MISSION STUDY OF MARS/VENUS PROBES DRIVEN BY ELECTRIC PROPULSION

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ABSTRACT

Mission studies are performed aiming at the applications of the electric propulsion to the scientific spacecraft for explorations on Mars and Venus. It is found that the hybrid transfer using both the conventional chemical rocket and the electric propulsion gains the extra weight without serious impacts to the existing spacecraft system of the Mars probe. The power throttling of the electric propulsion is required for the interplanetary missions.

INTRODUCTION

Various kinds of the electric propulsions (EP) have been developed in the world. And some of them are about to be applied to the space missions as the bus system. Institute of Space and Astronautical Science (ISAS) also has made R & D efforts on the EP systems applicable to the space missions[1]. Almost all our efforts have been concentrated on the improvements on the thrust performance and the endurance of the EP systems. As the results the EPs have had moderate performances enough to apply the space missions in the general meanings. On the other hand, a particular mission requires to the EP the specific performances: high specific impulse, or high thrust power ratio, thrust resolution, so on. The EP should be reviewed from the standpoint of the space mission.

In general the EP takes much longer time in the operation than the chemical rockets (CR). In the EP operation the spacecraft bus must provide the EP the electrical power, the attitude stability and the command. The EP is inferior to the CR in the ease of operation. The EP system should be developed so as to reduce the operational disadvantages. It is meaningful to consider the functions and the capabilities which the EP should equips in the aspects of the spacecraft system.

The state on the application of the EP in the space activity is not always good. Though a lot of the EP systems exist in the world, the EPs have rare flight chances. In order to break through the inactive state we have several ways to sell the EP systems. The first way is to replace the CR with the EP. In the present time a possible space mission is designed using the CR. The most reasonable way to get a flight chance for the EP is to gain extra payload weight and/or extra mission life time in comparison with the CR. In the strategy substituting the EP the spacecraft operation and the system must be similar to the mission with the CR so that the existing hardware and the software are available. In other words, it is the most important to gain the mission merits using the EP without serious rearrangements of the onboard and the ground systems existing. The other way is to establish the unique mission scenario for the EP. It is natural that the EP requires to the spacecraft system the different resources from the CR because the EP has the physical basis distinct from the CR's. In the second strategy it is necessary to consider ultra-high delta-V missions such as that out of the solar system or new hardwares as the nuclear power plant. These missions are significant big projects, which must be discussed on their invebitability previous to the usage of the EP. Anyway, the first way is the most realistic in the present time.

The ISAS has just started the development program of the new launcher M-5 rocket, which will have a capability to input a 2-ton satellite on the low earth orbit (LEO) and a 500-kg spacecraft in the interplanetary space[2]. As interplanetary missions in the mid 1990's, Mars and Venus are very attractive in the aspects of the space science and the space technology. There are vast scientific interests on these earth type planets[3]. Once a spacecraft approaches one of these planets, it can reach farther planets using the technology of the swing-by without fuel consumption. Spacecraft bound to inner planets have different system from those to outer planets in the aspects of the electrical power, the thermal control and the communication. So, the missions to Venus and Mars are very reasonable challenges to the EP.

The Mars/Venus orbiter missions using the EP are studied in comparison with conventional missions[4, 5] using the CR under the conditions of the equivalent launch weight about 2 ton at 250 km altitude LEO. The first interest of the present study is focused on the designing the interplanetary missions using the EP without rearrangement of the existing system. And the second is to list up the capabilities and the penalties of the EP system required for the interplanetary missions.

REFERENCE MISSION

The working groups in ISAS propose the Mars and the Venus orbiter missions[4, 5]. The main purposes of these missions are to explore the outer atmospheres around non-magnetized planets. In this section the proposed projects are introduced as the reference missions on the basis to take account of the application of the EP.

Mars Mission

The spin-stabilized spacecraft with total weight 450kg would be launched at November 1996. It would reach Mars at September 1997 and be input the ellipse orbit with 300km altitude and ten times of the Martian radius. The weight break down is tabulated in Table 1. The spacecraft equips 4 solar cell panels, a parabola antenna and several long wire sensors as seen in Fig.1. The solar cell panel and the parabola antenna have no gimbal systems. The surfaces of the solar cell panels face the same direction of the antenna beam axis, which is located in parallel to the spin axis. On the view from Mars, Earth can be seen on the eclipse less than 45 degrees apart from Sun. The spacecraft can get enough electrical power when the spin axis tracks Earth so as to communicate. So the spacecraft is operated with the spin axis in the eclipse surface. The solar cell panels of 1kW at Earth generate minimum 300W at Earth pointing on the Martian orbit. After the insertion to the Mars circulation orbit the wire sensors will be extended. The wire sensors are so flexible that their structural stability would be assured by spinning of the spacecraft. The spacecraft equips a bipropellant thruster for the Mars capture and hydrazine gas jets for the attitude control.

Venus Mission

The proposed Venus probe is also a spin-stabilized spacecraft. The spin axis is perpendicular to the eclipse surface. The despun-controlled parabola antenna tracks Earth independently on the spinning of the spacecraft body as seen in Fig.2. The solar cells are mounted around the cylindrical body so that the fraction $1/\pi$ of the total solar cells generates electrical power. The configuration difference between the Mars and the Venus probes is originated from the locations of the planets from Earth, that is, whether the outer or the inner planet. Total electrical power is 240W at Venus. The main specifications are listed in Table 2. The probe would be launched at March 1996 and be inserted on a circulation orbit with 300km altitude and 5 times of the Venusian radius at September 1996.

MISSION CENARIO by EP

The interplanetary mission is divided to the following six phases: launch from the ground, escape from the earth potential, input to a transfer orbit, capture in an objective planet, maneuver

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Table 1 Specifications of Mars probe at escape from Earth.

Total Weight	450kg	
payload	40kg	
common equip.	205kg	
propulsion	205kg	
CR hardware		45kg
fuel for capture		129kg
fuel for attitude control		20kg
fuel to adjust C_3		6kg
fuel margin		5kg
Communication	S/X bands fixed parabola antenna	
Electrical Power	fixed solar cell panels 300W at Earth point on Mars 1kW at Sun point on Earth	
Main Propulsion	bipropellant thruster 500N/312sec	
Attitude Control	spin hydrazine gas jet 3N/200sec	

Table 2 Specifications of Venus probe at escape from Earth.

Total Weight	420kg	
payload		30kg
common equip.		205kg
propulsion		185kg
Communication	S band despun parabola antenna	
Electrical Power	body mounted solar cells 240W at Venus 130W at Earth	
Main Propulsion	solid motor 296sec/β0.086	
Attitude Control	spin hydrazine gas jet 3N/180sec	

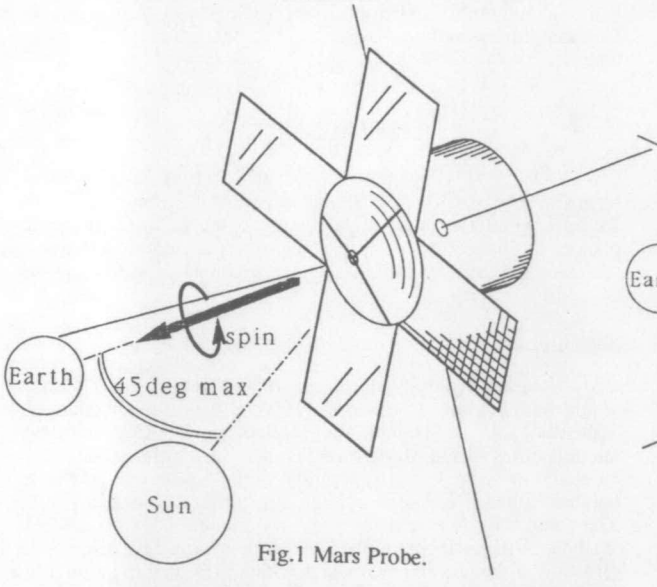


Fig.1 Mars Probe.

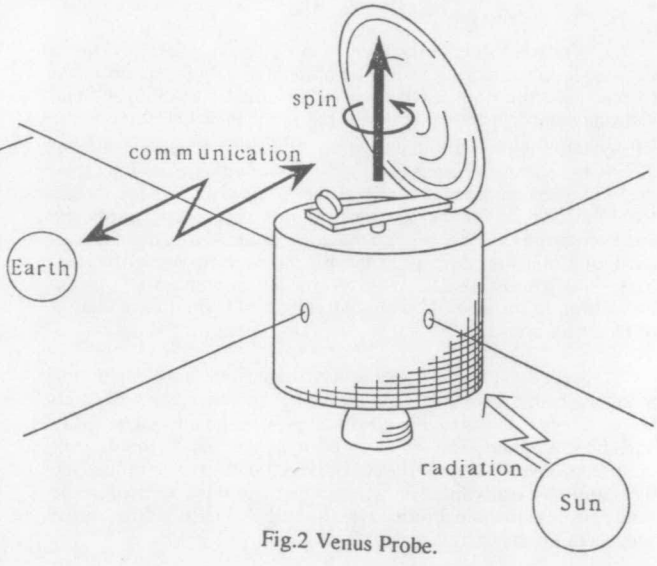


Fig.2 Venus Probe.

in a circulation orbit, and attitude control as shown in Fig.3. The EP can not adapt to the launch, the escape and the capture phases because of the requirement of high thrust in a short time. The EP operations at the transfer, the maneuver and the attitude control are considered in this paper.

Transfer Phase

In the CR mission, the earth escape and the input to the transfer orbit are conducted simultaneously at LEO after the launch from the ground. A spacecraft requires delta-V 3.21km/s for the earth escape from LEO at 250km altitude. And it needs the root C_3 2.68km/s for the insertion to the Mars transfer orbit from the Earth revolution orbit. When a kick motor gains the delta-V 3.53km/s at LEO, the spacecraft is able to escape the Earth potential and be put on the transfer orbit. The mission energy is reduced by giving the delta-V at a position with low potential energy, that is the gravity assist. The relation between the velocity and the potential energy is illustrated in Fig.4. The EP operation can not utilize the Earth gravity assist at the insertion to the transfer orbit because it takes a long time to give the required delta-V.

Suppose the case to transfer by the EP from out of the earth potential after the Earth escape phase. The EP boosts the spacecraft 2.68km/s after the kick motor gives the spacecraft 3.21km/s at LEO in the case of the Mars mission. The specific impulse 2500sec of the EP operation is equivalent in the fuel consumption to the CR with 300sec specific impulse because CR can complete the insertion to the transfer orbit by an extra 0.32km/s delta-V. The EP operation at over 2500sec in the specific impulse is able to gain an extra payload on the transfer phase in comparison with the CR mission.

On the transfer orbit the spacecraft changes its location in the solar system so that the intensity of the solar radiation varies. If the electrical power depends on the solar radiation, the electrical power is function of the distance from Sun. The solar radiation at Venus is 1.9 times as large as that at Earth and 0.43 at Mars. The power throttling is an important function of the EP for effective usage of the electrical power in the transfer orbit.

The transfer to Mars driven by the EP was calculated. The spacecraft is 500kg weight and zero C_3. The electrical power is 1kW at Earth and dependent on the distance from Sun. The EP produces the thrust force 25mN/kW with the specific impulse 3500sec with azimuth direction of the transfer orbit. The spacecraft reaches Mars taking about 5 years after 3.5 revolutions. In order to reduce the mission time, the hybrid transfer using both the EP and the CR is proposed. The CR gives the spacecraft a delta-V less than the C_3 for the Mars transfer at Earth. And the EP boosts the spacecraft for the rest of the C_3 on the transfer orbit. It is possible to design optional mission time depending on the energy distributions to the EP and the CR. If the EP mission is designed so as to match the reference mission with the CR in the transfer time, the spacecraft with the EP is able to be developed in the same schedule and be launched in the same window as the reference mission.

It is worth noting the reduction of the approaching velocity to an objective planet by the EP transfer. In the reference mission the spacecraft approaches Mars with 2.9km/s and needs 1.0km/s delta-V for circulating around Mars. If the spiral transfer by the EP realizes zero approaching velocity to Mars, only 0.24km/s delta-V is required for the input to the same circulation orbit as the reference mission. It is provable to have a merit in the payload

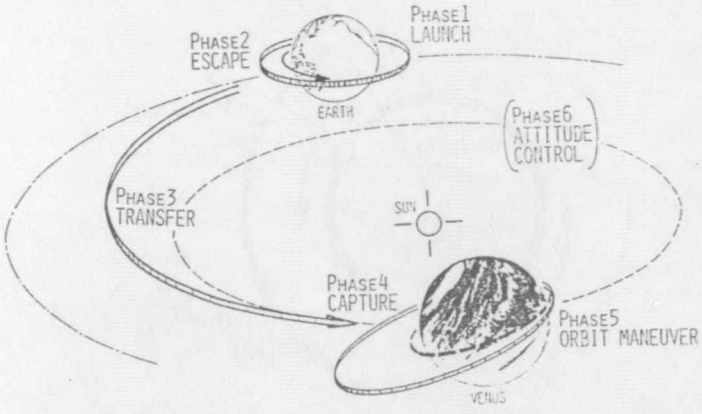


Fig.3 Mission Scenario.

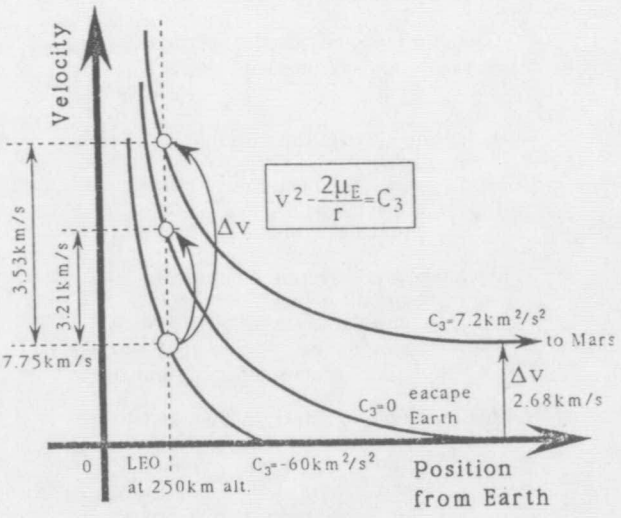


Fig.4 Gravity assist for Earth escape

weight of the orbiter by the EP spiral transfer which is not energy minimum relative to the Hormann transfer.

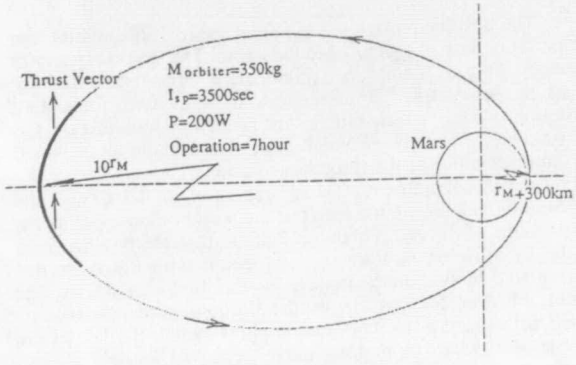
The functions of the EP required for the interplanetary missions are as follows: the automatism, the redundancy and the initial operation. It takes over several months to go to planets. A tracking site can communicate with the spacecraft in the interplanetary space during half of day. It is possible but not realistic to watch the spacecraft during all mission time using several tracking sites. So, the autonomous operation of the EP is significant important. One of the directions for the designing of the EP is "simplification" in order to adopt the automatism. Several approaches are proposed on the simplifications of the EP system[6]. The hybrid transfer is completed by the perfect operations of the EP and the CR. The redundant design is required for the EP as the bus system of the spacecraft. Whereas, the "light weight" on the EP system is also necessary in conflict with the redundancy. In general, high voltage equipments onboard spacecraft are not turned on in several weeks after the launch as far as the sufficient outgas from the equipments makes the gas breakdown miner. It is advantage for the hybrid transfer to operate the EP, which is the high voltage equipment, as soon as possible.

Circulation Orbit Maneuver

In the reference mission the orbiter circulates around Mars in 25hours. Consider the mission to descent the periapse from 300km altitude to 200km. The EP operation near the apoapse in 7hours completes the mission in about 10 days on the assumptions of available electrical power 200W, thrust power ratio 30mN/kW and thrust perpendicular to the main axis of the ellipse orbit. The EP operation is shown in Fig.5. The consumed propellant is only 60g. The same mission would performed by the CR with fuel about 1kg.

Attitude Control

Actuators for the satellite attitude control are required to generate a control torque enough to compensate an external disturbance. The spacecraft launched by ISAS equipped the magnetic torquer (MTQ) for the Earth circulation satellites and the hydrazine gas jet (RCS) for the interplanetary spacecraft. The MTQ could generate 1.4×10^{-3}Nm torque. The RCS had a capability 1.5Nm with 180sec specific impulse. The specifications of the attitude control actuators are summarized in Table 3. Even the electro-static type EP can produce a control torque one order of magnitude higher than that of the MTQ, and can reduce propellant consumption only one-twentieth of that of the RCS. Then, it is advantage to substitute the EP with the CR for the attitude control to compensate the external force on the fuel consumption. Restrictions on the attitude and/or the operation time exist for the attitude control by the EP. In the case of the solar-powered spacecraft the sun pointing of the solar collector is a significant necessary condition for the EP operation. It is possible to operate



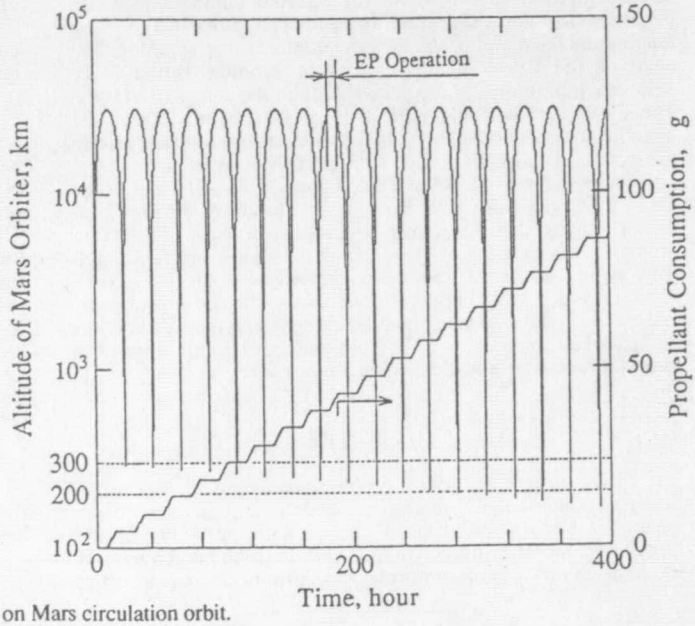


Fig.5 Maneuver on Mars circulation orbit.

Table 3 Actuators for attitude control

Magnetic Torquer (Satellite: Hakucho, 420kg)

magnetic moment	$48ATm^2$
max. torque	$1.4 \times 10^{-3}Nm$

Hydrazine Gas Jet (Spacecraft: Suisei, 140kg)

thrust	3N
specific impulse	180sec
spin torque	1.5Nm
min. pulse width	50msec

Electro-Static Electric Propulsion

specific impulse	3500sec
thrust power ratio	30mN/kW
max. torque	$1.5 \times 10^{-2}Nm$
(at power 0.5kW, arm 1m)	

Electro-Thermal Electric Propulsion

specific impulse	450sec
thrust power ratio	140mN/kW
max. torque	$7 \times 10^{-2}Nm$
(at power 0.5kW, arm 1m)	

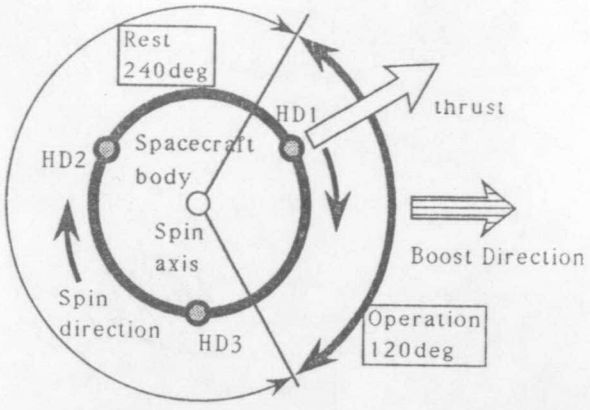


Fig.6 Sequential intermittent operation of EP.

the EP by battery backup in a short time. In the reference mission the battery specifications are 35Ah capacity and 50% maximum depth of discharge with 19V discharge voltage so that 1.2MJ electrical energy is available. Under the restricted available energy the electro-thermal EP generates 170Nms angular momentum and the electro-static EP does 40Nms. The spacecraft of ISAS were operated at angular momentum less than 40Nms. The capability of the electro-static EP is not enough for the attitude control powered by the battery, especially at a contingency operation. It is hard to gain the extra weight by substituting the EP with the CR attitude control system because the weight ratio of the attitude control actuator and fuel to the whole system is small.

Spacecraft Configuration

In is concluded that the EP application to the transfer phase is expected to gain the extra weight of the spacecraft system through the above investigations. Within the restriction to keep the basis of the reference missions as much as possible, several inventions are needed for the application of the EP to the interplanetary missions. The spin operation of the spacecraft around the objective planets is the most significant requirement for the structural stabilization of the long wire sensors as the payload. On the other hand, the EP boosting in the transfer phase needs to keep the thrust vector on the azimuth direction in the eclipse surface. In order to resolve the inconsistency, two approaches exist. The first way is to operate the spacecraft with non-spin in the transfer orbit. In that situation the weight (about 13kg) and the electrical power (about 45W) for the momentum wheel[7] are penalties for the spacecraft system. The other way is the intermittent operation of the EP synchronous to the spinning. The head of the EP is located so as to produce thrust force perpendicular to the spin axis and through the center of gravity. The EP is operated when the axis of the EP head faces the direction to generate the thrust in the spinning of the spacecraft. In the case of 2r.p.m. spin rate, a single head can be operated at 10sec after the rest of 20sec accompanying 17% cosine loss of the thrust force in a unit cycle. It is possible to utilize the electrical power during the rest using a condenser bank. These are problems: efficiency and life cycle of the condenser bank, high peak power of the EP. Sequential operation seen in Fig.6 of several heads is probable in order to resolve the above problems. For example, three heads, which are mounted on the side body of the spacecraft apart 120 degrees from each other, produce thrust in turn as powered by a single electrical processor.

MARS MISSION DESIGN

The proposed Venus probe is designed at the body-mounted solar cells, which is not efficient on the power generation, because the solar radiation is enough at Venus. On the other hand, the Mars probe equips the large solar cell panels. At the transfer orbit, the Mars probe has surplus electricity, using which the EP transfer is profitable in the standpoint of the system design. In this section, the Mars mission driven by the EP is designed on the assumptions:

1) The capabilities on 250km altitude LEO of the launcher are 1980kg for the Mars mission in order to compare with the reference mission driven by the CR. The specific impulse and the structure coefficient of a kick motor are 296sec and 0.086, respectively.

2) The spacecraft arrives at Mars after a half revolution around Sun by the hybrid transfer of the CR and the EP. The firing of the bipropellant thruster with 312sec specific impulse inserts the spacecraft to the Mars circulation orbit.

3) The mission schedules and the system configurations are almost same as the reference mission except for the transfer way and the propulsion system.

4) The spin-stabilized spacecraft are driven by the EP system with three heads of the sequential operation on the transfer orbit.

5) The EP system utilizes the electrical power which is left by 100W for house keeping and communication subtracted from the total generated power.

6) The EP produces the thrust of the specific impulse 3500sec with the ion production cost 300V and the propellant utilization efficiency 90% along the azimuth direction of the transfer orbit. The EP is operated by the sequential intermittent way with 17% cosine loss on the operation duty 90%. The power supply efficiency is assumed 80%.

The designed mission is summarized in Tables 4 and 5 in comparison with the reference mission. The hybrid transfer increases 20kg in the escape weight of the spacecraft. The EP boosts the spacecraft 770m/s delta-V to Mars with propellant consumption 13kg, which contains the propellant to adjust the C_3. The spacecraft driven by EP will be able to reach at Mars in almost the same schedule as the reference mission. The amount of the fuel for the Mars capture is reduced because of the decrease of the approaching velocity in the result of the weight gain 24kg as the Mars orbiter. If the EP system has 25kg in dry, the payload gain for the EP mission is 10kg in comparison with the reference mission due to weight reductions of the fuel and its storage system. It is probable to maneuver the Martian circulation orbit by the EP with slight propellant consumption so that the life of the probe at Mars would be prolonged.

According to the Venus mission, the restrictive conditions are too severe for the EP to gain an extra weight of the system. It is necessary to consider missions to transfer to Venus with several revolutions.

Table 4 Specifications of Mars probe driven by EP at escape from Earth.

Total Weight	470kg	
payload	50kg	
common equip.	205kg	
propulsion	215kg	
CR hardware		40kg
fuel for capture		112kg
fuel for attitude control		20kg
fuel margin		5kg
EP hardware		25kg
EP propellant		13kg
Electrical Power	fixed solar cell panels 300W at Earth point on Mars 1kW at Sun point on Earth	
Electric Propulsion	24mN/kW 3500sec 3 heads operation duty 90% sequential intermittent op.	
Chemical Rocket	bipropellant thruster 312sec	
Transfer	hybrid of CR and EP	
Attitude Control	spin hydrazine gas jet 200sec	

Table 5 Comparison with Mars missions

	CR Mission	EP Mission
Launch Date	Nov., 1996	Nov., 1996
Transfer Time	299day	313day
Escape Velocity	2.68km/s	2.27km/s
DV at Earth	3.53km/s	3.44km/s
Escape Weight	450kg	470kg
Propellant for EP	-	13kg
Flyby Weight	450kg	457kg
Approach Velocity	2.86km/s	2.51km/s
ΔV at Mars	1.03km/s	0.77km/s
Fuel for CR	129kg	112kg
Orbiter Weight	321kg	345kg

SUMMARY

The interplanetary missions using the EP were studied on the basis of the proposed missions driven by the CR. In order to compete with the CR, the EP mission was designed within the severe restrictions to keep the proposed systems and the schedules as much as possible. It was found that the hybrid transfer using both the CR and the EP gains the extra weight because of the reductions of the escape velocity from Earth and the approaching velocity to the objective planet. The functions of the EP for the hybrid transfer were: the power throttling, the automatism, the redundancy and the initial operation. The sequential intermittent operation of the EP was proposed to adopt the spin-stabilized spacecraft. The Mars and the Venus probes have the different system designs on the electrical power and the communication sub systems. It was found that the Mar mission has advantage to the EP because the electricity is surplus on the transfer orbit. The Mars mission driven by the EP was designed with the results of the 10kg extra payload in the same mission schedule as the reference CR mission.

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