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Mars-Jupiter Aerogravity Assist Trajectories for High-Energy Missions

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A Mars aerogravity assist for high-energy missions (solar flyby and Pluto missions) is considered. It is found that for high-energy missions, Mars aerogravity assist alone will involve very high Earth launch energy and heating rate. The potential of an alternative technique of combining Mars aerogravity assist with Jupiter gravity assist for high-energy missions is examined. The analysis shows that the use of this technique could reduce Earth launch energy and heating rate by 50 and 85%, respectively. Some problems regarding the actual implementation of this alternative technique are also discussed.

Nomenclature

A = $\rho_0 S r_0 C_L^*/2m$
b = R/r_0
C_D = drag coefficient
C_D^* = value of C_D at $(L/D)_{\max}$
C_{D_0} = value of C_D when $C_L = 0$
C_L = lift coefficient
C_L^* = value of C_L at $(L/D)_{\max}$
D = magnitude of drag force
E^* = $(L/D)_{\max}$
e = orbit eccentricity
H_{T_1} = hyperbolic path of vehicle before aerogravity assist (AGA)
H_{T_2} = hyperbolic path of vehicle after AGA
$\mathcal{H}$ = Hamiltonian
J = performance index
K = induced drag factor
k = constant dependent on composition of the atmosphere
L = magnitude of lift force
L/D = lift-to-drag ratio
M = Mach number
m = vehicle mass, kg
q_c = convective heating rate, W/cm^2
q_r = radiative heating rate, W/cm^2
R = radius of planetary spherical atmosphere
R_e = Reynolds number
R_J = radius of Jupiter
r = radial distance from the planet's center
r_n = radius of stagnation region or nose radius, m
r_p = closest approach or periapsis distance
r_r = reference altitude $+r_0$
r_0 = radius of the planet; 6050 and 3483 km for Venus and Mars, respectively
S = effective vehicle surface area, used to define C_L and C_D
S_h = scale height, km
t = time, s
V = planetocentric speed
V_p = velocity of the planet
V_{r_s} = radial component of V_s
V_s = heliocentric speed
V_{θ_s} = transverse component of V_s
V_∞ = planetocentric speed far away from the planet
v = $V/\sqrt{(\mu/R)}$
v_s = $V_s/\sqrt{(\mu/R)}$
v_∞ = $V_\infty/\sqrt{(\mu/R)}$
z = $1/S_h$
γ = planetocentric flight-path angle, deg
ΔV_s = change in V_s
δ = deflection angle, deg
η = ρ/ρ_0
θ = rotation angle for planetary atmospheric flight; also true anomaly
λ = normalized lift control
μ = gravitational constant multiplied by mass of the planet, km^2/s^2
ρ = density, kg/m^3
ρ_0 = value of ρ at reference altitude, kg/m^3

Subscripts

E = Earth
e = atmospheric entry
f = atmospheric exit; final condition
J = Jupiter
M = Mars
S or s = heliocentric or suncentric orbit
V = Venus
0 = initial condition
1 = where H_{T_1} ends
2 = where H_{T_2} starts

Superscripts

$-$ = before AGA
$+$ = after AGA

Introduction

PLANETARY gravity assist (GA) has been used several times for exploring the solar system successfully, including the recent Galileo and Ulysses missions. The concept of GA is based on the phenomenon that energy can be transferred from a planet to a spacecraft or vice versa when the spacecraft's heliocentric trajectory is perturbed as a result of the gravitational field of the planet. The perturbation causes rotation of the planetocentric velocity vector of the spacecraft, which results in a change in the direction and magnitude of the heliocentric velocity of the spacecraft. The change in the magnitude of V_s depends on the deflection angle δ and velocity V_∞ relative to the planet at a far away distance. Large changes in V_s are possible either by large δ and V_∞ [e.g., Jupiter GA (JGA)] or by

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multiple GA, such as Venus-Earth-Earth-gravity-assist (VEEGA) or Venus-Earth-gravity-assist. However, Jupiter is a remote planet and considerably high Earth launch velocity V_{∞_E} is needed to send a spacecraft from Earth to Jupiter to obtain a large V_∞ relative to Jupiter. An Earth launch energy of about 120 km^2/s^2 is needed to send a spacecraft on a solar mission using JGA, which is quite high. Furthermore, multiple GAs may be unable to get a sufficient V_∞ needed at Jupiter for a solar probe or a Pluto mission.

One technique that can be more attractive for solar probes or Pluto missions is the aerogravity assist (AGA), proposed by McRonald and Randolph[1] and Randolph and McRonald.[2] In AGA, a spacecraft with high lift-to-drag ratio (L/D) flies through the atmosphere of a planet to rotate around it using the lift force in addition to the gravitational force. This maneuvering can lead to quite a large velocity increase, ΔV_s, compared with the conventional GA alone, and can be used for high-energy missions (HEM), such as solar flyby or a Pluto mission.

The basic events of an AGA mission are as follows. The spacecraft is launched from a low Earth orbit (LEO) into a heliocentric elliptic transfer orbit and arrives at an appropriate planet with the planetocentric velocity V_∞^-. The spacecraft passes by the planet along a hyperbolic trajectory H_{T_1} with the periapsis inside the planetary atmosphere. At point A (Fig. 1), the vehicle enters the atmosphere and flies within it using aerodynamic lift for the maneuvering. When it leaves the atmosphere at point B, the deflection angle δ has been sufficiently increased (which results in an increase or decrease in V_s^+ depending on the objective), while as a result of the aerodynamic drag V_∞^+ is slightly less than V_∞^-. The spacecraft has now a new hyperbolic trajectory H_{T_2}.

Earlier studies on AGA have used very simple analytical models and assumed a circular and coplanar atmospheric trajectory.[1–4] Recent studies by the authors presented a more realistic mathematical model and obtained optimal trajectories to maximize ΔV_s both without and with a heating rate constraint.[5,6] It was also shown that Mars would be a better candidate for AGA than Venus, firstly because of the lower heating rate during the atmospheric maneuvering, and secondly because of a greater aphelion of the spacecraft after Mars AGA for the same V_∞^-. The studies also concluded that the spacecraft using only AGA for HEM will encounter an enormous amount of heating rate during maneuvering through the Martian atmosphere, which may change the structure of the spacecraft as a result of ablation and spalation.

In the present paper, the subject of AGA for HEM is dealt with in greater detail and an alternative technique combining Mars AGA with JGA (MAGAJGA) is discussed. With the use of this technique, high heating rates can be avoided and travel time as well as V_{∞_E} can be reduced substantially.

It is assumed here that the orbits of the planets are circular and the spacecraft is initially launched from an LEO with an application of ΔV_E.

Equations of Motion

The equations of motion for planar atmospheric flight[7] are

$$\frac{\mathrm{d}r}{\mathrm{d}t} = V \sin\gamma \tag{1a}$$

$$\frac{\mathrm{d}\theta}{\mathrm{d}t} = \frac{V\cos\gamma}{r} \tag{1b}$$

$$\frac{\mathrm{d}V}{\mathrm{d}t} = -\frac{V^2 \rho S C_D}{2m} - \frac{\mu}{r^2}\sin\gamma \tag{1c}$$

$$\frac{\mathrm{d}\gamma}{\mathrm{d}t} = \frac{V\rho S C_L}{2m} - \left(\frac{\mu}{Vr^2} - \frac{V}{r}\right)\cos\gamma \tag{1d}$$

It is assumed here that the planetary atmosphere is not rotating and the atmospheric density varies exponentially with the altitude. Also, when $r > R$, where R is the radius of the sensible atmosphere (see Fig. 1), the flight is Keplerian.

The drag polar used in the analysis is of the form[7]

$$C_D = C_{D_0} + K C_L^n \tag{2}$$

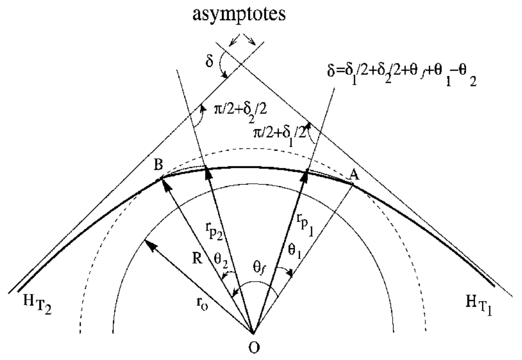

Fig. 1 Geometry of AGA trajectory.

In general, K and n are functions of the Mach number and the Reynolds number. At high Mach numbers, however, K and C_{D_0} are almost independent of the Reynolds number in laminar continuum conditions[7,8]; since that is the case in the major part of the atmospheric trajectory here, we will assume K and C_{D_0} to be constant. Moreover, from the drag polar we can derive the following useful relationships:

$$C_L^* = \left[\frac{C_{D_0}}{(n-1)K}\right]^{1/n} \tag{3a}$$

$$E^* = C_L^*\big/C_D^* = (1/n)(1/K)^{1/n}\left[(n-1)\big/C_{D_0}\right]^{(n-1)/n} \tag{3b}$$

Now one can introduce a new variable $\lambda = C_L/C_L^*$, which represents a normalized lift coefficient. In the following analysis λ will be used as an aerodynamic control parameter.

Although the drag polar is being used for the first time for AGA by the authors, it has been used earlier in aeroassisted orbit transfer and re-entry problems.[9,10] These investigators usually assumed a parabolic drag polar ($n = 2$). However, a parabolic drag polar is basically valid for the subsonic regime, although some experimental results show its validity in the low hypersonic regime as well.[7,11] The velocity of the vehicle in the present case will be higher than those considered in previous studies because the planetocentric trajectory of the vehicle is hyperbolic. There are no experimental results available at very high velocity; hence the drag polar based on the Newtonian theory valid for the hypersonic regime, i.e., $n = 1.5$, will be used.[12] The Newtonian theory is valid for both slender and blunt bodies at very high Mach number when the ratio of specific heats, c_p/c_v, is close to 1 because of the ionization caused by high temperature.[12]

Using C_L or λ as a control parameter corresponds physically to using pitch modulation to shape the trajectory. In this paper, C_L is allowed to assume both positive and negative values. A negative C_L value can be interpreted as resulting either from a negative pitch angle or from a positive pitch angle with the vehicle flying upside down.

The independent variable is changed in this analysis from t to θ, because that makes it easier to solve the problem. One then has to solve three equations of motion instead of four, and adjoint equations are also reduced by one, which saves computing time in determining the optimal solution. The equations of motion in nondimensionalized form with θ as the independent variable are given next for the planar case:

$$\frac{\mathrm{d}h}{\mathrm{d}\theta} = h\tan\gamma \tag{4a}$$

$$\frac{\mathrm{d}v}{\mathrm{d}\theta} = -\frac{A\eta f(n,\lambda)vh}{E^*\cos\gamma} - \frac{b\tan\gamma}{hv} \tag{4b}$$

$$\frac{\mathrm{d}\gamma}{\mathrm{d}\theta} = \frac{Ah\eta\lambda}{\cos\gamma} + 1 - \frac{b}{v^2 h} \tag{4c}$$

By using Eqs. (3) we have $f(n, \lambda) = (1 + 2|\lambda|^{1.5})/3$ for $n = 1.5$, where $f(n, \lambda) = C_D/C_D^*$. The term $\mathrm{d}t/\mathrm{d}\theta$ is not included in the dimensionless equations of motion as time t does not appear explicitly in the equations of motion as well as in the performance index, as we will see later. The dimensionless variables and parameters used in Eq. (4) are defined as follows:

$$h = r/r_0; \qquad v = V/\sqrt{\mu/R}; \qquad b = R/r_0$$

$$\eta = \frac{\rho}{\rho_0} = e^{-z(r-r_r)}; \qquad z = \frac{1}{S_h}; \qquad A = \frac{\rho_0 S r_0 C_L^*}{2m}$$

Heating Rate Constraint

During the flight of the spacecraft in the planetary atmosphere, its kinetic energy decreases. A fraction of this energy is converted into heat absorbed by the vehicle. Here it will be assumed that only the highest temperature in the stagnation region of the vehicle is of concern. To control this temperature, it suffices to control the heating rate in that region. Moreover, it is observed that if in the optimization problem we impose a constraint on the convective heating rate, then the integrated heating and the radiative heating rate will automatically be reduced. Hence in the calculations constraint is put only on the convective heating rate, whereas the maximum radiative heating rate is calculated only for the sake of completeness.

The convective heating rate q_c along the atmospheric trajectory is computed according to the equation[12]

$$q_c = \frac{k\rho^{\frac{1}{2}} V^3}{(r_n)^{\frac{1}{2}}} \tag{5}$$

In the numerical calculations, AGA of Mars has been considered, whose atmosphere is assumed to contain 85% CO_2 and 15% N_2, for heating rate calculations (in reality, the atmosphere of Mars contains 95% CO_2, 5% N_2, but the value of k for 95/5 mixture is not available in the literature).

The value of k for the 85/15 mixture, which is 1.8425×10^{-8} (taken from Ref. 13), should be close to the real value, since even for 50/50 mixture $k = 1.7765 \times 10^{-8}$, which is fairly close to that for 85/15 mixture.

The maximum radiative heating rate is computed according to the equation[14]

$$q_r = F r_n^a \rho^p f(V) \tag{6}$$

where $f(V)$ are tabulated values that are functions of the flight velocity V and the atmospheric composition. The exponents a and p are functions of ρ and V. According to Page and Woodward,[15] at high velocities (e.g., 10 km/s) the radiative heating rate is more or less the same for the atmospheres of Earth, Venus, and Mars. The velocities of spacecraft in the present case will be higher than the aforementioned velocity. Actually, at a very high Mach number the atmospheric gases dissociate and become a partial ionized plasma, and the radiative heating rates for the atmospheres of Earth, Venus, and Mars become approximately equal. Therefore the following values, which are for Earth (air), are used in the analysis:

$$F = 4.736 \times 10^4; \qquad a = 1.072 \times 10^6 V^{-1.88} \rho^{-0.325}; \qquad p = 1.22 \tag{7}$$

The nose radius r_n has been taken as either 0.5 or 1 m. (Randolph and McRonald have used $r_n = 1$ m in Ref. 2, and in the studies of the aeroassisted orbital transfer mission r_n is taken as 1 m as well.) Since only a rough estimate of the heating rate is of concern here, these models will be sufficient.

Performance Index

Let V_s be the final heliocentric velocity of the spacecraft after completing the maneuvering inside the atmosphere through an angle θ_f (Fig. 1). The objective is to maximize V_s subject to the differential constraints represented by the equations of motion and the heating rate constraint. Thus one can define the performance index

$$J = V_s^+ \tag{8}$$

subjected to the differential equation constraints represented by the equations of motion, Eqs. (4). In addition, the heating rate constraint $q_c \le q_{c_{\max}}$ can be included in the optimization problem either analytically or numerically, depending on the type of the problem.

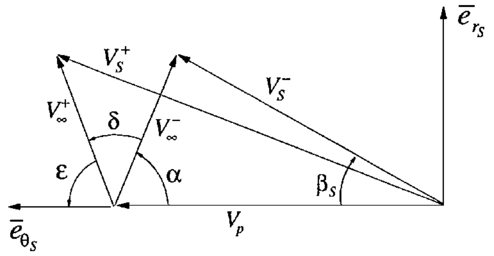

Fig. 2 Vector diagram of velocities for GA or AGA.

Referring to Fig. 2, the value of V_s^+ is given by

$$V_s^+ = \left[\left(V_p + V_\infty^+ \cos\epsilon\right)^2 + \left(V_\infty^+ \sin\epsilon\right)^2\right]^{\frac{1}{2}} \tag{9}$$

where

$$\epsilon = \pi - \alpha - \delta; \qquad V_\infty^+ = \left[V_f^2 - (2\mu/R)\right]^{\frac{1}{2}} \tag{10}$$

The term α is the angle between V_p and V_∞^- (Fig. 2). The last equation in Eqs. (10) can be rewritten in dimensionless form as

$$v_\infty^+ = \left(v_f^2 - 2\right)^{\frac{1}{2}} \tag{11}$$

where all of the velocities have been divided by $(\mu/R)^{1/2}$.

From Fig. 1, one can obtain

$$\delta = (\delta_1/2) + (\delta_2/2) + \theta_f - |\theta_1| - \theta_2 \tag{12}$$

where

$$\theta_i = \tan^{-1}\left(\frac{v_i^2 \sin\gamma_i \cos\gamma_i}{v_i^2\cos^2\gamma_i - 1}\right) \tag{13a}$$

$$\delta_i = 2\sin^{-1}(1/e_i) \tag{13b}$$

$$e_i = \left[\left(v_i^2 - 1\right)^2 \cos^2\gamma_i + \sin^2\gamma_i\right]^{\frac{1}{2}} \tag{13c}$$

with $i = 1$ and 2. Subscripts 1 and 2 denote quantities for hyperbolic planetocentric trajectories H_{T_1} and H_{T_2} at A and B, respectively. The terms θ_1 and θ_2 are the angles between their hypothetical closest approaches, r_{p_1} and r_{p_2}, and lines AO and BO, respectively, as shown in Fig. 1. In previous studies,[1,2] θ_1 and θ_2 were assumed to be zero. However, for more accurate results these quantities should be included in the calculations.

At the entry point we have

$$\theta_e = 0 \qquad \text{and} \qquad h_e = R/r_0 \tag{14}$$

and at the exit θ_f is unspecified and

$$h_f = R/r_0 \tag{15}$$

The optimization of the atmospheric trajectory is performed using Pontryagin's maximum principle. The optimal control problem is solved by determining $\lambda(\theta)$ to maximize V_s^+ subject to the inequality convective heating rate constraint along the atmospheric trajectory for a set of initial conditions determined by the heliocentric velocity at the planetary entry. The problem is a two-point-boundary-value-problem, in which some conditions are specified at the planetary atmospheric entry, whereas the others are specified at the atmospheric exit; the problem is solved by the shooting method.

The optimization method is not discussed here in detail; the interested reader is referred to the previous studies by the authors on this subject.[5,6]

Table 1 Various relevant quantities related to the Pluto mission with $V^{+}_{\infty_M}$ = 14.5 km/s

Relevant quantities	Values of E^*: 5	Values of E^*: 7
V^{-}_{∞}, km/s	18.94	17.60
V_{pp}[a], km/s	19.57	18.28
V_{∞_E}, km/s	11.05	10.30
For $r_n = 1$ m		
$q_{c\max}$ constraint, W/cm^2	634	516
$q_{r\max}$, W/cm^2	480	313
For $r_n = 0.5$ m		
$q_{c\max}$ constraint, W/cm^2	903	729
$q_{r\max}$, W/cm^2	330	215

[a] V_{pp} = velocity at periapsis.

Problems Associated with Mars AGA for High-Energy Missions

McRonald and Randolph[1] suggested $V^{+}_{\infty_M} = 14.5$ km (subscript M stands for planet Mars) for Mars AGA (MAGA) for a Pluto mission; the travel time for this mission would be 6.6 years from Mars to Pluto. It was estimated by the aforementioned authors that this speed of 14.5 km/s would be almost within the reach of the present technology. We have analyzed this scenario using an optimization method with heat constraint, and Table 1 gives the corresponding convective heating rate ($q_{c\max}$, the convective heating rate with respect to the $C_{L\max}$ of the vehicle) as well as the maximum radiative heating rate associated with this mission; it is clear that both $q_{c\max}$ and $q_{r\max}$ are very high. It may not be easy to handle such heating rates for about 8 min without damaging the spacecraft made with currently available materials. On the other hand, heating rates associated with $V^{+}_{\infty_M} = 20$ km/s, required for a solar flyby using MAGA, will be so high that it will be difficult even to calculate them because of the occurrence of ablation and spalation. Therefore, it can be concluded that MAGA alone will not be sufficient for any HEM at this time; however, the potential of AGA cannot be ruled out completely. Therefore, in the following section, some alternative techniques related to AGA will be discussed.

Alternative: MAGAJGA

In this section we will discuss several alternative techniques for HEM. Then these techniques will be compared with respect to the travel time for the Pluto and solar flyby missions.

Pluto Missions

The difficulties associated with using MAGA alone can be avoided by using MAGAJGA. In Fig. 3, a comparison of MAGAJGA and MAGA with respect to travel time is given, which shows that MAGAJGA is considerably superior to MAGA up to $V^{+}_{\infty_M} \approx 15$ km/s in terms of travel time to Pluto. The closest approach r_p taken in Fig. 3 is 6 R_J, and the advantage of MAGAJGA could be more if r_p at Jupiter is lowered from 6 R_J. Since the present technology restricts us from using such a high $V^{+}_{\infty_M}$, a safe $V^{+}_{\infty_M}$ will be 10 km/s or less. In that velocity region, MAGAJGA is significantly advantageous. For example, with $V^{+}_{\infty_M} = 7$ km/s (corresponding to $V_{\infty_J} \approx 10$ km/s), the travel time from Mars to Pluto is around 12 years, whereas in the case of simple MAGA, it is not even possible to reach Pluto with this $V^{+}_{\infty_M}$. One reason for the lower travel time by adding a JGA is because we can get a higher V_{∞_J} for a comparatively lower $V^{+}_{\infty_M}$, as shown in Fig. 4. For a given $V^{+}_{\infty_M}$ we are able to get a V_{∞_J} that can be 1.6 times as high. Another factor is that the heliocentric escape velocity at Jupiter is considerably lower than that at Mars. Hence, even a small V_{∞_J} can reduce the travel time to Pluto considerably.

One can notice in Fig. 3 that, at $V^{+}_{\infty_M} = 10$ km/s, the difference between the travel times of MAGAJGA and MAGA is very high, but this difference diminishes gradually and becomes negligible at $V^{+}_{\infty_M} = 35$ km/s. There are two causes for this. 1) The escape velocity with respect to the sun at Mars is 34.52 km/s. Thus, for $V^{+}_{\infty_M} = 10.37$ km/s, we have, after AGA, V_s equal to the escape velocity with respect to the sun (i.e., parabolic trajectory), but for $V^{+}_{\infty_M} = 11$ km/s, we have a hyperbolic heliocentric trajectory after AGA with $V_{\infty_S} = 8.42$ km/s. Furthermore, V_{∞_S} increases significantly with a relatively small increase in $V^{+}_{\infty_M}$ so that for $V^{+}_{\infty_M} = 15$ km/s we obtain a post-AGA hyperbolic heliocentric trajectory with $V_{\infty_S} = 19.81$ km/s. Hence, we can see in Fig. 3 that for $V^{+}_{\infty_M} > 15$ km/s the travel time difference between MAGAJGA and MAGA is reduced considerably since at this $V^{+}_{\infty_M}$ the values of V_{∞_S} after MAGAJGA and MAGA are quite close. 2) For MAGA we have assumed that the perihelion is at Mars, i.e., $\epsilon = 0$ deg. But we have chosen $r_p = 6R_J$ at Jupiter for MAGAJGA. From Eqs. (11) and (13), when V_{∞_J} increases, δ decreases, and hence the advantage of GA also decreases. If V_{∞_J} continues to increase, then at some point δ will be so small that $\cos\epsilon \approx 0.5$ and V^{+}_{s} will be much smaller than its ideal value corresponding to $\cos\epsilon = 1.0$. Therefore, in Fig. 3, $V^{+}_{\infty_M} = 35$ km/s is a point where, as a result of high V_{∞_J}, V^{+}_{s} after MAGAJGA is much smaller than that of the ideal value and the difference between the travel times with MAGAJGA and MAGA is negligible.

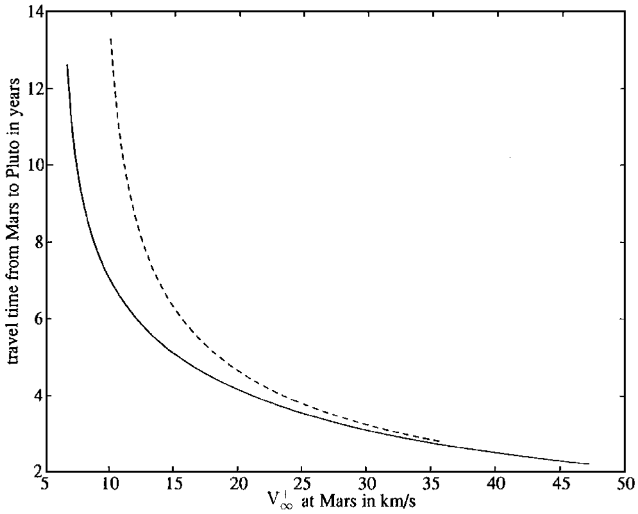

Fig. 3 Travel time from Mars to Pluto vs $V^{+}_{\infty_M}$: ——, MAGAJGA or VGAMAGAJGA when $r_p = 6.0\,R_J$ at Jupiter and – – –, MAGA only, with perihelion at Mars.

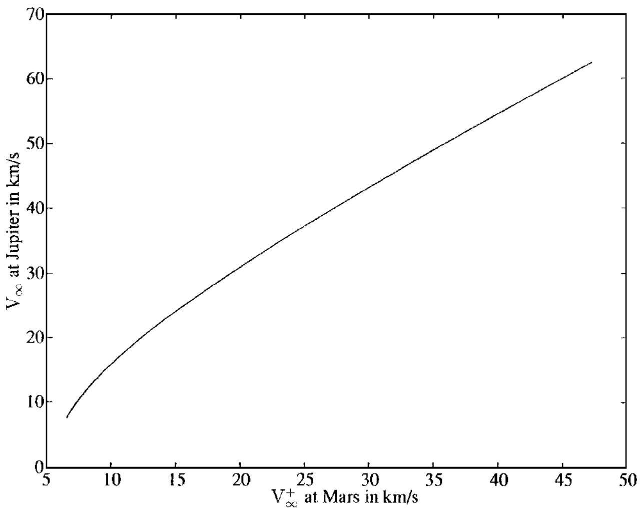

Fig. 4 Variation of $V^{+}_{\infty_M}$ with respect to V_{∞_J} after MAGA or VGAMAGA.

Table 2 shows that when $V^{+}_{\infty_M} = 12$ km/s the travel time for a Pluto mission using MAGAJGA can be seven years from Mars with substantially smaller heating rates and a V_{∞_E} that is considerably less than what was suggested by Randolph and McRonald (64% less in the case of MAGAJGA). If we use MAGA alone for the same $V^{-}_{\infty_M} = 12$ km/s, it is found that travel time will be around 14 years, which is quite high. One can also notice that a Pluto mission in seven years using JGA needs around double the V_{∞_E} compared with MAGAJGA.

Table 2 Comparison of JGA, MAGAJGA, and VGAMAGAJGA for Pluto missions[a]

Type of mission	$V_{\infty J}$, km/s	$V^{+}_{\infty M}$, km/s	$V^{-}_{\infty M}$, km/s	$V_{\infty E}$, km/s	r_p, R_J	$q_{c\max}$, W/cm^2	$q_{r\max}$, W/cm^2
Pluto mission, 8.5 yr from Mars							
JGA	13.1	N/A	N/A	10.7	5.0	N/A	N/A
MAGAJGA	13.1	8.65	10.2	5.6	6.0	120	10
VGAMAGAJGA	13.1	8.65	10.4	4.2	6.0	130	10
Pluto mission, 7 yr from Mars							
JGA	15.7	N/A	N/A	11.7	5.0	N/A	N/A
MAGAJGA	15.7	9.86	12.0	6.6	6.0	200	65

[a] $E^* = 7$ and $r_n = 1$ m; heating rate constraint is $q_{c\max}$; for $r_n = 0.5$ m the values of $q_{c\max}$ are 170, 183, and 283, respectively.

Table 3 Potential of JGA, MAGAJGA, and VGAMAGAJGA for solar missions[a]

Type of mission	i_s, deg	$V_{\infty J}$, km/s	$V^{-}_{\infty M}$, km/s	$V_{\infty E}$, km/s	r_p, R_J	$q_{c\max}$, W/cm^2	$q_{r\max}$, W/cm^2
Solar probe at 4 R_S							
JGA	0.0	12.0	N/A	10.3	11.1	N/A	N/A
	90	13.1	N/A	10.7	7.4	N/A	N/A
MAGAJGA	0.0	12.0	9.8	5.5	8.1	110	10
	90	13.1	10.2	5.6	5.8	120	6
VGAMAGAJGA	0.0	12.0	10.1	3.8	8.1	110	10
	90	13.1	10.4	4.2	5.8	130	6

[a] $E^* = 7$, and $i_s =$ vehicle orbit inclination; heating rate constraint is $q_{c\max}$; for $r_n = 0.5$ m the values of $q_{c\max}$ in ascending order are 155, 170, and 183, respectively.

Solar Flyby Missions

MAGAJGA can also be used effectively for solar flyby missions, with the perihelion at four solar radii (R_S), without having extremely high heating rates, as would be encountered if MAGA alone is used. Table 3 shows that $V^{-}_{\infty M} = 10.2$ and 9.8 km/s ($V^{+}_{\infty M} = 8.65$ and 8.15 km/s) will be required for the solar polar ($i_s = 90$ deg) and the ecliptic flyby mission ($i_s = 0$), respectively, by using MAGAJGA, assuming $E^* = 7$ during the maneuvering through the Martian atmosphere. It can also be seen in Table 3 that the associated heating rates are substantially lower compared with those with MAGA alone.

Venus-GA-MAGAJGA

Venus-AGA-MAGAJGA can be very complicated, since it involves two planetary atmospheric flybys, i.e., first by Venus and then by Mars. Moreover, Venus-AGA is advantageous only when $V^{-}_{\infty V} \geq 10$ km/s, which leads to high heating rates as was shown in Refs. 5 and 6. Nevertheless, Venus GA (VGA) along with MAGAJGA (VGAMAGAJGA) could be used for HEM as it will reduce $V_{\infty E}$. For example, in Table 2 note that a travel time of 8.5 years from Mars to Pluto needs $V_{\infty E} = 5.6$ km/s when MAGAJGA is used, but with VGAMAGAJGA it is reduced to 4.2 km/s for the same travel time. Figure 5 shows that the maximum value of $V^{-}_{\infty M}$ that can be achieved by utilizing VGA is 10.4 km/s and it can be sufficient for a solar flyby using a subsequent MAGAJGA with $E^* \geq 7$. VGAMAGAJGA, however, may impose restrictions on launch opportunities compared with MAGAJGA.

Discussion

MAGAJGA and VGAMAGAJGA have been discussed and compared, and it was shown that they can be alternate techniques for HEM with low heating rates and low $V_{\infty E}$. But one might raise some objections to both MAGAJGA and VGAMAGAJGA missions, which are discussed next.

1) A closer flyby of Jupiter, to get large gravitational bending for achieving a higher V_s, subjects the spacecraft to considerable electron and proton bombardments, hence requiring extra shielding for the electronic parts.

The preceding objection is valid but needs to be examined carefully. Table 4 presents closest approaches of various spacecraft sent to Jupiter for GA to date from Pioneer 10 to Ulysses.[16,17] Of these spacecraft, only Pioneer 11 showed damage to its electronic parts as a result of intense radiation because its r_p was 1.6 R_J (i.e., an altitude of 0.6 R_J). In our analysis that led to Fig. 3, $r_p = 6\ R_J$ was used, which is higher than the majority of the missions sent to Jupiter. Therefore the Pluto mission under discussion may not require greater shielding than that used in the past. Table 3 shows that for the case of the solar polar flyby (perihelion = 4 R_S) r_p will be 5.8 R_J, which is only slightly smaller than that of the Pluto mission discussed earlier. Furthermore, r_p was equal to 6.28 R_J in the recent Ulysses mission, which is slightly higher than the suggested 6 R_J. Hence some additional shielding to electronic parts will prevent the damage from the radiation.

Table 4 Closest approaches of spacecrafts sent for JGA

Name of spacecraft and launch	Date of launch	r_p/R_J
Pioneer 10	March 1972	2.86
Pioneer 11	April 1973	1.60
Voyager 1	September 1977	4.80
Voyager 2	August 1977	10.05
Ulysses	October 1990	6.28

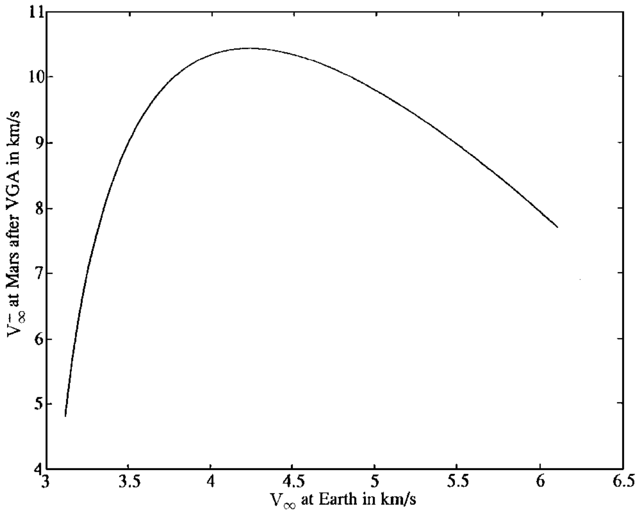

Fig. 5 $V^{-}_{\infty M}$ **vs** $V_{\infty E}$ **after VGA.**

2) The launch window for a Pluto mission using JGA comes after relatively longer time than that of MAGA, i.e., 23 months. Hence any Pluto mission that involves JGA will restrict opportunity of launch dates.

This objection is also based on facts, but Pluto missions will not be launched very often, at least at this stage. Therefore, this objection is not very a serious one considering that MAGAJGA decreases heating rates, travel time, and $V_{\infty E}$ compared with MAGA alone. It is worth mentioning here that MAGAJGA and VGAMAGAJGA require $V_{\infty E} = 5.6$ and 4.2 km/s, respectively, for a solar polar flyby mission, whereas $V_{\infty E}$ used in the case of the Galileo mission (VEEGA) was only 3.7 km/s. This shows the need of MAGAJGA and VGAMAGAJGA missions; without these assists Pluto or solar missions will be difficult to realize with a small Earth launch energy. Moreover, the travel time from Earth to Jupiter for these missions will be around 2.0 and 2.3 years, respectively, whereas in the case of the VEEGA mission it was six years.

3) The final period of the solar probe in the case of MAGA will be 0.7 years, whereas in the case of MAGAJGA it will be 4.5 years.

A solar flyby mission with MAGA alone is not possible at present, because of the intense heating, high $V_{\infty E}$, and guidance problems. Furthermore, the Ulysses solar probe was launched with $V_{\infty E} = 15.4$ km/s ($\Delta V_E = 11.16$ km/s) with the perihelion = 1.3 AU; it used JGA and its final period will be approximately six years. The $V_{\infty E}$ used for the Ulysses mission is more than twice that required for a MAGAJGA solar polar mission; the perihelion for the latter was chosen in this study as 4 R_S, which is much lower than that of the Ulysses solar probe. Table 3 also shows that $V_{\infty E}$ for MAGAJGA is almost half of that needed for a JGA solar probe mission.

Nevertheless, the required lifetime of a spacecraft for the MAGA-JGA mission must be longer compared with that for the MAGA mission.

Conclusion

Use of MAGA alone for HEM is not feasible because of the associated high heating rate during the atmospheric maneuvering. Thus MAGAJGA is recommended for HEM because of its advantages of low heating rate during aerodynamic maneuvering and low Earth launch velocity. For example, a solar polar flyby mission with the perihelion at 4 R_S that uses only MAGA requires a $V_{\infty_E} \approx 18$ km/s and the associated heating rate will be thousands of watts per square centimeter. However, the same mission using MAGA-JGA requires only $V_{\infty_E} = 5.6$ km/s and the associated heating rate will be 120 W/cm^2. Furthermore, using VGAMAGAJGA instead of MAGAJGA will reduce V_{∞_E} to 4.2 km/s, although this may limit opportunities for the mission.

A 6.6-year Pluto mission by using MAGA alone encounters very large heating, which may damage the structure of the spacecraft. But using MAGAJGA for a 7-year Pluto mission can decrease the Earth launch velocity and the associated heating rate substantially, by 64 and 70%, respectively.

The radiation problem during JGA can be tackled by increasing the closest approach of the spacecraft and by using extra shielding of the electronic parts. In any case, it was found that the closest approaches required for the solar flyby and Pluto missions will be higher than several previous missions involving JGA and slightly less than the recent Ulysses mission. Hence, the radiation problem is a comparatively minor one considering the merits of the MAGAJGA technique.

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