Mars Surface-to-Orbit Vehicles for Sample Return Missions

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THE Langley Research Center and the Jet Propulsion Lab. have recently conducted a joint study to define conceptual system designs for Mars surface sample return (MSSR) missions. A study to evaluate some of the factors which affect the performance of the Mars surface-to-orbit launch vehicle is the subject of this Note. The importance of having a high-performance Mars surface-to-orbit launch vehicle is underscored by the fact that this vehicle must be boosted to the vicinity of Mars and then landed on the surface of the planet. The Mars launch vehicle mass will thus have a substantial impact on the mass and systems requirements for all mission phases which are upstream from the Mars launch phase. In Ref. 1 it was shown that single-stage, dual-burn vehicles (DVB's) launched to circular orbits have higher performance efficiencies than continuous-burn vehicles (CBV's), and in the present Note it is shown that the efficiencies of the DBV's are further increased for large values of thrust-to-launch-mass ratios (T/M). In this study, the effects of T/M are evaluated in terms of both velocity losses and inert engine mass penalties. Velocity-loss data for the DBV's suggest a vehicle design which may significantly reduce the vehicle guidance and control requirements. The study utilizes an upgraded Mars density profile.

The trajectory program of Ref. 2 was adapted for use in this study. This program employs a steepest-ascent technique to optimize vehicle angle of attack and produce maximum burnout mass. Initial flight-path angle and coast periods between burn phases are adjusted to achieve this result. The program employs a spherical rotating planet model with an atmospheric density profile which varies with altitude. The atmospheric density profile used is shown in Fig. 1. It is essentially the same as the Viking Project mean density profile and includes some data obtained by the recent Mariner IX spacecraft. Much higher density profiles have been used in past studies[1] as indicated in the figure. A ballistic drag coefficient of 2060 kg/m^2, a value representative of Mars ascent vehicles, was assumed in the study. All launches were made eastwardly from the equator.

One measure of a surface-to-orbit launch vehicle's performance is velocity loss—the difference in actual velocity expended by the vehicle and the minimum velocity required to achieve orbit insertion in the ideal case. For a single-stage vehicle, this difference can be written

$$\Delta V = I_{sp} \ln (M_I/M_a)$$

where M_I and M_a denote the in-orbit masses for the ideal and actual cases, respectively. Note that specific impulse (I_{sp}) as used herein denotes impulse-per-unit mass or specific momentum (units of velocity) instead of impulse-per-unit Earth weight (unit of time). Figure 2 presents velocity-loss data as a function of T/M for single-stage (continuous-burn and dual-burn) vehicles launched to Mars circular orbits. The three values of specific impulse shown correspond to 260, 315, and 425 sec. The data illustrate the effects of T/M for the CBV's and DBV's and the effects of the design velocity of the first burn (V_1) for the DBV's. The CBV's are seen to be much more sensitive than the DBV's to orbit altitude. The values of T/M for the most efficient (least velocity loss) CBV's are about 2 g's (Mars $g = 3.71$ m/sec) whereas the T/M for the efficient DBV's are generally greater than 4 g's. The losses of the DBV's appear essentially insensitive to changes in I_{sp}, and the sensitivity of the CBV's is quite small at low values of T/M. For a constant value of $T/M = 5$ g's, the values of V_1 producing minimum velocity losses for the DBV's are seen to increase with orbit altitude.

Figure 3 shows velocity-loss data as a function of circular orbit altitude for single-stage vehicles. In Fig. 3a, losses for a typical CBV are compared with those of two different DBV's. Notice that the velocity loss (Fig. 3a) is nearly constant between 100 and 300 km altitude for the DBV's but velocity loss increases sharply for the CBV. Figure 3b shows curves for the maximum-efficiency vehicles of the present study together with data from Ref. 1. The significant difference in velocity losses at all altitudes

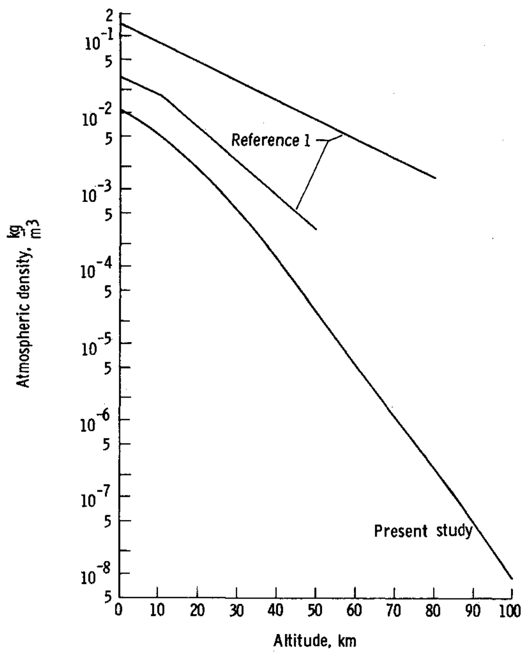

Fig. 1 Mars atmosphere density profiles.

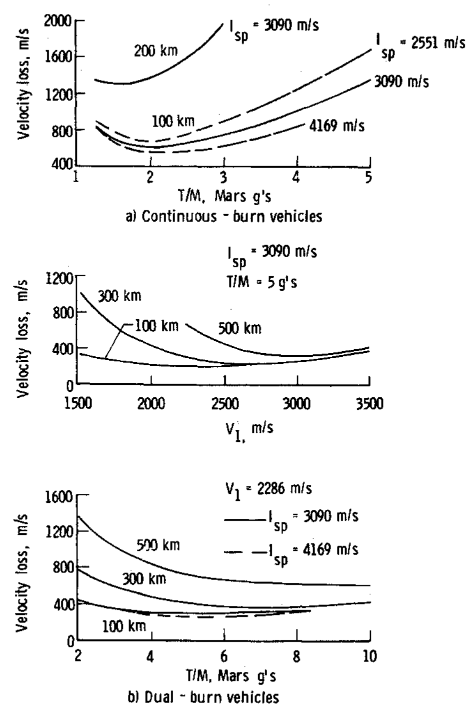

Fig. 2 Surface-to-circular-orbit velocity losses as function of vehicle parameters.

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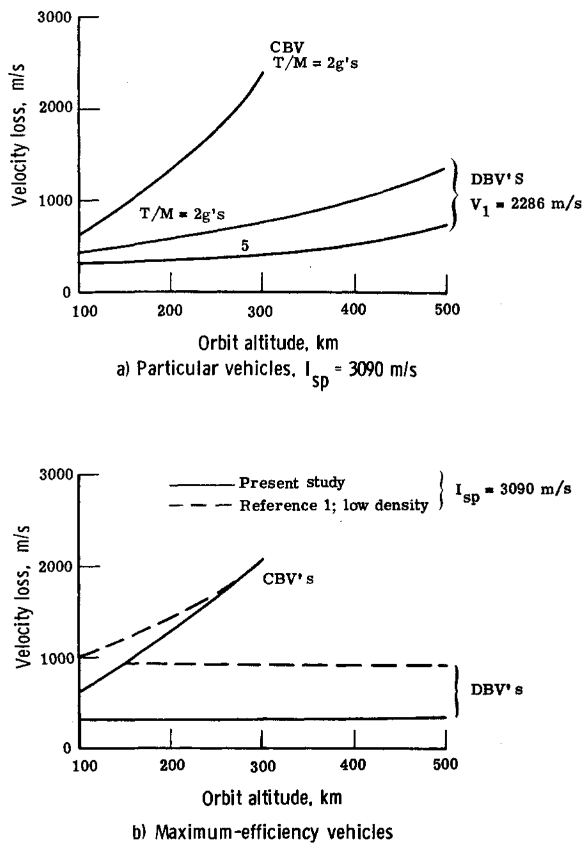

Fig. 3 Surface-to-circular-orbit velocity losses as function of orbit altitude.

between the DBV's of the present study and those of Ref. 1 is due primarily to the fact that the higher T/M values were not considered in that paper.

Table 1 compares the altitude at the beginning of the second burn phase and velocity losses between single-stage and two-stage DBV's launched to circular orbits. These data show that essentially all the velocity loss occurs prior to initiation of the second burn phase. Thus for an optimum trajectory, the first stage boosts the vehicle to about the desired orbit altitude where the second burn provides essentially a horizontal impulse for orbit insertion. This interesting and important result points to a mass-saving launch vehicle design which would employ a guided first stage and an unguided second stage which would be stabilized by spinning it up just prior to second-stage ignition.

Because the results of Figs. 2 and 3 show that velocity losses are very sensitive to T/M, it is instructive to compare payload ratios between vehicles utilizing liquid-propellant engines whose inert masses are sensitive to T/M and vehicles utilizing solid propellant. Payload ratio as used here is the ratio of the total injected mass less engine inerts to launch mass. The following relations which are typical for the two types of propellants were used to compute the inert engine and tankage masses:

$$M_I = \begin{cases} 0.10\,M_p & \text{Solid propellant} \\ 0.145\,M_p + 0.00388\,T(\text{sec}^2/\text{m}) & \text{Liquid propellant} \end{cases}$$

where M_p is propellant mass consumed and T is thrust.

Figure 4 presents payload-ratio data as a function of T/M for CBV's, DBV's, and TSV's. Because it is generally impractical to restart solid-propellant engines, no data are shown for solid-propellant DBV's. The main data were generated assuming a value of $I_{sp} = 3090$ m/sec (a typical value for an efficient Earth-storable liquid propellant) for both liquid and solid-propellant systems. On this basis, the solid-propellant CBV's are seen to produce payload ratios which are greater than those produced by the liquid-propellant CBV's and DBV's and which are about equal to those produced by the liquid-propellant TSV's. Data

Table 1 Altitude at second ignition and velocity losses for two-burn vehicles

Orbit altitude (km)	Altitude (km) at second ignition DBV	TSV	First burn velocity loss (m/s) DBV	TSV	Second burn velocity loss (m/s) DBV	TSV	T/M (g's)
100	99.3	99.6	301	294	2	2	5
200	199.6	196.9	348	338	1	6	5
500	497.9	497.5	704	697	2	5	5
500	500.2	500.2	605	598	0	10	10

points are also presented for the solid-propellant systems where a typical value of an efficient solid propellant is assumed. The solid-propellant CBV is seen to lose most of its advantage over the liquid-propellant CBV's when compared on this basis. However, the solid-propellant TSV is still seen to be superior to all the liquid-propellant vehicles. Thus a two-stage solid-propellant vehicle using a spin-stabilized second stage offers an attractive option for the design of a Mars surface-to-orbit launch vehicle.

In summary, a Mars surface-to-orbit launch vehicle study using an upgraded atmospheric density profile and an ascent trajectory optimization program produced the following results. Orbit insertion was achieved with velocity losses less than those found in an earlier study. Values of thrust-to-mass ratio which minimize velocity losses for dual-burn vehicles were found to be significantly greater than the values of T/M for the minimum-velocity-loss continuous-burn vehicle. Two-stage solid-propellant vehicles provide greater payload ratios than liquid-propellant vehicles which are penalized because their engine masses increase with increased values of thrust-to-mass ratio. The use of a two-stage solid-propellant vehicle employing a spin-stabilized second stage offers an attractive design option for a Mars surface-to-orbit launch vehicle.

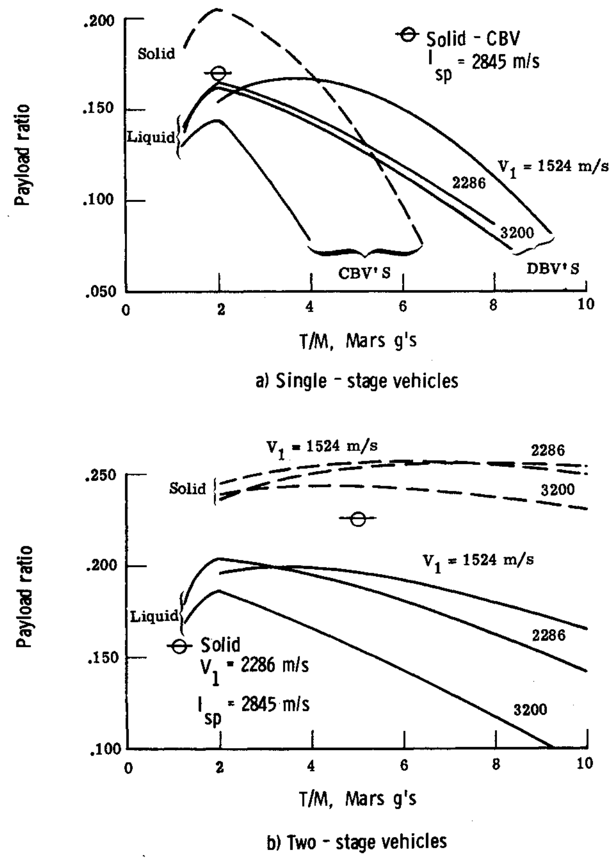

Fig. 4 Payload ratios for vehicles launched to 100 km circular orbits; I_{sp} = 3090 m/sec.

References

[1] Helgostam, L. F., "Requirements for Efficient Mars Launch Trajectories," *Journal of Spacecraft and Rockets*, Vol. 1, No. 5, Sept.–Oct. 1964, pp. 539–544.

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Ignition of Hybrid Rocket Fuels with Fuming Nitric Acid as Oxidant

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Introduction

IGNITION phenomena in hybrid combustion have received little attention and practically no work of fundamental nature has been carried out in this field. Minimum ignition delay (I.D.) coupled with smooth burning is one of the major considerations which govern the choice of a hybrid propellant system. Because start-stop operation is a potential advantage of the hybrid rocket, it is essential that the ignition of fuel with oxidizer should occur very quickly. Most of the work connected with the understanding of the combustion of hybrid propellants involves measurements of regression rates and suggestions of suitable model for hybrid combustion.[1–7] But the work on ignition delay studies, which is very important, has largely been ignored. The hypergolic hybrid propellants using solid fuel and liquid oxidizer are always preferred because the rocket motor can be stopped and restarted at will. The ignition of rocket motors using nonhypergolic hybrid propellants can be accomplished by using different techniques like hot gas, injection of hypergolic fuel or pyrogen igniters but it is difficult to restart the ignition. In order to have an insight into the combustion of hybrid propellants, it is of interest to investigate the ignition delay of these systems.

In this Note, experimental results related to ignition delay measurements of formaldehyde-type hybrid fuels are reported. The effect of different parameters like relative amounts, temperature, additives, and compactness of fuel on ignition delay has been included.

Experimental

Materials

Fuming nitric acid (specific gravity 1.5 g/cm^3) was used as an oxidizer. The hybrid fuels used in the investigation were aniline formaldehyde, o- & m-toluidine formaldehydes, and o-anisidine formaldehyde. All these fuels are hypergolic with fuming nitric acid. Aniline formaldehyde was prepared by mixing equal volumes of ice cooled purified aniline and formaldehyde solution. The mixture was stirred and temperature was not allowed to rise more than 20°C. Precipitated aniline formaldehyde was washed with distilled water and then recrystallized from benzene: o- & m-toluidine formaldehydes and o-anisidine formaldehyde were prepared in similar manner. Ammonium metavanadate, vanadium pentoxide, ammonium dichromate, potassium dichromate, and potassium permanganate were used as additives.

Measurement of Ignition Delay

Ignition delay (I.D.) of the hybrid fuels with fuming nitric acid was measured by the cup-test method.[8,9] However, the results obtained by this method will not tally exactly with the results obtained in actual rocket motor firing but will definitely give an idea about ignition delay of this hybrid propellant, because in actual rocket firing the physical factors like injection velocity, degree of atomization etc. will affect I.D. The weighed quantity of hybrid fuel in the form of powder was taken in a china dish (2″ diam) and to this was added the requisite quantity of fuming nitric acid with the help of a graduated dropping tube. The particle size of hybrid fuel was 150 μ. The experiments were done at room temperature (26 ± 2°C).

Experiments were done to find out the critical amounts of oxidizer and fuel which gave minimum ignition delay in case of each hybrid fuel. This was done by varying the quantity of fuel (0.2 g to 1.5 g) and keeping the quantity of fuming nitric acid fixed (1.0 cm^3).

Ignition delay measurements were done at 10°, 15°, 18°, 20°, and 25°C to study the effect of temperature on I.D. In case of all fuels, the O/F ratio was 1:1 except in case of aniline formaldehyde where the ratio was 1:2.

Various soluble and insoluble additives in fuming nitric acid were tried to reduce the ignition delay. The concentration of additive in acid was 4.5 g in 100 cm^3 of acid.

The effect of compactness of fuel on I.D. was investigated by measuring ignition delay of compressed fuel grains as such obtained by applying pressures of 6720, 8960, 11200, and 13440 lb on surface area 3.143 in.^{2} of the fuel. No powder was placed on the surface of the compressed grain to start ignition.

The experimental results are given in Tables 1–5.

Results and Discussion

Effect of Relative Amounts of Fuel and Oxidizer on Ignition Delay

Experiments to determine the critical relative amounts of fuel and oxidant for minimum ignition delay show (Table 1) that in case of aniline formaldehyde, o-toluidine formaldehyde, m-toluidine formaldehyde, and o-anisidine formaldehyde, the critical amounts are 0.5 g, 1.0 g, 1.0 g, and 0.30 g, respectively, the corresponding critical amount of fuming nitric acid (FNA) for each fuel is 1.0 cm^3. The minimum ignition delay values of aniline formaldehyde, o- and m-toluidine formaldehydes and o-anisidine formaldehyde, are 1.20, 0.30, 3.4, and 1.1 sec, respectively. The results of these experiments are very helpful in designing an injector.

Dependence of Ignition Delay on Temperature

Results given in Table 2 clearly show that ignition delay decreases with the increase in temperature of the system. The reason for this behavior may be attributed to the fact that reaction rates are affected by temperature. As the temperature rises, the reaction rate becomes faster which ultimately reduces

Table 1 Effect of relative amounts of fuel and oxidizer on I.D.[a]

Weight of fuel (g)	Average I.D., sec: Aniline formaldehyde	o-Toluidine formaldehyde	m-Toluidine formaldehyde	o-Anisidine formaldehyde
0.20	x[b]	0.50	x	1.2
0.30	x	0.40	x	1.1
0.50	1.2	0.35	4.1	2.1
1.00	1.7	0.30	3.4	2.5
1.50	1.9	0.40	3.6	...

[a] Quantity of FNA = 1.0 cm^3; temperature = 27 C.
[b] x stands for no ignition.

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