Table 2 ESCA analysis of TlBrI windows

Sample	Si concentration, atomic %	Tl:Br ratio
Control	0	1:1
Exposed, min. damage[a]	17	5:1
Exposed, max. damage[b]	6	>26:1

[a]Areas of lesser clouding and lesser loss of transmissivity.
[b]Areas of greater clouding and greater loss of transmissivity.

Conclusions

This experiment has shown that pyroelectric detectors made of lithium tantalate or strontium barium niobate are suitable for long-term space use. The LT and SBN detectors survived six years of storage plus almost six years of exposure to space with little or no loss of performance. The detectors made of TGS cannot be recommended because of their apparently short shelf life. Seven of the eight TGS detectors failed to respond after storage and/or flight.

The T1BrI windows experienced noticeable damage. The damage was not uniform and was limited to the detector windows that had direct exposure to space. The inverse relationship between the concentration of silicon on the window surface and the amount of bromine lost suggests that the silicate acted as a shield which lessened the loss of bromine and iodine. Since there was no damage to either the ZnS or the Ge windows, these materials can be recommended as window or lens material for long-term use in space.

This experiment shows that the choice of window and lens materials are of major importance. When used in space, a detector will be part of a system and will be located behind a lens or window of some sort. Damage to the lens or windows will most likely play a larger role in loss of system performance than will damage to the detector material.

References

[1]O'Neal, R. L., and Lightner, E. B., "Long Duration Exposure Facility—A General Overview," *LDEF—69 Months in Space, First Post-Retrieval Symposium*, NASA CP-3134, Part 1, June 1991, pp. 3–48.

Ronald K. Clark
Associate Editor

Mars Aerocapture: Extension and Refinement

P. F. Wercinski*
NASA Ames Research Center, Moffett Field, California 94035
and
J. E. Lyne†
University of Tennessee, Knoxville, Tennessee 37996

Nomenclature

A = vehicle reference area for aerodynamic coefficients, m^2
C_D = drag coefficient
D = drag, N
L = lift, N
L/D = lift over drag
m = vehicle mass, kg
V_e = atmosphere entry velocity, km/s
ΔV = velocity change, km/s
β = ballistic coefficient, kg/m^2

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*Aerospace Engineer, Thermosciences Division. Member AIAA.
†Assistant Professor, Deptartment of Mechanical and Aerospace Engineering. Member AIAA.

Introduction

SPACE missions that use the Martian atmosphere for decelerating both manned and unmanned spacecraft arriving along hyperbolic trajectories have been studied for decades. Previous studies,[1–4] as well as a more recent one by Lyne[5] have shown the effects of varying mission design parameters such as ballistic coefficient (m/C_DA), entry velocity (V_e), lift over drag (L/D), and deceleration limits on the selection of entry vehicle shapes and thermal protection systems. This note is intended to extend and revise previous aerocapture parametric studies for both low-energy (500-km circular) orbits and for high-energy (1-sol period, 500-km periapsis) orbits at Mars.

Specifying the atmosphere entry velocity for a desired vehicle configuration, i.e., L/D and β, will determine the allowable entry corridor for a vehicle aerocapturing at Mars. The entry corridor is defined as the range of atmospheric entry angles determined by the limiting conditions of undershoot and overshoot that will permit capture to a desired parking orbit as shown in Fig. 1. The overshoot trajectories are the shallowest entries that assure atmospheric capture with the vehicle entering in a lift-down attitude. Undershoot trajectories are flown in a lift-up attitude and are constrained either by g-loading or peak heating rates. The difference between the overshoot and undershoot trajectory entry flight path angles defines the entry corridor width.

Discussion

Recent manned Mars mission design studies[6] have proposed conjunction-class interplanetary trajectory missions that would

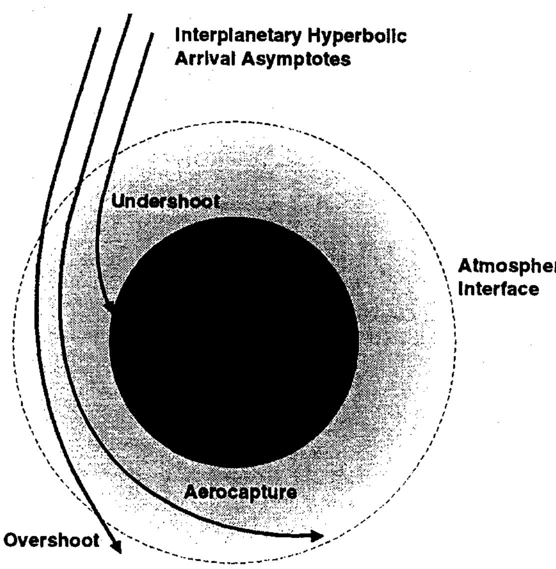

Fig. 1 Aerocapture at Mars.

arrive at Mars in the years 2010–2014. The spacecraft will arrive at Mars with entry velocities ranging from 6–8 km/s and will capture to a high-energy parking orbit with a period of 1 sol. Such an orbit offers significant ΔV savings for the trans-Earth injection maneuver (a reduction of 30–50% in ΔV compared to a 500-km circular parking orbit). Even higher energy parking orbits (orbit periods > 1 sol) offer only small additional improved savings in ΔV. Another issue that limits the choice of the parking orbit is orbit nodal regression[7] which is used for establishing in-plane injection maneuvers. However, this problem becomes less critical for conjunction-class missions with surface stay times of about 500 days as compared to short stay-time, opposition-class missions.

The variation of entry corridor width as a function of entry velocity is shown in Fig. 2 for aerocapture to both low- and high-energy orbits having a 5-*g* limit placed on the entry trajectory. Previous studies by Lyne[5] have looked primarily at low-energy orbits and have recommended 5 *g* as the nominal maximum *g*-load for crews subjected to long periods of weightlessness.[8] The 5-*g* trajectory limit line (see Fig. 2) forms a border between entry velocities constrained by the undershoot trajectory in retaining sufficient energy to achieve the desired orbit to the left and are constrained by the 5-*g* limit to the right. The choice of ballistic coefficient of 300 kg/m^2 is representative of current estimates for Mars payload masses and aeroshell diameters. Similar analyses were performed using ballistic coefficients of 200 and 400 kg/m^2; the results differed only slightly from those shown in Fig. 2. However, higher ballistic coefficients ($\beta > 400$ kg/m^2) tend to decrease the corridor width at arrival velocities between 6.5–8.5 km/s because the vehicle decelerates less rapidly and is more difficult to capture.

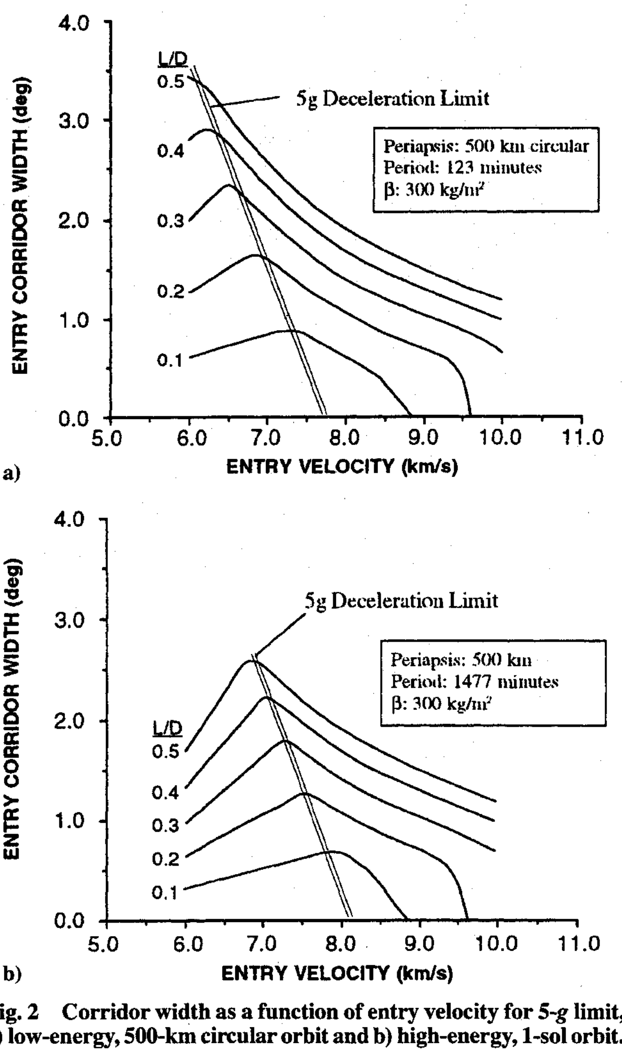

Fig. 2 Corridor width as a function of entry velocity for 5-*g* limit, a) low-energy, 500-km circular orbit and b) high-energy, 1-sol orbit.

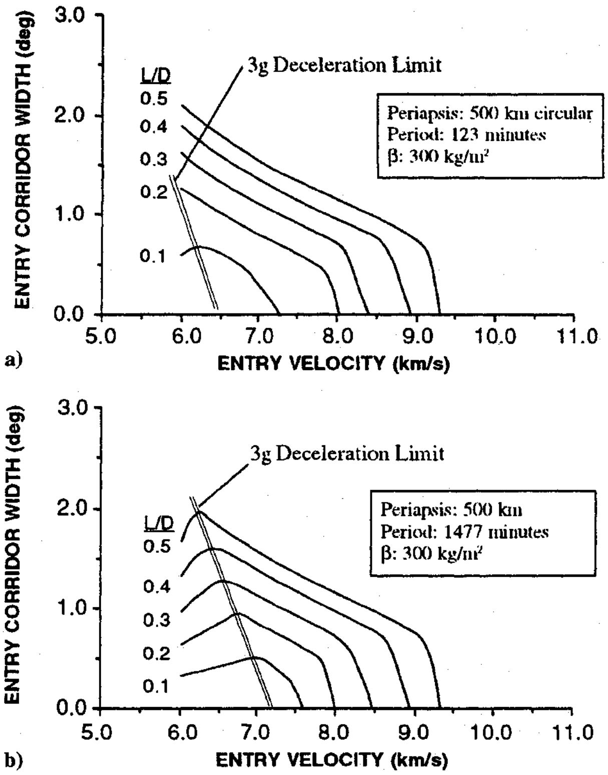

Fig. 3 Corridor width as a function of entry velocity for 3-*g* limit, a) low-energy, 500-km circular orbit and b) high-energy, 1-sol orbit.

The significantly lower entry corridor widths for aerocapture to a high-energy orbit is apparent at the low entry velocities of less than 7 km/s. This is due to the relatively small change in orbital kinetic energy that occurs for low entry velocity vehicles capturing to a high-energy orbit and thus requiring higher *L/D* to provide enough lifting force to establish a given-entry corridor. In addition, a 1-deg entry corridor width (generally used as a minimum value[5,8,9]) requires an *L/D* of about 0.17 for a 500-km circular orbit for entry velocities from 6–8 km/s (Fig. 2a). For the 1-sol parking orbit, the minimum *L/D* increases to almost 0.3 (Fig. 2b).

Similar analyses were performed using deceleration limits of 3 *g* and 7 *g* for $\beta = 300$ kg/m^2 and are presented in Figs. 3 and 4, respectively. for the 3-*g* case, the entry corridor is limited by the deceleration constraint rather than the parking orbit energy over the range of entry velocities, resulting in a general similarity between Figs. 3a and 3b. For the 3-*g* case, a minimum *L/D* of 0.3 is necessary to achieve a 1-deg entry corridor. When a 7-*g* limit is used (Fig. 4), the parking orbit energy constrains much of the 6–8 km/s entry velocity range for the *L/D*s shown. For a 500-km circular orbit, the minimum *L/D* is about 0.15 for the 6–8 km/s range (Fig. 4a) and the minimum *L/D* increases to about 0.3 for the 1-sol orbit (Fig. 4b). Once again, only small variations in the required *L/D* with entry velocity were seen for $\beta = 200$ and 400 kg/m^2.

Conclusions

Aerocapture to high-energy parking orbits has a large influence on the minimum *L/D* that is required to achieve a 1-deg entry corridor width. At the lower entry velocities characteristic of conjunction-class missions to Mars, higher energy parking orbits necessitate vehicles with increased *L/D*. The high-resolution figures from the present analysis clearly show that the entry velocity at the *g*-load trajectory limit line corresponds to the maximum entry corridor width at a given *L/D*. It is also evident that low entry velocities, $V_e \leq 6.5$ km/s, require relatively high *L/D* due to smaller changes in orbital kinetic energy to achieve a desired entry corridor width. therefore entry veloci-

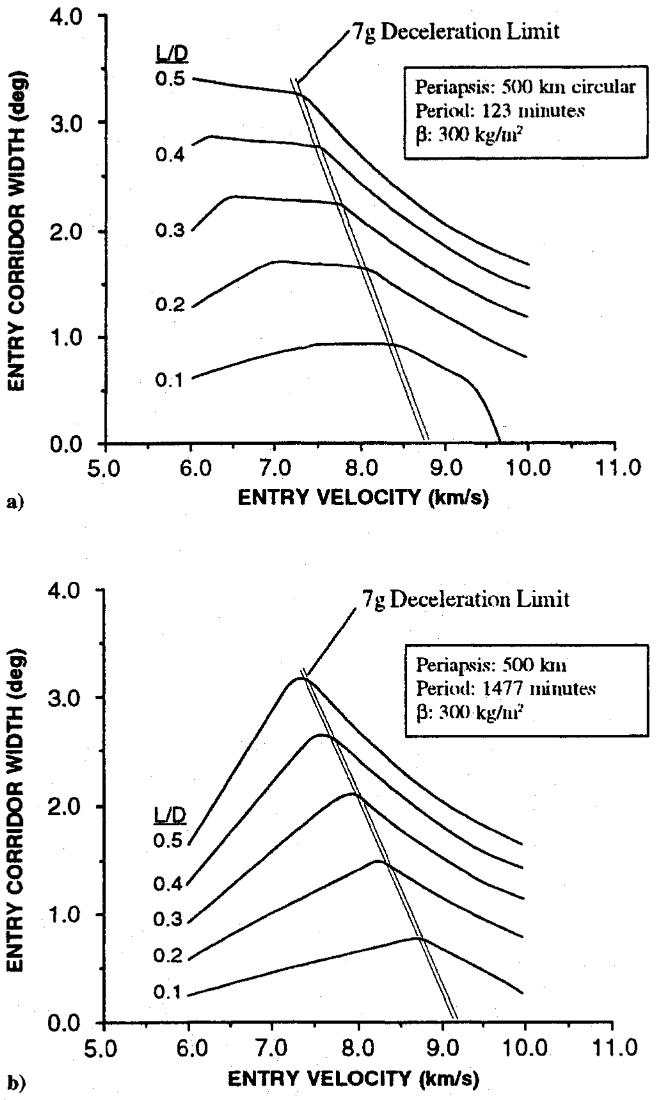

Fig. 4 Corridor width as a function of entry velocity for 7-*g* limit, a) low-energy, 500-km circular orbit and b) high-energy, 1-sol orbit.

ties between 6.5–8.0 km/s are more attractive for minimizing the vehicle *L/D*. For conjunction-class missions in general, an aerocapture vehicle with an *L/D* of 0.22–0.28 will achieve high-energy orbits for entry velocities between 6.5–8.0 km/s.

References

[1] Chapman, D. R., "An Analysis of the Corridor and Guidance Requirements for Supercircular Entry Into Planetary Atmospheres," NASA TR R-55, 1960.

[2] Cruz, M. I., "The Aerocapture Vehicle Mission Design Concept—Aerodynamically Controlled Capture of Payload into Mars Orbit," AIAA Paper 79-0893, May 1979.

[3] Tauber, M. E., Bowles, J. V., and Yang, L., "The Use of Atmospheric Braking During Mars Missions," *Journal of Spacecraft and Rockets,* Vol. 27, No. 5, 1990, pp. 514–521.

[4] Braun, R. D., Powell, R. W., and Hartung, L. C., "Effect of Interplanetary Trajectory Options on a Manned Mars Aerobrake Configuration," NASA TP-3019, Aug. 1990.

[5] Lyne, J. E., "The Effect of Parking Orbit Selection on Manned Aerocapture at Mars," AAS/AIAA Paper 92-103, Feb. 1992.

[6] Duke, M. B., Keaton, P., Weaver, D., Roberts, B., Briggs, G., and Huber, W., "Mission Objectives and Comparison of Strategies for Mars Exploration," AIAA Paper 93-0956, Feb. 1993.

[7] Desai, P. N., and Braun, R. D., "Mars Parking Orbit Selection," *The Journal of Astronautical Sciences,* Vol. 39, No. 4, 1991, pp. 447–467.

[8] Lyne, J. E., "Physiologically Constrained Aerocapture for Manned Mars Missions," NASA TM-103954, Aug. 1992.

[9] Tauber, M., Chargin, M., Henline, W., Chiu, A., Yang, L., Hamm, K. R., Jr., and Miura, H., "Aerobrake Design Studies for Manned Mars Missions," AIAA Paper 91-1344, June 1991.

Wing Effects on Missile Asymmetric Vortex Behavior

R. M. Howard,* M. H. Lung,† J. J. Viniotis,‡
D. A. Johnson,§ and J. A. Pinaire‡
Naval Postgraduate School,
Monterey, California 93943

Introduction

THE introduction of a vertical launch capability for ships carrying surface-to-air missiles represents a major advancement in weapon system technology. But a characteristic of vertically launched missiles is their inability to point at the target intercept point prior to launch. Because the missile cannot rely on the fire-control radar beam to guide it into its initial intercept trajectory, it must be maneuvered into this flight path. The requirement for certain trajectories can place high angle-of-attack demands of up to 50 deg on the missile.[1]

The existence of an induced side force on slender bodies caused by the formation of asymmetric vortices at high angles of attack has been characterized by numerous investigators.[2–4] Unpredictable side forces may be generated under certain flow conditions for the vertically launched missile, posing a potential threat to flight stability. In 1987 a flight test was performed on the Vertical Launch Anti-Submarine Rocket (ASROC) (VLA) standoff weapon, requiring a hard pushover maneuver to a 70-deg angle of attack. During the maneuver, control of the missile was lost because of yawing moments 150% greater than those measured in wind-tunnel tests. A short time before departure, the nose cap tip became unseated; about 0.25 seconds later, the tip reseated itself, but the yawing moment continued, and control could not be regained. The incident was attributed to asymmetric vortex shedding.[5]

Although many studies have been performed considering slender bodies at high angles of attack, much less work has been done treating the influence of realistic wing planforms on

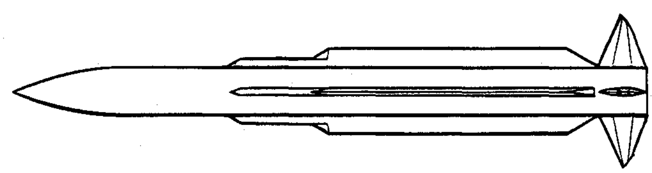

Fig. 1 Vertical launch surface-to-air missile model.

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*Associate Professor, Department of Aeronautics and Astronautics. Senior Member AIAA.

†Graduate Student, Weapons Engineering; Lieutenant, Republic of China Taiwan Navy.

‡Graduate Student, Weapons Engineering; Lieutenant, U.S. Navy.

§Graduate Student, Aeronautical Engineering; Captain, U.S. Army.