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EVALUATION OF AERODYNAMIC AND PROPULSIVE TERMINAL
PHASE SYSTEMS FOR AN UNMANNED MARS SOFT LANDER

by
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Summary

The terminal phase of an unmanned Mars soft lander is defined as that portion of the descent trajectory bridging the gap between the high speed entry trajectory and the very low speed soft landing. This paper presents the results of a parametric analysis comparing the performance and capability of several candidate deceleration systems considered for use during the terminal phase. System comparison is made on the basis of total decelerator system weight requirements and system capability to cope with the mission uncertainties. The mission mode is entry from orbit.

Two general types of terminal phase decelerator systems are analyzed: aerodynamic and all-retro systems. The aerodynamic decelerators considered include both subsonic type parachutes and (supersonic) ballutes. Subsonic type parachutes are limited to a maximum deployment Mach No. of 1.6. Supersonic ballutes are assumed deployed at Mach Nos. from 3.0 to 5.0. Both groups use a propulsive retro vernier system for final deceleration and landing. The all-retro system analysis assumes a rocket propulsion system with two phases - initial braking followed by a vertical descent.

The terminal phase initial conditions are derived from entry trajectories starting at 800,000 ft with a velocity of 4.5 km/sec and flight path angles between -13 and -20 deg. The most critical of the 10 VM atmospheres are used. System characteristics and size variations are reflected in an entry ballistic coefficient range of 0.20 sl/ft$^2$ to 0.40 sl/ft$^2$. A terrain height of 6000 ft above mean planet surface is assumed. Deceleration initiation based on altitude and altitude versus a component of velocity are compared, where applicable, for a given system. The most advantageous result is used in comparison with other systems.

These results show good agreement with the more detailed system studies, indicating low sensitivity to the simplifying analytical assumptions used. The aerodynamic decelerators are capable of handling uncertainties beyond those assumed; the all-retro system requires additional impulse to do this. The all-retro system is more sensitive to parameter assumptions. Aeroshell staging may be accomplished without design problems or effect on system optimization for the parachute vernier; this is not true for the ballute or all-retro systems. The parachute/vernier system is recommended for terminal phase deceleration for the mission considered in this analysis.

Symbols

- $A$ (ft$^2$) Aerodynamic reference area
- $B$ (sl/ft$^2$) Ballistic coefficient = Mass/$C_D A$
- $C_D$ Aerodynamic drag coefficient
- $F$ (lb) Vernier or retro system thrust
- $g$ (ft/sec$^2$) Acceleration due to gravity
- $h$ (ft) Altitude (above mean surface or terrain, as indicated)
- $I_{sp}$ (sec) Vernier or retro system specific impulse
- $M$ Mach number
- $m$ (slugs) Mass
- $t$ (sec) Time
- $W$ (lb) Weight
- $V$ (ft/sec) Velocity
- $\gamma$ (deg) Flight path angle
- $\Lambda$ Vernier or retro system propellant mass fraction

Subscripts

- $A/S$ Aeroshell
- $B$ Braking phase (retro)
- $C$ Vertical coast (retro- where applicable)
Subscripts (cont)

D Deployment conditions
DEC Aerodynamic phase of decelerator system
DS Total decelerator system (i.e., parachute + vernier system)
E Entry conditions
O Initial conditions of phase (aerodynamic, braking, vertical, etc.)
P Propellant
T Terrain (i.e., \( h_T \) = altitude of terrain above mean surface)
V Vernier phase of aerodecelerator system or vertical phase of all-retro system
♂ Mars
♀ Earth

Mission Description

A soft landing on the Martian surface from an orbiting Spacecraft is accomplished by a deorbit maneuver, a ballistic entry into the atmosphere, and a terminal descent with a final propulsive vernier to the surface. The entry vehicle consists of a Capsule Bus payload (typically including an entry science package, surface laboratory, terminal phase decelerator system, and structure) protected by a high drag aeroshell. The aeroshell assumed in this study is a 70 deg half-angle cone, 19 ft in diameter. Initial atmospheric deceleration is provided by the aeroshell drag characteristics. Because of the relatively thin Martian atmosphere, this, by itself, is insufficient to reduce entry velocity to the level necessary for a soft landing. Therefore, the entry phase is followed by a terminal phase to further decelerate the payload prior to a soft landing. The aeroshell is assumed released at or shortly after terminal phase initiation. The mission profile from deorbit to landing is shown pictorially in Figure 1.

This paper presents the results of a parametric analysis comparing the performance and capability of several candidate terminal phase decelerator systems.

Candidate Systems

The candidate terminal deceleration systems are classified generally as aerodynamic or all-retro; both types include a retro vernier system for final deceleration and soft landing. Aerodynamic decelerators are broken into two groups, subsonic and supersonic types. The subsonic parachutes are of the type tested in the Planetary Entry Parachute Program (PEPP) by the Martin-Marietta Corp., Denver Division, under contract to NASA/Langley Research Center. They are limited to a maximum Mach Number of 1.6 at deployment in this study. However, recent PEPP tests showed that deployment Mach Numbers of 2.0 and above may be possible. Deployment Mach Numbers of 3.0 and 5.0 were investigated for the supersonic type ballutes. Two stage aerodecelerators incorporating a ballute followed by a subsonic parachute were not considered for analysis herein because of the added complexity and the results of earlier studies which showed no performance gain. The all-retro decelerator assumes a rocket propulsion system analyzed in two phases - initial braking and vertical vernier. Vernier propulsion characteristics are identical with those of the braking phase except for a throttled thrust. Sketches of the terminal phase decelerators are shown in Figure 2. The ballute shown is the tucked-back type.

Atmospheric Models

A parametric analysis is required because of the wide range of Martian atmospheric uncertainty, terminal phase initial conditions, decelerator system characteristics, and general assumptions such as initiation logic, terrain height, etc. The Martian atmosphere is represented by ten models, designated VM-1 thru VM-10. The density-altitude characteristics of these are shown in Fig. 3. Although the decelerator must be capable of performing in all ten models, the VM-7 and VM-8 atmospheres are used to define limiting conditions. VM-7 has the lowest density from approximately 44,000 ft down and therefore results in the highest terminal velocity. VM-8 has the lowest scale height-tropopause altitude combination resulting in the highest velocity entering into the terminal phase region. Therefore, depending upon the altitude of terminal phase initiation, VM-7 and VM-8 will result in the highest relative velocity at initiation. Since the all-retro deceleration requirements in terms of thrust and propellant are primarily a function of the velocity to be taken out by the system, VM-7 and VM-8 will define the maximum all-retro system requirements. For the aerodecelerators, VM-8 fixes the deployment altitude (Mach number limited) because of its low speed of sound and low upper atmosphere scale height. The lower density of VM-7 below 44,000 ft sizes both the parachute and vernier systems.

Entry Trajectory Characteristics

Terminal phase decelerator system deployment conditions are established by entry trajectory characteristics. The entry trajectory characteristics are determined by entry flight path angle, entry velocity, vehicle ballistic coefficient and atmosphere model. Entry conditions
are established by orbit geometry, targeting requirements and deorbit accuracy. The targeting analysis assumes a nominal entry flight path angle associated with each entry velocity. To be conservative, in the face of atmosphere uncertainty, an idealized entry flight path angle versus entry velocity is selected which desensitizes the velocity-flight path angle variations at terminal phase initiation altitudes (10,000 to 20,000 ft) due to entry flight path angle uncertainties and targeting variables. This de-orbit-entry condition logic is used in this analysis. All entry conditions quoted are inertial. Entry conditions are post-grade entries into a rotating atmosphere in the equatorial plane; these trajectories are generally the critical ones. From these considerations and the skipout boundary limitation, a range of flight path angles (i.e., entry velocities of 3.5 and 4.5 Km/sec) are established as shown in Fig. 4. The "shallow" and "steep" entry corridors shown are representative of two degrees of orbit ephemeris uncertainty. The uncertainties inherent in analyses of the entry conditions come from the fact that tracking of a spacecraft in Mars orbit with Earth-based radar has not yet been done. Once the first spacecraft has been placed in orbit and the first lander landed, the error analyses can be performed with considerably greater confidence. On this basis, from Fig. 4, the assumption of entry flight path angles up to -20 degrees (V_E) is used throughout this report for the first mission. After the first mission, when better estimates of the orbit ephemeris can be established and the atmosphere uncertainty is considerably reduced, maximum entry flight path angles of -16 degrees or less appear practical. Thus, maximum flight path angle limits of -16 degrees and -20 degrees are used throughout this analysis and labeled "optimistic" and "conservative", respectively.

The terminal phase systems analysis presented in this report is based upon an entry velocity of 4.5 Km/sec and entry flight path angles from -13 to -20 degrees. These data are directly applicable to other entry velocities as well if the deorbit maneuver strategy is designed to result in a nominal flight path angle which generally parallels the skipout versus entry velocity boundary. This is illustrated in Figure 5 for representative entry trajectories with entry velocities of 3.5 and 4.5 Km/sec. The velocity-altitude profiles merge closely together in the 40,000 to 60,000 foot altitude regime, resulting in nearly identical flight conditions at the terminal phase initiation altitudes. The important factor leading to these results is the positioning of the entry corridor as a function of entry velocity. This strategy is consistent with the entry error analysis results.

A range of entry weights from 3000 to 4800 lbs was used for this study for initial and growth (later) unmanned Mars missions. Corresponding entry ballistic coefficients, including the aerodynamic characteristics of the aeroshell described earlier, are 0.20 si/ft^2 and 0.32 si/ft^2. These values are used extensively in this analysis as representative of light and heavy mission weights. Assuming that the light weight is most likely to be associated with the first mission, the values of E_b = 0.20 and 0.32 si/ft^2 are associated with the γ_E = -16 and -20 deg, respectively. A E_b range of 0.20 si/ft^2 to 0.40 si/ft^2 is used in combination with the total γ_E range discussed above to provide a complete parametric analysis.

Terminal Phase Initial Conditions

Figures 6 and 7 illustrate two forms of terminal phase initial condition data used in the analysis. Figure 6 illustrates altitude and dynamic pressure as a function of entry conditions. Over the range of initiation Mach numbers considered, altitudes at a given Mach number in VM-7 are higher than in VM-8 for all Mach numbers. Dynamic pressure is less than 20 psf for all conditions. Figure 7 presents altitude histories of velocity as a function of entry conditions in VM-7 and VM-8. The terminal phase initiation velocities are seen to be higher in VM-7 at lower altitudes; the altitude at which VM-8 velocities become larger decreases with increasing entry flight path angle (more negative values) and ballistic coefficient. Other data, not illustrated, show that VM-7 flight path angles are greater (more negative) for all altitudes over the range of entry flight path angle and ballistic coefficient considered.

Ground rules and assumptions more pertinent to the individual decelerator systems will be discussed in the appropriate place. To provide a system comparison under comparable conditions, a design terrain height of 6000 ft above mean planet surface is assumed throughout the analysis.

System Evaluation

The candidate terminal phase decelerators are evaluated within the framework outlined above. The systems are compared with respect to performance (i.e., weight requirements) and entry uncertainty capability (i.e., limiting entry flight path angle). Sensitivity to growth is also of importance. Thus performance and capability of the candidate system are compared over the range of E_b discussed above. Also of interest are sensitivity to unknowns, and sensitivity to verification by testing. Because of its parametric nature, the investigation is pursued using simplifying assumptions where applicable; the results are presented in normalized form.

A detailed systems evaluation and comparison is not attempted here. Rather, this analysis provides the basis for a more detailed comparison.
Aerodynamic Decelerator Systems

Decelerator Ballistic Coefficient

Decelerator ballistic coefficient for the aerodynamic portion, $B_{DEC}$, is varied over a range consistent with the aerodecelerator under consideration to provide the parametric data range for analysis.

$$B_{DEC} = \frac{m}{C_D A}$$

where $m$ is the mass of the capsule bus and aerodeceleration system (total wt. on the aerodecelerator). $B_{DEC}$ may also be related to the terminal velocity, $V_T$, by

$$V_T = \sqrt{\frac{2g_f B_{DEC}}{\rho}}$$

allowing the formulation of the relationship between parachute and vernier system, as will be shown.

Vernier System

The purpose of the vernier system is to decelerate the capsule after parachute separation to near zero velocity at the ground. The vernier is assumed ignited on a velocity-altitude trigger. A monopropellant vernier system is used having a propellant mass fraction of 0.5 and a specific impulse of 222 sec. Constant thrust provides deceleration from ignition conditions to zero velocity at zero altitude; several values of thrust to initial weight are investigated. In addition to constant thrust, other idealized assumptions include zero drag effect and no losses for maneuver requirements. The use of a constant mass fraction assumes a variable-sized motor.

The equations of motion used to define general retro system performance are based on ground rules and assumptions as follows:

1. Flat planet
2. Constant gravitational acceleration
3. Constant thrust
4. Constant thrust direction
5. Zero aerodynamic forces
6. Horizontal wind component of 220 fps

Nomenclature and sign convention are identified by the sketch below in addition to the nomenclature presented earlier.

The ground rules, assumptions, and sketch above are applicable to both the vernier used with aerodecelerators and the all-retro decelerator. Using a conventional Newtonian approach and defining

$$\Delta X = X - X_0 \quad \Delta Z = Z - Z_0$$

The vernier system performance is described by

$$\ln \left(1 - \frac{F_t}{m o C_j} \right) = \frac{\Delta X}{C_j \cos \theta} = \frac{(\Delta X + g t)}{C_j \sin \theta} \quad \text{Eq. (1)}$$

$$t = 1 - g \left(\Delta X \tan \theta + \Delta Z \right) \quad \text{Eq. (2)}$$

$$\Delta Z = t \left[ Z_0 - \frac{1}{2} g_f t + C_j \sin \theta \left(1 - \ln \left(1 - \frac{F_t}{m o C_j} \right) \right) + \frac{F_t}{m o C_j} \right] \quad \text{Eq. (3)}$$

The value of $\theta$ is determined by iteration using Equation (3) and varying $\theta$ until the $\Delta Z$ computed is equal to the $\Delta Z$ required by initial conditions.

A somewhat more efficient vernier system would include two phases - coast at low thrust followed by high thrust final braking. With a two-phase vernier, more pitchover would be required at ignition to compensate for lateral wind drift. This adversely affects the communication link geometry.

The vernier system characteristics used in this analysis are shown in Fig. 8. The minimum allowable ignition altitude used in the analysis is 1500 ft; this is to allow altitude for rough terrain avoidance. For the ignition altitudes of interest $h_o \leq 4000$ ft, vernier system weight
fraction varies from approximately 0.12 to 0.18 at optimum \( \frac{W}{W_0} \) values. Note that the weight fraction values shown must be normalized to aero-decelerator deployment or entry weight to be combined with aero-decelerator weight.

Subsonic Aerodecelerator

Effect of Atmosphere Uncertainty. The terminal phase system considered here is a subsonic type parachute (cross, disc-gap-band, ringsail, etc.) coupled with a vernier retro system for final descent and landing. A pyrotechnic mortar is used to eject the chute package from the capsule prior to opening. The Planetary Entry Parachute Program (PEPP) has demonstrated that parachutes of the type considered can be deployed at Mach numbers of 2.0 and higher. A limiting Mach number of 1.6 is used in this analysis. A second criteria used for the parachute system is that it must deliver the Capsule Bus to a (no wind) flight path angle, \( \gamma_p \), of -60 degrees or steeper. This flight condition, coupled with winds, will allow the Terminal Descent and Landing Radar (TD&LR) to lock up. The final constraint put on the parachute-vernier system design is that it must be able to land on a 6000 ft (above mean surface) level plateau over the 3\( \sigma \) range of possible entry flight path angle dispersions. The maximum flight path angles at entry are optimistically -16 degrees and conservatively -20 degrees.

The system comparisons and capabilities are made on the basis of total decelerator (parachute plus vernier) weight fraction of entry weight, i.e.,

\[
\frac{W_{DS}}{W_E} = \frac{W_0}{W_E} \left[ \frac{W_{DEC}}{W_0} + \frac{W_V}{W_0} \left( 1 - \frac{W_{DEC}}{W_0} \right) \right]
\]

where

\[ W_0 = W_E - \frac{W_A}{S} \]

or

\[
\frac{W_{DS}}{W_E} = \frac{W_{DEC}}{W_E} + \frac{W_V}{W_0} \left( 1 - \frac{W_{DEC}}{W_E} - \frac{W_A}{S} \right)
\]

Case 1 is for aeroshell ejected at parachute deployment. Case 2 is for aeroshell ejected at vernier motor ignition. The total parachute system weights (including mortar) used here are based on:

\[
W_{PARACHUTE} = 0.0243 D_0^2 \quad (D_0 = \text{reference dia. in feet})
\]

This equation was derived from the PEPP program parachute weights and other available parachute weight data.

These data, combined with the vernier motor characteristics presented above, result in the idealized decelerator weight fraction shown in Fig. 9. The data are idealized in the sense that it is assumed that the entry vehicle and entry conditions are such that the parachute can be deployed and do its job before reaching vernier ignition altitude. Vertical velocity at vernier ignition is assumed to be 1.25 x terminal velocity (worst case boundary from many trajectory runs). A lateral velocity due to assumed cross-winds of 220 fps was also added to the vernier ignition velocity. The curves are useful to establish the range of parachute sizes which should be considered. The optimum \( B_{DEC} \) is approximately 0.032 sl/ft\(^2\) without the aeroshell and 0.035 to 0.037 sl/ft\(^2\) with the aeroshell.

The weight fraction characteristics are also used to evaluate aeroshell separation. In order to minimize the parachute size required to assure aeroshell separation, staging is delayed until a Mach number of 0.8 is reached in the most severe atmosphere (VM-8). The aeroshell \( C_A \) at that point is approximately 1.15; the \( B_{DEC} \) is 0.037 sl/ft\(^2\). Reference to Figure 9 indicates that this is larger (smaller chute size) than required for optimization. Refining the requirement to assume an aeroshell separation distance of 100 ft in 3.0 seconds for radar non-interference (conservative) results in the \( B_{DEC} \) requirements shown in Table 1. Since these values are still larger than the optimum value of Figure 9 the optimum values may be used for parachute/vernier design and analysis.

Table 1

<table>
<thead>
<tr>
<th>Maximum Parachute Ballistic Coefficient Required for Aeroshell Staging</th>
<th>( B_E ) (sl/ft(^2))</th>
</tr>
</thead>
<tbody>
<tr>
<td>Without Aeroshell</td>
<td>0.0344</td>
</tr>
<tr>
<td>With Aeroshell</td>
<td>0.0395</td>
</tr>
</tbody>
</table>

The Mach number in VM-7 and the velocities in both VM-7 and VM-8 are shown in Figure 10 for deployment conditions corresponding to \( M = 1.6 \) in VM-8. These data are taken directly from entry trajectories and show that the VM-7 Mach numbers and velocities are generally lower than those in VM-8.
Parachute trajectories are calculated at effective ballistic coefficients of .03, .045 and .10 sl/ft² at γ_E = -13, -16, -20 and -24 degrees in VM-7 and VM-8. Deployment in VM-7 and VM-8 is at the altitude corresponding to M = 1.6 at VM-8. The deployment altitude for M = 1.6 in VM-8 and the altitude at γ_F = -60° for B_DEC = 0.030 sl/ft² are shown in Figure 11. In general VM-8 has the greater altitude loss for the low B_E and VM-7 for the high B_E. The exception is B_E = .20 sl/ft² and B_DEC = .03 sl/ft² where VM-8 is critical for γ_E above -18 degrees.

These characteristics reflect, in turn, on the allowable terrain height, or, if the maximum design terrain height is specified, the maximum allowable entry flight path angle. Representative data are shown in Figure 12 for B_DEC = .030 sl/ft². The results are plotted to show the terrain height and decelerator system weight fraction as a function of entry flight path angle and vernier motor thrust level. The maximum landing altitude or terrain height is found by subtracting the altitude loss required for vernier deceleration from the altitude at γ_F = -60°. The vernier ignition vertical velocity is considered equal to the relative velocity of the capsule with parachute at γ_F = -60°. In most cases the VM-7 atmosphere is critical for altitude loss and vernier velocity. Fig. 12 shows h_T ≥ 13,300 ft for B_E = .20 sl/ft², γ_E = -20 deg and h_T ≥ 9200 ft for B_E = .32 sl/ft² and γ_E = -16 deg. It should be understood that the decelerator weight fractions shown in data represented by Fig. 12 are for landing at the maximum landing altitude or terrain height. At higher B_E, the V_O (vernier) is lower at γ_F = -60° deg. Thus, the lower B_E and F/W_0_d require the higher decelerator weight fractions. The low F/W_0_d also have less γ_E capability because of the higher V_O (vernier) at higher γ_E. Over the range of B_DEC investigated, .030 to .10 sl/ft², B_DEC = .030 sl/ft² shows the lowest weight fractions and the highest γ_E capability. Figure 13 shows terrain height and decelerator weight fractions for vernier ignition at γ_F = -80°. This figure shows two disadvantages of γ_F = -80° compared to γ_F = -60°. The maximum landing-altitude or terrain height capability and the entry flight path angle capability are reduced due to the longer time (larger Δh) required to reach -80°. For example, the h_T ≥ 13,300 (γ_F = -60°) is reduced to 8300 ft (γ_F = -80°). However, the γ_F = -80° case shows reduced decelerator weight fractions because of the reduced vernier requirements starting from a lower velocity. The effect of two design terrain heights on γ_E capability as a function of system growth (increasing B_E) is shown in Figure 14. γ_E capability is reduced by a minimum of 4 deg for B_DEC = .03 sl/ft² up to B_E of .32 sl/ft². The effects of increased terrain height, increased B_DEC and higher system weight all result in a reduction in γ_E capability.

The above discussion is concerned with terrain height capability as a means of showing performance capability. Having been assured that h_T capability is more than adequate (≥6000 ft), we now turn to the performance at the desired maximum h_T. Representative parachute/vernier performance for a design altitude of 6000 ft (plus a 500-ft margin) is shown in Figures 15 and 16, comparing two values of B_DEC. Weight fractions are calculated using a vernier velocity found at the intersection of the parachute trajectory and the h-V curve for vernier ignition. The limitation due to γ_F = -60° at vernier ignition is found at 6000 ft terrain height from Figure 12. The limitation due to F/W_0_d at vernier ignition is the value of γ_E where the parachute fails to decelerate the system to a velocity on the appropriate vernier h-V performance curve. Values of weight fraction in Fig. 15 are lower than the corresponding values in Fig. 12 for γ_E below those corresponding to the limit of γ_F = -60 deg at h_T = 6500 ft. This is because of the lower vernier velocities and weight requirements. Using design values of B_DEC discussed above in the staging analysis and vernier design average F/W_0_d = 4.2 and 2.6 for entry ballistic coefficients of 0.20 sl/ft² and 0.32 sl/ft² respectively, the design performance capability is shown in Fig. 17.

Deployment altitude - altitude loss - γ_E capability relationships are used to define the parachute deployment altitude. The deployment altitude for the light mission is chosen to be as high as possible to facilitate meeting entry science requirements. This requirement is limited by the minimum entry uncertainty capability, chosen as at least -20 deg, γ_E. Some degradation of the minimum time-to-landing requirement for maximum on-the-ground communication results from these criteria. Later missions, without the entry science package, are deployed at the lowest altitude consistent with desired maximum entry uncertainty (γ_E). A value of γ_E = -16 deg is used as the constraint for later
missions, using the limiting $\gamma_E = -17.2\,\text{deg}$ due to the terrain height and flight path angle (vernier ignition) constraints, have a deployment altitude of $13,000\,\text{ft}$ above terrain. Design characteristics using these deployment altitudes are summarized in Table 2.

Table 2
Summary of Parachute/Vernier Decelerator Characteristics

<table>
<thead>
<tr>
<th>1) Design $B_{DEC}$ ($\text{sl/ft}^2$, optimum with aeroshell)</th>
<th>0.20</th>
<th>0.32</th>
</tr>
</thead>
<tbody>
<tr>
<td>2) $\gamma_E$ capability (deg)</td>
<td>-21.3</td>
<td>-17.2</td>
</tr>
<tr>
<td>3) Deployment alt. (ft. above terrain)</td>
<td>18,000</td>
<td>13,000</td>
</tr>
<tr>
<td>4) Terrain height (ft above mean surface)</td>
<td>6,000</td>
<td>6,000</td>
</tr>
<tr>
<td>5) Parachute size ($D_0$-ft)</td>
<td>85.5</td>
<td>110.0</td>
</tr>
<tr>
<td>6) Vernier $E/W_0d$</td>
<td>4.2 (avg)</td>
<td>2.6 (avg)</td>
</tr>
<tr>
<td>7) Aeroshell staging alt. (ft above terrain)</td>
<td>15,500</td>
<td>11,000</td>
</tr>
<tr>
<td>8) Minimum vernier ignition (ft above terrain)</td>
<td>1,500</td>
<td>1,500</td>
</tr>
<tr>
<td>9) Design $W_{DS}/W_E @ \gamma_E$</td>
<td>.150 @ -20°</td>
<td>.182 @ -16°</td>
</tr>
<tr>
<td>10) Approximate max. time (sec) from deployment to:</td>
<td>12</td>
<td>9</td>
</tr>
<tr>
<td>A/S separation</td>
<td>111</td>
<td>90</td>
</tr>
<tr>
<td>Vernier ignition</td>
<td>121</td>
<td>100</td>
</tr>
<tr>
<td>Landing</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Deployment Trigger Considerations. The generalized parachute/vernier analysis above led to deployment altitudes as a result of atmosphere uncertainty and other requirements and considerations. The parachute vernier decelerator deployment altitudes recommended are at 18,000 ft and 13,000 ft above terrain for the $B_E = 0.2$ and 0.32 sl/ft$^2$ systems. Trajectories were calculated for VM-7, 8, 9 and 10 for the $B_E = 0.20$ sl/ft$^2$ system at a fixed deployment altitude of 18,000 ft above terrain. It was found that the system has reached a minimum flight path angle of $-78.2^\circ$ or near terminal conditions and a maximum velocity of 282 ft/sec. The minimum time on the parachute was 36.4 sec in VM-7 and the maximum time was 110.8 sec in VM-10. Time to staging at 15,500 ft above terrain ranged from 3.7 sec to 12.1 sec. A timer for aeroshell separation could also be used; a time of 6.0 sec is sufficient to meet the $M \leq .8$ requirement.

An alternative deployment method to the altitude trigger is deployment based on an h-h curve. The desired result is a shorter and fixed time on the parachute for all conditions. It has been postulated that payload oscillation angles due to crosswinds, wind shears, and gusts may cause a problem for TDNL radar lockup to initiate the final vernier phase. The fixed altitude deployment method has parachute phase times about three times as long in VM-10 as in VM-7. A fixed parachute time will allow a shorter deployment time to be used for all atmospheres diminishing the time the parachute is exposed to winds. More important, it will allow the use of a backup timer to be used for vernier ignition.

Several times on the parachute were investigated. It was found for a time of 35 seconds the shortest looked at, meets the requirements above, the ground rules of the investigation, and performance levels (weight) of the altitude deployment case. With a minimum vernier ignition altitude of 4000 ft above terrain, all final altitudes of the end of the parachute phase are within 1800 ft of that altitude. The minimum flight path angle is $-76.7^\circ$ compared to $-78.7^\circ$ for the fixed altitude case. Assuming both h and h-h deployment use the same parachute, performance may be compared using vernier weight fractions. The maximum value is 0.175 compared to 0.171 for fixed altitude deployment. These data were generated using the h-h curves from VM-7, VM-8, and VM-10 trajectory data and $B_{DEC} = 0.032$ sl/ft$^2$ as shown in Fig. 18.
It is evident that no performance advantage is indicated for the h-h concept. Further, the h-h trigger is more complex to implement and the shorter time on the parachute is not consistent with the concept of maximum time for entry science experiments. Finally, the results of separate analyses indicate no problem with TD&LR lock-up as a result of capsule oscillation. For these reasons, the altitude trigger is recommended over the h-h concept.

**Supersonic Aerodecelerator**

The supersonic aerodynamic decelerator consists of a tucked-back ballute coupled with a vernier retro system for final descent and landing. A range of deployment Mach numbers from 3.0 to 5.0 is used. The lower limit was arbitrarily chosen to provide a significant increase over parachute conditions. The upper limit is defined as that above which deployment conditions (dynamic pressure, heating) become excessive, decreasing the ballute design efficiency. Other constraints on the ballute-vernier system design are the same as for the parachute-vernier system.

a) Flight path angle at vernier ignition \( \gamma_F = -60 \) deg or steeper

b) Landing terrain height capability = 6000 ft above mean surface level at maximum entry flight path angle

System comparisons and capabilities were made on the basis of \( W_{TB}/W_E \) as for the parachute system with the aeroshell ejected at vernier ignition. The basic weight equation for the tucked-back ballute is:

\[
W_{TB} = C_P q_D R_{TB}^3 \left( \frac{1.29 \left( 0.537 - K \right) + 2.15}{K_c} \right) \left( \frac{R_{TB}}{16.62} \right) (F.S.) \]

where

- \( C_P = \) design pressure coefficient \( q_D = 2.5 \)
- \( q_D = \) design dynamic pressure
- \( R_B = \) ballute radius
- \( F.S. = \) factor of safety = 2.0

The constants \( K_i, K_e \) and \( K_c \) are fabric strength to weight ratio factors. Nominal values for Nomex ballutes at room temperature are:

- \( K_c = 96,000 \) ft
- \( K_e = 46,000 \) ft

These are corrected for design deployment Mach number, \( M_D \), by:

\[
M_D = 1.5 \quad K_1 = K_1 x 1.0 \\
M_D = 3.5 \quad K_1 = K_1 x 0.9 \\
M_D = 5.5 \quad K_1 = K_1 x 0.8 \\
\]

The \( K \) factor is

\[
K = 0.4, \; 4 < M_D < 5 \\
K = 0.6, \; 2 < M_D < 3 \\
\]

These factors were inserted in the above weight equation and curve fitted to obtain

\[
W_{TB} = 4.06 q_D R_{TB}^3 \left( \frac{0.0001106 - 0.00001257 M_D + 0.00000263 M_D^2}{16.62} \right) \\
\]

This equation also includes factors of 1.25 for deployment and attachment system and 1.3 for a 10% burble fence. The factor at the end of the equation is a shape factor accounting for the ballute diameter to capsule diameter ratio. For certain low dynamic pressure-Mach number combinations, the above weight equation yields weights less than minimum weight fabric. A minimum weight fabric including coating is assumed to be 1.5 oz/yd. This yields the following minimum weight equation:

\[
W_{TB} = 0.245 R_{TB}^2 - 4 \\
\]

The trailing ballute has not been considered in this analysis even though it offers certain advantages over the tucked-back ballute in staging. Previous studies have shown it to be significantly heavier than the tucked-back ballute for the same drag area.

The combination of the ballute weight characteristics and the vernier characteristics presented earlier resulted in the decelerator weight fractions shown in Figure 19. These data are idealized in the same sense as corresponding parachute data. The deployment dynamic pressures were chosen as the most severe to be encountered for the particular ballistic coefficient. The fact that ballute weights are proportional to diameter cubed as opposed to diameter squared for subsonic type parachutes leads to optimum sizes which are smaller than for corresponding subsonic designs. Optimum \( B_{DEC} \) values are .057 and .070.
slugs/ft$^2$ for $B_E$ of 0.20 and 0.32 slugs/ft$^2$, respectively, for deployment at $M = 3.0$. Corresponding values for $M_D = 5.0$ are substantially higher; weight fraction values for $M_D = 5.0$ are also higher than for $M_D = 3.0$. This trend is a function of the heavier ballute weights required for the more severe design requirements at Mach 5.0 deployment.

The weight fraction characteristics of Figure 19 are also used to evaluate aeroshell separation. At a subsonic Mach number for $B_E = 0.2$ slugs/ft$^2$, a ballute $B_{DEC}$ of 0.037 slugs/ft$^2$ would be required for aeroshell separation. This results in a decelerator weight fraction increase of 12% over the optimum value shown in Figure 19. The increase is 26% for a $B_E$ of 0.32 slugs/ft$^2$. It is therefore assumed that aeroshell separation takes place at vernier ignition, taking advantage of the effect of vernier thrust impingement. System design to assure complete separation and limit blocking of the TDLR is complicated by this requirement.

The Mach number in VM-7 and velocities in VM-7 and VM-8 for deployment conditions of $M = 3.0$ in VM-8 are shown in Figure 20. The VM-7 and VM-8 atmospheres are critical for the reasons discussed earlier in connection with the parachute analysis. As with lower deployment Mach numbers, the Mach numbers and velocities are lower in VM-7 than in VM-8. The resulting altitude loss effects from deployment to the 70-deg altitude are shown in Figure 21 for $B_{DEC} = 0.03$ slugs/ft$^2$. VM-7 is critical for all $B_E$ and $B_E$ considered.

The altitude loss characteristics combined with vernier altitude requirement lead directly to the maximum landing terrain height shown in Figure 22. This shows the effect of vernier thrust to weight and the effect of $B_E$, respectively. For a given terrain height, the maximum entry angle can be determined from these data. At $h_T = 6500$ ft, $W_{o_d} = 2.0$, $\gamma_E > -24^\circ$ for $B_E = 0.20$ slugs/ft$^2$ and 0.32 slugs/ft$^2$. The decelerator weight fractions associated with the maximum landing terrain height are also shown. Figure 23 shows the effect of two design terrain heights on the entry angle uncertainty as a function of entry angle uncertainty as a function of entry ballistic coefficient (system growth) for deployment at $M = 3.0$ and 5.0. The maximum $\gamma_E$ advantage of the higher deployment Mach number is clearly evident. However, this advantage is offset by the significantly higher decelerator weight fraction requirements at high Mach number shown earlier (Fig. 19). This leads to a choice of the lowest deployment Mach number which meets the $\gamma_E$ capability required. On this basis, a value of $M_D = 3.0$ is used for ballute design using maximum $\gamma_E = -16^\circ$ for growth missions.

Deployment altitude and altitude loss characteristics are also used to define the ballute deployment altitude. As for the parachute case, the deployment altitude for the light mission is chosen as high as possible, consistent with $\gamma_E = -20^\circ$ as a limit. A deployment altitude of 25,000 ft above terrain is chosen, assuring a $\gamma_E$ capability of -21 deg. Landing terrain height capability is well in excess of 6000 ft. Deployment at 21,500 ft above terrain for a $B_{DEC}$ of 0.70 slugs/ft$^2$ assures a $\gamma_E$ capability of -17.9 deg. The $\gamma_E$ capability due to vernier $W_{o_d}$ is limiting in this case. The design ballute/vernier system performance is shown in Figure 24 for light and heavy missions. The results indicate a slightly higher entry uncertainty capability at $B_E = 0.32$ slugs/ft$^2$ but higher decelerator weight fractions for all $B_E$ when compared to design parachute data. This is due primarily to heavier ballute weights required. The characteristics of the design ballute/vernier system are summarized in Table 3.

**All-Retro Decelerator Systems**

**Study Approach**

The objective of the analysis of the all-retro system is to define performance and capability over a wide range of parametric conditions. As a result, a somewhat idealized approach is taken as opposed to a more detailed design concept. This analysis assumes a two-phase decelerator flight profile:

1) Braking phase - constant thrust, constant attitude to zero horizontal velocity;
2) Vertical phase - constant thrust in one or two steps to zero vertical velocity at zero altitude above the terrain.

The ground rules and assumptions for the simplified approach are as presented earlier for the parachute vernier. In spite of the simplifying assumptions, the results are in good agreement with data developed using more sophisticated models on a large digital computer. The constant attitude, $\theta$, during the braking phase is taken equal to $\gamma_0$. This approach results from an investigation made to determine the optimum value of $\theta$, where optimum results in the minimum decelerator weight required for a specific set of entry and retro ignition conditions. It was found that as ignition altitude approaches the minimum, optimum $\theta$ approaches $\gamma_0$. Minimum ignition
Table 3

Summary of Ballute/Vernier Decelerator Characteristics

<table>
<thead>
<tr>
<th></th>
<th>$P_{\text{DEC}}$ (si/ft$^2$ - optimum w/aeroshell)</th>
<th>$P_{\text{E}}$ (si/ft$^2$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.20</td>
<td>0.32</td>
</tr>
<tr>
<td></td>
<td>0.057</td>
<td>0.070</td>
</tr>
<tr>
<td></td>
<td>2) $\gamma_E$ capability (deg)</td>
<td>-21.0</td>
</tr>
<tr>
<td></td>
<td>3) Deployment altitude (ft above terrain)</td>
<td>25,000</td>
</tr>
<tr>
<td></td>
<td>4) Terrain height (ft above mean surface)</td>
<td>6,000</td>
</tr>
<tr>
<td></td>
<td>5) Ballute diameter (ft, excl. burble fence)</td>
<td>40.0</td>
</tr>
<tr>
<td></td>
<td>6) Vernier $F/W_0$</td>
<td>4.2 (avg)</td>
</tr>
<tr>
<td></td>
<td>7) Minimum vernier ignition altitude (ft above terrain)</td>
<td>1,500</td>
</tr>
<tr>
<td></td>
<td>8) Design $W_{DS}/W_E$</td>
<td>0.155 @ -20°</td>
</tr>
<tr>
<td></td>
<td></td>
<td>0.216 @ -16°</td>
</tr>
<tr>
<td></td>
<td>9) Approximate max time (sec) from deployment to Vernier ignition</td>
<td>86.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>70.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>96.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>80.</td>
</tr>
</tbody>
</table>

Altitude is defined as that altitude resulting in braking phase termination at zero altitude above the terrain. In addition to performance criteria, the use of $\theta = -\gamma_0$ is also justified from a systems viewpoint. The aeroshell trims at zero angle of attack. At retro ignition, then, there is no requirement to determine orientation relative to some reference and maneuver to some $\theta \neq -\gamma_0$. Thus, the goals of simplicity and reliability are maintained.

The retro system assumes a bipropellant motor with an average Isp of 285 sec and a propellant mass fraction of 0.50. Mass fraction variation with total impulse requirement is neglected. A range of braking phase thrust to initial mass ratio of 50 to 150 lb/slug is investigated, consistent with propulsion system capability and preliminary studies results.

The system performance is presented in terms of the ratio of decelerator system weight to total capsule weight at initiation. The aeroshell is assumed ejected at retro ignition; its weight is not included in the initiation weight. The weight fractions may be corrected to an entry weight base by applying a factor of 0.871 and 0.913 for the 0.20 si/ft$^2$ and 0.32 si/ft$^2$ systems, respectively. System capability is presented in terms of entry angle uncertainty for an entry ballistic coefficient.

Complete parametric data are presented for an altitude ignition philosophy with the basic two-phase decelerator flight profile described above. The performance for a two-part vertical phase including a low thrust coast, and a velocity-altitude initiation are also presented.

**Braking Phase Equation of Motion**

For the special case where $\theta = -\gamma_0$; $x_f = 0$

and using

$$\dot{x}_0 = V_0 \cos \gamma_0,$$

$$\dot{\gamma}_0 = V_0 \sin \gamma_0$$

From eq. (1) where $\dot{x}$ at the end of the phase $= \dot{x}_f = 0$

$$\ln \left(1 - \frac{F_{x}}{m_0 C_j} \right) = -\frac{V_0}{C_j} \left(\dot{x}_f = 0\right)$$

or

$$\frac{F_{x}}{m_0 C_j} = 1 - \exp \left(-\frac{V_0}{C_j}\right) \left(\dot{x}_f = 0\right)$$

From Eq. (2)

$$\tau = -\frac{k_f}{g} \left(\dot{x}_f = 0\right)$$

$g = g_d$ unless otherwise specified.
From Eqs. (1), (4), and (5)
\[ \dot{z} = -g \frac{C_j}{F/m_0} \left[ 1 - e^{-V_0/C_j} \right] \]
and Eqs. (3), (4), and (5)
\[ \Delta z = \frac{\dot{z}_f}{g} \left[ C_j \sin \gamma_0 \left( 1 - \frac{V_0}{C_j} - \frac{1-e^{-V_0/C_j}}{1-e^{-V_0/C_j}} \right) \right] \]
\[ \Delta z = \frac{\ddot{z}_f}{2} \left( \dot{x}_f = 0 \right) \]
Equations (4), (5), (6), and (7) completely describe the braking phase, keeping in mind that mass of propellant used, \( m_p \), may be described by:
\[ m_p = \frac{F}{C_j} \]
Note that \( F/m_0 \) must be input as an independent variable.

Vertical Phase Equations of Motion (Constant Thrust, No Coast)
In general, the above equations do not work for vertical descent. Therefore, a slightly different approach must be taken. The vertical phase initial conditions are obtained directly from braking phase final conditions. Therefore:
\[ Z_0V = Z_f \; \dot{x}_f = 0 \]
\[ Z_0V = Z_f \; \dot{x}_f = 0 \]
Final conditions for vertical descent are, of course, zero vertical velocity at zero altitude above terrain. From Eq. (1)
\[ \frac{F t}{M_0 C_j} = 1 - e^{-g t / C_j} \; \left( \dot{Z}_f = 0 \right) \]
and the relationships between thrust, time and propellant mass are as shown above. Since \( \sin \theta = \sin 90^\circ = 1 \), Eq. (3) may be written as
\[ Z = Z_0 + Z_0 - \frac{1}{2} g t^2 + C_j \; \left[ 1 - \ln \left( 1 - \frac{F t}{m_0 C_j} \right) + \right. \]
\[ \left. \frac{\ln \left( 1 - \frac{m_f}{m_0 C_j} \right)}{C_j} \right] \]
It is found that substitution of (8) into (9) does not give a correct solution; the \( \frac{m_p}{m_0} \) and \( \frac{F}{m_0} \) combination must mutually satisfy both (8) and (9). Therefore, an iterative approach, applicable to digital solution, is used. An initial value of \( t \) is taken
\[ t_{initial} = \frac{\Delta Z}{\frac{1}{2} g t^2} \]
where
\[ \Delta Z = Z - Z_0 \]
and \( Z \) is zero altitude above terrain. Eqs. (8) and (9) are solved using the initial value of \( t \); the solution of (9) is compared to the required value of \( \Delta Z \) computed above. If the comparison is not within the required accuracy (10 feet was used in this analysis) the value of \( t \) is corrected until the required accuracy is attained. The correction for \( t \) is obtained by
\[ \Delta t = \frac{\Delta (\Delta Z)}{d(\Delta Z)/dt} \]
where \( \Delta (\Delta Z) = Z - Z_0 - \Delta Z \) from Eq. (9) and \( d(\Delta Z)/dt \) is obtained by differentiating Eq. (8).
Equations (8) and (9) then completely describe the vertical phase. To relate \( \frac{m_p}{m_0} \) to the
initial conditions of the braking phase, it is necessary to apply a factor equal to \( \left( 1 - \frac{m_p}{m_0} \right) \) braking
\[ \frac{m_p}{m_0} = \frac{m_p}{m_0} \left( 1 - \frac{m_p}{m_0} \right) \]
and likewise for the \( \frac{F}{m_0} \) parameter.

Effects of Atmosphere Uncertainty
Data representative of the effect of atmosphere uncertainty on system performance (decelerator weight fraction) is shown in Figures 25 and 26. System limitations in terms of minimum initiation altitude for each \( R_e \), \( \gamma_e \) are indicated. It is evident that, to be competitive, the all-retro de detector must be initiated at the minimum possible altitude above terrain. The minimum initiation altitude is defined as that altitude resulting in braking phase termination (-90 deg flight path angle) at zero altitude above the terrain. It is seen that over the \( R_e \), \( \gamma_e \) range of interest, VM-7 defines the minimum initiation altitude. However, at any given initiation altitude, VM-8 defines the minimum required decelerator system size (propellant load). The VM-7 altitude limitation is due to the fact that for the \( R_e \), \( \gamma_e \) of interest, VM-7 velocities at initiation.
are higher at the lower altitudes. The VM-8 decelerator size requirement results from the relatively long VM-8 descents from VM-7 initiation altitudes.

Figure 27 shows the system performance as a function of entry flight path angle for ignition at the minimum altitude. It is seen that the entry uncertainty capabilities (limiting \( \gamma_E \)) for a \( \gamma_E \) is defined by VM-8 initial conditions. This is due again to the velocity at retro system ignition. At higher \( \gamma_E \), \( \gamma_E \) velocities at initiation become increasingly higher above some crossover altitude. The VM-8 velocity increase is sufficiently rapid to result in definition of the \( \gamma_E \) limit as shown. This is also responsible for the rapidly increasing slope of the weight fraction vs \( \gamma_E \) curves.

The combination of the minimum allowable initiation altitude (VM-7) and the minimum allowable decelerator weight fraction (VM-8) results in the performance and system capability shown in Figure 228. For example, from Fig. 25 at \( \gamma_E = 0.20 \) sl ft\(^2\) at that altitude, a value of \( W_{DS}/W_0 \) of 0.27 must be used based on VM-8. This is the value plotted in Fig. 28. Thus, the combined performance in Fig. 28 is worse than the performance in each atmosphere shown in Fig. 27. Entry uncertainty capability, however, is no worse than for the limiting VM-8 case shown in Fig. 27.

The all-retro system is sensitive to the type of initiation trigger used and the control logic during the vertical descent. A system using a fixed altitude trigger coupled with a fixed thrust vertical descent displays generally poor performance, as shown above. The trigger altitude must be set high on the basis of VM-7 trajectory characteristics resulting in inefficient, long, low speed vertical descents in VM-8. This condition is alleviated by the use of a two-step vertical phase incorporating a low thrust coast followed by a high thrust (approaching impulsive) final deceleration to zero velocity at landing. The equations of motion for this maneuver assume constant acceleration in each step.

For a two-phase (coast/vernier) descent, where, for each phase

\[
\frac{F}{W_{0_d}} = n
\]

the vernier phase equations will be

\[
t_v = -\frac{\hat{z}_{f_B}}{\dot{g}(n_v^{-1})}; \text{ (for } n_c = 1.0 g) \tag{10}
\]

\[
h_{0_v} = \frac{1}{2} \frac{\hat{z}_{f_B}^2}{g(n_v^{-1})}; \text{ (for } n_v = 0) \tag{11}
\]

\[
m_{p_v} = \dot{m}_n \left[ 1 - \frac{n_v}{\dot{g}(n_v^{-1})} \left( \frac{\hat{z}_{f_B}}{C_j} \right) \right] \tag{12}
\]

The results herein assume \( n_v = 30 \) g.d.

For a 1.0 g.d. coast immediately following the braking phase, the equations are

\[
h_{f_C} - h_{0_C} = h_{0_v} - h_{0_C} = V_{0_C} = \frac{\hat{z}_{f_B}}{C_j} \tag{13}
\]

\[
t_C = \left( h_{0_v} - h_{0_C} \right) / \hat{z}_{f_B} \tag{14}
\]

\[
m_{p_C} = \dot{m}_n \left[ 1 + \frac{g_d(h_{0_v} - h_{0_C})}{C_j \hat{z}_{f_B}} \right] \tag{15}
\]

Finally

\[
\frac{m_{p_v}}{m_{0_C}} = \left( \frac{m_{p_v}}{m_{0_v}} \right) \left( \frac{m_{p_C}}{m_{0_C}} \right) = \left( 1 - \frac{m_{p_v}}{m_{0_v}} \right) \left( 1 - \frac{m_{p_C}}{m_{0_C}} \right) \tag{16}
\]

and for the total deceleration system mass fraction

\[
\frac{m_{D_{2}}} {m_{0}} = \frac{1} {\lambda} \left\{ \left( \frac{m_0}{m_{D_{2}}} \right) + \left( 1 - \frac{m_{p_v}}{m_{0_v}} \right) \left( 1 - \frac{m_{p_C}}{m_{0_C}} \right) \right\} \tag{17}
\]

The guidance law for the two-step vertical phase need not be more complex than the one-step approach shown above. A constant acceleration, for example, is not strictly required; constant low and high thrust steps would work equally well with the same proportional throttling as required for the no-coast case. There is, in fact, no reason for accepting the low performance of the no-coast case shown above. It is presented here only as a convenient means of displaying the effects of atmosphere uncertainty. The results of employing a vertical coast are shown in Fig. 29.

At \( \gamma_E = 0.20 \) sl ft\(^2\), \( \gamma_E = -16^\circ \), \( F/m_0 = 50 \) lb/sl, the weight fraction of .27 from Fig. 28 is reduced to 0.22 in Fig. 29. Note that the improvement in decelerator weight fraction is not accompanied by any change in entry uncertainty capability.
Performance may also be improved by the use of a velocity-altitude trigger for braking phase ignition. This approach can be most rewarding from a performance standpoint but requires considerable care in establishing the trigger logic. The contours of velocity vs minimum initiation altitude for VM-7 and VM-8 as an example are not colinear. This is shown in the sketch below. If an intermediate atmosphere (VM-7½) is postulated, the most adverse V-h trigger contour must be selected, as shown in the sketch. This eliminates much of the potential performance gain.

![Required trigger contour](image)

The performance resulting from the use of a V-h ignition capability trigger is shown in Fig. 30. Again the $\gamma_E$ capability is unchanged although performance is improved over the no-coast altitude initiation case. A comparison with the vertical coast performance (Fig. 29) shows a slight advantage for the V-h trigger at lower $B_E \gamma_E$. A weight fraction of .215 is shown for the case used above as an example. At higher $B_E$, the intermediate $\gamma_E$ show a slight performance advantage for the vertical coast; the reason for this is evident from the sketch above. This effect would be magnified by inclusion of VM-9, and VM-10 requirements for V-h trigger initiation. In that case the intermediate VM would be treated in a manner similar to the VM-7½ postulated above. It is emphasized that the V-h trigger concept used here is a "first cut" effort directed toward defining all retro performance suitable for comparison with other systems.

### Sensitivity to Parametric Variations

The results presented above are based on a number of study ground rules, simplifying assumptions, and arbitrary system characteristics. It is necessary to determine the sensitivity of the system performance and capability to changes in the parameters defined by ground rules, etc. The effects of constant attitude $\theta = \gamma_0$ and initiation philosophy are discussed above. This section deals with the sensitivity to drag, terrain height, and retro system characteristics.

The generalized performance data presented above assumes that drag is zero $\left( C_D = 0 \right)$ in the presence of a forward firing rocket engines. The total decelerating force (thrust + drag) experienced by a vehicle during the firing of retro-rockets is dependent upon the geometry of the vehicle, the location of the retro-rockets, the thrust to free stream drag ratio, and the Mach number. The interaction between the rocket exhaust plume and the free stream is complex and not amenable to analytic solution. Experimental data have shown that the result of the flow interference can range from an increase in total drag to its complete elimination. There is not sufficient experimental data to allow rational estimates of effective drag for a given configuration due to the large number of variables involved.

The general approach used here in the investigation of drag effects is to solve a simplified axial force equation with and without drag and ratio the resulting propellant mass ratios, $\frac{m_p}{m_0}$. Where drag is included, the drag during retrofire is assumed to be equal to the free stream drag. The effect of gravity is neglected as a second order effect in the solution. The approach and gravity assumption are justified by the use of a ratio for comparison with and without drag and because the desired result is a "ball park" magnitude of drag effect.

Adding a drag term to the basic axial acceleration expression and neglecting the effect of gravity, we have

$$\frac{dV}{dt} = -\frac{E}{m} + \frac{1}{2} \frac{C_A V^2}{m}$$

Letting $m = m_0 - \dot{m} t$ and $K_1 = \frac{1}{2} \rho C_D A / F$

then

$$-V_f \frac{C_j}{C_j} + \tan^{-1} \frac{V_0}{V_f} = \ln \left( 1 - \frac{m_p}{m_0} \right)$$

If we solve without the drag term, we have

$$\frac{V_f - V_0}{C_j} = \ln \left( 1 - \frac{m_p}{m_0} \right)$$

or

$$\ln \left( 1 - \frac{m_p}{m_0} \right) = \frac{\Delta V}{C_j}$$

(the ideal velocity equation)
For \( V = 0 \) the ratio of \( \frac{m_{CD}}{m_{CD}^0} \) is a convenient way of determining the effect of drag.

Rewriting \( K_1 \)

\[ K_1 = \frac{1}{2} \rho \frac{C_D}{F} = \frac{1}{2} \frac{D}{B_D} \frac{1}{\rho_0} \]

where \( B_D = \frac{m}{C_D} \)

Assuming \( \rho \approx \rho_0 \), we can use values of \( \rho_0 \) corresponding to average values of \( h_0 \) as follows:

\[ \rho_0 = 1.2 \times 10^{-5} \text{ sl/ft}^3 \text{ at } h_0 = 10,000 \text{ ft} \]

\[ \rho_0 = 4.0 \times 10^{-5} \text{ sl/ft}^3 \text{ at } h_0 = 8000 \text{ ft} \]

We may estimate values of effective decelerator ballistic coefficient, \( B \), as follows, using values of \( m_0 \) and \( C_D A \) corresponding to a 10 ft dia. flat face cylinder where

\[ C_D = 0.95 \text{ to } 1.2 \approx 1.0 \]

\[ A = 78.5 \text{ ft}^2 \]

\[ \frac{B \rho}{B_0} \left( \frac{s}{	ext{ft}^2} \right) = 0.85 \]

\[ \frac{1}{B_0} \left( \frac{s}{	ext{ft}^2} \right) = 1.34 \]

\[ \frac{1}{B_0} \left( \frac{s}{	ext{ft}^2} \right) = 1.89 \]

Using these values with values of \( \frac{V_0}{m_0} \) corresponding to \( h_0 \) we can approximate the performance ratio as follows: \( \left( \frac{F}{m_0} = 100 \text{ lb/sl} \right) \)

\[ \frac{B_E}{B_0} \left( \frac{s}{	ext{ft}^2} \right) \quad \gamma_E \quad \theta_E \quad \frac{m_{CD}}{m_{CD}^0} \quad \frac{m_{CD}}{m_{CD}^0} = 0 \]

\[ \frac{B_E}{B_0} \left( \frac{s}{	ext{ft}^2} \right) \quad \gamma_E \quad \theta_E \quad \frac{m_{CD}}{m_{CD}^0} \quad \frac{m_{CD}}{m_{CD}^0} = 0 \]

\[ \frac{B_E}{B_0} \left( \frac{s}{	ext{ft}^2} \right) = 0.85 \]

\[ \frac{B_E}{B_0} \left( \frac{s}{	ext{ft}^2} \right) = 1.34 \]

\[ \frac{B_E}{B_0} \left( \frac{s}{	ext{ft}^2} \right) = 1.89 \]

From the above, it is seen that the incorporation of drag effects results in an approximate 1.5% to 14% reduction in propellant usage. For the light mission, a rough average is 5%; this would reduce the previously mentioned value of \( \frac{W_{DS}}{W_0} \) from 0.215 to 0.204.

The all-retro system is sensitive to other parameters, notably \( h_L \), \( I_{SP} \), and \( \lambda \). Performance sensitivities to \( h_L \) and \( I_{SP} \), are of small significance; they are in the order of 2%/1000 ft and 0.5%/sec, respectively. The sensitivity of entry uncertainty capability to terrain height is also very small for the low \( B_E \) used in this analysis. Of considerable significance, however, is the sensitivity to propellant mass fraction, \( \lambda \). This is true because propulsion system weight is inversely proportional to \( \lambda \) and because the value of \( \lambda \) for all-retro applications may vary over a relatively wide range (approximately 0.3 to 0.6 depending upon a number of systems and design considerations). A value of 0.5 was chosen somewhat arbitrarily for this study as indicative of an optimistic and sterilizable system suitable for comparison with aerodynamic systems. It is also important to note that a constant value of \( \lambda \) is not consistent with actual system design. In general, \( \lambda \) improves with increased total impulse (propellant) required. This effect would tend to reduce the slope of a system weight fraction curve, as indicated in the sketch below. The total effect of \( \lambda \) is, of course, highly dependent upon actual system design; thus, a performance improvement or decrease cannot be quoted here. Little or no change would be expected in entry uncertainty capabilities.

Effect of Throttling Ratio Considerations

Preceding results indicate the following, with regard to \( \frac{F}{m_0} \):

1) Higher \( \frac{F}{m_0} \) result in lower \( \frac{m_{DS}}{m_0} \);

2) Higher \( \frac{F}{m_0} \) result in higher \( \frac{\gamma_E}{\lambda} \) capability at a \( B_E \).

These results indicate that the highest practical value of thrust be chosen. However, higher thrusts require higher throttling ratios, especially if some fixed minimum thrust is required during the vernier phase. We can define throttling ratio, \( T_R \), as a function of \( \frac{F}{m_0} \) and \( \frac{m_{DS}}{m_0} \) as follows:
For a configuration which has engine-out capability, we can define

$$F_{\text{max}} = \left( \frac{F}{\gamma_0} \right) \left( \frac{m_0}{K_{\text{EO}}} \right)$$

where $K_{\text{EO}}$ is an engine-out constant ranging from 0.5 to 0.67 depending on philosophy for handling engine-out. For a minimum thrust to weight requirement (during vernier, for example)

$$F_{\text{min}} = K m f g_d$$

where $K_m$ is the minimum $F/W_d$ ratio desired, usually ranging from 0.6 to 0.9 $g_d$.

The final mass is

$$m_f = m_0 \left( 1 - \frac{m_{DS}}{m_0} \right)$$

The throttling ratio is then

$$TR = \frac{F_{\text{max}}}{F_{\text{min}}} = \frac{K_{\text{EO}} K_m g_d}{K_{\text{EO}} m_f g_d} \left( 1 - \frac{m_{DS}}{m_0} \right)$$

A carpet plot of $TR$ for representative values of $F/m_0$, $K_m$, $K_{\text{EO}}$, and $m_{DS}/m_0$ is presented in Figure 31.

Previously presented data indicate a minimum value of $F/m_0 = 100$ lb/sl to achieve a $\gamma_0$ capability of -20 deg for a $R_E$ of 0.40 sl ft$^2$. Figure 31 indicates a value of $K_{\text{EO}} \times K_m$ of 0.50 to maintain a throttling ratio of less than 20 for $F/m_0 = 100$ lb/sl and $m_{DS}/m_0 < .40$. Note that the choice of $F/m_0$ is defined by the system capability requirement as discussed earlier. The effect upon throttling ratio essentially defines the minimum $T/W_d$ for a desired maximum TR restriction. For the conditions outlined above, the minimum value for $K_m$ is 0.75. Note, however, if values of $TR$ had proved excessive (> 20) over the range of $K_m$, $K_{\text{EO}}$ for $F/m_0 = 100$ lb/sl, it would have been necessary to back off on $F/m_0$ and accept a reduced $\gamma_0$ capability.

System Comparison

A comparison of the terminal phase systems is shown in Figure 32 for $R_E$ of 0.2 and 0.32 sl ft$^2$. In terms of total decelerator weight fraction, the parachute system is somewhat lighter than the ballute or all-retro systems. The parachute/vernier system is 3% lighter than the ballute/vernier and 19% lighter than the all-retro system at $R_E = 0.20$ sl ft$^2$ and $\gamma_0 = -20$ deg. The parachute's advantage increases to 17% and 21% at $R_E = 0.32$ sl ft$^2$, $\gamma_0 = -16$ deg. This also indicates a lower sensitivity to system growth for the parachute/vernier decelerator. Entry uncertainty capabilities for all systems must be considered adequate. Entry angle uncertainties of -21 deg and -16 deg may be tolerated for the .20 sl ft$^2$ and .32 sl ft$^2$ systems, respectively. Note that this comparison is valid for a combination of assumed worst conditions of terrain height, atmosphere, and entry flight path angle uncertainties.

In view of the slight, but hardly overwhelming, performance advantage of the parachute/vernier system, other factors must be considered for additional comparison. In simplest terms, the terminal phase decelerator must "take out" a $\Delta V$ over some $\Delta t$. Mission uncertainties such as terrain height, atmosphere, entry conditions, system tolerances, and deployment tolerances boil down to a velocity-altitude uncertainty at terminal phase initiation. Both the aerodynamic and all-retro systems considered in this analysis are capable, by design, of handling the most unfavorable combinations of the uncertainties considered. If the terminal phase initial conditions result in a higher altitude or velocity than designed for (sensor accuracy, entry body c.g. offset, etc.) the aerodynamic systems and particularly the subsonic type parachute could very well still work successfully. The all-retro system, without additional propellant, would have a burnout before touchdown. To summarize, the all-retro system is limited by its designed impulse, whereas the aerodecelerators have some "grey area" in which to work.

A further disadvantage for the all-retro system lies in the optimistic propellant mass fraction assumed. Propulsion system standardization (i.e., same engines for light and heavy missions) results in mass fractions in the order of half the value used herein. Decelerator weights go up proportionally.

Problems in ballute attachment and release techniques are anticipated for the tucked-back configuration. Further, the aeroshell staging problem discussed for the ballute is equally applicable to the all-retro system. These problems have not been encountered in parachute/vernier design, and are solved without effect on the optimization of the decelerator. Finally, the parachute has been successfully tested under realistic conditions; this adds confidence not only to parachute feasibility and data used in the analysis but also to testing technique and feasibility.
Spacecraft/Capsule Separation

Communicate with Spacecraft
Align Capsule

Enter Mars Atmosphere

Chute Deployment & Heat Shield Jettison

Terminal Phase

Jettison Chute & Fire Vernier Engines
Vernier Engine Shutoff
Landing

Figure 1 Mission Profile for A Mars Soft Landing
Figure 2 Candidate Terminal Phase Decelerator Systems
Figure 3 Martian Atmosphere Models
a) Shallow Entry Corridor

Skipout Limit

2g Over Skipout

Nominal

±3g

b) Steep Entry Corridor

Skipout Limit

2g Over Skipout

Nominal

±3g

VM-8 Atmosphere
Entry at 800,000 ft
B = 0.4 s1/ft²
R₉ = 3393 km
Posigrade, Equatorial Entry

Entry Velocity (km/sec)

Nominal Range

Figure 4: Entry Corridors
Figure 5 Altitude Velocity Profiles

- $B_E = 0.2 \text{ sl/ft}^2$
- $V_E = 3.5 \text{ km/sec}$
- $\gamma_E = 9^\circ$
- $\gamma_E = 12^\circ$
- $\gamma_E = 14.5^\circ$
- $\gamma_E = 16^\circ$

Terminal Velocity ($V_T$)
1. Hypersonic Drag Coefficient
2. Drag Coefficient @ MN = 1.0
3. Subsonic Drag Coefficient
Figure 6  Parachute Deployment Conditions
Figure 1: Aeroball Relative Velocity vs. Altitude
Figure 8 Characteristics of Vernier System for Use With Aerodecelerators
Parachute System Weight
\[ W_{PS} = 0.0243 D^2 \]
Propulsion System
\[ I_{sp} = 222 \text{ sec} \]
\[ \gamma = 0.50 \]
\[ V_{wind} = 220 \text{ fps} \]
\[ \left( \frac{T}{q_f} \right) = 3.0 \]

Figure 9 Parachute/Vernier Decelerator System Weight Fraction
Deployment at Altitude for M = 1.6 in VM-8

A. Deployment Mach No. in VM-7

B. Deployment Velocity

Figure 10 Effect of Atmosphere Uncertainty on Parachute Deployment Conditions
$B_{DEC} = 0.030 \text{ sl/ft}^2$

A. $B_E = 0.20 \text{ sl/ft}^2$

B. $B_E = 0.32 \text{ sl/ft}^2$

Figure 11 Parachute Altitude Loss Characteristic
A. Altitude Capability

\[ B_{\text{DEC}} = 0.03 \, \text{s}^2/\text{ft}^2 \]

\[ B_E = 0.20 \, \text{s}^2/\text{ft}^2 \]

\[ B_E = 0.32 \, \text{s}^2/\text{ft}^2 \]

B. Decelerator Requirements

![Diagram of decelerator requirements]

Figure 12 Parachute/Vernier Terrain Height Capability \( \gamma = -60 \, \text{deg} \) at Vernier Ignition
$B_{DEC} = .03 \text{ sl/ft}^2$

A. Altitude Capability

B. Decelerator Requirements

Figure 13 Parachute/Vernier Terrain Height Capability $\gamma = -80 \text{ deg}$ at Vernier Ignition
$T/W_{\infty}$ (Vernier) = 3.0
$\gamma = -60$ Deg At Vernier Ignition

Figure 14 Parachute/Vernier Effect of Terrain Height on $B_E$, $\gamma_E$ Capability
Figure 15 Parachute/Vernier Decelerator Entry Uncertainty Capability

\[ B_{DEC} = 0.030 \text{ sl/ft}^2 \]
Figure 16 Parachute/Vernier Decelerator Entry Uncertainty Capability

\[ \mathbf{B}_{\text{DEC}} = 0.045 \text{ sl/ft}^2 \]
h_T = 6000 ft + 500' Margin

A. \( B_E = 0.20 \text{ sl/ft}^2 \)
   \( T/W_0 = 4.2 \)
   \( B_{DEC} = 0.037 \text{ sl/ft}^2 \)

Limit for \( h_0 \) @ \( M = 1.6 \) in VM-8

B. \( B_E = 0.32 \text{ sl/ft}^2 \)
   \( T/W_0 = 2.6 \)
   \( B_{DEC} = 0.035 \text{ sl/ft}^2 \)

Limit Due to \( T/W = 2.6 \)
Limit Due to \( \gamma = -60^\circ \) at Vernier Ign

Figure 17 Design Case Parachute/Vernier Performance and Capability
$B_{DEC} = 0.032 \text{ sl/ft}^2$

$\Delta t_{para} = 35 \text{ sec}$

Vernier Target Altitude = 4000 ft.

Figure 18 Parachute Deployment Conditions Based on $h$-$\dot{h}$
Figure 19 Ballute/Vernier Decelerator System Weight Fraction
Deployment at Altitude for M = 3 in VM-8

A. Deployment Mach No. in VM-7

B. Deployment Velocity

Figure 20 Effect of Atmosphere Uncertainty on Ballute Deployment Conditions
A. \( B_E = 0.20 \text{ sl/ft}^2 \)

B. \( B_E = 0.32 \text{ sl/ft}^2 \)

Figure 21 Ballute Altitude Loss Characteristics
A. Altitude Capability

\[ \frac{T}{W_{o_d}} \text{ Variable} \]
\[ \gamma = -60^\circ \text{ at Vernier Ignition} \]
\[ B_{\text{DEC}} = 0.03 \text{ sl/ft}^2 \]
\[ M_D = 3.0 \]

Max Landing Altitude ~ \( \text{ft x 10^{-3}} \) above Mean Surface

- \( B_E = 0.20 \text{ sl/ft}^2 \)
- \( B_E = 0.32 \text{ sl/ft}^2 \)

B. Decelerator Requirements

\[ \frac{\dot{W}_{DE}}{W_E} \]

Figure 22 Ballute/Vernier Terrain Height Capability
A. Deployment M = 3.0

\[ \frac{T/W}{O_d} \text{ (Vernier) } = 3.0 \quad \gamma = -60^\circ \text{ at Vernier Ignition} \]

\[ h_T = 0 \text{ ft} \]

\[ h_T = 6000 \text{ ft above Mean Surface} \]

\[ B_{DEC} = 0.03 \text{ sl/ft}^2 \]

\[ B_E \sim \text{sl/ft}^2 \]

B. Deployment M = 5.0

\[ B_{DEC} = 0.10 \text{ sl/ft}^2 \]

\[ B_E \sim \text{sl/ft}^2 \]

Figure 23 Ballute/Vernier Effect of Terrain Height on \( B_E, \gamma_E \) Capability
h_e = 6000 ft + 500 ft Margin
M_D = 3.0

A. \( B_E = 0.20 \text{ sl/ft}^2 \)
\( F/W_{o_d} = 4.2 \)
\( B_{DEC} = 0.057 \text{ sl/ft}^2 \)

Limitation Due to
\( h_o @ M = 3.0 \)

B. \( B_E = 0.32 \text{ sl/ft}^2 \)
\( F/W_{o_d} = 2.6 \)
\( B_{DEC} = 0.070 \text{ sl/ft}^2 \)

Limitation Due to
\( \gamma = -60^\circ @ h_m = 6000 \text{ ft} \)

Figure 24 Design Case Ballute/Vernier Decelerator Performance and Capability
Figure 25 All Retro Decelerator Effect of Atmosphere Uncertainty on Minimum Initiation Altitude

\( \gamma \) = -16 deg
Figure 26 All-Retro Decelerator Effect of Atmosphere Uncertainty on Minimum Initiation Altitude \( \gamma_E = 20 \text{ deg} \)
Figure 27 All-Retro Decelerator Effect of Atmosphere Uncertainty on Performance and Entry Uncertainty Capability
Figure 28 All-Retro Decelerator Performance and Entry Uncertainty Capability for Minimum Altitude Initiation, No Vertical Coast
Figure 29  All-Retro Decelerator Performance and Entry Uncertainty Capability for Minimum Altitude Initiation, with Vertical Coast
Figure 30  All-Retro Decelerator Performance and Entry Uncertainty Capability Velocity - Altitude Ignition Trigger
Figure 31 Throttling Ratio for $K = K_{E0} \times K_M = 0.50$
Figure 32 Decelerator Comparison
Conclusions

Based on the preceding system comparison, it is concluded that:

1. The parachute/vernier decelerator is lighter than the other candidate systems considered under the assumptions and ground rules of this analysis.

2. The parachute/vernier system is less sensitive to system growth (increased weight).

3. The aerodynamic decelerators as analyzed herein have more margin to cope with unknowns or uncertainties.

4. The all-retro system is more sensitive to system assumptions and analysis ground rules.

5. Aeroshell and aerodecelerator separation are accomplished without design problems and without effect on performance optimization for the parachute/vernier system.

6. The parachute/vernier system is preferred for terminal phase deceleration for the mission and within the ground rules discussed herein.

References


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The information contained herein in no way constitutes a final decision by NASA or JPL or any other Government agency relative to its use in a Voyager-type program.