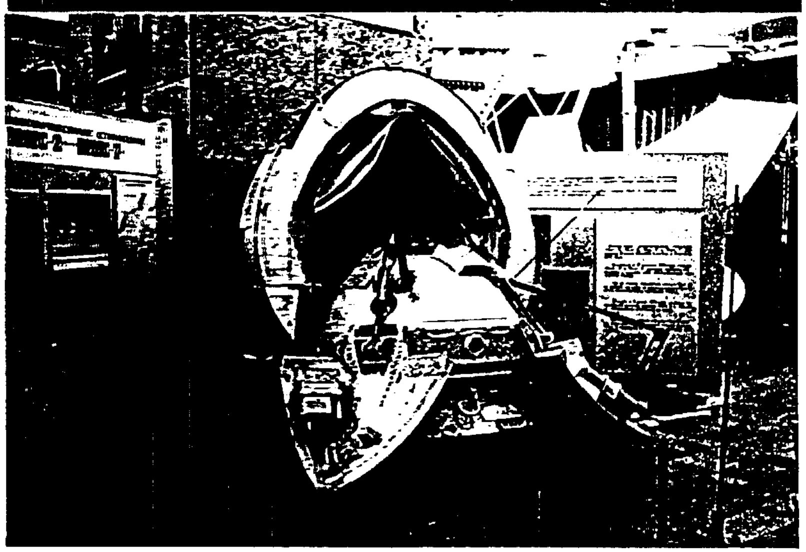
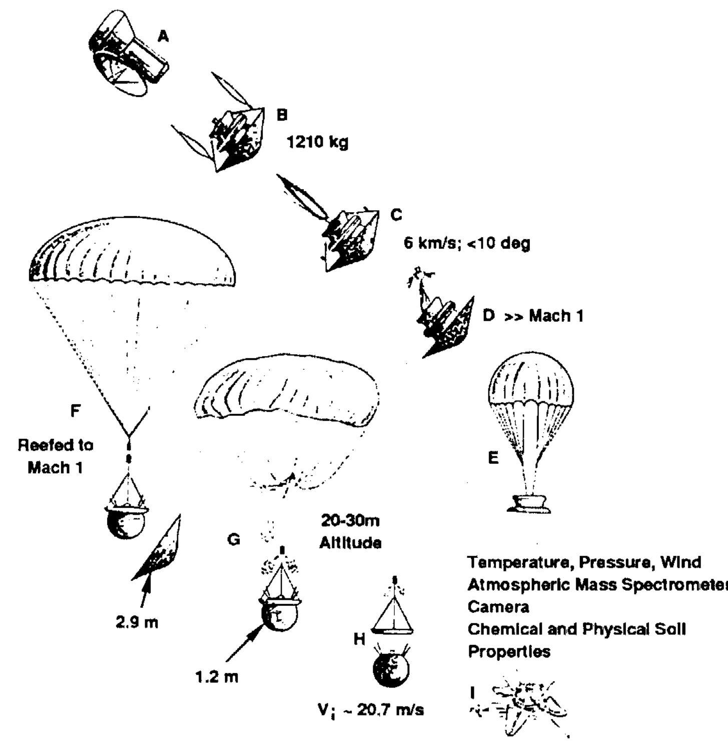
MARS 3 LANDER

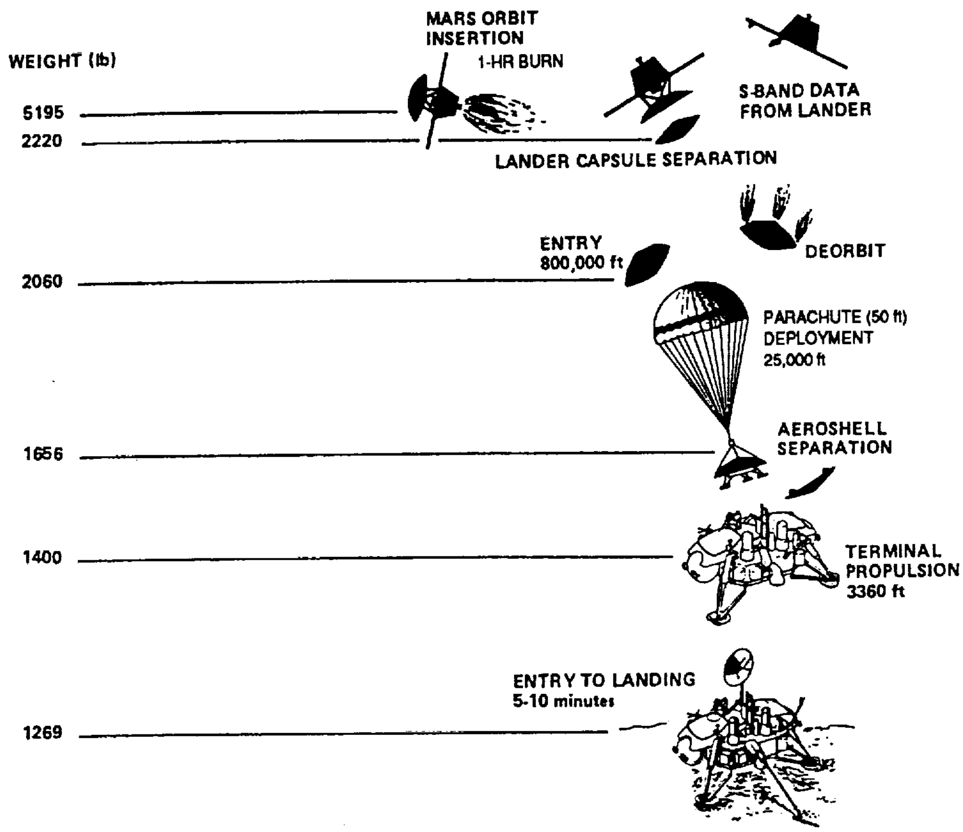


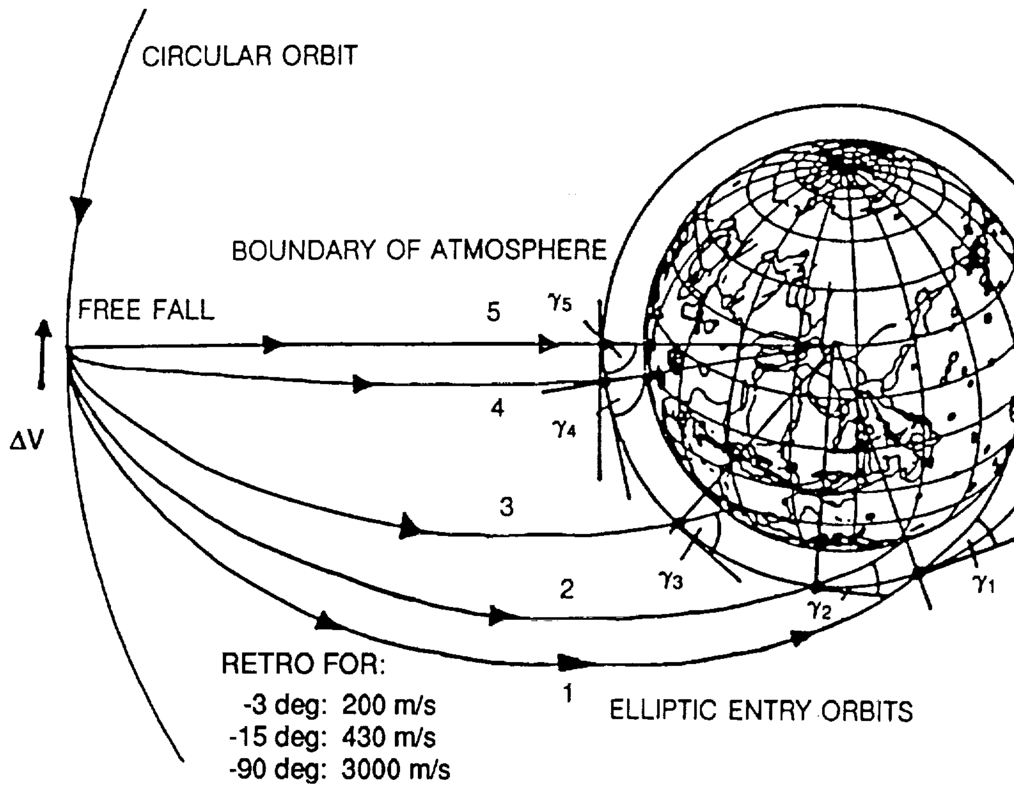


A. Mars orbiter after separation of descent apparatus
B. Descent apparatus pulling away from Mars orbiter -4 1/2 hr before MOI
C. Firing of main descent engine for entry into the Martian atmosphere
D. Deployment of drogue parachute
E. Separation of drogue parachute and upper portion of main parachute housing
F. Deployment of main parachute
G. Detachment of main parachute and ignition of final braking rocket
H. Separation of landing capsule
I. Landing capsule on the Martian surface with deployed instrumentation

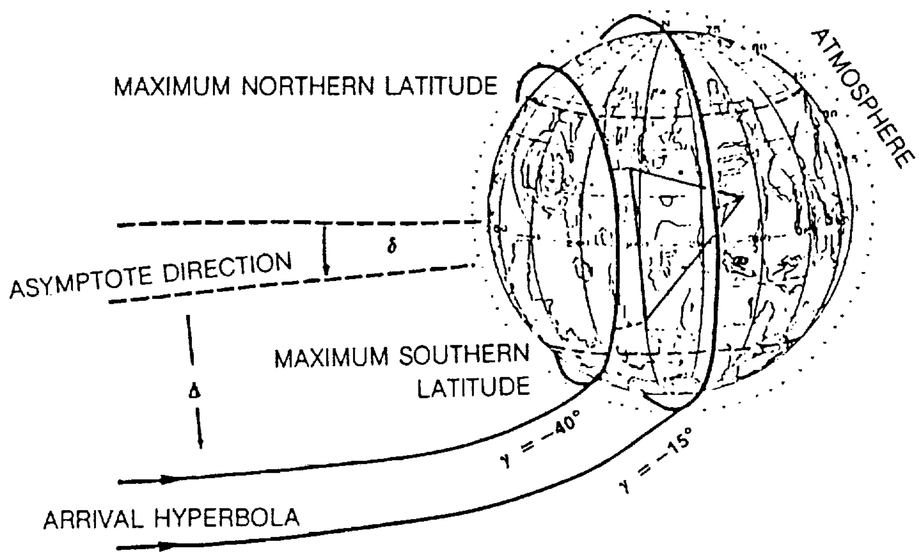
Landing sequence of second-generation Mars spacecraft
(R. Escarcega)

Viking Entry Through Landing Sequence

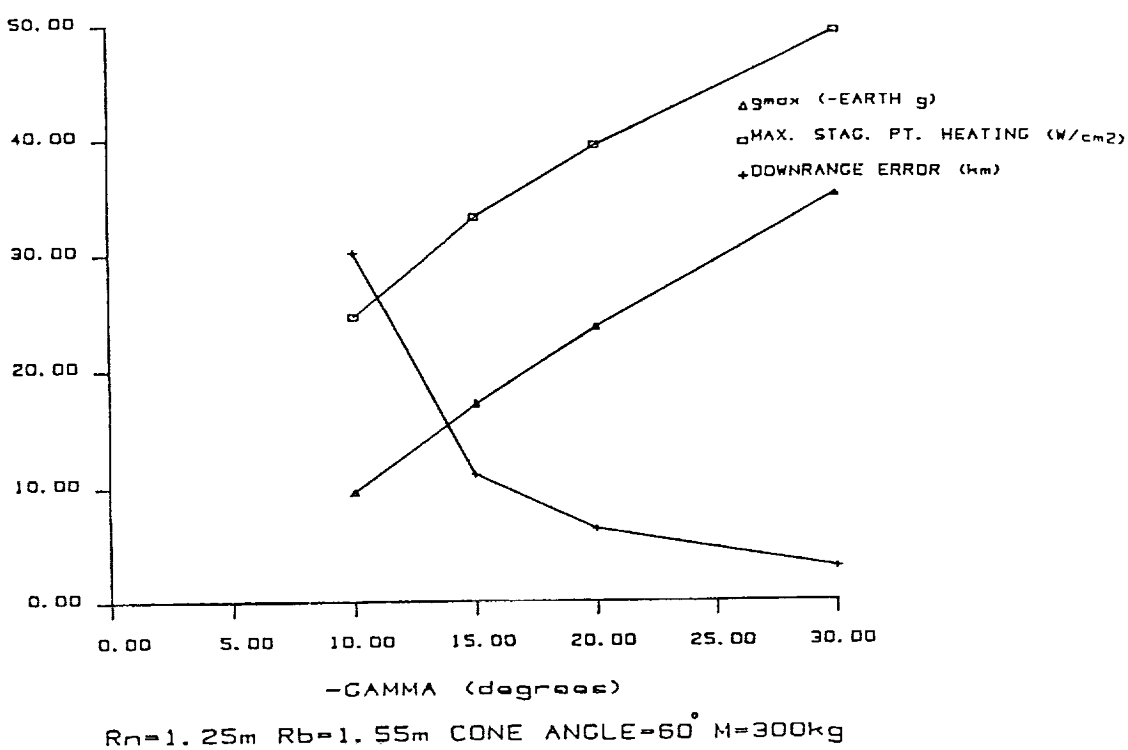




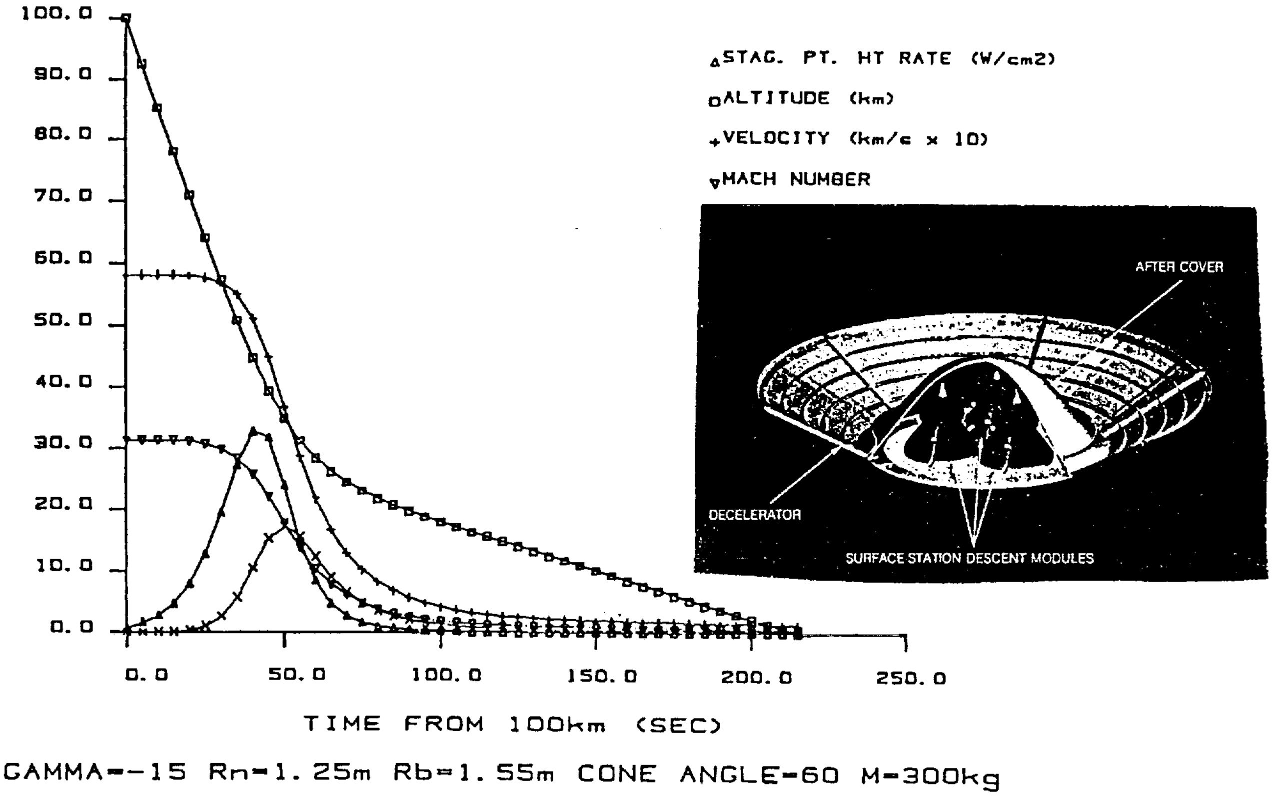
Deorbit manoeuvre from low circular orbit.



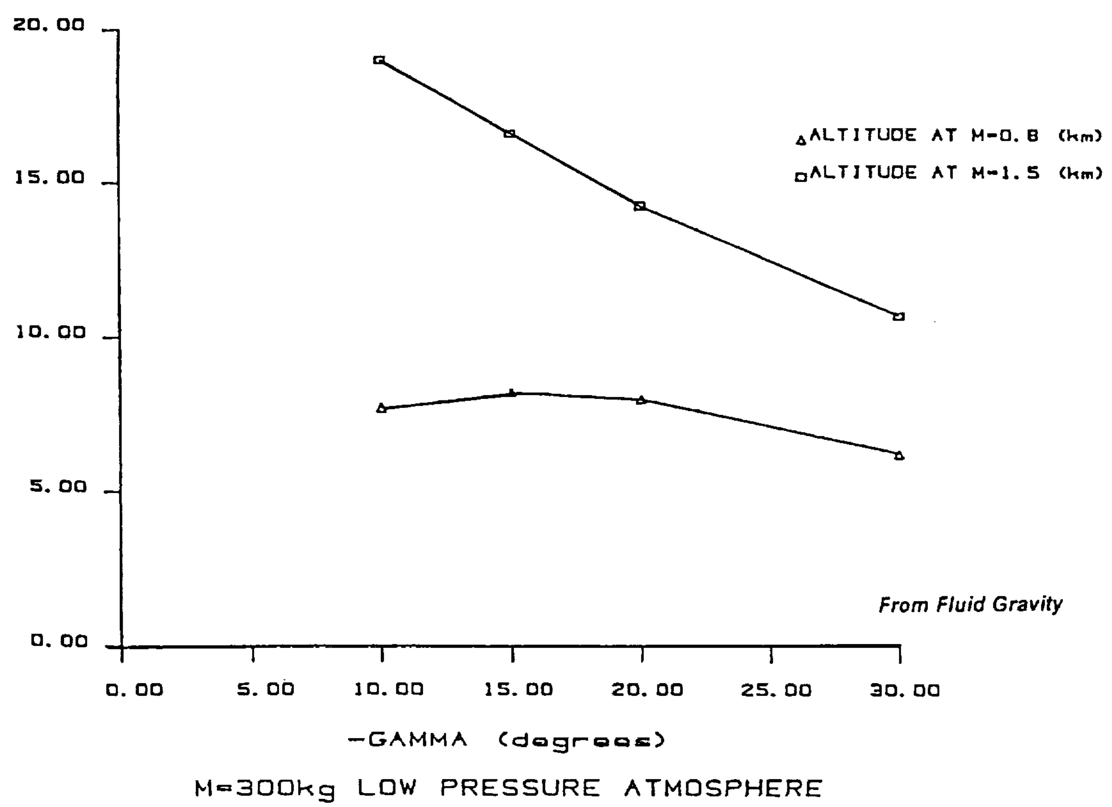
Location of entry points at Mars' atmosphere.



Ballistic entry from hyperbolic arrival (performance for different entry angles).



Ballistic entry from hyperbolic arrival (trajectory parameters).



Ballistic entry from hyperbolic arrival (parachute deployment altitude).

Session B, Submittal No. 11

Lester L. Sackett
The Charles Stark Draper Laboratory

GLOBAL NETWORK MISSION WORKSHOP

Achievable Accuracies for Targeting Landing Sites

Les Sackett (617/258-2289)
The Charles Stark Draper Laboratory
Cambridge, Massachusetts 02139

Workshop held at the Jet Propulsion Laboratory
February 6-7, 1990

Data provided by
John Higgins (617/2582433)
Ken Spratlin (617/258-2441)

CSDL

Global Network Workshop

Primary Question:

What are the desired and achievable accuracies for targeting the landing sites?

Subsidiary Questions:

What are the navigation (knowledge) uncertainties at the time of aeroshell firings?

What are the landing (guidance) dispersions of the penetrators?

What contributes to the errors in knowledge and targeting accuracy?

How can the errors be reduced?

For the approach targeted aeroshells, what are the errors as a function of the deployment time?

Does onboard nav help and how much?

What is the dispersion due to passage of the aeroshell through the atmosphere?

Due to the time on the parachute?

Due to the error introduced by the small rocket firing?

Does Viking experience help in estimating the targeting accuracies?

What is the effect on the trajectory of the despin from 60 to 15 rpm following the targeting firing?

What is the effect of changing the assumptions or parameter values (e.g., flight path angle, ballistic coefficient, etc.)?

EI FOOTPRINT SENSITIVITY TO BURN ERRORS

(Approach Deployed Aeroshell)

- With nominal entry-interface (EI) conditions
  - — h_{EI} = 125 km
  - — γ_{EI} = -20 deg
  - — V_∞ = 3.5 km/s

- Accelerometer quantization = 0.005 m/s (typical)

- Assume ability to effect burn is comparable to ability to measure burn

- Thus, assuming 1 accelerometer quant of ΔV error per axis

DAYS PRIOR TO EI	1	2	5	10	20
$\Delta\gamma$ (deg)	0.01	0.02	0.06	0.12	0.24
ΔDR (km)	1.06	2.08	5.13	10.19	20.32
ΔCT (km)	0.28	0.55	1.34	2.68	5.34

CSDL

MARS PENETRATOR VEHICLE

ENTRY DISPERSION ANALYSIS

John P. Higgins
(617)–258–2433

CSDL

ASSUMPTIONS

- PRE–DEORBIT TRAJECTORY
  - SEMI–MAJOR AXIS = 6986 KM
  - ECCENTRICITY = 0.485097
- ENTRY INTERFACE CONDITIONS
  - ALTITUDE = 125K KM
  - VELOCITY = 4.2 KM/S (DEORBIT PENETRATOR)
  - VELOCITY = 6.0 KM/S (DEPLOYED PENETRATOR)
  - GAMMA = –15 / –20 DEGREES
- BALLISTIC ENTRY (ZERO LIFT)
- PENETRATOR BALLISTIC COEFFICIENT = 30 KG / SQ METERS
- PARACHUTE BALLISTIC COEFFICIENT = 3 KG / SQ METERS
  - DEPLOYED AT 10 KM ALTITUDE
- ATMOSPHERE MODEL – MARS NORTHERN MEAN
- GRAVITY MODEL – CONIC

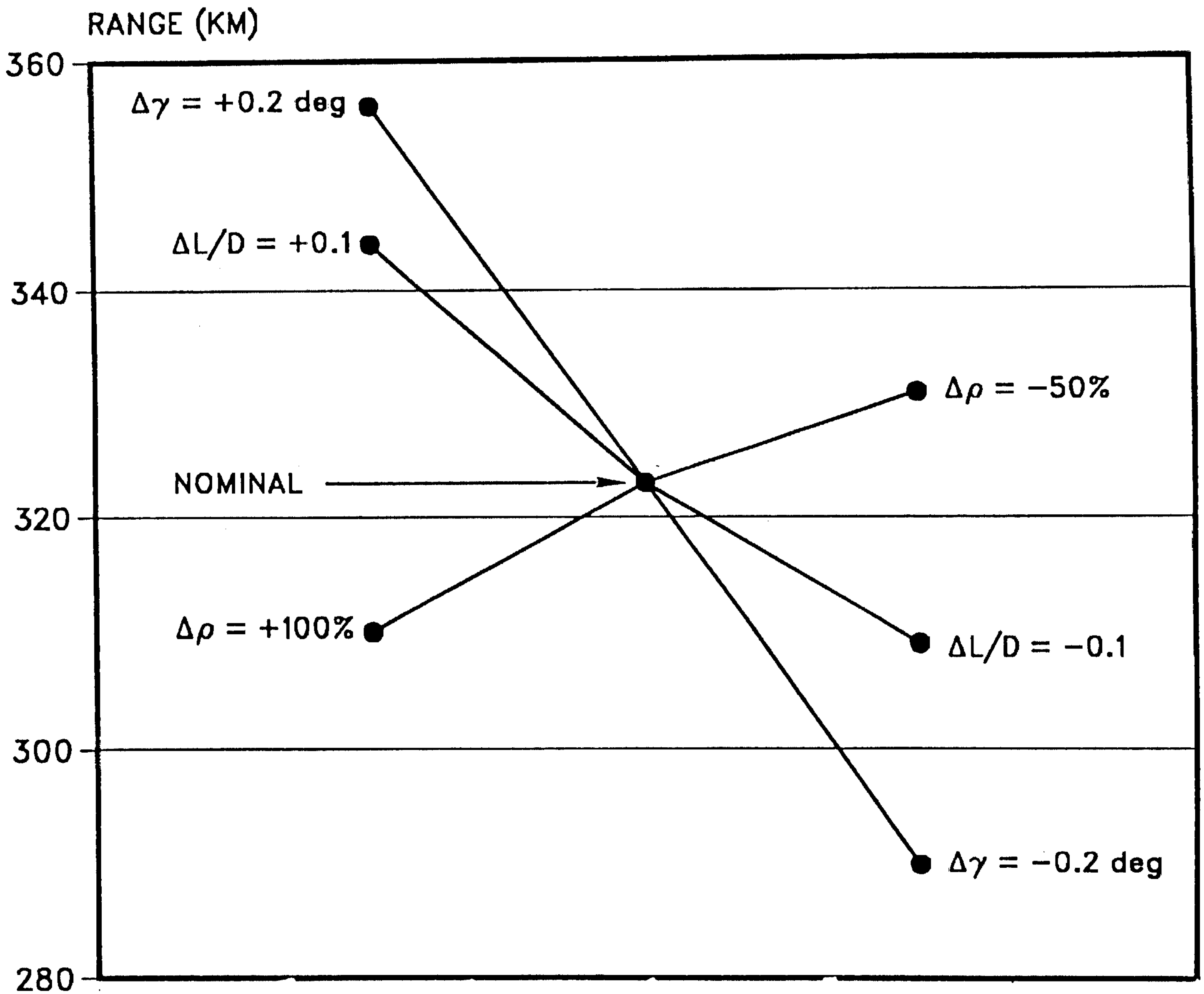
CSDL

ENTRY DISPERSION SOURCES

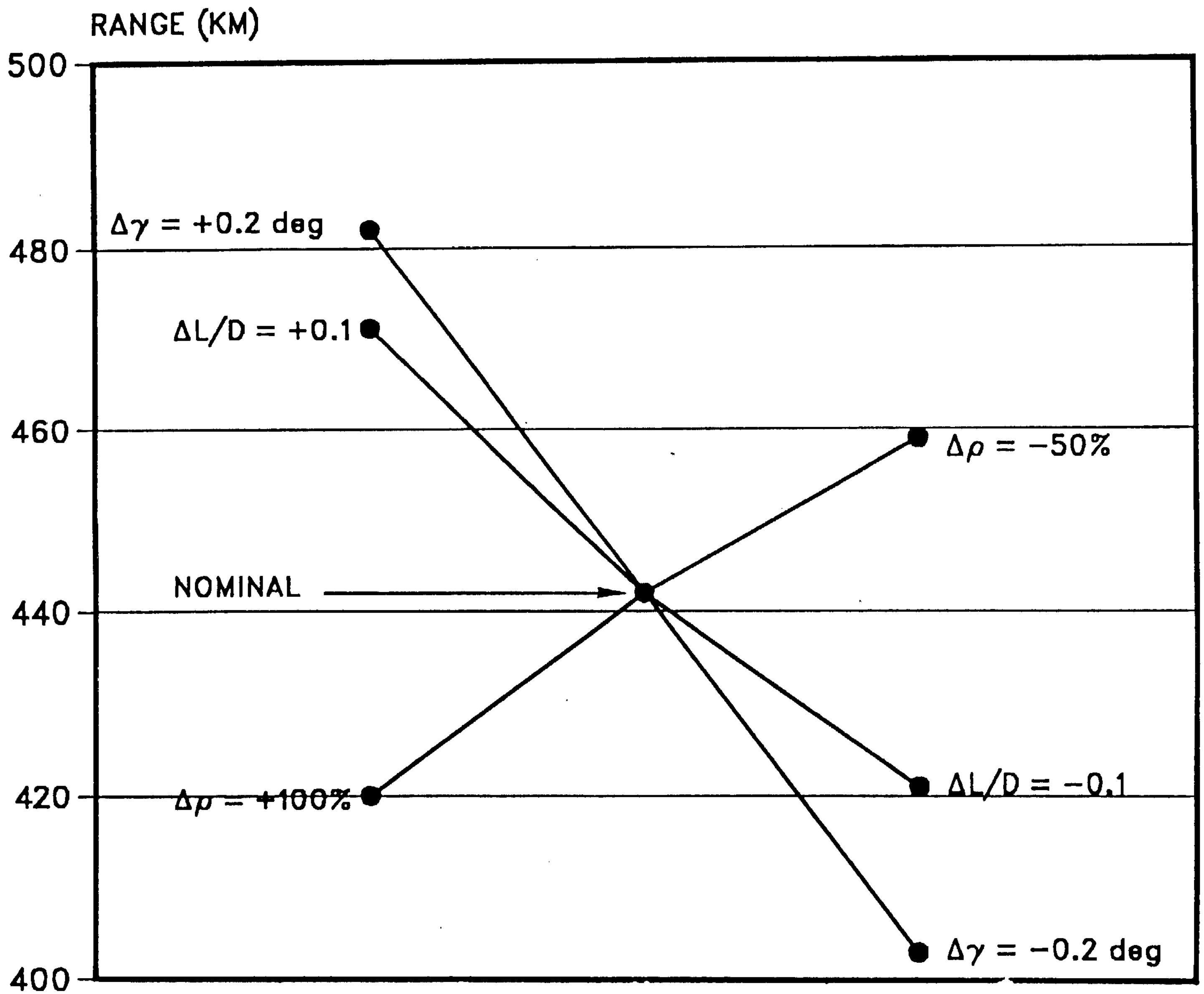
- LIFT–TO–DRAG RATIO
- ATMOSPHERIC VARIATIONS
- ENTRY INTERFACE FLIGHT PATH ANGLE
    - DUE TO GN&C DEORBIT BURN PERFORMANCE (PRIMARILY NAVIGATION ERRORS)
    - DOWNRANGE DISPERSION PRIMARILY DUE TO ENTRY PENETRATION POINT VARIATION

CSDL

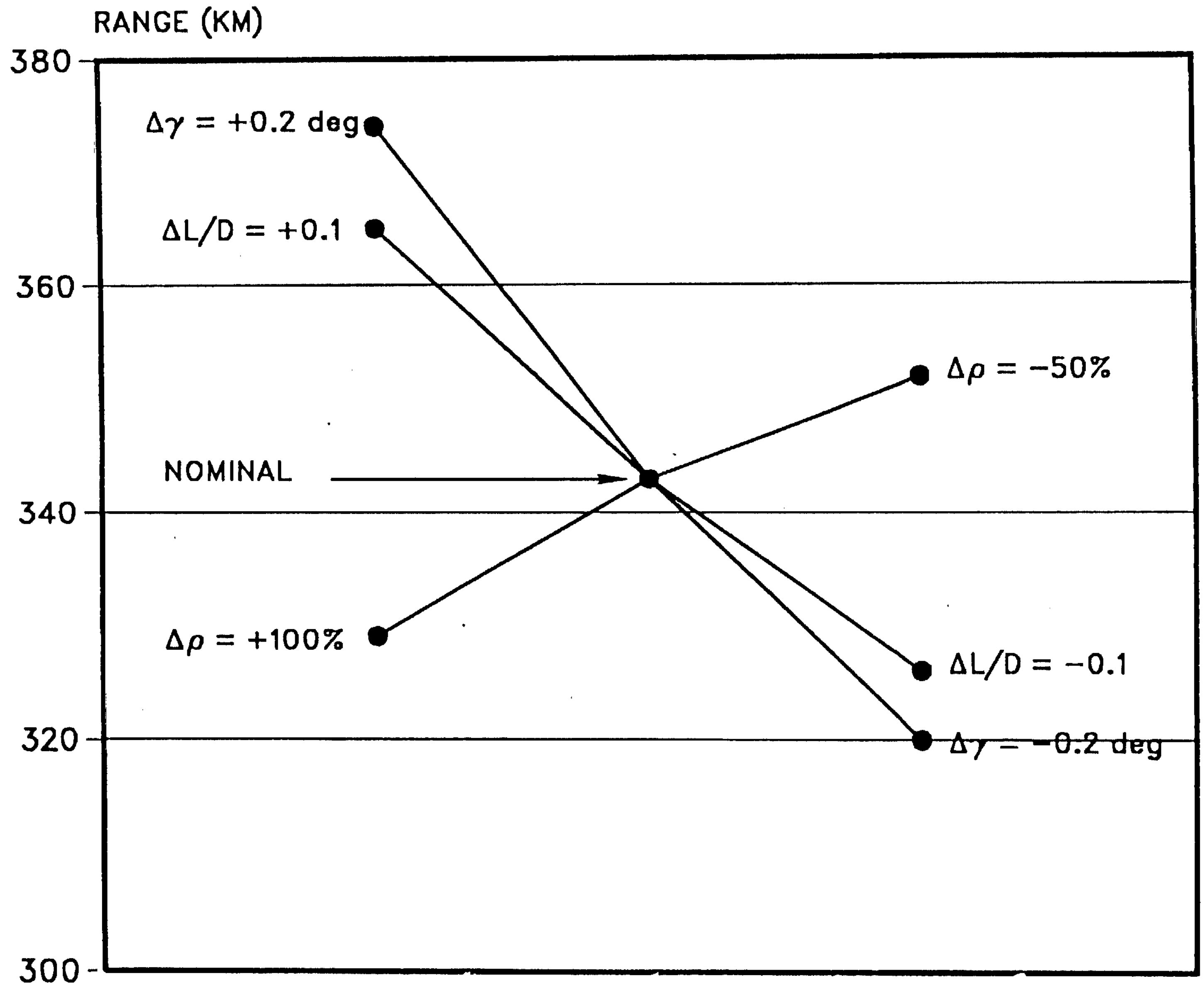
MARS ENTRY DISPERSION ANALYSIS
DEORBIT PENETRATOR – EI GAMMA = –20 deg



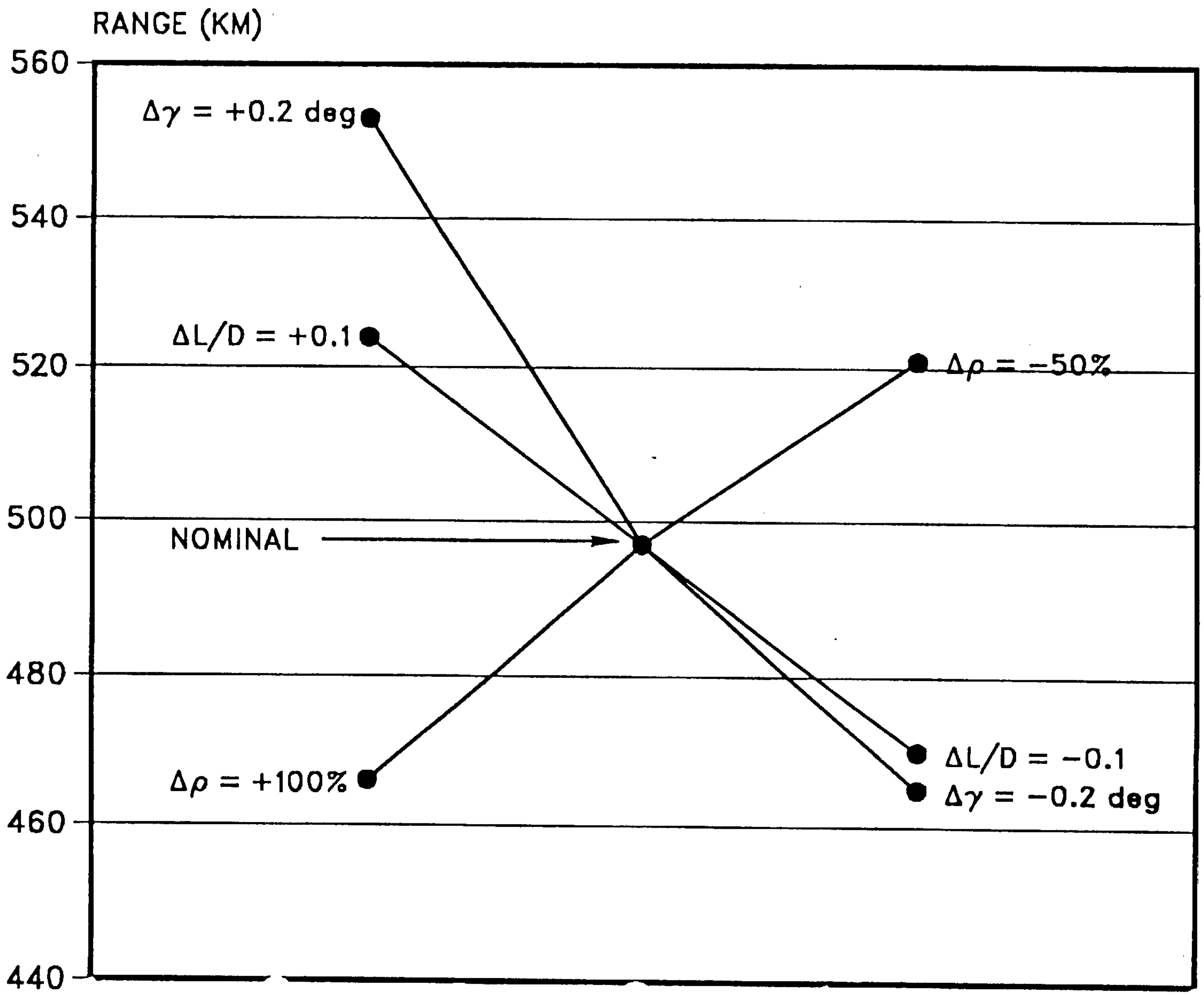
MARS ENTRY DISPERSION ANALYSIS
DEORBIT PENETRATOR – EI GAMMA = −15 deg



MARS ENTRY DISPERSION ANALYSIS
DEPLOYED PENETRATOR – EI GAMMA = –20 deg



MARS ENTRY DISPERSION ANALYSIS
DEPLOYED PENETRATOR – EI GAMMA = –15 deg



Following is a list of memos produced at the Draper Laboratory on guidance, navigation, and control for the Mars Rover Sample Return Mission. Some of these may be relevant to the Global Network Mission studies.

MRSR MEMO LOG
(MARS ROVER SAMPLE RETURN MISSION)

DATE	MRSR-89-/EGB	TO	FROM	SUBJECT
1/3/89	001/EGB-89-001	Distribution	T. Brand	MRSR 1987-1988 Memo Log
1/18/89	002/EGB-89-022	Distribution	S. Bauer	Executive Summary of MRSR Lander Navigation Performance
1/12/89	003/EGB-89-023	Distribution	S. Shepperd	Summary - Mars Communication Orbiter as a Pre-Aerocapture Navigation Aid
2/7/89	004/EGB-89-034	Distribution	T. Brand	Mars Mission - Pre-aerocapture Navigation Communication Orbiter as a Nav Aid Preliminary Study
2/27/89	005/EGB-89-035	Distribution	S. Bauer/ T. Brand	MRSR Deorbit/Entry Navigation Performance with Updates from a Communications Orbiter
3/15/89	006/EGB-89-037	File	A. Engel	DSMAC Discussions in January at the Naval Avionics Center
4/7/89	007/EGB-89-084	File	D. Fuhry	High Energy Aerobraking Task Status
4/11/89	008/EJC-89-065R	T. Brand	R. Greenspan	Electronics Burden of Radio Navigation for a Mars Lander
5/11/89	009/EGC-89-118	Distribution	H.L. Malchow	Qualitative MRSR Requirements Survey
5/18/89	010/EGC-89-127	Distribution	L. Sackett	Flight Control Presentation at JSC on April 20, 1989

The Charles Stark Draper Laboratory, Inc.

MRSR MEMO LOG

DATE	MRSR-89-/EGB-	TO	FROM	SUBJECT
5/19/89	011/EGB-89-115	File	K. Spratlin/ T. Brand	MRSR Entry and Landing Task Presentation to MRSR Videocon
6/21/89	012/EGB-89-137	Distribution	A. Engel	Briefing Charts - Autonomous Landing Study Probability of Safe Landing
6/21/89	013/EGB-89-139	File	A. Engel	Briefing Charts - Safe Landings for MRSR (April 20, 1989)
	014 CANCELLED			
6/21/89	015/EGB-89-141	Distribution	D. Fuhry	CSDL Presentation Given at MRSR Mars Approach Navigation Coordination Meeting
6/21/89	016/EGB-89-142	D. Smith/JPL	D. Fuhry	Representative Computer Throughput Requirements for CSDL Predictor/ Corrector Aerocapture Guidance
7/12/89	017/EGB-89-151	Distribution	S. Bauer	MRSR Deorbit/Entry Navigation Performance
7/31/89	018/EGB-89-169	Distribution	A. Engel/ S. Bauer	Briefing Charts - MRSR Probability of Landing and Deorbit/Entry Navigation Performance Studies
8/14/89	019/EGB-89-174	Distribution	T. Brand	Reduced IMU Alignment Error Prior to Deorbit and Entry
9/1/89	020/EGC-89-211	Dist.	H. Malchow	Attitude Control Configuration Options for the Trans-Mars Cruise Phase of the MRSR Misssion Sample Return Segment

MRSR MEMO LOG

DATE	MRSR-89-/EGB	TO	FROM	SUBJECT
10/2/89	**021**/EJC-89-157	T. Brand	R. Greenspan	Power Considerations for Radio Navigation Aids during Mars Approach
10/3/89	**022**/EGB-89-216	K. Spratlin	W. McClain	Comments on "Trajectory Predrediction Accuracy for Lower Orde Mars Gravity Models"
10/3/89	**023**/EGC-89-225	Distribution	D. McCallum	Charts of Performance Specifications for Attitude Control Actuators and Sensors Rendezvous and Docking Sensors and Accelerometers
10/6/89	**024**/EGB-89-219	Distribution	S. Shepperd	Mars Approach Navigation Constant Power Radiometric Navigation Performance
10/6/89	**025**/EGB-89-221	Distribution	S. Bauer	MRSR Pointing Accuracy During Entry
10/26/89	**026**/EGB-89-235	Distribution	K. Spratlin	1989 MRSR Final Report Inputs
11/14/89	**027**/EGB-89-255	File	K. Spratlin	CSDL Contribution to Lunar/Mars Transportation Study Requested by D. Long/JSC
11/15/89	**028**/EGC-89-275	Distribution	D. McCallum	Attitude Control Requirements for Aerocapture Phases of MRSR Mission
11/15/89	**029**/EGB-89-260	Distribution	S. Bauer	Mars Entry Navigation Using IMU without External Updates
11/17/89	**030**/EGB-89-299	Distribution	S. Bauer	Mars Orbit Navigation with Radiometric Updates from a Communication Orbiter

MRSR MEMO LOG

DATE	MRSR-89-/EGB	TO	FROM	SUBJECT
12/31/89	031/EGB-89-303	File	D. Brown	Presentation on Sensor Look Angles During Hazard Avoidance

Session B, Submittal No. 12

Bruce A. Crandall
Hughes Aircraft Company

LAUNCH VEHICLE ALTERNATIVE STRATEGIES

BRUCE CRANDALL

HUGHES AIRCRAFT CO.

HUGHES AIRCRAFT COMPANY

LAUNCH/MISSION OPTIONS

SYSTEM	COST	SATISFACTION
BASELINE TIV/CENTAUR	100%	100%
2-3 ATLAS/CENTAUR *	$\sim$50%	$\lesssim$100%
4-6 ATLAS/CENTAUR *	100%	>>100%

*EXISTING INTEGRATION FOR 6500+ LB CLASS, 8 FOOT BODY WITH INTEGRAL PROPULSION

LARGE TRADE SPACE EXISTS DUE TO HIGH COST OF TIV/CENTAUR

PROBE INTEGRATION OPTIONS
COMPARISON WITH PIONEER VENUS

	PROBE INTEG	INSERT PROP	ORBITAL SCIENCE	PROBE COMM	BUS CONTROL
GNM CAR/ORB	X	X	X	X	X
PIONEER VENUS					
MULTIPROBE	X			X	X
ORBITER		X	X		X

DIVIDING FUNCTIONS =LESS COMPLEX INTEGRATION
MORE ORBITAL FLEXIBILITY
DEPLOYMENT
COMMUNICATIONS

SUMMARY

MAINTAINING A BROAD VIEW OF THE GNM OPTIONS AT THIS TIME WILL LEAD TOWARD THE MOST OPTIMUM DESIGN

SIGNIFICANT MATURE TECHNOLOGIES EXIST IN THE AREA OF HIGH G ELECTRONICS, PROPULSION, IMAGING AND GUIDANCE...WITH DIRECT APPLICABILITY TO..

PENETRATOR DESIGN
DEPLOYMENT PHILOSOPHY
CARRIER INTEGRATION
TARGETING FIDELITY

BEST SATISFACTION TO COST PERFORMANCE MAY BE REALIZED USING SMALLER LAUNCH VEHICLES AND SATELLITES

SCIENCE MISSION PRIORITIZATION AND CHARACTERIZATION WILL HELP TO CONVERGE MISSION REQUIREMENTS AND DESIGN

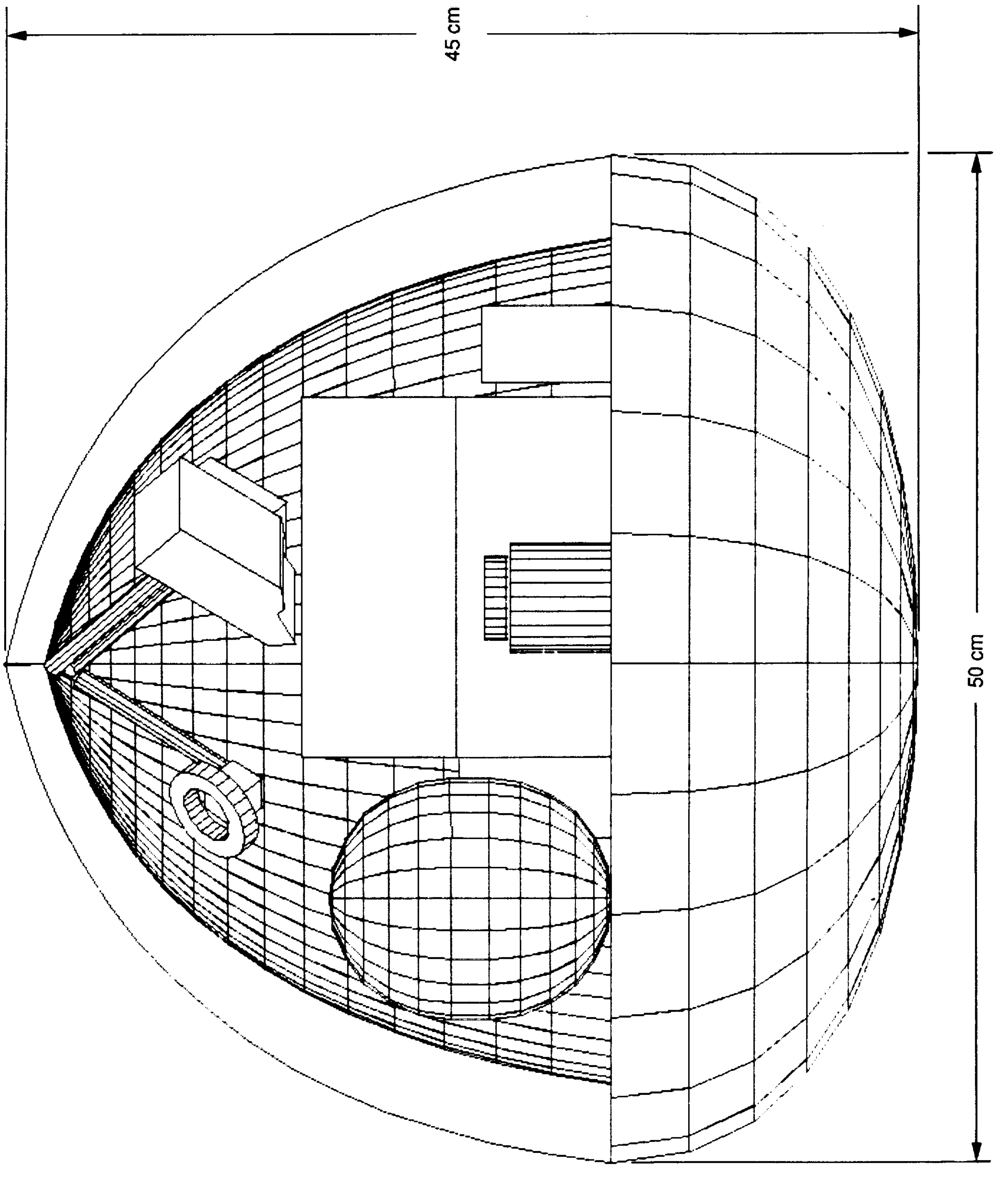
Session B, Submittal No. 13

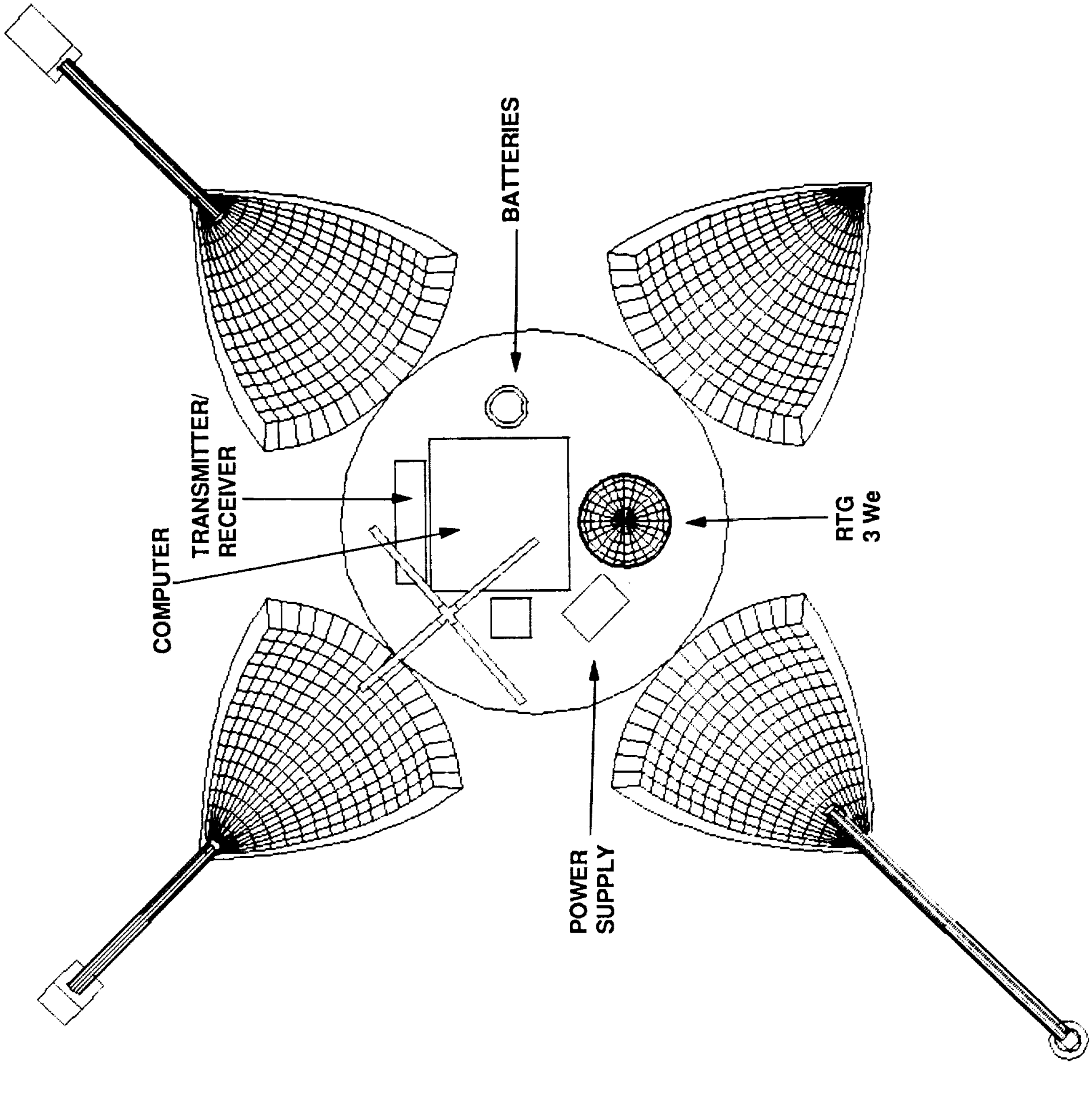
Carlos S. Moreno
Jet Propulsion Laboratory/California Institute of Technology

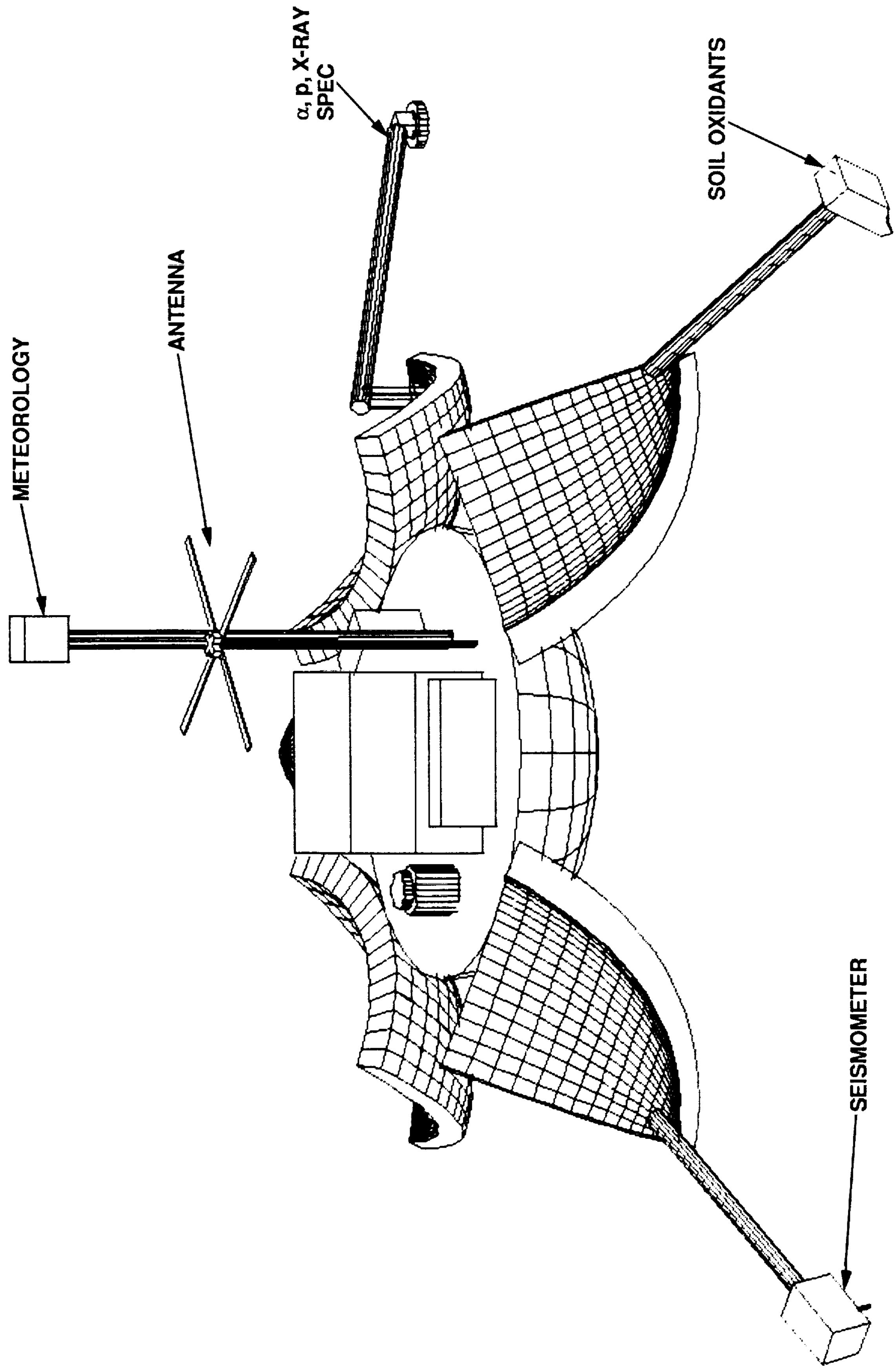
HARD LANDER CONCEPTUAL DESIGN

CARLOS MORENO

JPL


45 cm
50 cm


BATTERIES
TRANSMITTER/
RECEIVER
COMPUTER
RTG
3 We
POWER
SUPPLY



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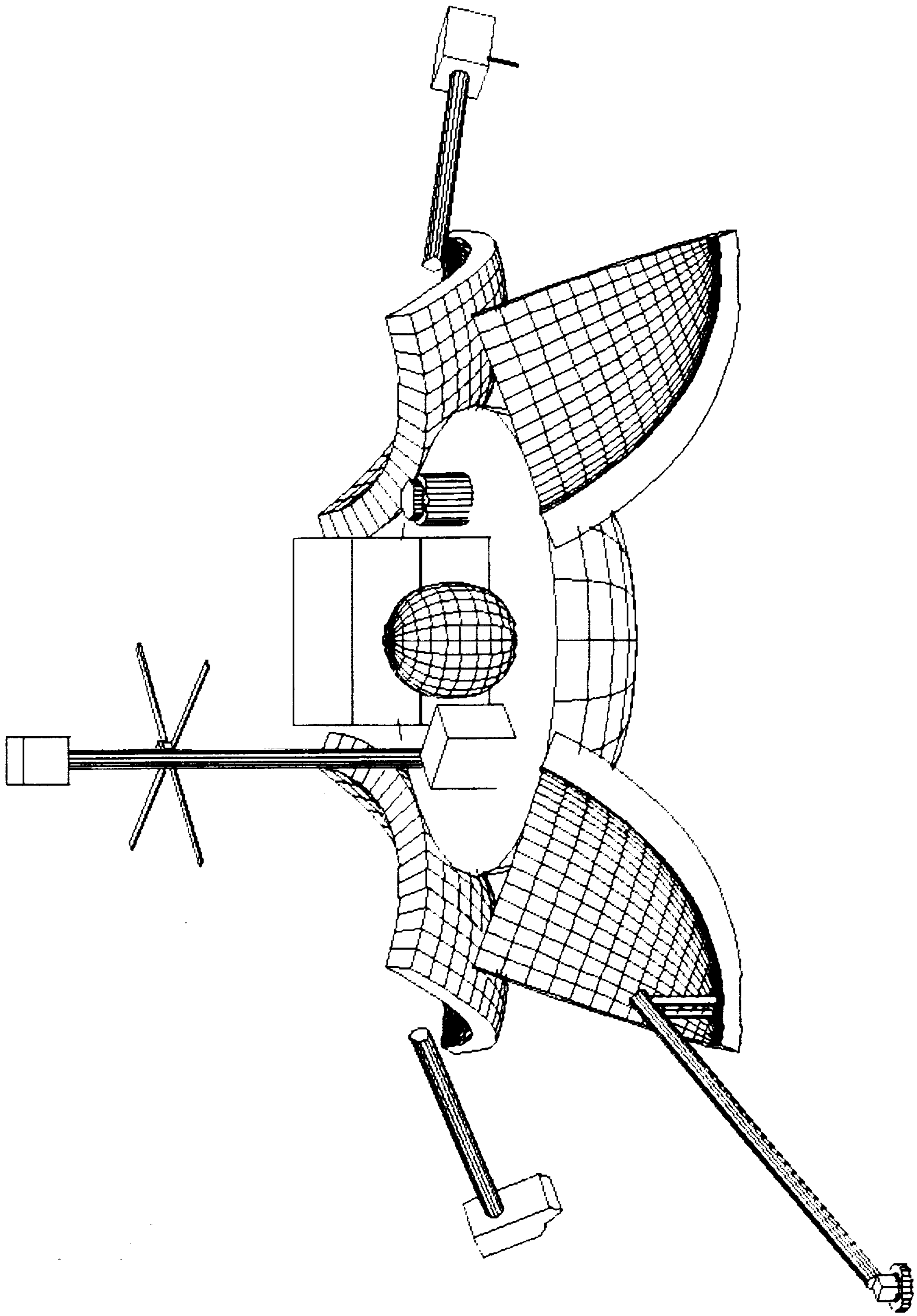
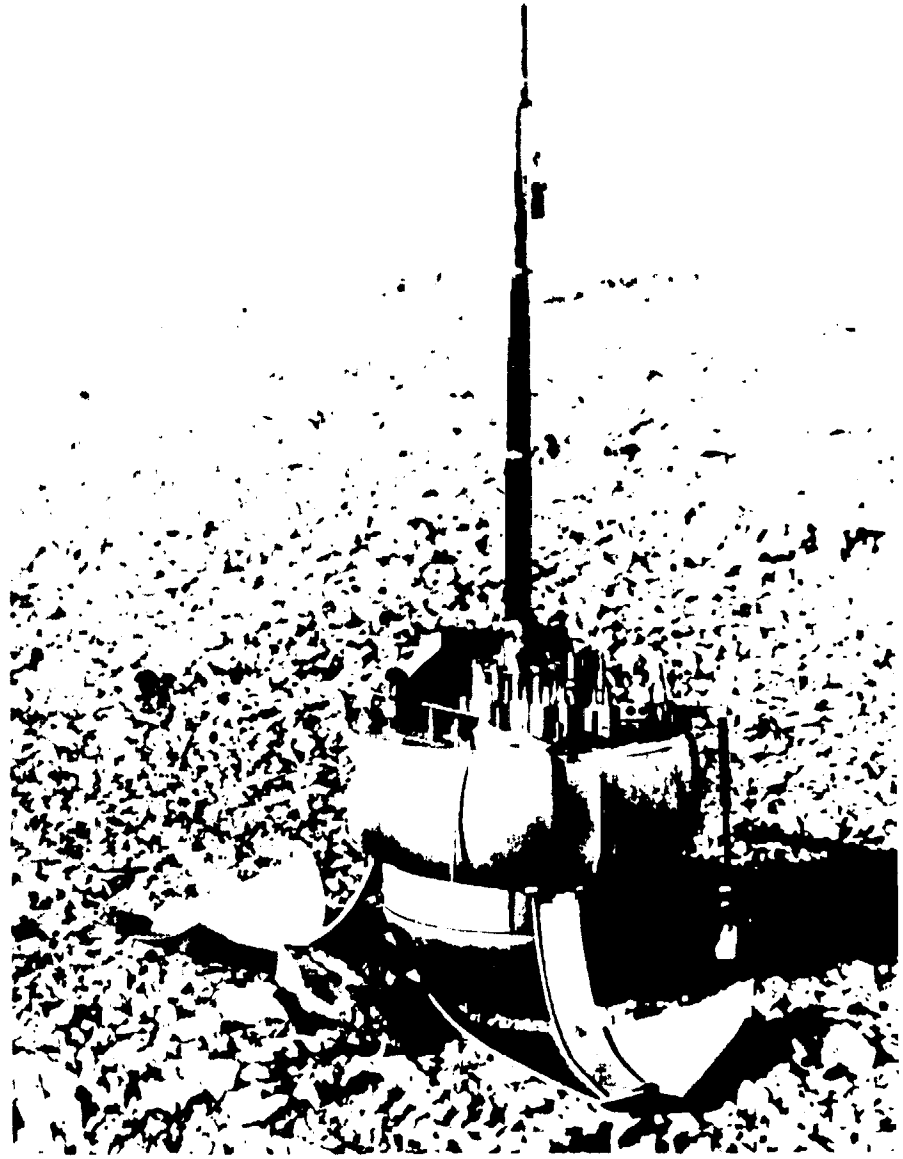


An automated biological laboratory was developed by Philco Aeronutronic Company to study the possibility of a hard-landing entry probe to make simple assays of the Martian environment. One of several studies for Voyager in the mid-1960s, it grouped science instruments that could be programmed for numerous experiments. None of the projects was flown, but they provided understanding of extraterrestrial biology detectors.



Session B, Submittal No. 14

John Garvey
McDonnell Douglas

DELTA LAUNCH VEHICLE STRATEGIES

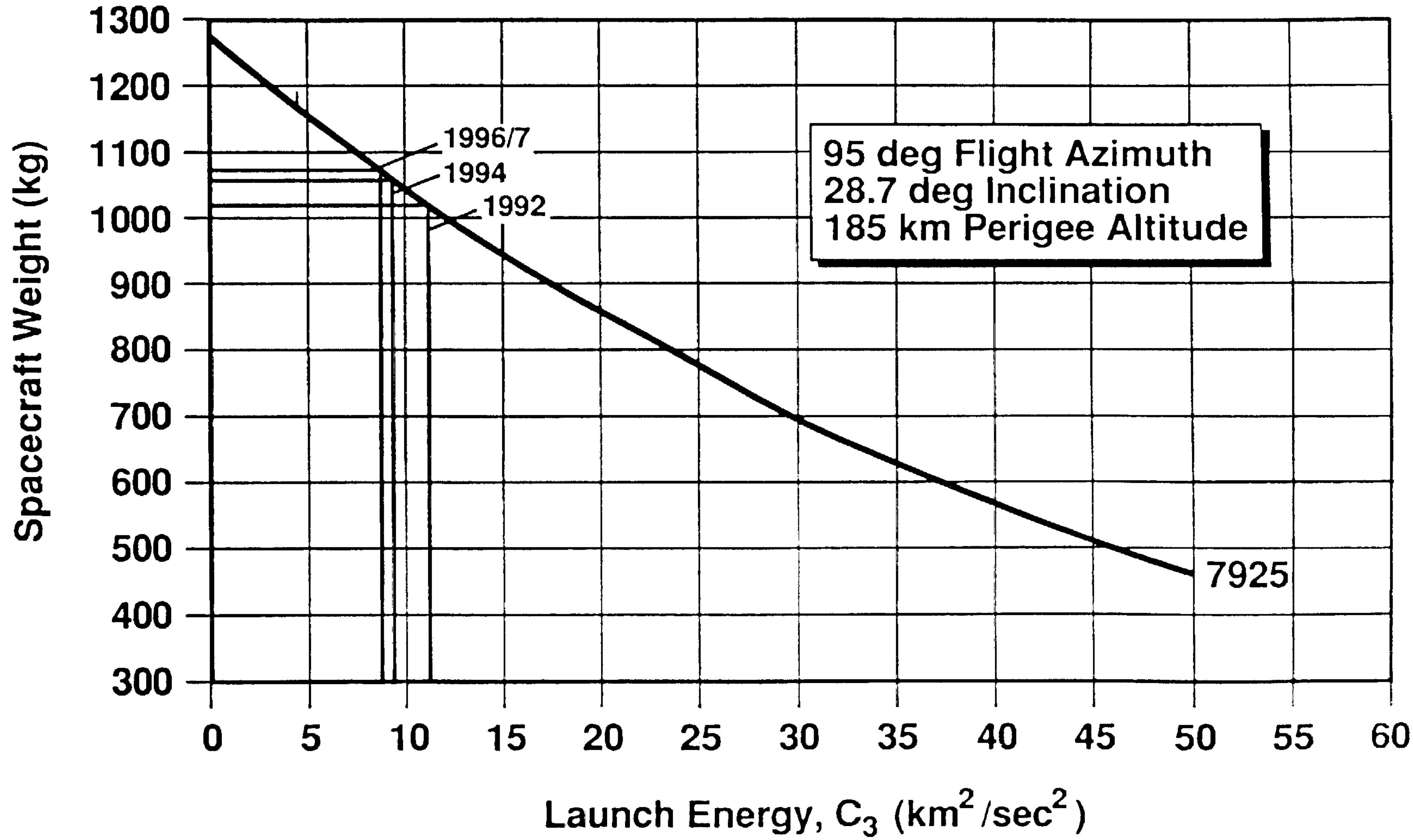
JOHN GARVEY

MDAC

DELTA THREE-STAGE PLANETARY CAPABILITY – ESMC

VJX098.1 T1CV

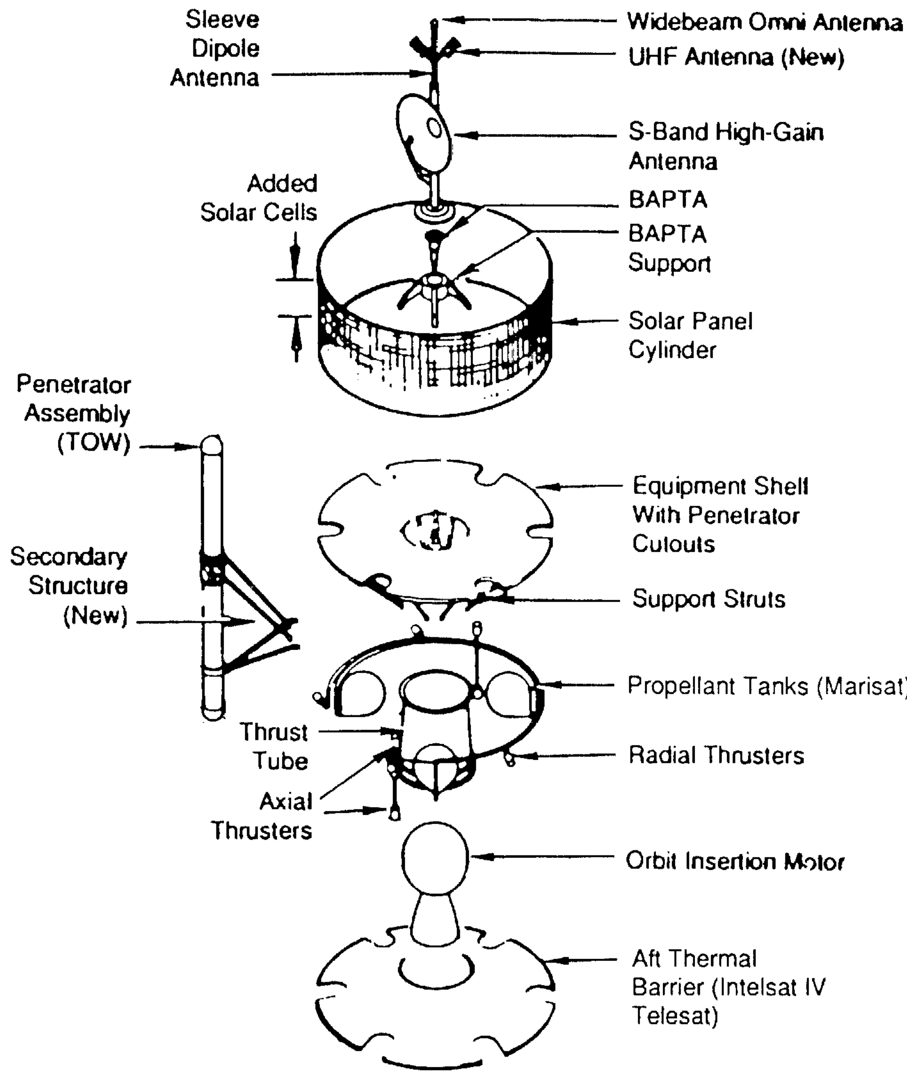
MCDONNELL DOUGLAS



MSP SPACECRAFT BASED ON PIONEER-VENUS BUS

MCDONNELL DOUGLAS

VJX144 T2CA

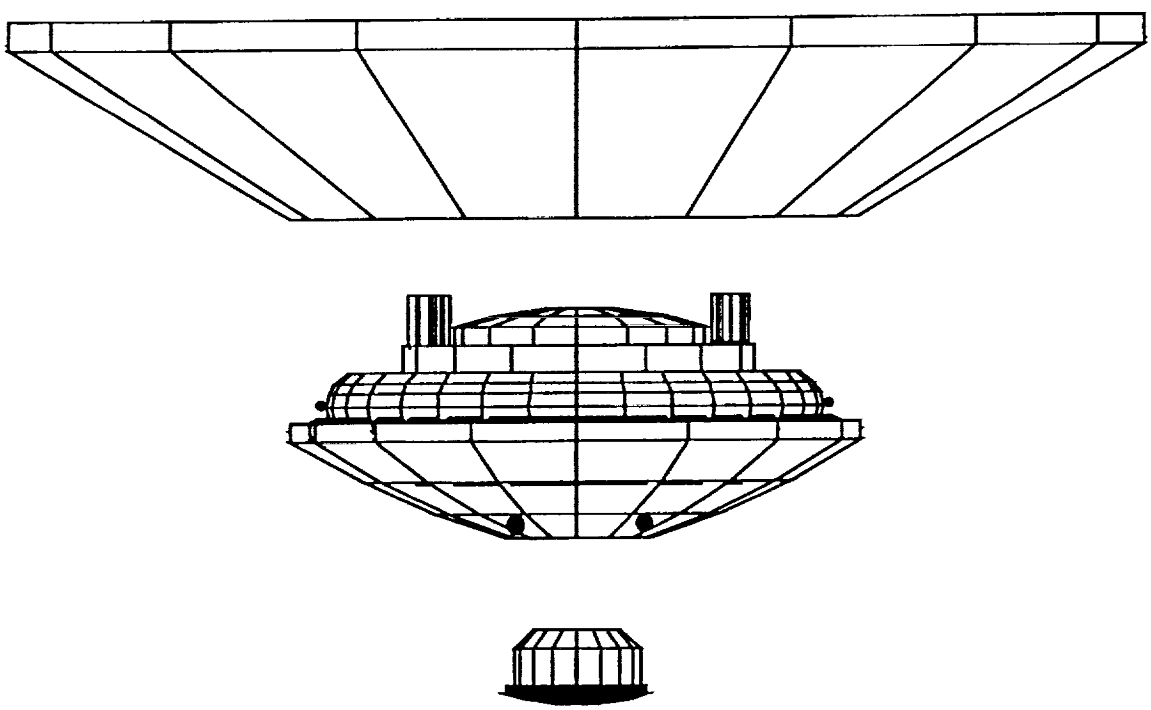


Session B, Submittal No. 15

Stephen Bailey
Johnson Space Center

GNM Mission and System Design Proposal

A Position Paper in Response to the Mars Global Network Workshop, Feb 6-7, 1990



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