Assessment of Aeroassist Orbital Maneuver Technologies for Next Mars Exploration

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In this article, preliminary feasibility assessment of aeroassist systems for JAXA's future Mars Exploration has been conducted. The aerobraking technology is found to bring about an increase in the payload ratio by considerable reduction of the propellant mass required for orbital descent after propulsive orbital insertion. The aerocapture technology, if accuracy in both the interplanetary orbit determination and guidance is improved, is shown to achieve a remarkable increase of the payload mass using the state-of-the-art technologies. Discussions are made on the guided entry systems which are expected to realize better landing mark precision than obtained by the ballistic flight.

Key Words: Aerobreak, Aerocapture, Aeroassist Maneuver, Guided Entry, Mars Exploration

1. Introduction

Exploration to Mars has come under review in Japan Aerospace Exploration Agency together with a lot of planetary scientific groups all over Japan¹⁾. Mars orbiters for aeronomical and meteorological researches²⁻⁴⁾, Mars landers for geoscientific researches⁵⁾, and aero-flyby sample return systems⁶⁾ are currently proposed, so that the next Mars exploration is expected to be a conglomerate mission, enjoying enhanced launch capability of the H-IIA or H-IIB rocket. In the early study of the mission design²⁾, the orbiters and the landers are transported altogether to Mars and inserted into an extended elliptic orbit, after which the orbiters are separated and maneuvered to their respective final orbit while the landers are ejected and flown into the Mars atmosphere. In such a scenario, aeroassist orbital maneuver techniques are expected to bring about an extended potential for mission design by reducing propellant mass required for Mars orbit insertion (MOI) and Mars orbit transfer (MOT), and by minimizing atmospheric flight trajectory dispersion for pinpoint landing. From this viewpoint, in order to evaluate effectiveness of the aeroassist technologies applied to the next Mars mission, and to assess the technical readiness of these technologies, the preliminary system analyses of aerobraking, aerocapture, and guided entry are conducted in this article.

2. Aerobraking

Since a large ΔV as well as a large thrust density is required at planetary orbit insertion, chemical thrusters such as solid rocket motors and bi-propellant orbital maneuver engines (OMEs) are in general used to decelerate the spacecraft. However, since the specific impulse of such chemical thrusters is not higher than 330 sec, a large amount of propellant is needed to achieve the ΔV required for orbit

insertion. Consequently, a large mass fraction of the spacecraft is occupied by the propulsion system, resulting in limited mass available for mission payloads. Let us consider an example in which the Mars approach velocity at infinite is 3.8 km/s and the spacecraft is inserted into an elliptic orbit having a 150 km periapsis altitude and a 3,000 km apoapsis altitude, as shown in Table 1. This is the nominal target orbit currently under consideration for the lower orbiter²⁾. The theoretical ΔV for MOI is 2.27 km/s. By using an OME system having a specific impulse of 310 sec and a structural factor of 0.15, the mass fraction of the propellant and the propulsion system is estimated to be 0.526 and 0.605, respectively, as summarized in Table 1. That is, only 0.395 of

Table 1. Comparison between payload mass ratios achievable by propulsive direct MOI and by primary orbit insertion followed by aerobraking.

Mars approach velocity, V_∞	3.8 km/s
Target orbit periapsis altitude, H_p	150 km
Target orbit apoapsis altitude, H_a	3,000 km
Theoretical ΔV for orbit insertion	2.27 km/s
OME specific impulse	310 sec
OME structural factor	0.15
Direct orbit insertion	
Propellant mass ratio	0.526
OME mass ratio	0.605
Payload mass ratio	0.395
Primary orbit insertion and aerobraking	
Primary orbit periapsis altitude, H_p^*	150 km
Primary orbit apoapsis altitude, H_a^*	$7 \times$ Mars radius
Theoretical ΔV for orbit insertion	1.59 km/s
Aerobraking altitude, h_{ab}	115 km
Propellant mass ratio (total)	0.409
for primary orbit insertion	0.407
for periapsis descent	0.0005
for periapsis ascent	0.0013
for periapsis adjust	0.010
OME mass ratio	0.480
Payload mass ratio	0.520

Fig. 1. The effective ΔV achievable per one flight path of aerobraking and the maximum heat transfer rate of orbiters.

the spacecraft mass is open to the mission payload including the structure when the spacecraft is inserted into the target orbit directly from the interplanetary orbit by the OME system.

On the other hand, let us consider a mission scenario in which the spacecraft is first inserted into the primary elliptic orbit having a 150 km periapsis altitude and a $7 R_m$ apoapsis altitude where R_m is the Mars average radius, and then transferred to the target orbit by aerobraking. By using the OME system having the same specific impulse and the same structural factor, the propellant mass ratio for insertion to the primary orbit is reduced to 0.407. Taking into account the propellant needed for periapsis descent, ascent, and adjustment during aerobraking, the total propellant mass ratio is expected to be 0.409 (see Table 1 for details). The resultant payload ratio of 0.520 is higher by 0.125 than that obtained by the propulsive direct MOI.

To apply the aerobraking technique to orbital maneuver, it is necessary to assess the deceleration performance and the heat transfer rate of the orbiter, in addition to the mass advantage of aerobraking described above. To clarify the aerobraking performance, trajectory calculations in the presence of aerodynamic drag were conducted using a constant drag coefficient. Here, the standard Mars atmosphere model was taken from Ref. 7. In Fig. 1, the effective ΔV obtained per one flight path of aerobraking and the maximum heat transfer rate on the stagnation point of the orbiter surface are plotted against the periapsis altitude. In the figure, β is the ballistic coefficient defined by $\beta = W/C_D S$ where W , C_D , and S are the mass of the orbiter, the drag coefficient, and the representative area, respectively, and h_a is the apoapsis altitude. $\beta = 47.6$ and 142.9 correspond to the smallest and the largest ballistic coefficient of the lower orbiter under consideration. It is seen that the effective ΔV considerably increases with decreasing β , and slightly increases as h_a decreases. The heat transfer rate is only a function of h_a since

the flight velocity at the peak heating point is almost independent of β in aerobraking. To enjoy the mass advantage of aerobraking, it is necessary to eliminate additional devices to protect the orbiter against the heat flux transferred from the ambient high-speed flow. Without such thermal protection devices, if the operational temperature of the orbiter surface is limited below 450 K, which is approximately the maximum operational temperature for aluminum alloys, the maximum acceptable heat transfer rate estimated by radiation equilibrium is approximately 2 kW/m^2 . In this case, the periapsis altitude for aerobraking should be maintained above 105 km, and the achievable maximum ΔV is not higher than 2 m/s for $\beta = 47.6$ and 0.4 m/s for $\beta = 142.9$, as seen in Fig. 1. This suggests that a large number of cycles are needed to transfer the orbiter from the initial to the final orbit by aerobraking. In the case of Table 1, it takes approximately one year to obtain $\Delta V = 680 \text{ m/s}$ which is necessary for orbit transfer when $\beta = 47.6$.

As just described, aerobraking is capable of considerably increasing the payload mass ratio in comparison to the propulsive direct MOI. Without additional devices, technical readiness of aerobraking is considered to be high. However, it requires the long-term and continuous operation and the repetitive orbit determination of the orbiter till the orbiter reaches the final orbit.

3. Aerocapture

The operational time for aerobraking can be shortened by decreasing the periapsis altitude so that the ΔV achievable per one flight path can be correspondingly increased. The extreme form of such approach is the aeroassist orbit transfer (AOT), in which the ΔV required for orbit transfer is realized by a single flyby through the planetary atmosphere. In such a situation, however, the thermal protection system is necessary to protect the orbiter from the increased heat flux, which derogates the mass advantage of aerobraking compared to the propulsive direct orbit insertion. Aerocapture is the orbit insertion method in which the spacecraft is directly inserted in a single atmospheric flyby from the interplanetary approach orbit into the target orbit by using the aerodynamic drag to obtain ΔV required for orbit insertion. The sequence of event is illustrated in Fig. 2. Since the most portion of ΔV required for orbit insertion is realized by the aerodynamic drag during the atmospheric flight, a small propulsion system with a small amount of propellant for periapsis raise maneuver and apoapsis adjust maneuver after the aero-flyby is only needed. This brings about drastic reduction in mass of the propellant as well as that of the propulsion device in comparison to the propulsive orbit insertion.

3.1. Fundamental characteristics

For the purpose of clearly identifying advantages of aerocapture in comparison to the propulsive direct orbit insertion and aerobraking, another MOI example is considered below. Here, the approach velocity at infinity is set to 4.8 km/s, which is a moderate approach velocity as high as that of the atmospheric entry system of Mars Science Laboratory⁸⁾.

Fig. 2. Schematic of aerocapture sequence.

Table 2. Comparison between payload mass ratios achievable by propulsive direct MOI, primary orbit insertion followed by aerobraking, and aerocapture.

Mars approach velocity, V_∞	4.8 km/s
Target orbit periapsis altitude, H_p	150 km
Target orbit apoapsis altitude, H_a	3,000 km
Theoretical ΔV for orbit insertion	2.93 km/s
OME specific impulse	310 sec
OME structural factor	0.15
Direct orbit insertion	
Payload mass ratio	0.289
Primary orbit insertion and aerobraking	
Primary orbit periapsis altitude, H_p^*	150 km
Primary orbit apoapsis altitude, H_a^*	$7 \times$ Mars radius
Theoretical ΔV for orbit insertion	2.25 km/s
Aerobraking altitude, h_{ab}	115 km
Payload mass ratio	0.386
Aerocapture (see Table 3 for details)	
Orbit insertion device mass ratio	0.474
Payload mass ratio	0.526

The target orbit having a 150 km periapsis altitude and a 3,000 km apoapsis altitude is chosen. In this case, the payload mass ratio achieved by the propulsive direct MOI is estimated to be 0.289. When the spacecraft is first inserted into the primary orbit having a 150 km periapsis altitude and a $7 R_m$ apoapsis altitude, and then transferred to the target orbit by aerobraking, the payload mass ratio is expected to be 0.386, as summarized in Table 2.

To clarify the aerocapture performance, trajectory calculations taking into account the aerodynamic drag and lift in the presence of Mars atmosphere were conducted. Here, the standard Mars atmosphere model was taken from Ref. 7. In Fig.3, the flight path angles for successful MOI and unsuccessful MOI are plotted against the ballistic coefficient of the orbiter. Here, L/D is the aerodynamic lift-to-drag ratio of the orbiter. The lift is defined to be positive when it is in the ascent direction relative to the Mars surface. The dash-dot line denoted by “escaped” is the upper limit of the flight path angle beyond which the orbiter cannot be captured by Mars gravitational sphere, while the dash-dot-dot line denoted by “dropped” is the lower limit below which the orbiter is dropped to the ground. The solid line denoted by “optimized”

Fig. 3. Variation of flight path angles at an altitude of 200 km ($V_\infty = 4.8$ km/s).

is the flight path angle which minimizes ΔV required for the periapsis raise maneuver and the apoapsis adjust maneuver after aero-flyby. The dotted line denoted by “ $\Delta V=200$ m/s” is the flight path angle which requires ΔV of 200 m/s after aero-flyby to transfer the orbiter into the target orbit. In this calculation, the mean temperature model taken from Ref.7 is used as the standard Mars atmosphere model.

In Fig.3, the range of the flight path angle between “dropped” and “optimized” is significantly narrow when $L/D = -0.3$, and broadens as L/D increases. In the ballistic aerocapture ($L/D = 0$), the tolerance of the flight path angle that realizes insertion to any Mars orbit is only 0.2 deg. When $L/D = 0.3$, the tolerance for insertion to any Mars orbit broadens to 1.2 deg, while the range of the flight path angle for $\Delta V < 200$ m/s is only 0.3 deg. These suggest that the lifting flight is essentially the necessary requirement to realize the successful aerocapture MOI. In addition, it is expected that the range of the flight path angle for the successful MOI, which is hereafter denoted as “the aerocapture corridor”, can

Fig. 4. Mars atmosphere models.

be considerably broadened by appropriately changing L/D during atmospheric flight. In the case of Fig.3, the region between the solid lines for $L/D = -0.3$ and 0.3 forms the theoretical maximum aerocapture corridor.

3.2. Influences of Mars atmosphere models

The above discussions only apply to the mean temperature model in Ref. 7 in the absence of uncertainties in the Mars atmosphere model. In Fig.4, density and temperature profiles of the representative Mars atmosphere models⁷⁾ are plotted. The atmospheric density profile, which has a dominant impact on the aerocapture corridor, may considerably change according to the solar activity, the season, the climate, the weather, and the geographical conditions. Among such affecters, the sand storm has the most significant impact on both the density and the temperature profile. In Fig.4, the atmosphere model affected by the sand storm is denoted by “Dusty profile”.

To assess the influence of the atmosphere models on the aerocapture characteristics, trajectory calculations were conducted using the different atmosphere models. As a result, the optimum flight path angle for each atmosphere model is shown in Fig.5. It is seen that the optimum flight path angle considerably depends on the atmospheric model, especially when the ballistic coefficient is small. When $L/D = 0.3$, the optimum flight path angle changes by more than 0.3 deg according to the atmospheric model, as seen in Fig.5. On the other hand, as seen in Fig.3, the range of the flight path angle for $\Delta V < 200$ m/s is only 0.3 deg. From these facts, it is difficult to achieve successful MOI by using a constant L/D during flight even for $L/D = 0.3$. A broad aerocapture corridor can be realized by appropriately changing L/D during atmospheric flight, as already described in 3.1. In Fig.5, the screened area between the dash-dot line (the optimum flight path angle for the low-temperature model) for $L/D = -0.3$ and the dash-dot-dot line (the one for the dusty model) for $L/D = 0.3$ forms the maximum aerocapture corridor under

Fig. 5. Variation of optimum flight path angles for different atmosphere models ($V_{\infty} = 4.8$ km/s).

uncertainty of the atmospheric density.

3.3. Aerocapture shells

The aeroshell is the primary component additionally required for the aerocapture system. It should not only protect the mission payloads against the hypersonic aeroheating environments but also precisely realize L/D as well as the ballistic coefficient as designed to accomplish successful orbit insertion for the given orbiter mass. To avoid an excessive temperature rise inside the orbiter after the hypersonic flight, the aeroshell should be separated from the mission payloads immediately after escape from the Mars atmosphere. To fulfill these functions, the aeroshell consists of the heatshield system, the thermal insulator, the structural skin and frame, and the separation apparatus. Since the aeroshell is an additional component peculiar to the aerocapture system, it is necessary to use the ultralight heat-resisting structure to maximize the mass advantage in comparison to propulsive MOI systems. A potential candidate currently under consideration is the carbon-fiber-reinforced-plastic (CFRP) honeycomb sandwich structure⁹⁾ covered with the thermal insulator and the ultralight CFRP ablator having a bulk density of 0.2 to 0.3 g/cm³. Such the ultralight CFRP ablator is currently under development in Japan Aerospace Exploration Agency¹⁰⁾.

The axisymmetric sphere-cone shape is one of the possible candidates for the aeroshell forebody. For example, using a sphere-cone aeroshell with a half cone angle of 70 deg and a nose radius of $0.4D$ where D is the base diameter, L/D can be set to 0.3 at an angle of attack of 18 deg¹¹⁾. L/D can be maintained at 0.3 by adjusting the trim angle of attack at 18 deg, which is realized by placing the center of gravity of the orbiter apart by $0.025D$ in the radial direction from the aerodynamic center. Using such an aeroshell, the effective L/D in the ascent direction can be controlled between -0.3 and 0.3 during flight by rotating the aeroshell around the aeroshell center axis using the reaction control systems (RCSs). This is the so called bank-angle modulation system. Conceptual

Fig. 6. Aeroshell examples for Martian aerocapture with L/D control.

design of an aeroshell with the bank-angle modulation system is illustrated in Fig.6a.

Since the bank-angle modulation method involves rotational motion of the orbiter around the center axis using the RCSs, considerable mass of propellant as well as propulsion devices is needed. In addition, attitude perturbation associated with rotational motion, flight path deviation in the crosswise direction, and interaction between the gas jet and the hypersonic flow may have negative influences on the aerocapture success rate. In an attempt to eliminate such disadvantages, the aeroshell attached with aerodynamic flaps is alternatively proposed, as illustrated in Fig.6b. The primary advantage of the flap-controlled aeroshell is that the effective L/D in the ascent direction can be controlled without rotational motion, which essentially eliminates the attitude perturbation originating from rotational motion, and that the effective L/D in the ascent direction can be continuously and accurately changed without flight path deviation in the crosswise direction. The disadvantage of the flap-controlled system may be that additional mass of flaps, actuators, and electric power sources is necessary, and that the system is more complex than the bank-angle modulation system. Unfortunately, the technology readiness level of the flap-controlled aeroshell is lower than that of the bank-angle modulation system. To apply the flap-controlled aeroshell system, further research and development is necessary in the future.

In both systems, aerodynamic performance may be assessed by wind-tunnel experiments with the aid of numerical analyses where nonequilibrium aerothermodynamic effects are incorporated. Since there are no ground test facilities that can realize the real flight environments for testing, uncertainties in aerodynamic performance remain before flight. The

Fig. 7. Success rate for target orbit insertion using analytic predictor-corrector (APC) algorithm.

aerodynamic uncertainties are expected to be larger in the flap-controlled system than in the bank-angle modulation system. In order to realize successful orbit insertion, the aerodynamic uncertainties should be compensated for by the guidance system described below. For this reason, from a mission designing viewpoint, it is of primary importance to accurately quantify the magnitude of uncertainties. Of course, a continual research is needed to reduce the aerodynamic uncertainties by improving numerical prediction technologies for nonequilibrium aerothermodynamic flows.

3.4. Guidance

As discussed in 3.2, the aerocapture MOI can be successfully realized by controlling the effective L/D during the atmospheric flight, using an appropriate guidance algorithm. To assess the success rate of the aerocapture MOI, the analytic predictor-corrector (APC) algorithm, which is one of the most common and effective guidance algorithms tested so far, was used in this study. The success rate was statistically computed by the Monte-Carlo method. In this calculation, the Mars atmosphere model was defined by a normal probability distribution using the mean temperature model as the center value, the low-temperature model as the center value $-3\sigma^-$, and the dusty model as the center value $+3\sigma^+$, where σ^- and σ^+ are negative and positive standard deviation, respectively. Uncertainties in the aerodynamic coefficient are taken into account by expressing the aerodynamic coefficient as a normal probability distribution with $3\sigma = 10\%$ of the center value. The ballistic coefficient is set to 100 kg/m^2 . The trial is judged as success if the net ΔV required for the periapsis raise maneuver and the apoapsis adjust maneuver after aero-flyby is smaller than 200 m/s . More detailed information of this calculation is found in Ref.12.

The success rate estimated by the Monte-Carlo calculation is plotted against the flight path angle in Fig.7. It is seen that

Table 3. Mass distribution of MOI system for aerocapture.

Total mass of orbiter		500 kg
MOI system mass for aerocapture		237 kg
Forebody shell	heatshield	48 kg
	skin, frame, and separation devices	47 kg
Aftbody shell	heatshield	30 kg
	skin, frame, and separation devices	49 kg
RCS	propellant	20 kg
	tank, valve, and propulsion device	16 kg
OME	propellant	21 kg
	tank, valve, and propulsion device	6 kg
Payload mass		263 kg
Payload mass ratio		0.526

the success rate greater than 99 % can be accomplished for flight path angles ranging from -15.75 to -15.1 deg. That is, the acceptable errors in the flight path angle for successful aerocapture are ± 0.325 deg. This corresponds to admissible errors of ± 4 km in orbit determination at an altitude of 200 km. Using the state-of-the-art technologies, such accurate orbit determination and guidance in the approaching phase are not impossible, though they are not readily achievable. In order to apply aerocapture to MOI, it may be necessary to develop innovative technologies for aerocapture as well as guidance and navigation in the approaching phase.

3.5. System configuration

In order to assess the payload mass ratio achievable by aerocapture, system analysis was conducted for an aerocapture system shown in Fig.6a, based on the results of the 6-degree-of-freedom dynamic analysis of the orbiter and the heatshield analysis along the flight trajectory⁹⁾. Here, the total mass, the ballistic coefficient, and the diameter of the orbiter are assumed to be 500 kg, 62.5 kg/m^2 , and 2.4 m, respectively. The result is summarized in Table 3. Since the atmospheric entry velocity is not higher than 7 km/s, the aerodynamic heat transfer rate at the stagnation point of the heatshield remains below 0.6 MW/m^2 during flight. For this reason, the heatshield mass ratio can be reasonably suppressed in comparison to the HAYABUSA and the Stardust reentry capsule which has a reentry velocity over 11 km/s. Finally, the payload mass ratio achievable by aerocapture is estimated to be 0.526, which is higher than that obtained by the propulsive direct MOI and the primary orbit insertion followed by aerobraking, as shown in Table 2.

4. Guided Entry

If rovers can be carried by the atmospheric entry and landing system to a desired location of geological and geographical interest, scientific observations of enhanced quality and extended freedom may be accomplished. This is brought about by using the aerodynamic guided entry system coupled with the landers having precise guidance and landing capacity⁵⁾. To clarify effectiveness of the guided entry, let us consider a landing mission scenario in which the entry system with $\beta = 100 \text{ kg/m}^2$ is flown into the Martian atmosphere from the primary elliptic orbit having a 150 km periapsis altitude and a $7 R_m$ apoapsis altitude by initiating an appropriate

Fig. 8. Dispersion of parachute deployment point (altitude = 13 km) for a) ballistic flight and b) guided entry using analytic predictor-corrector algorithm with $L/D = 0.2$.

deceleration ΔV to realize a 1270-km downrange at the altitude of parachute deployment, which is assumed to be 13 km above ground. Monte-Carlo trajectory calculations taking into account uncertainties in the atmospheric density, the aerodynamic coefficient, and the initial flight path angle show that, if the entry system is flown in a ballistic flight without guidance, the parachute deployment point may be scattered approximately in a 150×20 km ellipse around the aimed point, as shown in Fig.8a.

On the other hand, in order to estimate the aerodynamic maneuver capacity, atmospheric entry trajectory calculations were made by changing L/D of the entry system. In Fig.9, the achievable downrange of the entry system is plotted as a function of the entry flight path angle. It is clear that the downrange can be reasonably extended by increasing L/D . Uncertainties in the atmosphere model do not have significant impact on the downrange. These results suggest that the dispersion of the parachute deployment point can be

Fig. 9. Achievable down range of the entry system.

minimized in the presence of uncertainties in the atmospheric density and the aerodynamic coefficient by using a lifting aeroshell controlled by an appropriate guidance algorithm¹³⁾, which is essentially identical with the one for aerocapture described in 3.4.

Based on this fact, the guided entry performance was assessed by calculating the dispersion of the parachute deployment point using the Monte-Carlo method, taking into account uncertainties in the atmospheric density, the aerodynamic coefficient, and the initial flight path angle. Here, the APC guidance described above were incorporated into the trajectory calculation. The result for $L/D = 0.2$ is shown in Fig.8b. It is seen that the landing point dispersion for guided entry remains within a circle with a 10-km radius. This is significant improvement in adjustment of the parachute deployment point in comparison to that of the ballistic flight without guidance. It should be noted that, in order to realize pinpoint landing within a dispersion circle with a radius less than 1 km, additional guidance systems such as a parafoil and a lander should be activated during descent after parachute deployment.

5. Summary

Preliminary assessment of effectiveness and feasibility of aeroassist systems for JAXA's future Mars Exploration has been conducted. In comparison to the propulsive direct orbit insertion, aerobraking is capable of increasing the payload mass inserted into the final orbit by reduction of the propellant mass required for orbit insertion and transfer. Technical readiness of aerobraking is so high that it is strongly recommended to introduce aerobraking to the lower orbiter for orbit transfer. Aerocapture is shown to bring about a remarkable increase of the payload mass inserted into the final orbit. However, to actualize aerocapture system, accuracy in both the interplanetary orbit determination and guidance

should be improved. The guided entry systems are potential of realizing pinpoint landing of rovers by minimizing the landing point dispersion within a circle with a 10-km radius, which is favorable for transporting rovers to the desired location of geological and geographical interest.

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