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LANDING SITE DISPERSIONS OF AN UNCONTROLLED LANDER ON MARS

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SUMMARY

A study was made to determine the accuracy with which a lander, descending from a bus orbit, can reach a specified landing site on Mars without using a range control system. This report considers both a lifting (L/D = 0.5) and a nonlifting lander - each with two different values of mass. Several different entry angles were considered, along with two different procedures for the deorbit maneuver: the use of the minimum deorbit velocity increment and the use of a constant deorbit velocity increment.

The error sources considered are variations in the Martian atmosphere, variations in the actual lift of the vehicle, and errors in performing the deorbit maneuver. It is concluded that if minimum deorbit velocity increment is used with entry angles of -20° to -24°, the landing site accuracy will be on the order of 1°, which is of the same order as the present knowledge of the location of the landing site. Thus a range control system need not be included in the lander.

INTRODUCTION

One plan for the unmanned exploration of Mars is to use an orbiter bus to carry a lander into an elliptical orbit. After several orbits, during which the knowledge of the orbit is improved, the lander is detached and given a small ΔV to put it in a descent trajectory. The vehicle then enters the atmosphere and experiences some atmospheric braking. Just before touchdown a soft landing device, either a parachute or a retrorocket, or both, is used. The landing trajectory will be chosen so that the vehicle will touch down at some specific location on the planet.

Comparisons of landing vehicles with and without lift have shown (refs. 1 and 2) that if a lifting vehicle is used, the drag-to-weight ratio can be reduced, or equivalently, the landed payload can be increased. (Some considerations of entry requirements for lifting and nonlifting vehicles are given in refs. 2-4.)

In general, we can expect that the landing vehicle will not touch down at exactly the desired point, but instead the touchdown point will have some dispersion. If this dispersion is sufficiently small, then the landing will be satisfactory. If, however, the dispersion is too large, then some form of control must be used during the descent trajectory. This control might be applied either immediately after the application of the deorbit ΔV or else during atmospheric flight. The purpose of this report is to determine the size of the dispersion when no control is used, and from this data, to determine whether a control system is required. It is desirable, of course, to avoid the use of a control system, since without one the vehicle would be simpler and, therefore, more reliable.

The causes of landing errors considered are: variations in the Martian atmosphere, errors in making the velocity maneuver to enter the descent trajectory, and uncertainties in the lifting characteristics of the vehicle. Pragluski (ref. 3) did work similar to that reported here, but he did not carry the trajectory through the atmosphere, and used only a single entry velocity. Avco (ref. 4) has also done similar work, but restricted it to a nonlifting vehicle, a single entry angle, and a restricted set of entry ranges. This report considers the more general case of a lifting or nonlifting vehicle, with two different values of mass, entering the planetary atmosphere anywhere in the plane of the initial orbit, at several different entry angles. The initial orbit is assumed to be equatorial and posigrade. The effects of other orbit inclinations are considered in appendix A. The deorbit ΔV used is either the minimum value for entry at a particular range, or a fixed value of sufficient size to allow entry at any point. For both cases, the expected downrange error at impact with the surface is determined. The report presents results for the planar problem, and some theoretical considerations of out-of-plane motion are included in appendix B. The effects of Martian winds and of motion while the soft landing system is in operation are not considered.

In order to determine whether a particular landing site error is acceptable, the error must be compared with the available knowledge of the Martian surface. The knowledge of the location of desirable landing sites will be based on astronomical photographs, and on photographs taken from spacecraft such as Mariner IV or the orbiter bus itself, prior to detaching the lander. Slipher (ref. 5, p. 59), in discussing the quality of earth-based photographs of Mars, says "The finest linear markings found on our best images measure around $0''.1$ to $0''.5$ of arc, but their true width must be greater because irradiation will tend to narrow them down considerably. The smallest circular objects are about three times this diameter for dark spots, and even greater in case of white spots." These remarks refer to resolution, that is, the ability to distinguish between neighboring objects. The ability to specify the location of a particular feature is not that good. Thus, the location of desirable landing sites is known no better than about $0''.2$ of arc on Mars as seen from the Earth during close oppositions. For comparison this is equivalent to about 50 km over the surface of Mars, or about 1° of central angle. The Mariner IV photographs have much greater resolution, that of the best of them being about 1 km (ref. 6). However, the attitude control system of Mariner IV had an accuracy of about 0.5°. Since the closest approach was about 6000 km, the uncertainty in the location of the picture is again about 50 km. If the picture is taken from the orbiter itself, then some improvement in the knowledge of the location of the picture can be achieved. If we assume the same attitude control accuracy, and a 1000×5000 km altitude elliptical orbit, then at perigee the uncertainty of the picture location will be about 10 km, whereas at apogee it will still be about 50 km. Thus, it seems reasonable that for many situations a landing error of 50 km, or 1°, would not be considered large, and even 2° error may be acceptable.

NOTATION

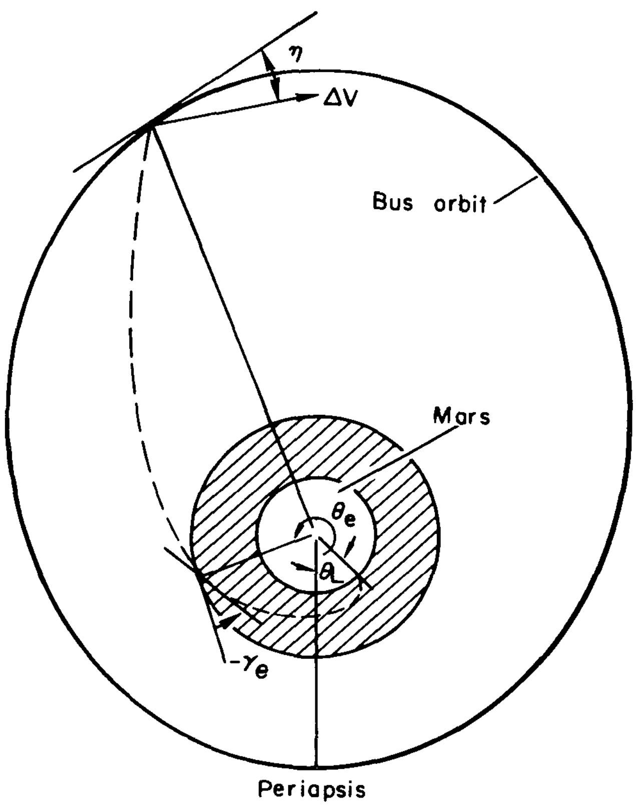
f	vehicle acceleration per unit mass, km/sec^2
f_y	lateral component of aerodynamic force per unit mass, km/sec^2
i	inclination of orbit
$\frac{L}{D}$	lift-to-drag ratio of lander
$\frac{m}{C_D A}$	ballistic coefficient, kg/m^2
r	radial distance from Mars, km
V_d	velocity on descent trajectory at intersection with bus orbit, km/sec
V_e	entry velocity with respect to inertial space, km/sec
Y_L	out-of-plane component of landing site, km
γ_e	flight-path angle at entry with respect to inertial space, positive upwards, deg
$\delta(\ )$	variation in ( )
ΔV	magnitude of velocity vector required to transfer from bus orbit to descent trajectory, km/sec
ΔV_y	out-of-plane component of transfer velocity vector, km/sec
η	angle between ΔV and local horizontal, deg
θ	range angle, deg
θ_e	range angle of lander at entry, measured in direction of motion from periapse of bus orbit, deg
θ_L	range angle of landing site, measured in direction of motion from periapse of bus orbit, deg
ξ	out-of-plane component of angle between ΔV and local vertical, deg
φ	vehicle roll angle, deg
ψ_L	out-of-plane component of landing site, deg
$(\ )_{3\sigma}$	three sigma dispersion of a statistical quantity

ANALYSIS OF RANGE ANGLE ERRORS

The problem is to determine the expected range angle error on landing, as a function of the three principal error sources: errors in making the velocity maneuver, variations in the atmosphere, and variations in the vehicle's lift-to-drag ratio. Here, range angle is defined as the angular distance traveled over the surface of the planet in the plane of motion.

Navigation errors (i.e., initial condition uncertainties) at the start of the entry are not considered. After several orbits of Mars, the expected value of these errors will be on the order of kilometers in position and meters per second in velocity. The position error propagates to the surface nearly 1:1, and is negligible compared to the other errors. The velocity error is smaller than the error in making the velocity correction, and therefore can either be ignored or combined with the velocity correction.

Let us arbitrarily break the trajectory into two parts, the descent trajectory, immediately after the velocity maneuver and continuing down to the entry altitude, and the atmospheric entry, from entry to touchdown. This division of the trajectory lessens the number of parameters to be considered in each part, since the first part is independent of the vehicle and atmosphere, and the second part is independent of the bus orbit and the range angle at entry. The sketch shows the geometry involved and defines a number of parameters used in this analysis.



Geometry of entry maneuver.

From the first part of the trajectory, one obtains entry conditions of V_e, γ_e, and θ_e as a function of the velocity maneuver. To a first-order approximation, deviations in these entry conditions are given by the expressions:

$$\left.\begin{aligned} \delta V_e &= \frac{\partial V_e}{\partial \Delta V}\,\delta\Delta V + \frac{\partial V_e}{\partial \eta}\,\delta\eta \\ \delta\gamma_e &= \frac{\partial \gamma_e}{\partial \Delta V}\,\delta\Delta V + \frac{\partial \gamma_e}{\partial \eta}\,\delta\eta \\ \delta\theta_e &= \frac{\partial \theta_e}{\partial \Delta V}\,\delta\Delta V + \frac{\partial \theta_e}{\partial \eta}\,\delta\eta \end{aligned}\right\} \quad (1)$$

where $\delta\Delta V$ and $\delta\eta$ are, respectively, the magnitude and direction errors in making the velocity maneuver.

From the second part of the trajectory, for a given atmosphere and vehicle, one can obtain range angle as a function of entry conditions. Using the first-order approximation as before, we have

$$\delta\theta_L = \frac{\partial\theta_L}{\partial\theta_e}\,\delta\theta_e + \frac{\partial\theta_L}{\partial V_e}\,\delta V_e + \frac{\partial\theta_L}{\partial\gamma_e}\,\delta\gamma_e \tag{2}$$

Since the altitude of entry is also an entry condition, we might expect this to appear in equation (2); and if the initial point of the trajectory is specified as a particular time, this would be true. However, we have specified the end point of the descent trajectory as that point at which the trajectory passes through the entry altitude, regardless of whether the vehicle is in a perturbed trajectory or not. As a result, of course, the partials of equation (2) are not dependent on altitude, but must be calculated with this particular starting condition.

Equations (1) and (2) combined give the range angle error due to errors in making the velocity corrections.

The errors in range angle due to atmospheric variations were obtained by numerical integration of the equations of motions, starting with a fixed set of entry conditions and using different atmospheres. A comparison of θ_L for the different atmospheres then determines the deviation due to atmosphere, $\delta\theta_{L_{atmos}}$.

Finally, the first-order error due to lift-to-drag variations was determined as

$$\delta\theta_{L(L/D)} = \frac{\partial\theta_L}{\partial(L/D)}\,\delta\,\frac{L}{D} \tag{3}$$

The error sources, $\delta\Delta V$, $\delta\eta$, $\delta(L/D)$, and the atmospheric variations are independent. The overall 3σ expected landing site error can then be obtained by adding the individual 3σ errors on a root-sum-square basis, giving

$$(\delta\theta_L)_{total_{3\sigma}} = \left[\begin{aligned} &\left[\left(\frac{\partial\theta_L}{\partial V_e}\frac{\partial V_e}{\partial\Delta V} + \frac{\partial\theta_L}{\partial\gamma_e}\frac{\partial\gamma_e}{\partial\Delta V} + \frac{\partial\theta_L}{\partial\theta_e}\frac{\partial\theta_e}{\partial\Delta V}\right)\delta\Delta V_{3\sigma}\right]^2 \\ &+ \left[\left(\frac{\partial\theta_L}{\partial V_e}\frac{\partial V_e}{\partial\eta} + \frac{\partial\theta_L}{\partial\gamma_e}\frac{\partial\gamma_e}{\partial\eta} + \frac{\partial\theta_L}{\partial\theta_e}\frac{\partial\theta_e}{\partial\eta}\right)\delta\eta_{3\sigma}\right]^2 \\ &+ \left[\frac{\partial\theta_L}{\partial(L/D)}\,\delta(L/D)_{3\sigma}\right]^2 + \left[\delta\theta_{L_{atmos}}\right]^2 \end{aligned}\right]^{1/2} \tag{4}$$

The quantity $\delta\theta_{L_{atmos}}$ was obtained from the expected extremes of the possible atmospheres, and it was felt that it could be considered as a 3σ number; thus it is of the appropriate size to combine with the other error sources.

COMPUTER SIMULATION

In order to determine the numbers to be used in equation (4), two simulation programs were written for use on an IBM 7094 computer, one for each portion of the trajectory. It was assumed that the descent trajectory and entry were posigrade equatorial orbits about Mars, so that only two-dimensional programs need be generated. A brief study of out-of-plane errors and of the effects of orbit inclination are included in the appendices.

Descent Trajectory

The first of these programs considered the descent trajectory, between the velocity maneuver and entry into the Martian atmosphere. This program was used to obtain the descent trajectories, and the partial derivatives of conditions at entry, V_e, γ_e, θ_e, with respect to the magnitude and direction of the corrective maneuvers, ΔV and η (eq. (1)). The procedure used in the program was as follows: First a bus orbit was selected. The orbit altitude was 5000 km at apoapsis and 1000 km periapsis. With the bus orbit selected, a descent trajectory was determined which used the minimum ΔV,[1] and which arrived at an altitude[2] of 244 km with specified values of entry angle, γ_e, and entry location, θ_e. This calculation was repeated for $-180^\circ < \theta_e < 180^\circ$, to allow considerations of landing at any point in the plane of the orbit. Zero range angle was chosen under the periapsis of the bus orbit. The entry angle was also varied from -24° to -16°. The entry velocity, V_e, was determined for each of these trajectories, and plots of ΔV_{min} and V_e vs. θ_e are presented in figures 1 and 2.

For operational reasons, it may be desirable to use a fixed value of ΔV, which will still allow landing at any location. The value chosen, $\Delta V_{min_{max}}$, would be the maximum of the ΔV_{min} shown on figure 1. The particular values used are given in the following table.

γ_e	-16	-20	-24°
$\Delta V_{min_{max}}$	0.94	1.07	1.24 km/sec

Thus, a trajectory was determined for each θ_e and γ_e which used the specified $\Delta V_{min_{max}}$. The corresponding V_e was also determined, and is plotted in figure 3.

[1]It is interesting to note that ΔV_{min} corresponds to the minimum V_e.

[2]The entry altitude was chosen as 244 km since this is just beyond the outer reaches of the highest atmosphere considered. The descent trajectory was stopped and the atmosphere entry was started as the vehicle passed through this altitude.

Having determined the trajectories to be considered, we next compute the partials of θ_e, γ_e, and V_e with respect to ΔV and η. This is done by using the two sets of trajectories just determined and perturbing ΔV and η slightly. The partials are then computed from the perturbed trajectories. Standard elliptical equations were used throughout this program.

Atmospheric Entry

The second program was used to determine the effects subsequent to entry. In this case, a single trajectory was integrated from the desired entry conditions (V_e, γ_e) to touchdown, where V_e and γ_e were obtained from the trajectories determined in the previous section. The vehicle is assumed to have a constant ballistic coefficient and a constant lift-to-drag ratio throughout the entry, with the lift applied in the upward direction. Entries were made using two different values of each of these parameters. The ballistic coefficients used were $m/C_DA = 62.9$ and 23.6 kg/m^2 (0.35 and 0.15 slug/sq ft) and these cases are referred to as the high and low mass cases, respectively. The values of L/D used were 0 and 0.5. The equations used in determining the entry trajectory are those of reference 7.

Thus a family of trajectories was obtained, from which range angle was plotted versus entry flight angle, initial velocity, and L/D. The slopes of these curves were then used to determine the partials of range angle with respect to velocity, flight-path angle, and L/D for a particular atmosphere. In addition, entries into different atmospheres, with all other parameters the same, were compared to obtain the dispersion due to uncertainty about the atmosphere. The atmospheres used in this program are assumed to be exponential above some transition altitude, and convective below. They were based on the JPL atmospheres VM3, 4, 7, and 8 (see ref. 2). It was also assumed that the atmosphere was rotating with the planet and that there was no wind relative to the surface.

Landing Conditions

It is of interest to consider the landing conditions that will exist for the variety of vehicle parameters and entries under consideration. In particular, we would like to be sure that the vehicle becomes subsonic at a sufficiently high altitude, so that the soft-landing system will operate properly.

Accordingly, the altitude at which the Mach number was 0.9 was determined for all entry trajectories and vehicles, and for the two extreme atmospheres. It was found that this altitude did not vary significantly with entry angle or velocity, but did vary with mass, L/D, and atmosphere as shown in the following table:

Mass	Atmosphere	Altitude, km	
		L/D = 0	L/D = 0.5
Low	VM8	5	11
High	VM8	-4	3
Low	VM3	22	27
High	VM3	7	14

From this it can be seen that the high mass vehicle with 0 lift is not practical, since on entering the VM8 atmosphere it will impact before slowing down sufficiently. However, in order to present a complete picture, the results of this case will be presented along with the others.

RESULTS AND DISCUSSION

The results are given in two parts. First the effects of the individual error sources on landing site dispersion will be presented. Secondly, these errors will be combined to give a picture of the overall dispersions.

The four error sources under consideration are, as mentioned earlier, the error in the magnitude of the deorbit impulse, $\delta\Delta V$, the error in the direction of the deorbit impulse $\delta\eta$, the error in the lift characteristic of the vehicle $\delta(L/D)$, and the error in the knowledge of the atmosphere at Mars. The particular values used here for these error sources are

$$\delta\Delta V_{3\sigma} = 0.75 \text{ percent}$$

$$\delta\eta_{3\sigma} = 1.5^{\circ}$$

$$\delta(L/D)_{3\sigma} = 0.05$$

while the atmosphere error represented the extreme error caused by the atmosphere uncertainty. These numbers are 3σ values of the expected error. The first two were obtained from reference 5 (p. 91), and the third number is 10 percent of the maximum L/D of 0.5. The first set of results, showing the effects on the landing site error, $\delta\theta_L$, caused by each of the error sources individually, are plotted in figures 4(a)-(h). The partials plotted in these figures are for entries into a VM8 atmosphere. Partials were also obtained for entries into a VM3 atmosphere, but the curves are quite similar, and are not presented. The dispersion due to atmosphere, however, is the extreme over all four atmospheres.

Several trends are apparent from the curves of figure 4. When ΔV_{min} is used (figs. 4(a)-(d)) with $L/D = 0$, the miss caused by $\delta(L/D)$ is negligible. At $L/D = 0.5$, however, $\delta(L/D)$ can cause significant errors. This increase in miss due to $\delta(L/D)$ with L/D is independent of the vehicle mass. Also, at $L/D = 0.5$, the effect of atmosphere is negligible for $\gamma = -24^{\circ}$ and low mass. It increases somewhat with mass, but much more so with decreasing entry angle. This increase with decreasing entry angle is general for all of the partials, and at the shallower angles any one of the four errors may be the largest, depending on the value of θ_e.

For $\Delta V_{min_{max}}$ (figs. 4(e)-(h)) the same general remarks hold, with the exception of the errors due to $\Delta\eta$. These errors are much larger than any other in the region from $\theta_e = -180$ to $\theta_e = -30$, which is the region where $\Delta V_{min_{max}}$ is considerably greater than ΔV_{min}.

The overall results are shown in figures 5(a)-(h) where the 3σ value of the total landing site error, $\delta\theta_{L_{total}}$, is plotted against entry range. The curves for atmosphere VM8 are the root sum square of the curves in figure 4. The curves for atmosphere VM3 are obtained from data similar to that of figure 4, in which the partials were computed for entry into the VM3 atmosphere, whereas the dispersion due to atmosphere variations was taken exactly from figure 4. Thus, the curves show that there is little difference in the dispersion when partials are calculated for the VM8 or the VM3 atmospheres.

From these curves one can also conclude that: the error is considerably larger for the shallower entry angles, it increases slightly with L/D at the steeper entry angles, and more so at the shallower entry angles, and is nearly independent of mass.

If ΔV_{min} and the steep entry angle $\gamma = -24^\circ$ are used, the 3σ entry error will be less than 1.5° for any entry range, and less than 1° for some entry ranges. Using $\Delta V_{min_{max}}$, however, may lead to a restriction in the choice of entry ranges, since the error is less than 1.5° for only these entry ranges between $\theta_e = -30^\circ$ and $\theta_e = -120^\circ$. If it is desired to remove this restriction, some control system will be needed. Since the only large error component in this region is that due to $\delta\eta$ (figs. 4(e)-(h)), it is sufficient to control only this error source. Thus, a system which measured η, and subsequently used a vernier thrust to correct for errors in η would be sufficient.

It is interesting that in the region of $\theta_e = 0$, the error associated with $\Delta V_{min_{max}}$ is less than the error associated with ΔV_{min}. The reason for this is that $|\delta\theta_L/\delta\eta|$ is smaller for $\Delta V_{min_{max}}$ and that term is the dominant error source in that region.

Similar data obtained for other orbits of different eccentricity indicate that the same general trends occur, and that similar conclusions can be drawn.

CONCLUSION

It has been shown that a landing vehicle entering Mars from a bus orbit with entry angles of -20° to -24° will land within about 1.5° of the desired landing site without the use of a range control system after deorbit. It has also been shown that at present our knowledge of the location of the desired landing site is approximately 1° to 2°, so that a landing site accuracy of that size appears quite reasonable.

If the minimum ΔV is used for the deorbit maneuver, then any location in the orbital plane can be reached with that accuracy. If, however, a fixed ΔV, equal to the maximum of the possible minimum ΔV is used, then the landing sites which can be reached with an accuracy of 1.5° are limited. The landing sites available, for this condition, are those between 30° before perigee of the bus orbit and 120° after perigee. If it is desired to use the $\Delta V_{min_{max}}$, and still require that any landing site be achievable, some corrective scheme must be used. An adequate scheme would be one that would reduce the error in the direction of the velocity correction by a factor of about 6, from 1.5° to 0.25°, 3σ. The other error sources are sufficiently small that they need not be corrected.

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APPENDIX A

EFFECT OF ORBIT INCLINATION

The main study has assumed that the vehicle will be entering the Martian atmosphere from an equatorial posigrade orbit. The effect of different inclination orbits will be discussed here. There are two different effects to consider.

First, as the inclination varies, the entry velocity with respect to the atmosphere will vary, causing variations in the flight time, and a resultant in-plane error of the landing site. The two extremes are posigrade and retrograde orbits, and the dispersion due to different atmospheres was studied for these two cases. For entry angles of -20°, this dispersion was greater for the retrograde orbits by about 0.2°. The change in dispersion was less for steeper entry angles. This is only a modest change so that the general results shown in the text apply for any inclination angle.

Second, any variation in time of entry will cause a landing site error due to the rotation of the planet. For an equatorial orbit, either posigrade or retrograde, this effect has been considered, since the landing site has been computed in Martian-fixed coordinates. For inclined orbits, however, there will be an out-of-plane component, with a maximum for $i = 90^{\circ}$. The maximum deviation in time of entry due to atmospheric variations is about 150 sec, with a corresponding maximum lateral miss of about 0.6°.

Thus, the effect of orbit inclination on landing site error is relatively small.

APPENDIX B

OUT-OF-PLANE ERRORS

The main study considers only the affects of in-plane errors. However, a brief study of out-of-plane errors was also made. Two possible out-of-plane sources were considered. The first was an error in the direction of the velocity maneuver, and the second was an error in the roll attitude of the vehicle during the entry.

Velocity Manuever Error

If the velocity maneuver has an out-of-plane error, the plane of the actual descent orbit will have an angle of inclination $i = \sin^{-1}(\Delta V_y/V_d)$, where ΔV_y is the out-of-plane component of the velocity maneuver, and V_d is the velocity of the descent orbit just subsequent to the maneuver. We then further assume that the vehicle remains in the new plane. The out-of-plane distance at the landing site due to this inclination is

$$Y_L = r \sin i \sin(\theta_L - \theta_e)$$

where $(\theta_L - \theta_e)$ is the angular distance traveled along the descent trajectory to touchdown, and r is the radius of the descent trajectory at the point where it passes over the landing site. We are interested in obtaining an approximation of the maximum value of Y_L. In so doing, we should use a large value for r, such as the entry radius, $r = 3472$ km. We can write $\sin i$ as $\Delta V/V_d \sin \delta\xi$, where $\delta\xi$ is the out-of-plane component of the directional error $\delta\eta$; and we can use the maximum for $\sin(\theta_L - \theta_e)$, namely, $\sin(\theta_L - \theta_e) = 1.0$.

With these assumptions and for $\delta\xi = \delta\eta = 1.5^\circ$

$$Y_{L_{max}} = 3472 \frac{\Delta V}{V_d} (0.026)$$

Converting this to an angular displacement over the Martian surface, in degrees, we have

$$\psi_{L_{max}} = \frac{3472(\Delta V/V_d)(0.026)}{3332} 57.3 \approx 1.5 \frac{\Delta V}{V_d}$$

For trajectories with ΔV_{min}, the maximum value of $\Delta V/V_d$, considering all entry ranges and angles, was $\Delta V/V_d = 0.5$. For trajectories with $\Delta V_{min_{max}}$ this maximum value was $\Delta V/V_d = 0.66$. The resulting maximum out-of-plane error due to the velocity is thus less than 0.75° with ΔV_{min}, and less than 1° with $\Delta V_{min_{max}}$.

Roll Attitude Error

If, however, the out-of-plane error is caused by a constant vehicle roll error φ, there will be a lateral acceleration, f_y, proportional to the vehicle acceleration f; that is

$$f_y = f \sin \varphi$$

Here it is assumed that f is the total vehicle acceleration, and that the vehicle roll axis is perpendicular to the total acceleration vector. The lateral displacement at landing will then be

$$Y_L = \iint_{entry}^{landing} f \sin \varphi \, dt^2 = \sin \varphi \iint_{entry}^{landing} f \, dt^2$$

This integration was performed for two different trajectories. Both had a shallow entry angle, $\gamma = -12^{\circ}$, resulting in very long flight time, and differed only in the atmosphere used. Thus the results give the maximum expected lateral range. The results were that

$$\psi_{L_{max}} = 0.6^{\circ} \text{ for VM3 atmosphere}$$

$$\psi_{L_{max}} = 0.42^{\circ} \text{ for VM8 atmosphere}$$

If the results for the two error sources are combined on an rss basis, the maximum lateral range will be less than 0.9° with ΔV_{min}, and less than 1.2° with $\Delta V_{min_{max}}$.

REFERENCES

1. Roberts, Leonard: Entry Into Planetary Atmospheres. Astronaut. Aeron., vol. 2, no. 10, Oct. 1964, pp. 22-29.

2. McLellan, C. H.; and Pritchard, E. B.: Use of Lift to Increase Payload of Unmanned Martian Landers. Proc. AIAA/AAS Stepping Stones to Mars Meeting (Baltimore, Md.), Mar. 28-30, 1966, pp. 351-356.

3. Pragluski, W. J.: Flight Mechanics of Unmanned Landers. Proc. AIAA/AAS Stepping Stones to Mars Meeting (Baltimore, Md.), Mar. 28-30, 1966, pp. 107-115.

4. Anon.: Comparative Studies of Conceptual Design and Qualification Procedures for a Mars Probe/Lander. Final Report: Volume V, Subsystem and Technical Analysis, Book 1, Trajectory Analysis: AVSSD-0006-66-RR, Avco Corp. (Contract NAS 1-5224), May 11, 1966.

5. Slipher, Earl C.: The Photographic Story of Mars. Northland Press, Flagstaff, Ariz. and Sky Pub. Co., Cambridge, Mass., 1962.

6. Nicks, Oran W.: Review of Mariner IV Results. NASA SP-130, 1967.

7. White, John S.: Investigation of the Errors of an Inertial Guidance System During Satellite Re-entry. NASA TN D-322, 1960.

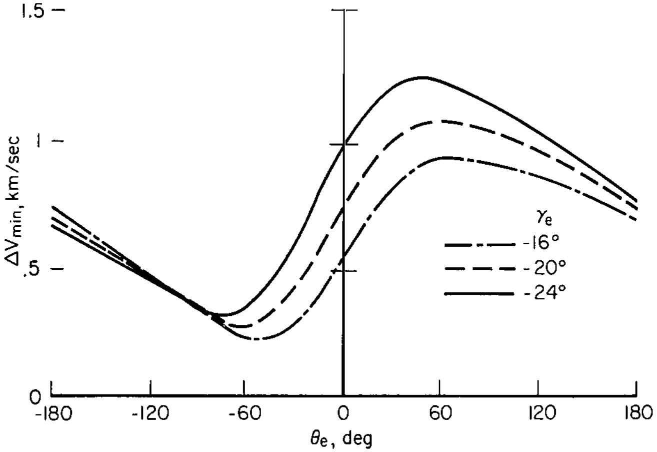


Figure 1.- Minimum ΔV as a function of entry range and flight-path angle.

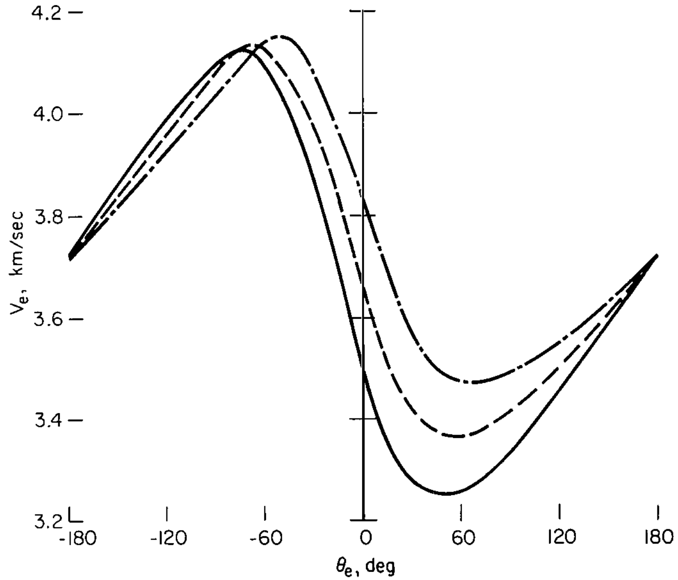


Figure 2.- Entry velocity associated with ΔV_{min} as a function of entry range and flight-path angle.

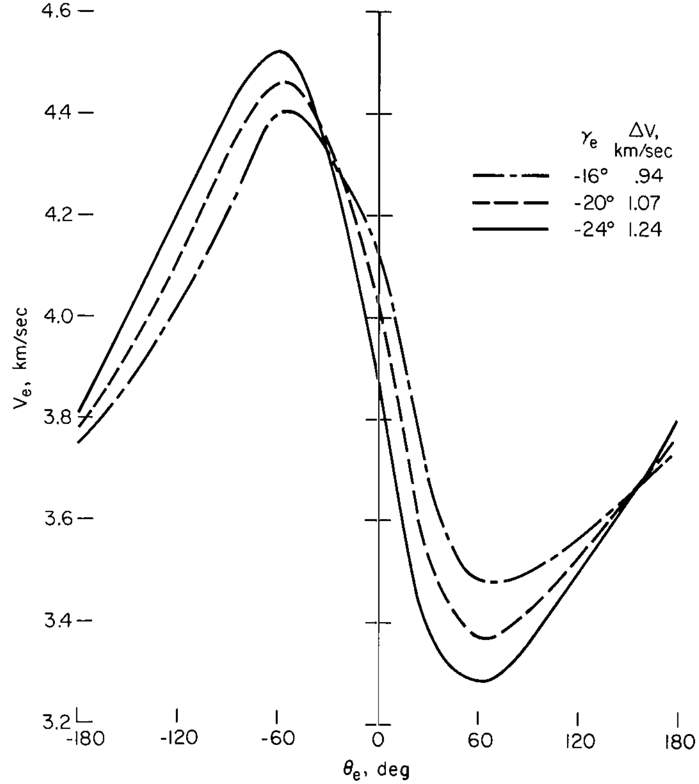
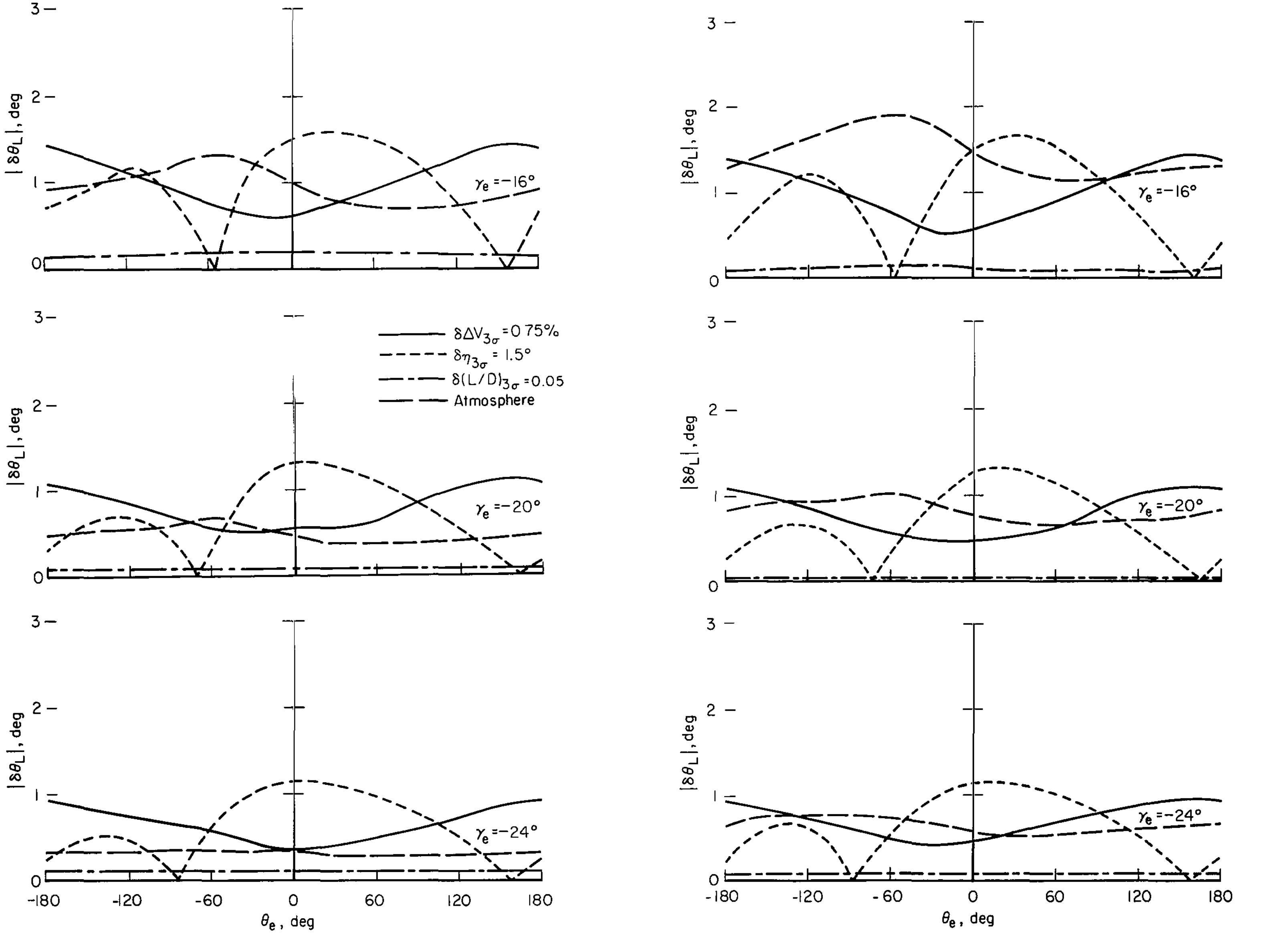


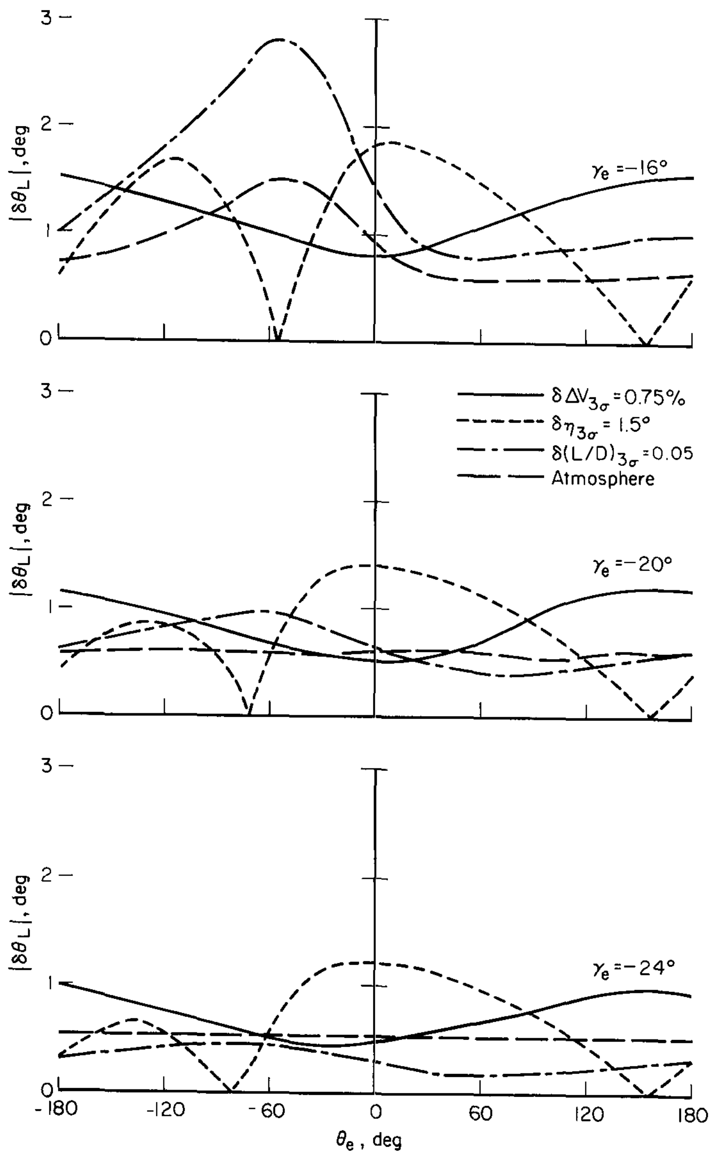
Figure 3.- Entry velocity associated with the maximum ΔV_{min} as a function of entry range and flight-path angle.



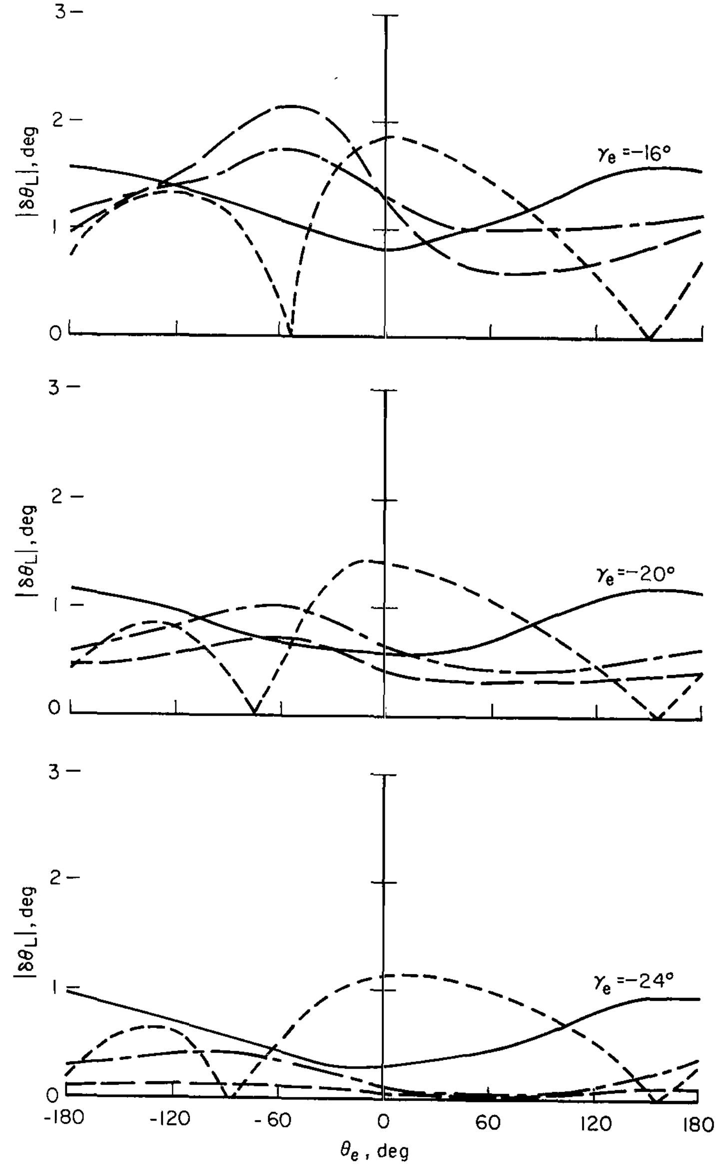
(a) $L/D = 0$, high mass, ΔV_{min}.

(b) $L/D = 0$, low mass, ΔV_{min}.

Figure 4.- Landing site errors due to individual error sources as a function of entry range and flight-path angle.

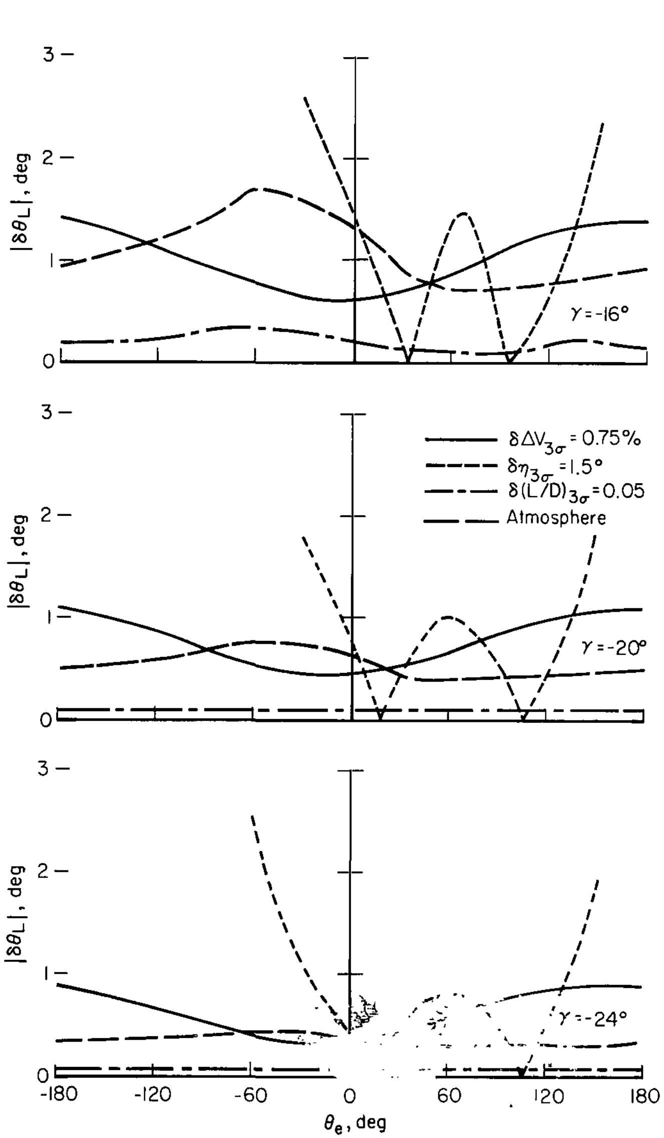


(c) L/D = 0.5, high mass, ΔV_{min}.

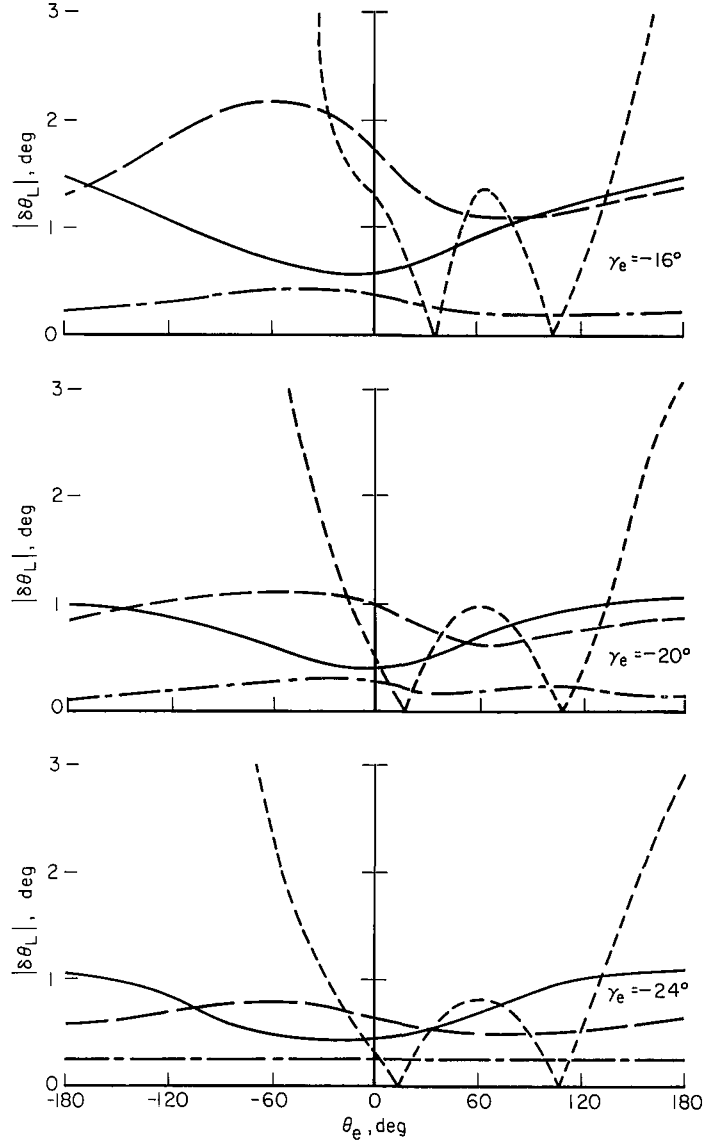


(d) L/D = 0.5, low mass, ΔV_{min}.

Figure 4.- Continued.

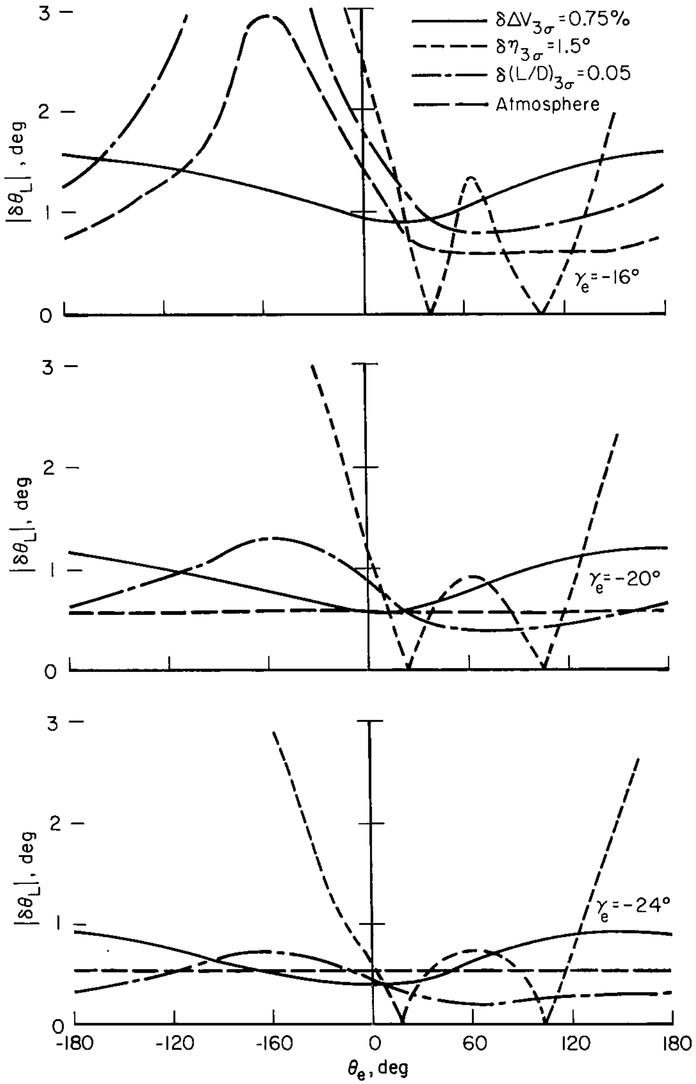


(e) L/D = 0, high mass, $\Delta V_{min_{max}}$.

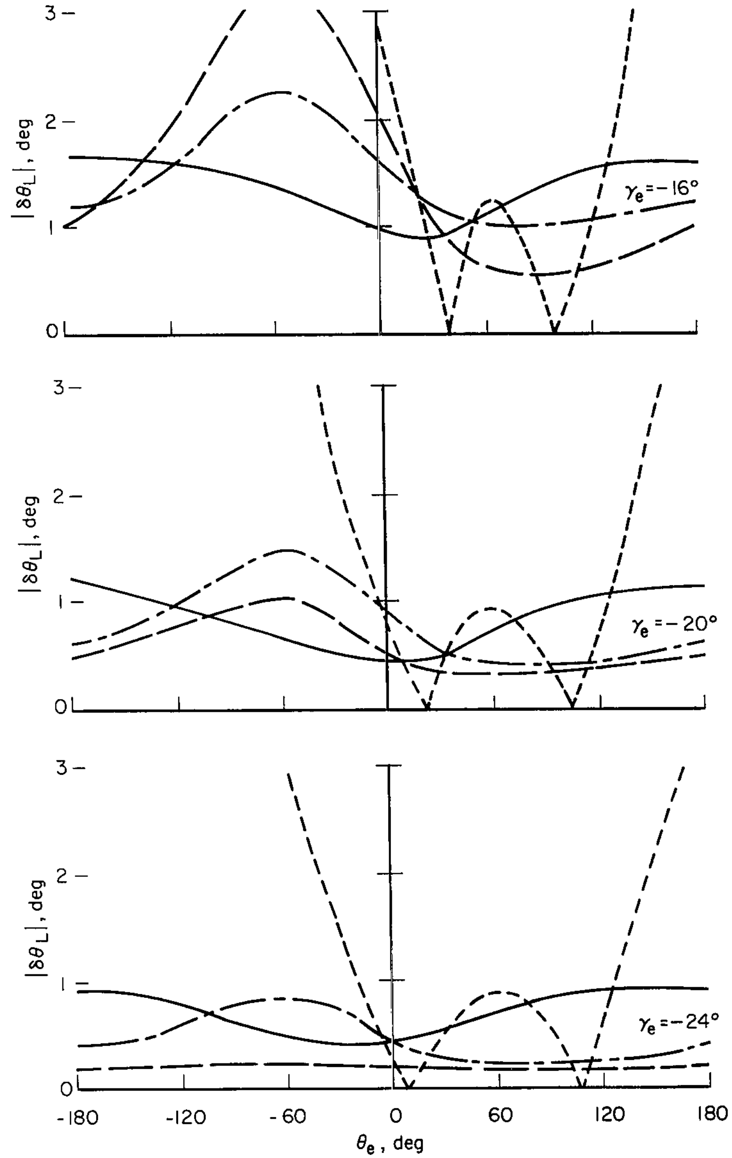


(f) L/D = 0, low mass, $\Delta V_{min_{max}}$.

Figure 4.- Continued.

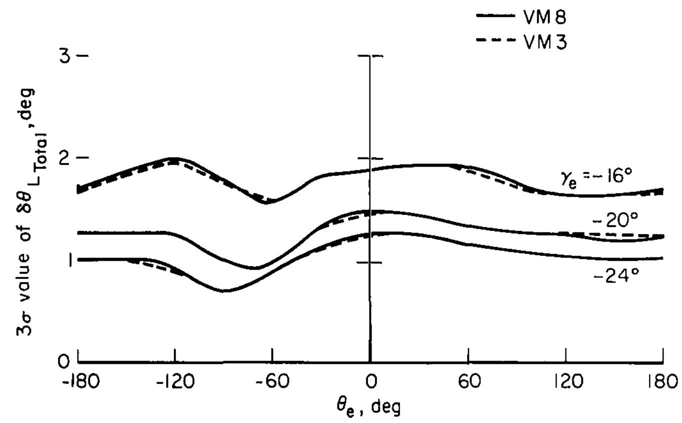


(g) L/D = 0.5, high mass, $\Delta V_{min_{max}}$.

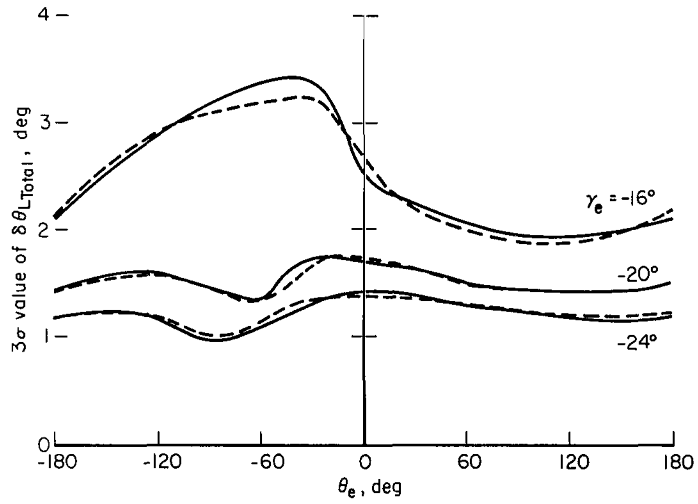


(h) L/D = 0.5, low mass, $\Delta V_{min_{max}}$.

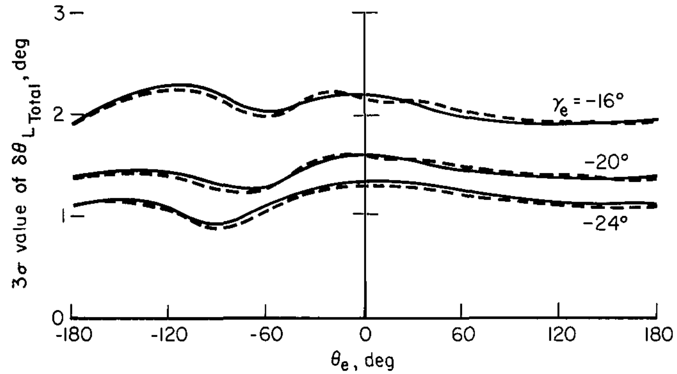
Figure 4.- Concluded.



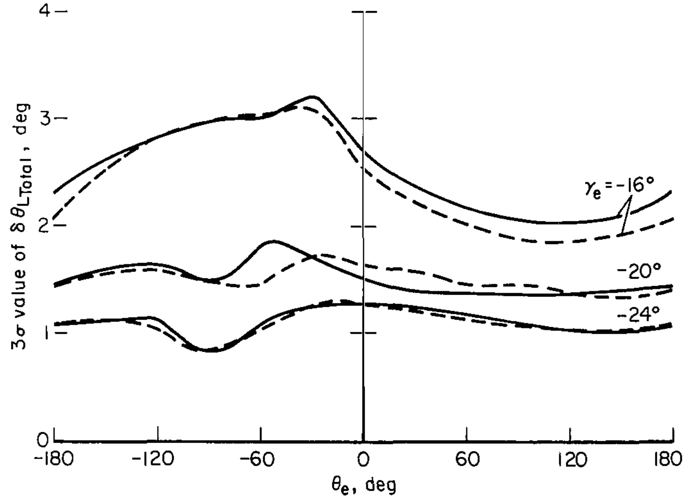
(a) L/D = 0, high mass, ΔV_{min}.



(c) L/D = 0.5, high mass, ΔV_{min}.

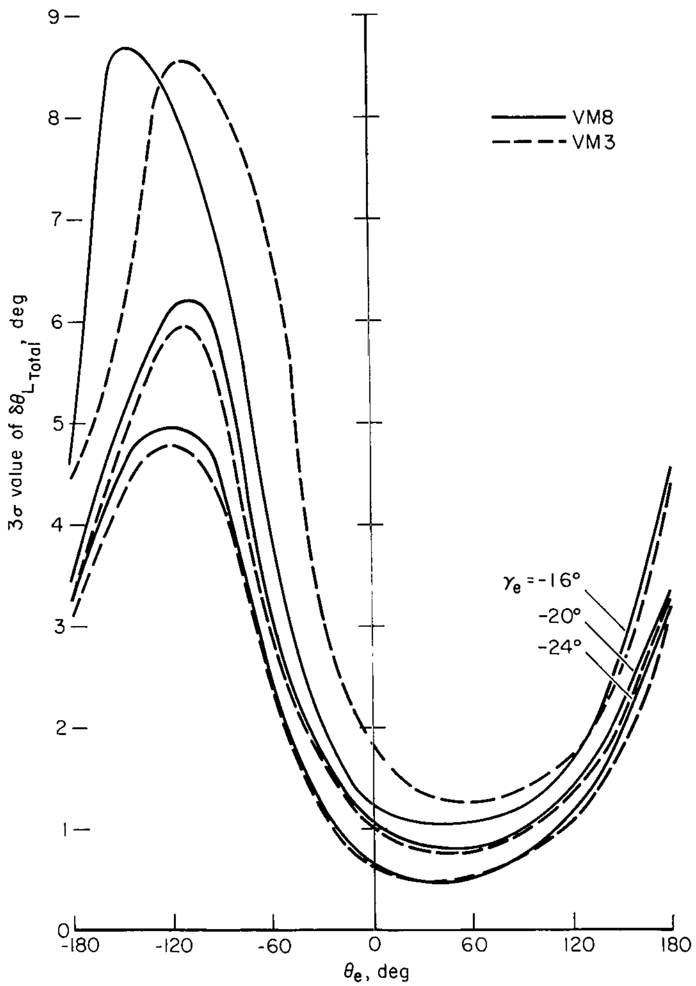


(b) L/D = 0, low mass, ΔV_{min}.

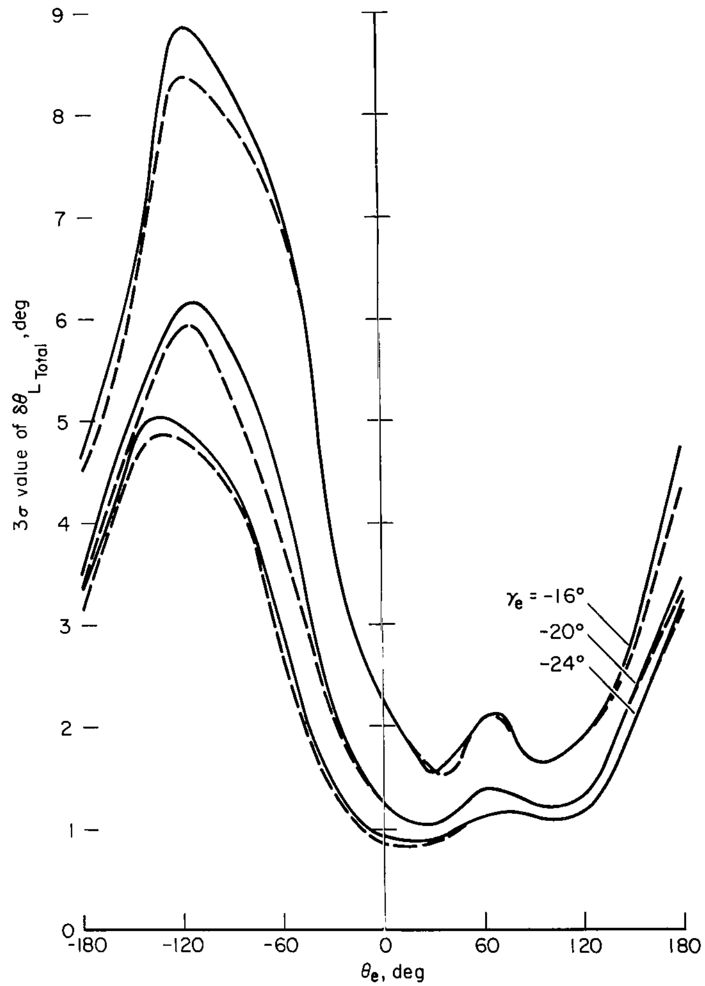


(d) L/D = 0.5, low mass, ΔV_{min}.

Figure 5.- Total landing site error as a function of entry range and flight-path angle.



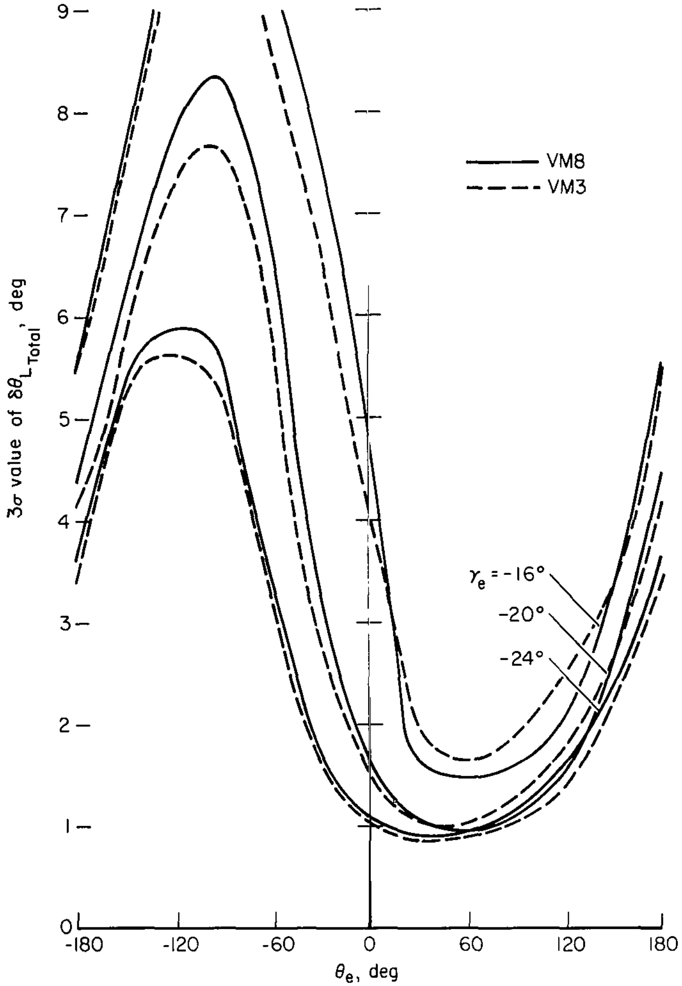
(e) L/D = 0, high mass, $\Delta V_{min_{max}}$.



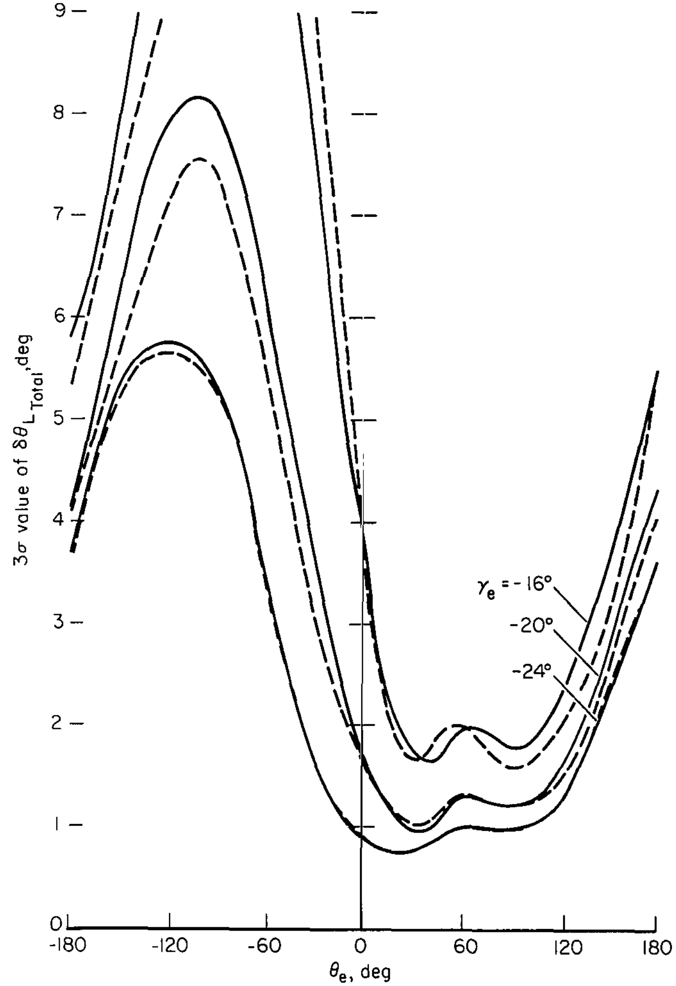
(f) L/D = 0, low mass, $\Delta V_{min_{max}}$.

Figure 5.- Continued.

NASA-Langley, 1968 — 30 A-2561



(g) L/D = 0.5, high mass, $\Delta V_{min_{max}}$.



(h) L/D = 0.5, low mass, $\Delta V_{min_{max}}$.

Figure 5.- Concluded.

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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