Design Study of a Korean Mars Mission

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ABSTRACT

In this paper we carried out a design study for an unmanned Mars mission suitable for Republic of Korea. The mission will use a KSLV series launch system, which is to place a one tonne payload into the LEO. We calculated the velocity increments(ΔV) required for departure from Earth and insertion into the orbit around Mars based on the mission opportunity data provided by NASA. Two types of Mars modules - entry type and orbiter type - were considered in this study. We calculated the mass of TPS(thermal protection system) for the entry type Mars module based on the heat transfer rate and heat load from the Mars atmosphere to the surface of the TPS. The heat transfer rate and heat load were obtained through an entry trajectory calculation. For the orbiter type Mars module, we calculated the mass breakdown of the additional spacecraft which is to insert the Mars module into the orbit around Mars. Other mass items were determined by proportioning from the existing Mars modules. This paper finally proposes the payload capacities for each types of Mars modules.

Key Words : Mars mission, KSLV, Mars Entry, TPS, Velocity Increment, Mass Budget

Introduction

A series of Mars exploration missions have been carried out by many space organizations in the world in response to the rising interest in that red planet. In this paper we carry out a design study for an unmanned Mars mission suitable for Republic of Korea.

Korea plans a series of launch systems named KSLV(Korean Space Launch Vehicle) to put several earth-bound satellites into an elliptic Earth orbit(EEO). The first one, KSLV-I, will put a 100kg payload into the 300km-1500km orbit. The KSLV-II and III will put a 1-tonne and 1.5-tonne payload into the same orbit, respectively([1],[2]). The KSLV-II is considered available for a Mars mission in this paper.

In order to send a spacecraft to Mars, the payload on top of a KSLV-II vehicle will fire a kick motor from the EEO. The resulting ΔV enables the spacecraft to enter the trans-Mars-orbit along which it makes a journey to Mars. When the spacecraft reaches Mars, it may or may not execute a burn depending on its mission. The total ΔV required for an overall mission maneuver and the specific impulse(Isp) of the spacecraft's engine define the so-called mass budget. Korea is currently capable of producing a solid rocket engine with an Isp of 290sec, which can be used as the kick motor.

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When the spacecraft approaches Mars, there come two choices for the mission. One is the so-called entry-type mission and the other the orbiter-type. In the entry-type mission, the spacecraft directly enters the Mars atmosphere from its trans-Mars orbit. It decelerates with the help of the aerodynamic drag force in the Mars atmosphere to finally land on the ground. It requires a thermal protection system(TPS) to protect itself from the aerodynamic heating during the entry flight. The Mars Pathfinder[3] is one of the entry-type missions. Because the journey to Mars lasts only about 7~14 months for entry-type missions, the power to the spacecraft can be supplied by a primary battery.

On the other hand, in the orbiter-type mission, the spacecraft transfers into the orbit around Mars from the trans-Mars orbit. To do so, it requires an additional ΔV. It also needs a solar panel to power the communications equipments while it keeps orbiting around Mars throughout its life. The Mars Express[4] by the European Space Agency and Mars Odyssey[5] by NASA are the examples of the orbiter-type mission.

In this paper we determine the ΔV required for both mission types (entry and orbiter) by using the orbit dynamics equations and the mission opportunity data provided by NASA[6]. Other parameters are determined by utilizing the Mars Pathfinder data for the entry-type mission, and Mars Express and Mars Odyssey for the orbiter-type mission. We then calculate the atmospheric entry trajectory for the entry type mission. The heat transfer rate and heat load are calculated via the Fay-Riddell[7] method to estimate the dimensions and mass of the TPS. Through these, the conceptual design of the Korean Mars mission is completed.

Calculation of ΔV requirements

We conducted the analysis of the spacecraft's maneuvering path to estimate the ΔV requirements based on the mission opportunity data[6]. It provides us with the C3L and VHP by launching dates. The values are tabulated in Table 1.

Table 1. C3L and VHP values by launch dates

Departure Date	C3L	Arrival Date	VHP
1990/09/10	14.3890	1991/10/05	2.3958
1992/09/30	11.7320	1993/09/19	2.3987
1994/10/24	9.4674	1995/08/27	2.5142
1996/11/21	8.9323	1997/09/29	2.8683
1999/02/07	8.4403	1999/12/31	3.3391
2001/04/16	7.8538	2002/01/27	3.7766
2003/05/10	12.5640	2003/12/29	2.7652
2005/09/02	15.4450	2006/10/08	2.4668

The C3L and VHP represent the characteristic of the trans-Mars orbit. The former represents the energy required to transfer from an Earth orbit into the trans-Mars orbit at departure from Earth and the latter the velocity relative to Mars on the trans-Mars orbit at arrival at Mars.

The ΔV requirements for each orbit transfer can be calculated through the two-body equation of motion[8], wherein the primary body is Earth or Mars and the secondary body is the spacecraft.

$$\ddot{\vec{r}} + (\mu r^{-3})\vec{r} = 0 \ , \ \mu = GM \tag{1}$$

where G and M represent the universal constant of gravitation and the mass of the primary body, respectively. See Fig. 1.

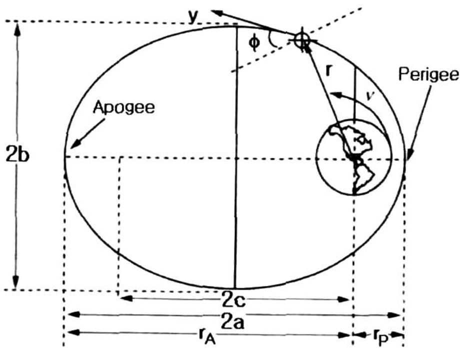

Fig. 1. Geometry of a typical elliptical orbit.[8]

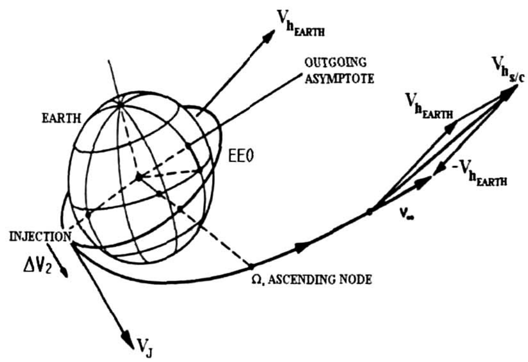

Fig. 2. The trans-Mars orbit at departure.[6]

The derived $\triangle$Vs are,

$$\triangle V_2 = \sqrt{C3L + \frac{2\mu_e}{r_{P,e}}} - \sqrt{\frac{2\mu_e r_{A,e}}{r_{P,e}(r_{A,e}+r_{P,e})}} \tag{2}$$

from an EEO to the trans-Mars orbit and

$$\triangle V_3 = \sqrt{VHP^2 + \frac{2\mu_m}{r_{P,m}}} - \sqrt{\frac{2\mu_m r_{A,m}}{r_{P,m}(r_{A,m}+r_{P,m})}} \tag{3}$$

from the trans-Mars orbit to the orbit around Mars, where r_P and r_A represent the periapsis and apoapsis of Earth(e) and Mars(m), respectively. See Fig. 1 and 2.

Earth and Mars get in the same relative positions every 2.14 years(synodic period). There come the optimal days for the mission requiring minimum departure energy once during this synodic period. Each cycle of 7 consecutive Martian mission opportunities amounts to 5499.55 days and shows nearly repetitious characteristics. A much closer repetition occurs after 11699.031 days[6]. Using this repetitive feature of Earth-Mars system, we could calculate the $\triangle$V requirements for the future mission opportunities up to the year 2037 from the NASA opportunity data covering 7 consecutive opportunity cycles, say, from the year 1990 to 2005. In Fig. 3, the resulting $\triangle V_2$ and $\triangle V_2 + \triangle V_3$ at each optimal days are shown from the year 1990 to 2037. The orbital parameters used in the calculation are tabulated in Table 2. The values at Mars were inferred from those of the Mars Express[4].

Table 2. Orbital parameters at Earth and Mars

Earth		Mars[4]	
$r_{P,e}$	300km	$r_{P,m}$	258km
$r_{A,e}$	1,500km	$r_{A,m}$	11,560km

The procedure of our Mars mission is,

(1) The KSLV places the spacecraft on the 300km-1500km EEO. The mass of the spacecraft is 1-tonne at this EEO.

(2) The spacecraft is boosted into the trans-Mars orbit to set for Mars. It burns fuel to acquire the $\triangle V_2$ of 3.2~3.6km/s. See Fig. 3.

The amount of fuel consumed is represented by the ratio of the initial mass to the final mass. This mass ratio is calculated from the force-free rocket equation.

$$\frac{M_{\text{initial}}}{M_{\text{final}}} = \exp\left[\frac{\Delta V}{I_{sp} * 9.8}\right] \tag{4}$$

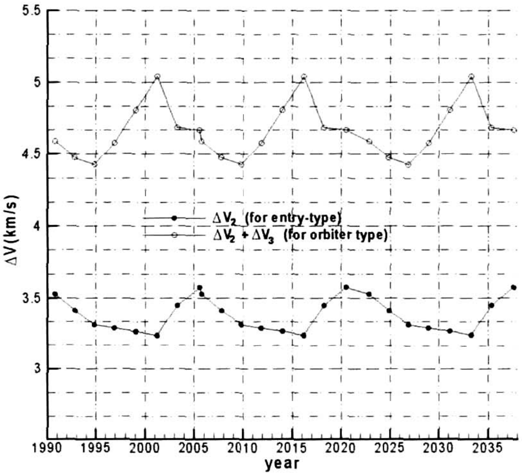

Fig. 3. ΔV requirements for the entry and orbiter-type missions

The solid rocket engine with an Isp of 290sec will be considered in this study.

(3) Mid-course correction is made with a small hydrazine engine. This engine is the same as that used for attitude control of Earth-bound satellites launched by Korea.

(4) Cruising toward Mars for about 200 days. When it is near Mars:

(5-1) In the orbiter-type mission, the spacecraft carries out a retro-burn and inserts itself into the orbit around Mars. The geometry of the orbit around Mars is similar to that of Mars Express. The resulting ΔV_3 is about 1~1.8km/s. See Fig. 3.

(5-2) In the entry-type mission, the entry, descent, landing(EDL) sequence commences.

Mass Budget

From Fig. 3, one can determine the optimal dates when the ΔV requirements are minimum for each of the entry and orbiter type missions. The optimal dates for the entry-type mission are when ΔV_2 is the minimum. The optimal dates for the orbiter-type mission are when $\Delta V_2+\Delta V_3$ is the minimum. Note that the orbiter-type mission requires higher ΔV because of the Mars-orbit insertion. At one of these optimal dates, we considered the mass particulars of the spacecraft system.

The spacecraft system for the Mars mission consists of the bus and the Mars module. The bus is responsible for carrying the Mars module to Mars. The bus is equipped with a propulsion system for the ΔV needed for departure. When spacecraft system arrives near Mars, the bus is separated from the Mars module and abandoned. The structural mass of the bus is determined from the mass of the propellant. Generally, it is about 10% of the propellant mass. Referring to the past Mars missions([9],[10]), we presume that the trajectory correction maneuvers (TCM) usually requires total ΔV of no more than 50m/s. See Table 3. We added a ΔV margin of 1km/s for safety.

Table 3. ΔVs of TCMs for the past Mars missions

	ΔV required for TCMs, m/s				
	TCM1	TCM2	TCM3	TCM4	TCM5
Pathfinder[9]	33.3	2.08	0.432	0.138	0.2~2.
Mars Global Surveyor[10]	21.1	5.6	0.197	0.26	

The mass of each part is calculated from the following equation[8]. See Table 4.

$$
\begin{aligned}
v_f - v_i &= -C\ln r_m \\
&= -C\ln[r_s(1-r_l)+r_l] \\
&= -C\ln[r_l(1-r_s)+r_s]
\end{aligned}
\tag{5}
$$

$$
r_m = \frac{m_f}{m_i}\ ,\ r_s = \frac{m_s}{m_p + m_s}\ ,\ r_l = \frac{m_l}{m_i}
$$

where $m_i = m_p + m_s + m_l\ ,\ m_f = m_s + m_l$

and the subscripts i, f, p, s, and l represent initial, final, propellant, structure and Mars module, respectively.

Table 4. Mass of each component(ΔV includes a margin of 1km/s)

ΔV(km/s)		4.242
mass(kg)	propellant	775
	bus	86
	Mars module	139
	Total	1,000

The Mars module is the one which finally survives to complete the mission objectives at Mars. Now we consider the Mars module to be used for each of the entry and orbiter type missions.

Mars module - entry type

A typical EDL scenario is;

(1) The module enters the Mars atmosphere at a chosen entry angle.

(2) The module descends toward the ground decelerating due to the aerodynamic drag force. The TPS endures aerodynamic heating.

(3) A parachute deploys for further deceleration at Mach number of about 2. The TPS, having finished its function, is jettisoned.

(4) The payload lander gets out of the module with a bridle connecting them.

(5) Near the ground, the lander is completely separated from the module. The lander finally lands on the ground with the help of an airbag system.

(6) The lander is deployed, and begins its functions. The data obtained by the lander is transmitted to one of the orbiters circling Mars.

We calculated the EDL trajectory of the Mars module of the entry type mission to determine the proper specification of the TPS. We used the atmospheric density profile from the Martian atmospheric model provided by the NASA Glenn Research Center[12]. We considered the trajectory before the deployment of the parachute(step (3)). The geometry of the module was determined by downsizing from that of the Pathfinder. See Fig. 4. The module was assumed to move in 3 degrees of freedom and the parameters needed for the calculation were determined by referring to the values of the Pathfinder[11]. See Table 5.

The heat transfer rates and heat load at the stagnation point corresponding to the previously calculated EDL trajectory were calculated via the Fay-Riddell method[7]. The results are shown in Fig. 5.

We considered the SLA-561V used for the Pathfinder as the TPS material. We estimated the TPS thickness by proportioning from that of the Pathfinder based on the heat load value. We then reduced this thickness by 10% because the Pathfinder flight data indicated that the bondline temperature rose only to about 260K[13]. The resulting TPS thickness and the corresponding TPS mass for the forebody are 2.18cm and 11kg, respectively. Other components such as the parachute system, bridle, air bag, etc. were determined by proportioning from those of the

Table 5. Parameters used in the trajectory calculation

mass at entry*a	139kg
base area of vehicle*b	1.298m^2
drag coefficient of vehicle*b	1.7
nose radius of vehicle*b	0.322m
initial height	125km
VHP	3.777km/s
inertial entry velocity*c	6.866km/s
relative entry velocity*c	6.657km/s
initial entry angle*b	-14.06°
latitude of entry location*b	30°

*a: Result of eq.(5). See Table 4.
*b: Inferred from the Pathfinder.[11]
*c: Intermediate results of the trajectory calculation.

Pathfinder and Mars airplane vehicle[14]. The final mass breakdown for the entry-type Mars module is tabulated in Table 6.

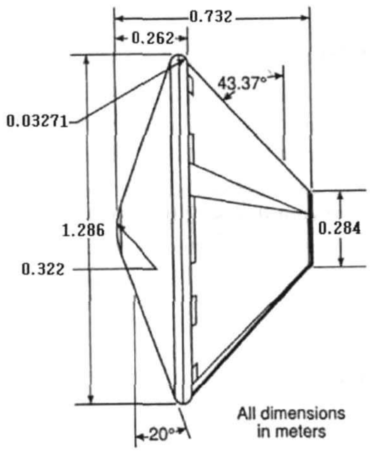

Fig. 4. Geometry of the Mars module - entry type

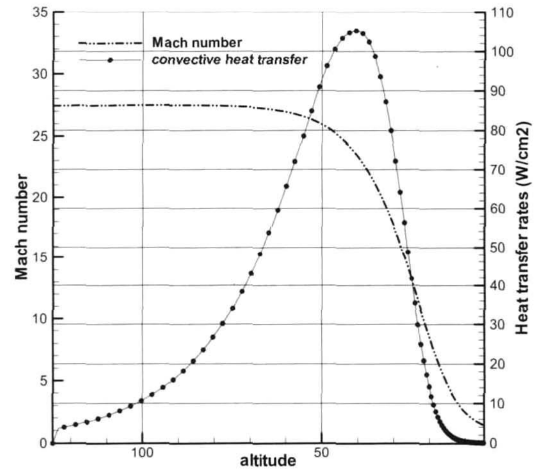

Fig. 5. Mach number and the stagnation point heat transfer rate as functions of the altitude(heat load=4.92kJ/cm^2)

Mars module - orbiter type

After separation from the bus, the Mars module for the orbiter-type mission fires its own engine to transfer to the orbit around Mars. A ΔV of 1.22km/s is needed for this orbit transfer including the ΔV of 0.1km/s for safety margin. The amount of propellant to acquire the ΔV and the corresponding masses of the structure and the payload were calculated again through the equation (5). The final mass estimates are tabulated in Table 7. These mass values were determined by proportioning from those for Mars Odyssey[5]. Note that the mass of the electrical power system occupies a much greater portion of the total mass of the orbiter-type Mars module than the entry-type one. During the cruise from Earth to Mars, only a low-power, low-gain transmitter is needed because the amount of data to be transmitted is small.

Table 6. Mass breakdown for the entry-type mission

Component		Mass Estimate(kg)
TPS	forebody heatshield	11
	separation assembly	10.5
	Afterbody heatshield	3.5
Structure	Forebody	6.7
	Afterbody	10.7
Airbag system	Material	20
	Gas generator	2.4
	Airbag retraction motor	1.5
Chute & Bridle		6
Thermal control		1.4
Batteries+electronics[14]		4
Payload		61.3
Total		139

Table 7. Mass breakdown for the orbiter-type mission

Component		Mass Estimate(kg)
Propulsive system (ΔV=1.22km/s)	Propellant	48.5
	Structure	5.5
Electrical power (solar panel & batteries)		46
Thermal control		8
Mechanisms (to operate the solar panel, high gain antenna, etc.)		10
Payload		21
Total		139

The entry-type spacecraft needs only such a small transmitter[14]. A high-power, high-gain transmitter is needed to transmit the scientific data obtained at Mars to Earth. The orbiter-type Mars module needs such a large transmitter.

Discussion

As shown above, the entry-type mission allows a payload of 61.3kg on the surface of Mars. The payload is similar to those of Spirit[15] and Opportunity[16] by NASA. The orbiter-type mission allows a payload of 21kg. This study does not specify what kind of payload the entry-type Mars module should carry. There are three possibilities. They are; 1) surface rover, 2) soil penetrator, and 3) airborne payload such as airplane[14], helicopter[17], or balloon. The predicted payload of 61kg allows all these choices. This topic will have to be pursued in the future.

Conclusion

We carried out a conceptual design of a Mars mission feasible with the KSLV-II launch system for both the entry-type and orbiter-type mission. The required ΔV and corresponding mass budget were first determined. Then the entry trajectory and the resulting heat load have been calculated to determine the dimensions and mass of the TPS. Through these, the mass breakdown for each of the entry-type and orbiter-type missions was derived. The useful mass of the entry-type and orbiter-type mission were determined to be 61.3kg and 21kg, respectively.

References

1. 오범석, 이준호, 노웅래, 조미옥, 박정주, 조광래, "발사체의 개념설계", 한국항공우주학회지, 제30권, 제6호, 2002, pp.130-141.

2. 이효근, 류정주, "나로 우주센터 소개", 한국항공우주학회지, 제32권, 제1호, 2004, pp.123~130.

3. http://nssdc.gsfc.nasa.gov;National Space Science Data Center - Master Catalog : Spacecraft-Pathfinder.

4. http://sci.esa.int/science-e/www/area/index.cfm?fareaid=9;ESA website.

5. Saunders, R. S., Arvidson, R. E., Badhwar, G. D., Boynton, W. V., Christensen, P. R., Cucinotta, F. A., Feldman, W. C., Gibbs, R. G., Kloss Jr, C., Landano, M. R., Mase, R. A., Mcsmith, G. W., Meyer, M. A., Mitrofanov, I. G., Pace, G. D., Plaut, J. J., Sidney, W. P., Spencer, D. A., Thompson, T. W., and Zeitlin, C. J., "2001 Mars Odyssey Mission Summary", *Space Science Reviews*, Vol. 110, 2004, pp.1-36.

6. Sergeyevsky, A. B., Snyder, G. C., and Cunniff, R. A., "Interplanetary Mission Design Handbook, Volume I, Part2 - Earth to Mars Ballistic Mission Opportunities, 1990-2005", *NASA JPL Publication 82-43*, Sep. 15, 1983.

7. Fay, J. A. and Riddell, F. R., "Theory of Stagnation Point Heat Transfer in Dissociated Air", *Journal of Aeronautical Science*, Vol. 25, No.2, Feb. 1958, pp.73-85,121.

8. Wertz, J. R. and Larson, W. J., "Space Mission Analysis and Design", *Space Technology Library, 3rd edition*, 1999.

9. http://mpfwww.jpl.nasa.gov/MPF/mpf/mpfnavpr.html.

10. http://mars.jpl.nasa.gov/mgs/pdf/405.PDF, "Mars Global Surveyor Project Mission Plan Document", Final version, Rev.B, Nov. 1996.

11. Spencer, D. A., Kallemeyn, P. H., and Thurman, S. W., "Mars Pathfinder Entry, Descent, and Landing Reconstruction", *Journal of Spacecraft and Rockets*, Vol. 36, No. 3, May-June 1999, pp.357~366.

12. http://www.grc.nasa.gov/WWW/K-12/airplane/atmosmre.html.

13. Milos, F. S., Chen, Y. K., Congdon, W. M., and Thornton, J. M., "Mars Pathfinder Entry Temperature Data, Aerothermal Heating, and Heatshield Material Response", *Journal of Spacecraft and Rockets*, Vol. 36, No. 3, May-June 1999, pp.380~390.

14. Gage, P. J., Allen Jr, G. A., Park, C., Brown, J. D., Wercinski, P. F., and Tam, T. C., "A Loaf-Shaped Entry Vehicle For a Mars Airplane(The Best Thing Since Sliced Bread?)", *AIAA paper* 2000-0634.

15. http://nssdc.gsfc.nasa.gov;National Space Science Data Center - Master Catalog : Spacecraft-Spirit.

16. http://nssdc.gsfc.nasa.gov;National Space Science Data Center - Master Catalog : Spacecraft-Opportunity.

17. Young, L. A., Gulick, V., Aiken, E. W., Mancinelli, R., and Briggs, G. A., "Rotorcraft as Mars Scouts", *2002 IEEE Aerospace Conference*, Big Sky, MT, Mar 9-16, 2002.