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MANNED MARS MISSION VEHICLE DESIGN REQUIREMENTS FOR AEROCAPTURE

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ABSTRACT

The goal of this study was to define vehicle design requirements of a reusable system for manned Mars missions which employ aerocapturing techniques to obtain desired orbital velocities. Requirements for vehicle L/D and ballistic coefficient are determined for expected aerocapture velocities. This paper presents conclusions concerning g-loads environment and TPS requirements for a vehicle that aerocaptures at Mars and Earth. Although the goal of a reusable system (based on current state-of-art technologies) was not obtained, the viability of aerocapture at Mars and Earth was established.

INTRODUCTION

The deceleration of a vehicle from hyperbolic approach velocities to orbital velocity at Mars and Earth can be accomplished by propulsive braking or atmospheric braking (aerocapture). Many authors have shown that aerocapture is more advantageous than propulsive braking in terms of initial departure mass in low-Earth-orbit (LEO). Therefore, to take advantage of aerocapture at Mars and Earth for a manned Mars mission, vehicle design requirements must be defined in terms of external configuration (L/D), size and mass (m/CDA), entry velocity, aerodynamic heating, and g-loads. The goal of the aerocapture analysis was to define vehicle design requirements for a reusable aerocapture system.

MARS AEROCAPTURE

Trajectory analyses of Earth to Mars transfers for arrival dates from 1999 to 2028 have determined the entry velocity requirement to be approximately 17,700 ft/sec to 30,000 ft/sec. This velocity range corresponds to two classes of missions: conjunction class (<20,500 ft/sec) and opposition class (>20,500 ft/sec).

In order to minimize the scope of the entry trajectory analysis, the analysis of external configuration and mass requirements made use of recent and previous Mars mission studies. Raked-off elliptical cone configurations provide a range of L/D's which were assumed to be adequate for aerocapture. Previous Mars mission studies provided estimates of vehicle mass. With these estimates, an aerocapture analysis was conducted with a modified version of the guidance logic from reference 1. The aerocapture vehicles were assumed to be trimmable within $\pm$ 4.0 degrees of the desired angle-of-attack.

The aerocapture guidance was required to achieve the target apoapsis altitude in the presence of all combinations of the following system dispersions: (1) Flight path angle dispersion of $\pm$ 0.30 degree; (2) Angle of attack dispersion of $\pm$ 4.00; and (3) Mars atmosphere density models from reference 2. A minimum altitude constraint of 100,000 feet at Mars was utilized.

An aerocapture is a guided deceleration through an entry corridor in a planet's atmosphere to achieve a desired orbital velocity. The entry corridor is defined by those trajectories which have flight path angles steep enough to avoid skipping out of the atmosphere (remaining at hyperbolic velocity) and shallow enough to achieve a desired apoapsis while maintaining desired g-load and aerodynamic heating levels. The vehicle L/D is the parameter which controls the width of the entry corridor for a vehicle using lift vector modulation for control. Figure 1 shows the required vehicle L/D to meet the aerocapture velocity requirements at Mars. An L/D of 0.6 is required to satisfy the complete aerocapture velocity range requirement. Within the aerocapture corridor the minimum altitude of a trajectory is important for control of aerodynamic heating, g-loads and other considerations such as obstacle avoidance. For a specified guidance logic, the vehicle ballistic coefficient, m/CDA, is the primary driver of the minimum altitude of an aerocapture trajectory (Figure 2). The aerocapture analysis demonstrated that a ballistic coefficient greater than 100 lbm/sq ft would violate the minimum altitude constraint at Mars (Figure 3). Therefore, the vehicle design requirement for external configuration, size and mass is an L/D of 0.6 with a ballistic coefficient less than 100 lbm/sq ft. The effect of these conclusions on the stagnation heat flux and g-load environments must also be studied to determine thermal protection system requirements and crew environment.

Figure 4 presents the reference stagnation heating rate for a one foot radius sphere as function of ballistic coefficient and entry

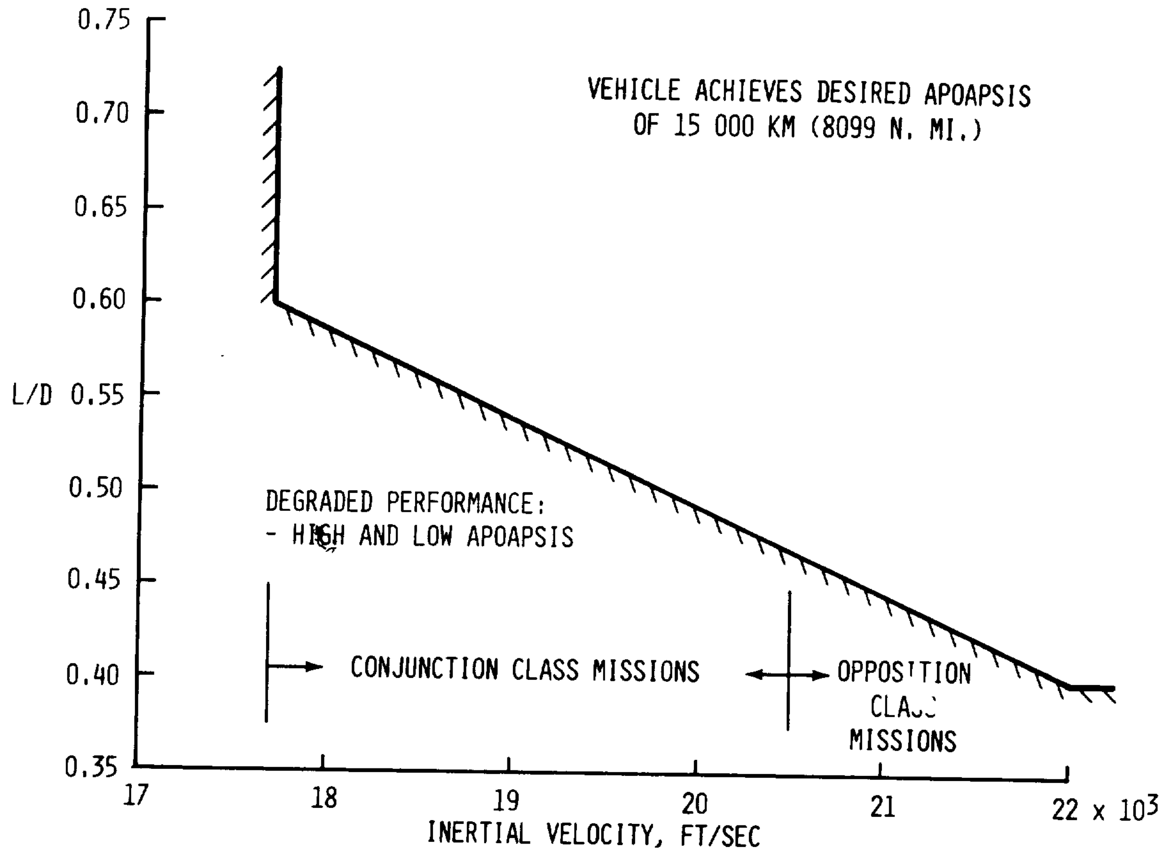

Figure 1.- Mars aerocapture L/D requirements.

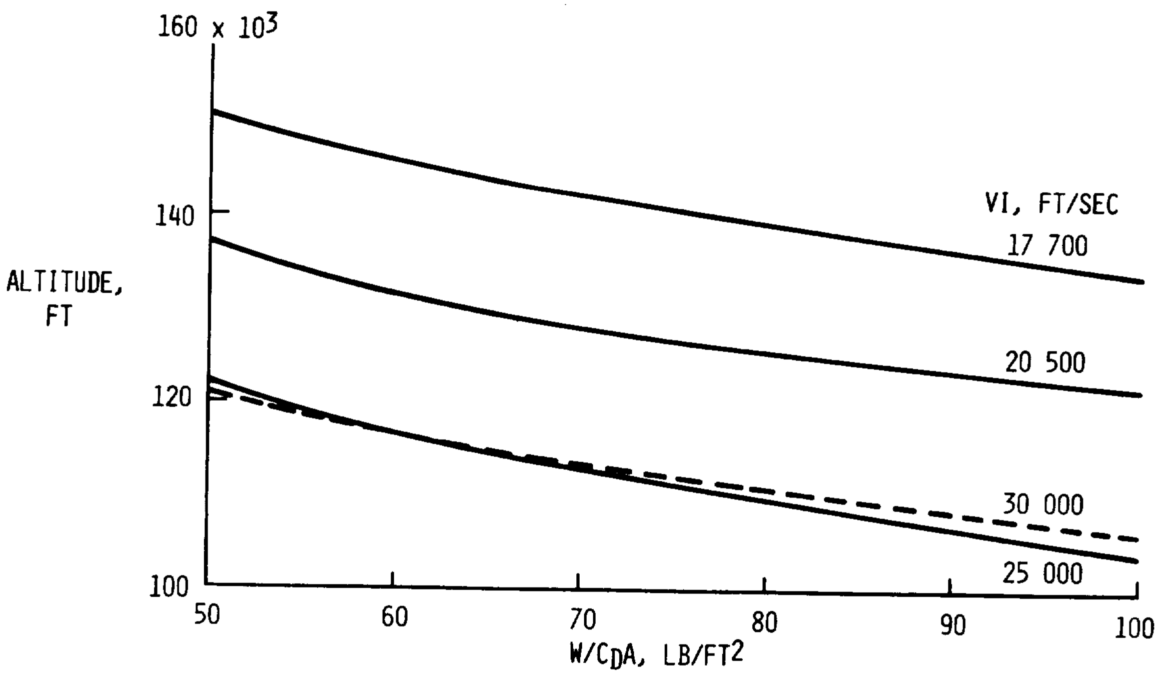

Figure 2.- Minimum altitude during aerobraking at Mars.

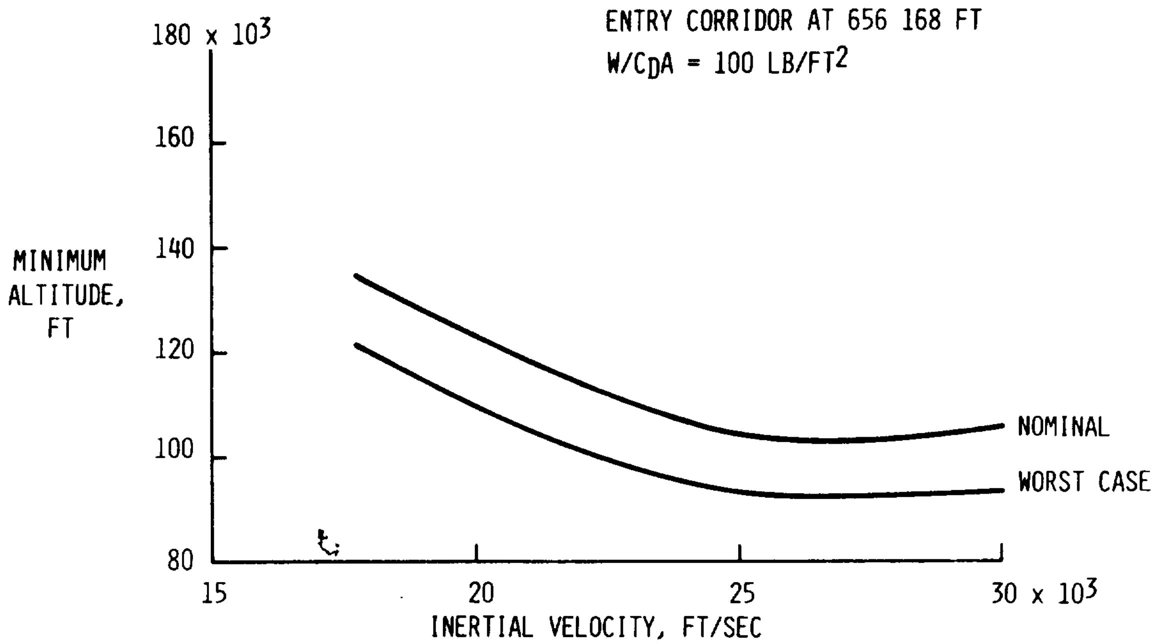

Figure 3.- Minimum altitudes for Mars aerocapture maneuver.

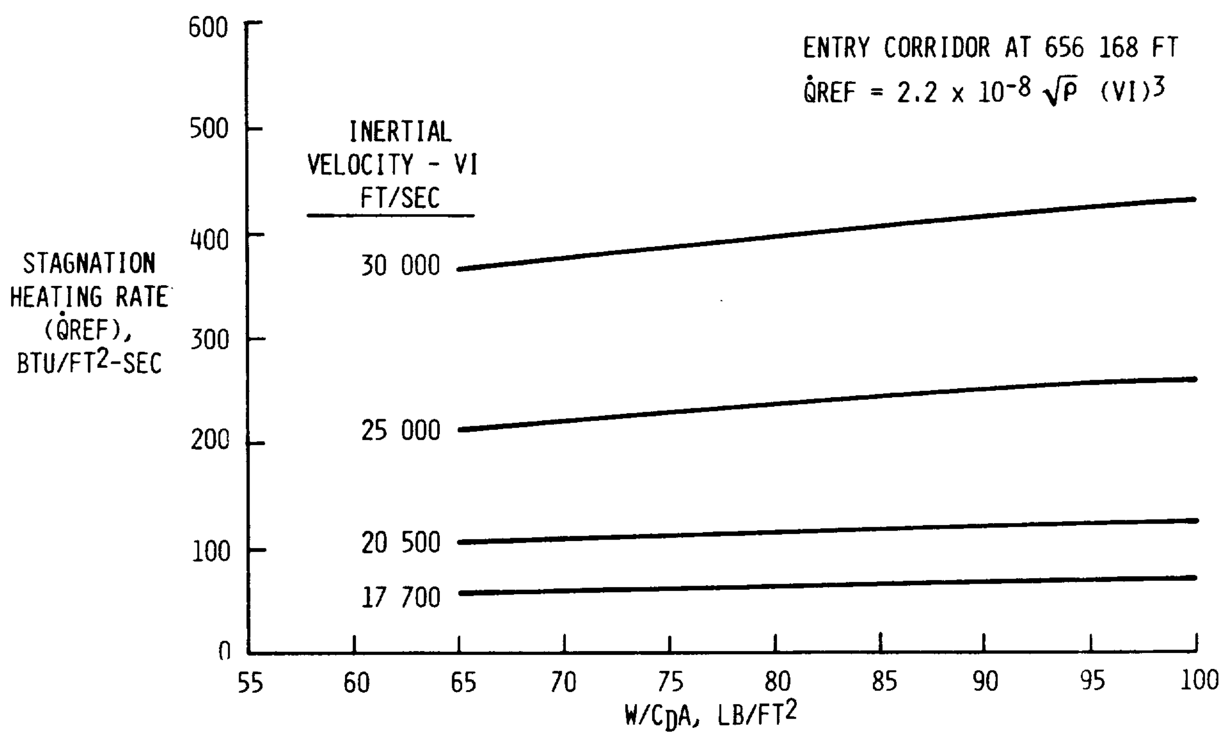

Figure 4.- Mars stagnation heating rates versus ballistic number.

velocity for aerocapture at Mars. When these reference heating rates are assessed for an 85 foot diameter aerobrake, the conclusion can be drawn that an ablative or advanced state-of-the-art TPS is required for opposition class missions and may be required for conjunction class missions.

Figures 5 and 6 present the expected g-load for conjunction and opposition class missions, respectively, within the acceptable Mars entry corridor. The expected g-loads for conjunction class missions appear to be acceptable, while the g-loads for opposition class missions approach intuitively unacceptable values. However, life scientists will have to identify acceptable g-load requirements.

The most severe conditions for the aerocapture maneuver are produced by analyzing a vehicle which has a ballistic coefficient of 100 lbm/sq ft. Tables 1 through 4 present the detailed results of the Mars aerocapture analysis for the complete range of approach velocities which cover conjunction, opposition and Venus swingby missions.

EARTH AEROCAPTURE

Trajectory analyses of Mars to Earth transfers have determined that the maximum expected entry velocity for conjunction class missions is 38,000 ft/sec and that opposition class entry velocities significantly exceed 38,000ft/sec. The aerocapture analysis at Earth was limited to vehicles that satisfied the Mars aerocapture requirements because the same vehicle was assumed to perform the Mars and Earth aerocaptures. The analysis was also limited to conjunction class missions because the conclusions drawn from this conjunction class analysis would only be amplified by the more severe vehicle environment of opposition class missions. Figures 7 and 8 present the g-load and reference stagnation heating rates across the aerocapture corridor for a vehicle which has an L/D of 0.6 and ballistic coefficient of approximately 55 lbm/sq ft (greater than expected ballistic coefficients for actual vehicle designs). From the calculated g-load environment and extrapolations to opposition class entry velocities, it can be concluded that the crew would experience intuitively unacceptable g-loads. Furthermore, when thermal protection system requirements are assessed using the data on Figure 8 for a vehicle with an 85 foot diameter aerobrake, the conclusion can be drawn that an ablative or advanced state-of-the-art TPS is required. Since g-loads and a reusable TPS appear unacceptable, a

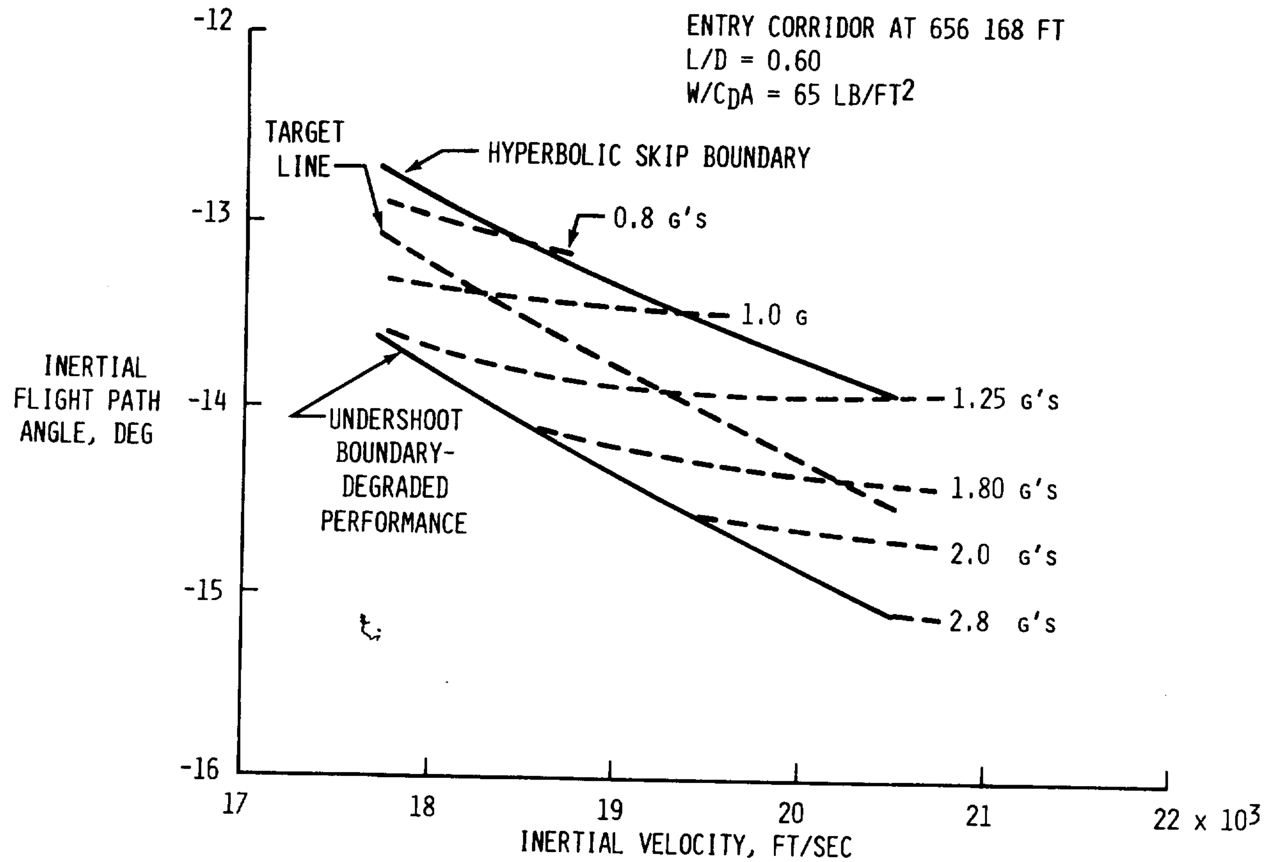

Figure 5.- Mars entry corridor for conjunction class missions.

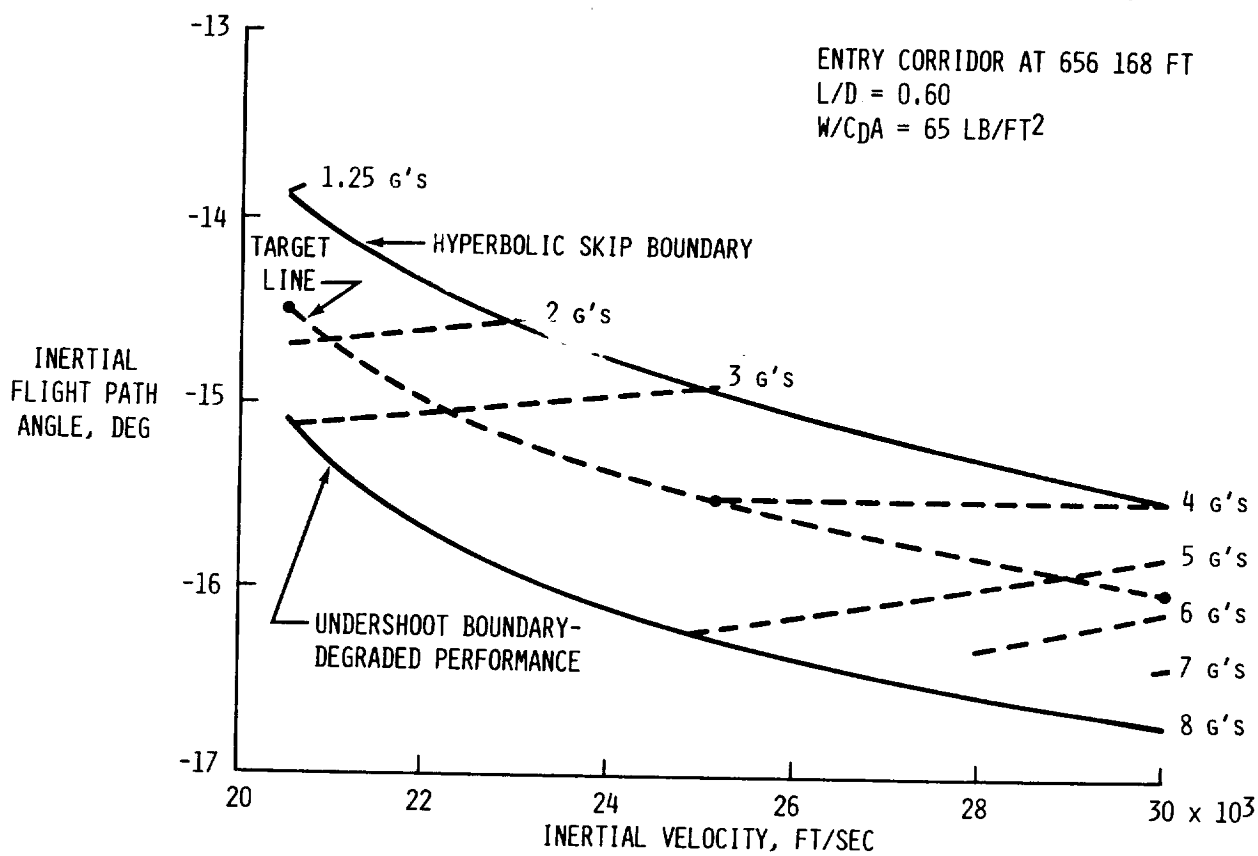

Figure 6.- Mars entry corridor for opposition class missions.

TABLE 1.- MARS AEROCAPTURE PERFORMANCE FOR VI = 17 700 FT/SEC

V_I = 17 700 FT/SEC　　L/D = 0.60　　TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γ_I = -13.22 DEG　656168 FT　W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		134 642	0.86	69.8	8104.0
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	126 222	1.02	73.0	8346.1
	$\Delta\alpha$ = 4.0 DEG	122 485	0.96	78.3	8295.3
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	137 240	0.65	57.0	8121.1
	$\Delta\alpha$ = 4.0 DEG	136 721	0.57	59.8	7925.3
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	138 094	1.27	79.6	8025.7
	$\Delta\alpha$ = 4.0 DEG	137 699	1.10	83.6	8119.7
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	152 147	0.80	63.8	8090.1
	$\Delta\alpha$ = 4.0 DEG	146 353	0.80	69.4	8085.9

TABLE 2.- MARS AEROCAPTURE PERFORMANCE FOR VI = 20 500 FT/SEC

VI = 20 500 FT/SEC   L/D = 0.60   TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γI = -14.5 DEG   656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		120 702	1.69	124.4	8082.9
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	113 368	1.92	129.4	8837.3
	$\Delta\alpha$ = 4.0 DEG	108 225	1.90	139.0	8194.9
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	124 222	1.26	102.9	8393.1
	$\Delta\alpha$ = 4.0 DEG	117 808	1.29	111.1	8344.9
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	130 277	2.10	135.1	8244.4
	$\Delta\alpha$ = 4.0 DEG	125 529	1.95	144.7	7981.7
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	138 194	1.56	114.1	8114.9
	$\Delta\alpha$ = 4.0 DEG	133 196	1.52	123.1	8067.1

TABLE 3.- MARS AEROCAPTURE PERFORMANCE FOR VI = 25 000 FT/SEC

V_I = 25 000 FT/SEC L/D = 0.60 TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γ_I = -15.6 DEG 656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		104 362	3.87	258.7	8087.2
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	98 260	4.23	259.8	8246.9
	$\Delta\alpha$ = 4.0 DEG	93 168	4.26	285.6	8211.5
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	106 735	3.12	218.3	8295.6
	$\Delta\alpha$ = 4.0 DEG	99 508	3.34	243.7	8235.8
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	113 351	4.41	270.9	7997.1
	$\Delta\alpha$ = 4.0 DEG	107 954	4.48	293.6	8089.2
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	120 239	3.68	240.0	8086.5
	$\Delta\alpha$ = 4.0 DEG	114 810	3.65	265.6	8079.2

TABLE 4.- MARS AEROCAPTURE PERFORMANCE FOR VI = 30 000 FT/SEC

V_I = 30 000 FT/SEC   L/D = 0.60   TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γ_I = -16.05 DEG 656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		106 283	5.07	431.1	8097.3
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	98 731	5.86	441.2	8348.4
	$\Delta\alpha$ = 4.0 DEG	93 464	5.90	484.3	8345.2
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	118 689	2.93	338.6	8351.2
	$\Delta\alpha$ = 4.0 DEG	101 632	3.12	359.8	8161.4
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	109 743	6.89	469.2	8315.0
	$\Delta\alpha$ = 4.0 DEG	105 189	6.83	510.3	8135.6
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	122 084	4.78	400.0	8118.6
	$\Delta\alpha$ = 4.0 DEG	113 481	4.80	439.1	8106.2

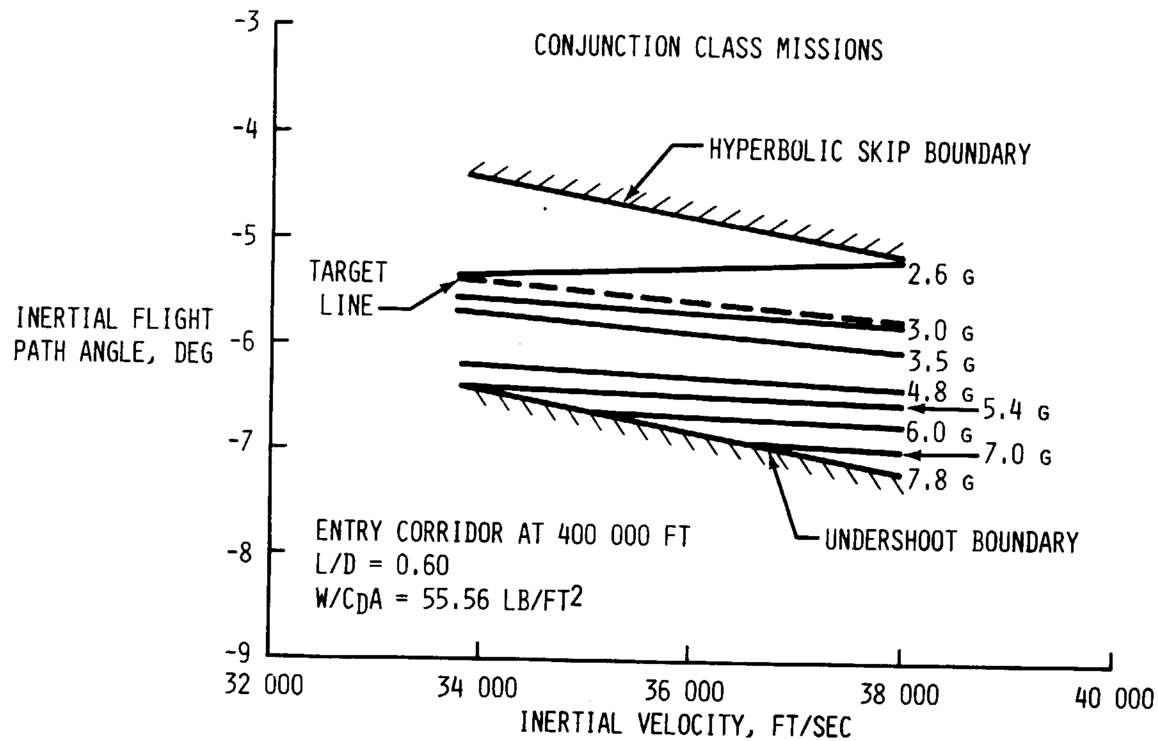

Figure 7.- Earth aerocapture corridor with g-loads.

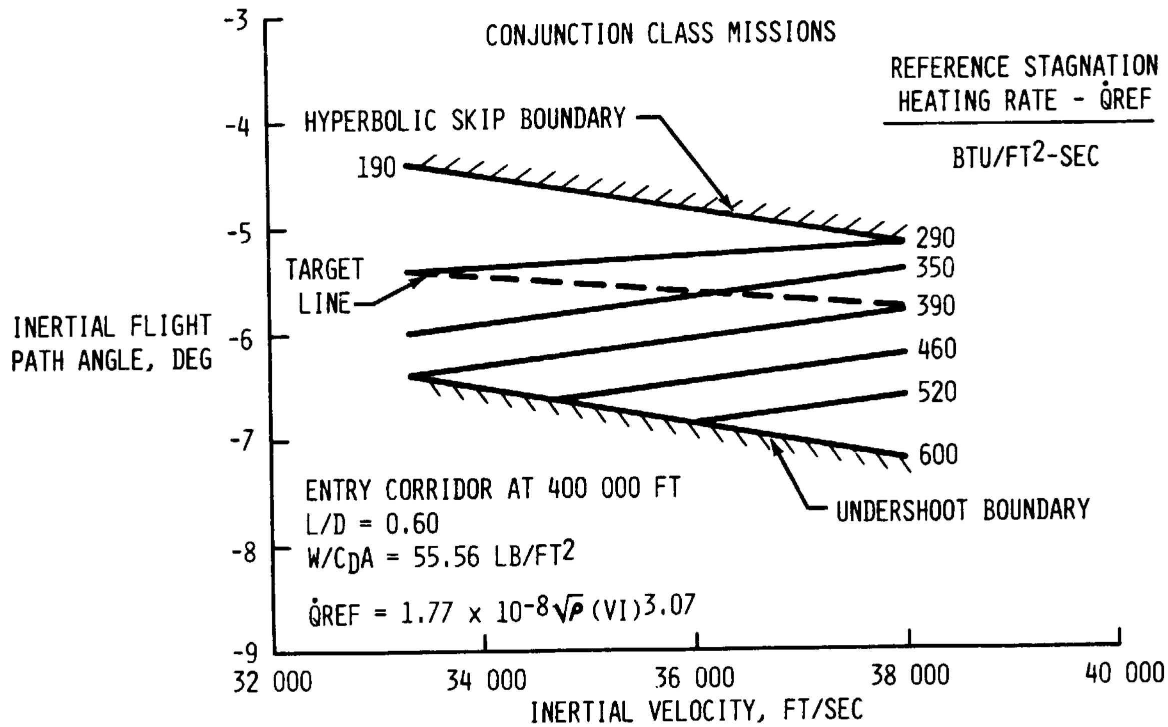

Figure 8.- Earth aerocapture corridor with stagnation heating rates.

propulsive braking system is required to augment the aerocapture system to reduce the aerocapture velocity and, thereby, relieve g-load and aero-heating environments of the aerocapture system.

Another approach to aerocapture at Earth is to aerocapture only part of the Earth return vehicle. A "small" crew and Mars sample module could be designed into the Earth return vehicle which would have a small ballistic coefficient. The advantage of this approach is that the minimum altitude during entry would be increased which would decrease the amount of aerodynamic heating. Figures 9 and 10 present the g-load and reference stagnation heating rates across the aerocapture corridor for a vehicle which has an L/D of 0.6 and ballistic coefficient of 10 lbm/sq ft. Several conclusions can be drawn from these plots. Propulsive braking may still be required for g-load control of the small module. However, the mass of propellent required to perform the braking of the small module would be less than the mass of propellent required to perform the same function for the complete Earth return vehicle. Also, reusable TPS may be acceptable only for conjunction class entry velocities for the small module.

CONCLUSION

The initial goal of the aerocapture analysis was to derive vehicle design requirements for a reusable system that could aerocapture at Mars and Earth. The aerocapture analyses have determined that a vehicle with L/D of 0.6 and ballistic coefficient less than 100 lbm/sq ft can be aerocaptured at Mars and Earth. However, the goal of a reusable system may be unrealistic. The TPS requirements point to non-reusable TPS or an advanced state-of-the-art TPS. Also the expected g-load environment at Earth points to aerocapture systems which have some propulsive braking capability for control of the vehicle g-loads. Since TPS requirements are affected by vehicle ballistic coefficient, reduction in ballistic coefficient can be obtained by studying separate aerocaptures at Mars of the Mars transfer vehicle and staged Mars landers; and at Earth by considering aerocapturing only a small crew/sample module.

RECOMMENDATIONS

The approach to this study was to make use of previous Mars mission studies and recent raked-off cone vehicle studies. The next step will be to take a more parametric approach to vehicle design requirements defini-

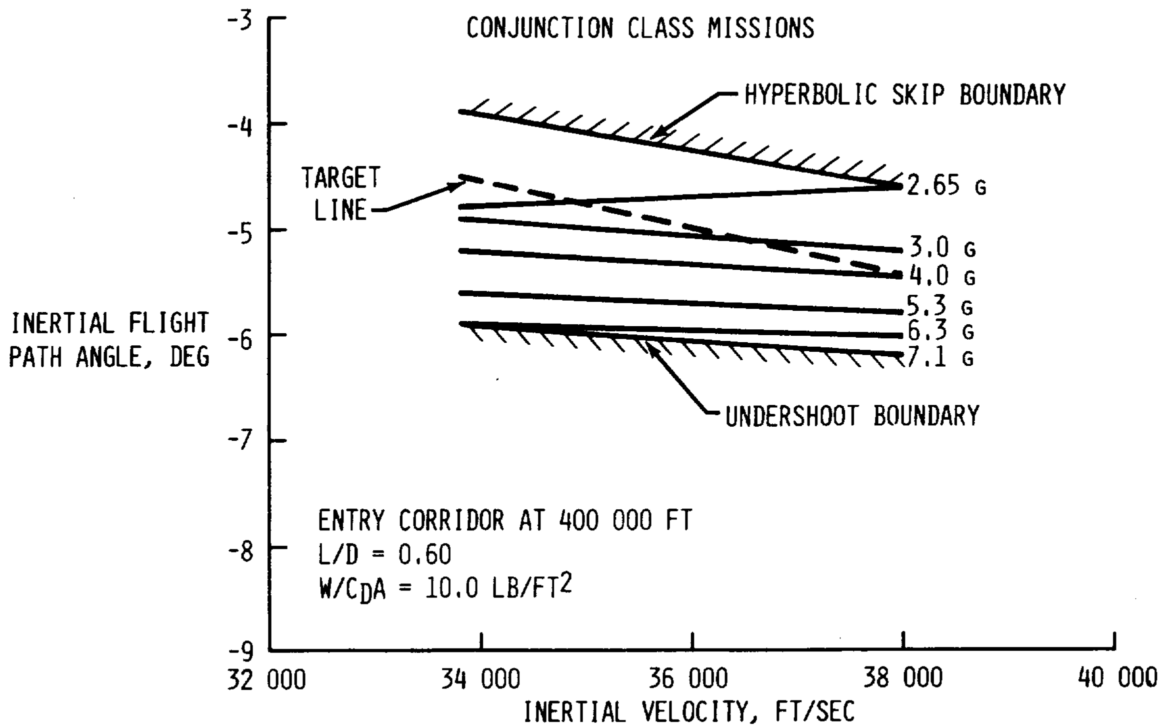

Figure 9.- Earth aerocapture corridor with g-loads.

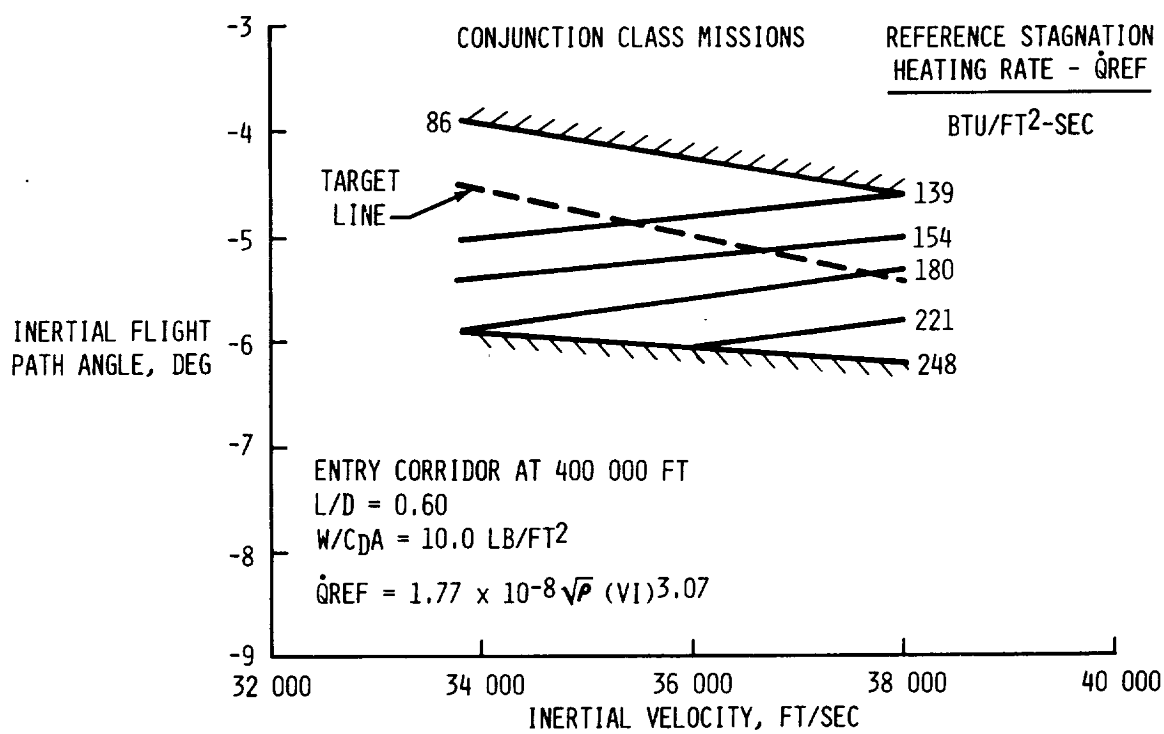

Figure 10.- Earth aerocapture corridor with stagnation heating rates.

tion by assessing a larger range of L/D, ballistic coefficient, and external configuration. Preliminary analyses indicate that an advancement in the state-of-the-art TPS technology is required to make a reusable system possible. Therefore, further TPS studies are recommended. Finally, the allowable crew entry g-load levels require definition for the case of long exposure to zero g or low level g. Physiological tests could be performed during an Apollo type entry from Space Station for a crew made up of personnel who have had long exposure to zero g and personnel who have not.

REFERENCES

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