ISAS MERCURY ORBITER MISSION TRAJECTORY DESIGN STRATEGY

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Abstract

This paper shows the recent results of ISAS Mercury orbiter mission study from trajectory design point of view, conducted by ISAS Mercury Exploration Working Group. Three options were studied; 1) Multiple Mercury flyby mission via Solar Electric Propulsion (SEP), 2) SEP orbiter, which uses SEP as a primary propulsion system for interplanetary transfer phase as well as Mercury orbit insertion phase and 3) Conventional bi-propellant chemical propulsion option. Detailed mission design description for SEP flyby and orbiter mission are noted in Ref.1 and Ref.2 respectively, while this paper focuses on the trajectory design strategy for the ballistic option, which is considered as the current nominal scenario (see also Ref.3 and Ref.4).

Key Words: Mercury Orbiter, Multiple Swingbys

Introduction

The planet Mercury is not well scientifically disclosed and Mercury is left one of the most attractive planets in the solar system exploration. Only flybys were attempted before by the 1973-1975 Mariner 10 mission. Since then, several Mercury missions have been proposed including JPL's Mercury Dual Orbiter (1999 Titan-IV launch[5]), JPL&TRW's Discovery Hermes Orbiter (1999 Delta-II launch[6]), ESA's Mercury Orbiter (2004 Ariane V launch[7]) based on C.W.Yen's ballistic trajectory with multiple Venus and Mercury swingbys[8-9] and recent Delta and Ariane class Solar Electric Propulsion (SEP) mission proposals[10,11].

Interplanetary Trajectory

The reference interplanetary sequence from Earth departure to Mercury orbit insertion is based on JPL's C.W.Yen's multi-revolution and multiple swingby trajectory[8-9]. Launch is postulated in August of 2005 (C3.dep=16 km^2/s^2), which is followed by two Venus swingbys and two Mercury swingbys (see Fig.1). The spacecraft orbit from Venus to Venus is one-to-one synchronous with Venus orbit. This synchronous Venus swingbys are dedicated to lower the aphelion of the orbit nearly to the radius of Venus's orbit, and the perihelion distance to that of Mercury.

Two Mercury swingbys are introduced in order to reduce the relative velocity with respect to Mercury, by changing the Mercury arrival position with a small amount of Deep Space Maneuver (DSM) in the vicinity of aphelion passage. This is called reverse delta-V Mercury gravity assist technique, where small DSM results in a large saving of Mercury capture delta-V. The trajectory leg between the 1st and 2nd Mercury swingby is two-to-three synchronous, which means the spacecraft makes two revolutions around the Sun while Mercury makes three. The next trajectory arc between the 2nd and 3rd Mercury swingbys is in three-to-four synchronization with Mercury orbit. The Mercury Orbit Insertion (MOI) occurs in September of 2009 and total flight time would be 4.2 years. Orbital information of the interplanetary phase is summarized in Table 1 through Table 4 and in Fig.2 and Fig.3. The proposed trajectory design method in the following section was utilized to design this trajectory assuming Sun-spacecraft two-body motion.

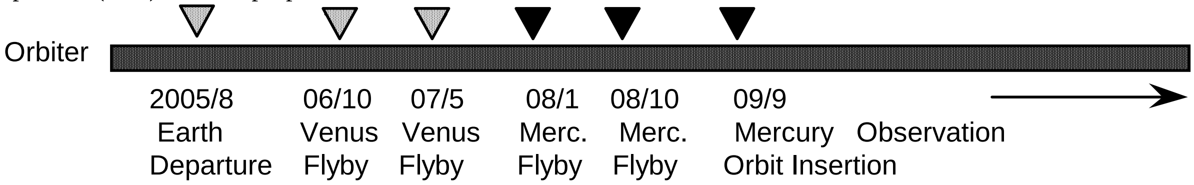

Fig.1: Interplanetary Trajectory Sequence Outline

Table 1: Interplanetary Trajectory Sequence

Event	S/C-Sun distance (AU)	Epoch (Year Month Day)
Earth	1.0144	2005 08 05
1st DSM	.60675	2005 12 15
1st Venus	.72228	2006 10 26
2nd DSM	.88987	2007 01 08
2nd Venus	.72228	2007 06 08
3rd DSM	.74283	2007 11 11
1st Mercury	.34945	2008 01 17
4th DSM	.69052	2008 03 21
2nd Mercury	.33980	2008 10 08
5th DSM	.62659	2008 12 07
3rd Mercury	.31184	2009 09 30

* DSM: Deep Space Maneuver

Table 2: Flight Time (day / year /revolution)

Event-Event	day	year	revolution
EARTH -1st DSM	130.84	.358	.49128
1st DSM-1st VENUS	314.62	.861	1.1457
1st VENUS -2nd DSM	74.111	.203	.33019
2nd DSM-2nd VENUS	150.59	.412	.67108
2nd VENUS -3rd DSM	156.46	.428	1.0859
3rd DSM-1st MERCURY	66.141	.181	.46451
1st MERCURY-4th DSM	64.814	.177	.49189
4th DSM-2nd MERCURY	200.88	.550	1.5196
2nd MERCURY-5th DSM	61.019	.167	.52411
5th DSM-3rd MERCURY	297.92	.816	2.5317
Total Time	1517.4	4.15	

Table 3: Swingby Conditions

	Venus-1	Venus-2	Mercury-1	Mercury-2
ALT	3449.4	300.00	200.00	200.00
VEL	8.8270	8.8265	5.7763	5.1436
I.DEC	14.122	-17.608	0.4007	6.0829
I.RAS	-142.73	-158.91	116.93	135.29
O.DEC	-17.607	-8.5769	6.3840	6.3683
O.RAS	-158.90	153.92	139.27	163.21

ALT: swingby altitude (km)
VEL: relative velocity (km/s)
I.DEC: in-coming asymptote declination (deg)
I.RAS: in-coming asymptote right ascension (deg)
O.DEC: out-going asymptote declination (deg)
O.RAS: out-going asymptote right ascension (deg)
(with respect to Earth Ecliptic Plane)

Table 4: Mercury Arrival Geometry

Epoch	2009 09 30	
C3	(km2/s2)	11.368
Relative Velocity	(km/s)	3.3717
Declination	(deg)	6.9406
Right Ascension	(deg)	145.69

(with respect to Earth Ecliptic Plane)

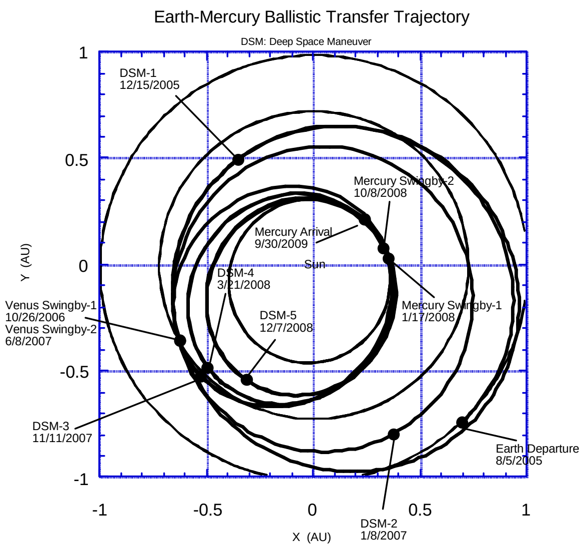

Fig.2: Earth-Mercury Interplanetary Trajectory

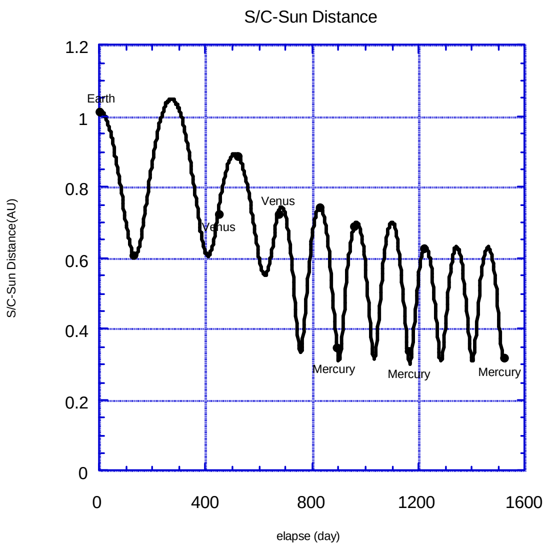

Fig.3: S/C-Sun Distance History

Trajectory Design Algorithm

Three trajectory design algorithms were utilized for the design of the preceding Earth-Mercury multiple swingby trajectory (see Table 5-7 and Fig.4-6). The first algorithm refers to Ref.12, while the second and third algorithms were newly developed to take extremely high nonlinearity into account. The numbers listed in Table 5 through 7 correspond to the parameter number assuming Earth-Venus-Venus-Mercury-Mercury-Mercury trajectory.

Algorithm 1 (see Table 5, Fig.4 and Ref.12): The whole trajectory is divided into several trajectory legs which start from a planet to a breakpoint or from a breakpoint to the next planet. Delta-V is applied at the breakpoints. The control parameters consist of six swingby parameters (swingby epoch, relative velocity, target plane angle, swingby altitude, approach asymptote right ascension and declination), terminal conditions at departure and arrival planets (epoch, relative velocity, asymptote right ascension and declination) and delta-V epochs between swingbys. Once these control parameters are given, the whole sequence is obtained by forward and backward integration from the swingby points, while there are positional discrepancies at breakpoints. Non-Linear Programming (NLP) method is utilized for this parameter optimization problem, in order to make these positional discrepancies at breakpoints converged to zero, while maximizing the spacecraft mass at arrival.

Table 5: Trajectory Design Algorithm-1 (Ref.12)

Control Parameters (37 parameters)
- Swingby parameters (6 x 4 = 24)
  - swingby epochs
  - relative velocity
  - target plane angle
  - swingby altitude
  - asymptote right ascension
  - asymptote declination
- Terminal parameters (4 x 2 = 8)
  - terminal epochs
  - relative velocity
  - asymptote right ascension
  - asymptote declination
- Delta-V parameters (5)
  - delta-V epochs

Inequality Conditions
- swingby altitude > minimum

Equality Conditions
- delta-V position: forward = backward

Objective Function
- Spacecraft mass at arrival

Algorithm 2 (see Table 6 and Fig.5-1 & Fig.5-2): The second algorithm was newly developed to decrease nonlinearity by making the whole optimization process into inner loop trajectory generation and outer loop NLP optimization.

Table 6: Proposed Trajectory Design Algorithm-2

INNER LOOP

Swingby: Control Parameters (6)
- swingby epochs
- relative velocity
- target plane angle
- swingby altitude
- asymptote right ascension
- asymptote declination

Swingby: Target Parameters (6)
- delta-V position (x, y, z) prior to swingby
- delta-V position (x, y, z) after swingby

Terminal: Control Parameters (3)
- relative velocity
- asymptote right ascension
- asymptote declination

Terminal: Target Parameters (3)
- delta-V position (x, y, z)

OUTER LOOP

Control parameters (22)
- Earth departure epoch (1)
- Arrival epoch (1)
- delta-V epoch (5)
- delta-V position (5 x 3 = 15)

Inequality Conditions
- swingby altitude > minimum

Equality Conditions
- None

Objective Function
- Spacecraft mass at arrival

For the inner loop, positions of the delta-Vs are given beforehand, and six swingby parameters (swingby epoch, relative velocity, target plane angle, swingby altitude, asymptote right ascension and declination) are obtained to satisfy the six delta-V position components before and after swingby. Therefore after the inner loop process, the multiple swingby trajectory is always generated with no positional discontinuous point. Then outer loop optimization process finds the optimal solution with maximum spacecraft mass (minimum total delta-V), where departure and arrival epochs, delta-V epochs and positions are the control parameters to be optimized. The merit of this method is, by introducing the inner loop process which assures continuous trajectory generation, the number of the control parameters for the outer loop optimization is diminished which increases convergence property of the optimization.

Algorithm 3 (see Table 7 and Fig.6-1 & Fig.6-2): The inner loop and outer loop structure of the optimization process is the same with that of Algorithm 2.

The difference is the number of the control parameters of the inner loop. Instead of six swingby parameters (swingby epoch, relative velocity, target plane angle, swingby altitude, asymptote right ascension and declination), three swingby parameters (relative velocity, right ascension, declination) are used. And the trajectory leg prior to and after swingby is treated separately which furthermore increases the convergence property of the inner loop. After the inner loop process, the multiple swingby trajectory is generated with no positional discontinuous point like Algorithm 2. The velocity discontinuity at swingby points, which is the incoming and outgoing relative velocity difference, is reduced by including in the equality constraint in the outer loop NLP optimization.

This Algorithm-3 shows the best convergence property and was used for Earth-Mercury multiple swingby trajectory design shown in the previous section.

Table 7: Proposed Trajectory Design Algorithm-3

INNER LOOP
Control Parameters (3)
relative velocity
asymptote right ascension
asymptote declination
Target Parameters (3)
delta-V position (x, y, z)
OUTER LOOP
Control parameters (26)
Earth departure epoch (1)
Swingby epochs (4)
Arrival epoch (1)
delta-V epoch (5)
delta-V position (5 x 3 = 15)
Inequality Conditions
swingby altitude > minimum
Equality Conditions
incoming = outgoing relative velocity
Objective Function
Spacecraft mass at arrival

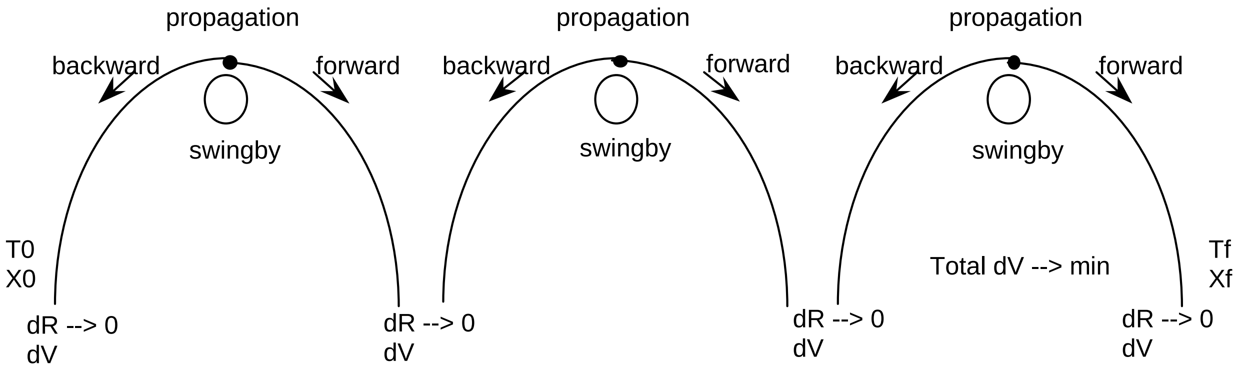

Fig.4: Trajectory Design Method-1 (Ref..12)

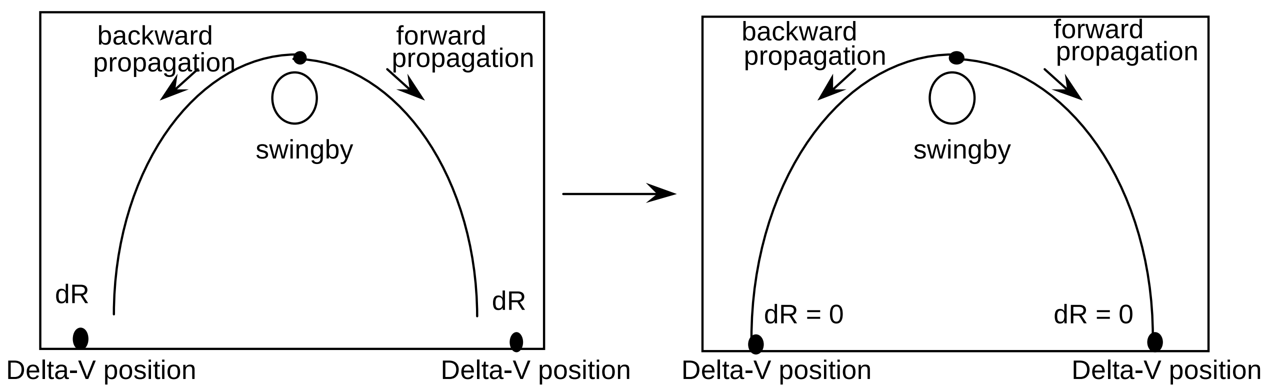

Fig.5-1: Trajectory Design Method-2 (Inner Loop)

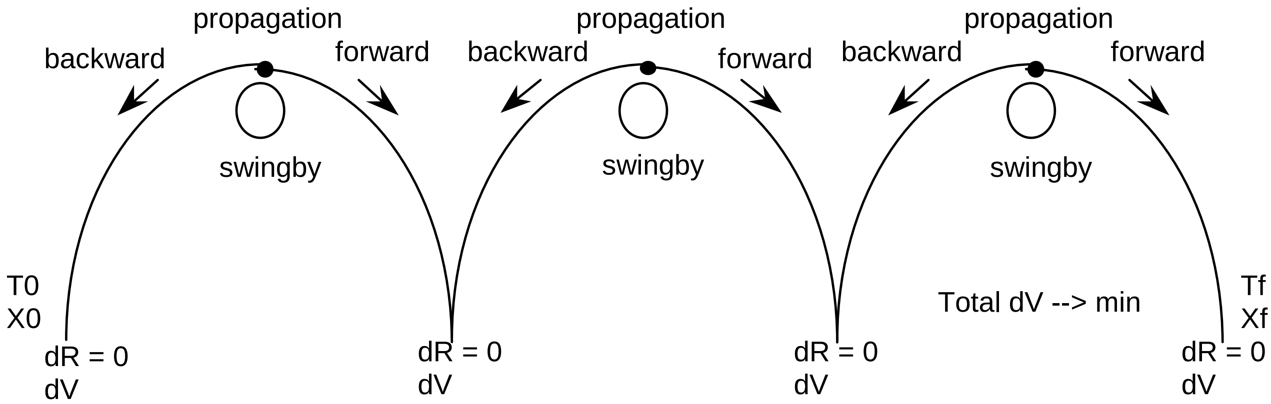

Fig.5-2: Trajectory Design Method-2 (Outer Loop)

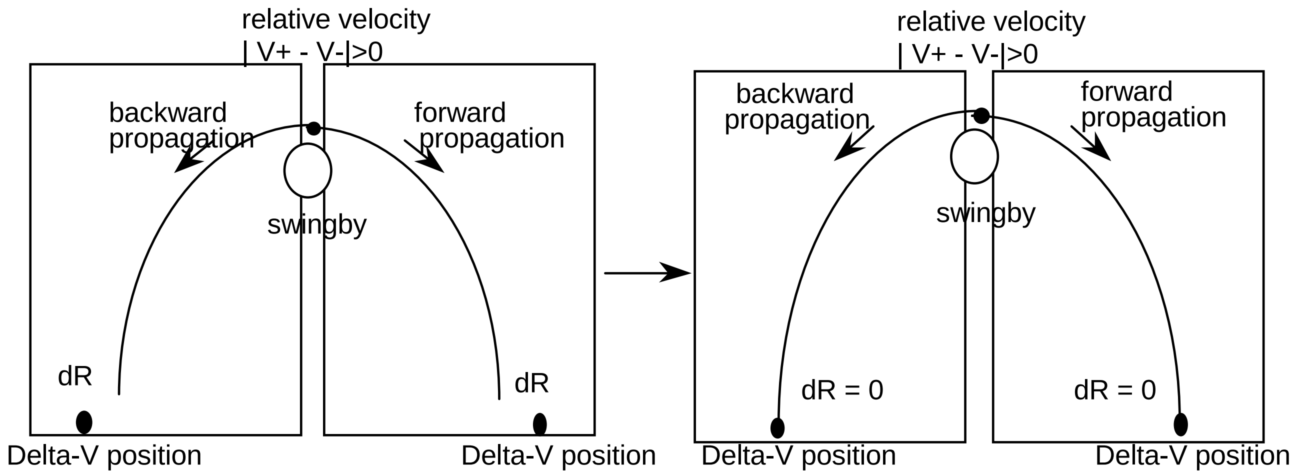

Fig.6-1: Trajectory Design Method-3 (Inner Loop)

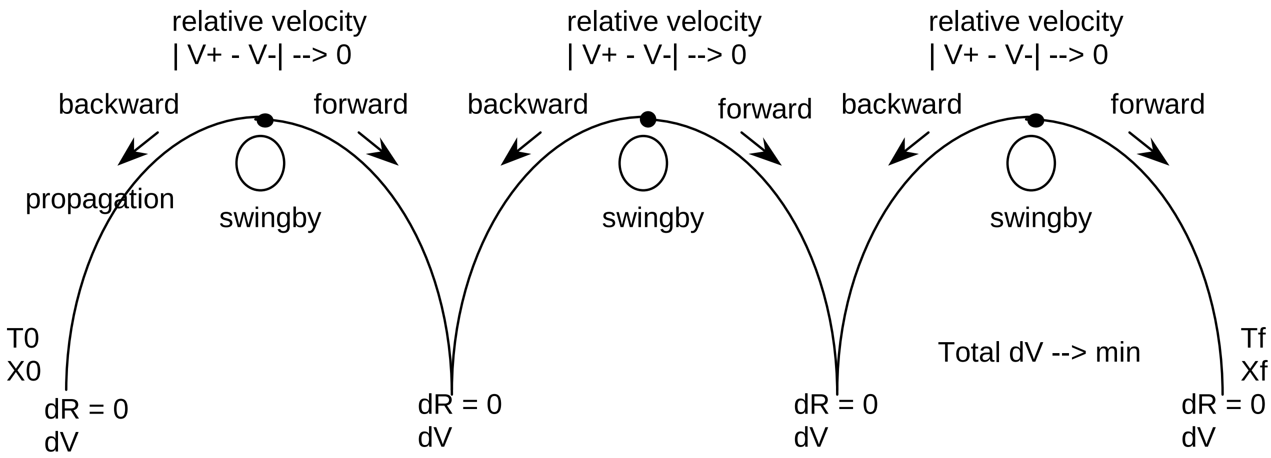

Fig.6-2: Trajectory Design Method-3 (Outer Loop)

Mercury Orbit Insertion

From fields & particles observation and planetology investigation points of view, 0.5 day period polar orbit (300 km x 6 Mercury radii altitude, 90 deg inclination) is tentatively selected for Mercury observation orbit. Lower apoherm is better since higher apoherm altitude suffers larger solar perturbation on the periherm altitude which results in larger delta-V for orbit maintenance. Argument of periherm is determined mainly to avoid long shadow and to get good scientific mapping capability (see Table 8). Table 8 summarizes the length of shadow duration in orbiting phase. It assumes Mercury at its aphelion position and apoherm direction in anti-Sun direction. The baseline 300 km x 6 rM 12 hour period orbit experiences 1.5 hour shadow if the argument of periherm is 30 deg, which is considered nominal for this mission.

There are basically two methods to finally attain polar 12 hour orbit (300km x 6 Mercury radii) with 30deg periherm latitude assuming an approaching asymptote with 7 deg declination (see Table 4).

1) Polar Capture Orbit: This strategy requires two impulses. Four day period polar orbit is tentatively assumed for initial capture orbit. The periherm latitude around 60deg is determined from the declination of the approaching asymptote. Then periherm latitude and apoherm altitude is controlled by the second maneuver to 30 deg and 6 rM respectively (axis change). Longer initial capture orbit period results in less delta-V.

2) Low Latitude Periherm Capture Orbit: This requires three impulses. Initial capture orbit is four day period 30 deg periherm latitude orbit (near 40 deg inclination). Then orbital plane is controlled by maneuver at the apoherm passage from 30 deg to 90 deg. Then finally apoherm altitude of 6 rM is achieved by the 3rd maneuver at periherm passage.

Table 8: Mercury Orbiting Phase: Shadow Duration

i (deg)	arg (deg)	per.h (km)	apo.h (rM)	period (hour)	umbra (min)	penumbra (min)
90	0	300	6	12	127	146
90	30	300	6	12	91	100
90	60	300	6	12	52	55
90	90	300	6	12	32	33
90	0	300	10	21	185	231
90	30	300	10	21	114	130
90	60	300	10	21	57	61
90	90	300	10	21	33	34
90	0	300	15	36	240	337
90	30	300	15	36	133	156

i: inclination (deg), arg: argument of periherm (deg)
per.h: periherm altitude (km)
apo.h: apoherm altitude (Mercury radii)
umbra & penumbra: shadow per revolution (min)

When the relative velocity at Mercury arrival is 3.37 km/s, total delta-V for Mercury orbit insertion is 2221 m/s for the first sequence and 1652 m/s for the second sequence assuming a simple model. Therefore the second sequence with three impulses was adopted. It is summarized in Table 9 with the meaning of each delta-V. The trajectory was numerically integrated using full model with other planets' gravity terms and solar pressure effect. There are also operational requirements such as Sun angle constraint (90 deg +/- 20 deg) at delta-V maneuver, which are also satisfied in the sequence. Table 10 shows the initial conditions of scientific observation orbit, which is the consequence of the Mercury orbit insertion sequence noted in Table 9.

Table 9: Mercury Orbit Insertion Sequence

Epoch (UTC)	delta-V	Effect
2009 09/30 19h	1300 m/s	4 day Period apoherm 30 rM
2009 10/02 18h	3 m/s	Periherm altitude control
2009 10/06 16h	129 m/s	Plane change to polar orbit
2009 10/10 11h	23 m/s	Periherm altitude control
2009 10/12 06h	220 m/s	Apoherm reduction to 12h period

Table 10: Initial Conditions of Scientific Observation Orbit

Mercury Orbit Insertion +11.95 days	(2009 10/12 6h)
Period:	0.51 days
Periherm altitude:	313 km
Apoherm altitude:	6.3 rM
Inclination (Earth Ecliptic)	84.9 deg
Periherm latitude (Earth Ecliptic)	-37.1 deg

Mercury Orbiting Phase

Mercury orbiting phase will last about half Earth year to have complete mapping, taking into account of Mercury's rotation period's (59 days) two-thirds resonance with orbital period (88 days). Half Earth year is the minimum requirement from scientific observation point of view.

Fig.7 through Fig.10 show the orbital evolution during Mercury orbiting phase. The trajectory was numerically integrated using full model with other planets' gravity terms and solar pressure effect. The drag due to the tenuous atmosphere and the perturbation due to the higher order gravity harmonics are neglected since their effects are considered small enough compared to the other terms.

Fig.7 shows the periherm and apoherm altitude evolution. Without any orbit control, periherm altitude has an increasing tendency, which prevents the spacecraft from immediate crash on the Mercurian surface. Periherm altitude is controlled around 500 km in this case. Fig.8 is the required delta-V for periherm altitude control. The delta-V for orbit maintenance is 50 m/s for half year, and the mission life time depends on the fuel allotted for this orbit maintenance. Both the interval between the orbit maintenance delta-Vs and the target periherm altitude are arbitrary, and can be determined by the requirement such as the orbit determination precision for gravity harmonics estimation.

Fig.9 summarizes the length of shadow duration in orbiting phase. Maximum shadow duration is around 90 minutes per revolution which satisfies the battery requirement. Fig.10 is the average distance from Mercury during shadow passage. Shadow occurs when the spacecraft is at the distance of 1.5 or 4.5 Mercury radii from the Mercury center.

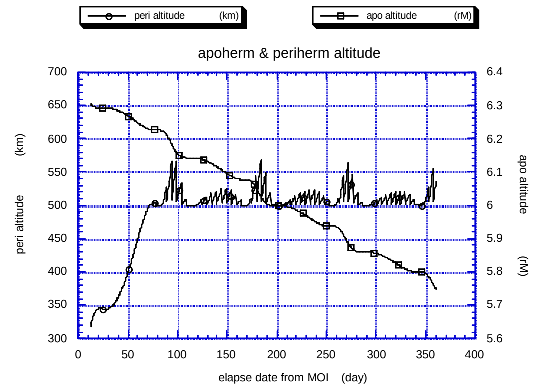

Fig.7: Periherm & Apoherm Altitude Evolution

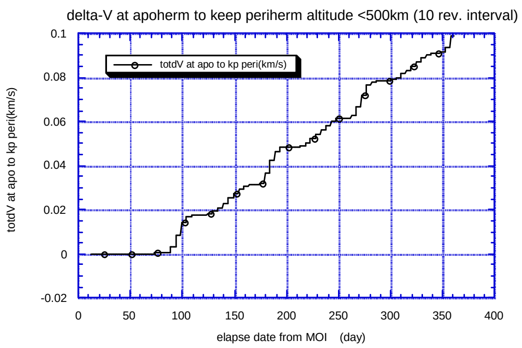

Fig.8: Delta-V for Periherm Altitude Control

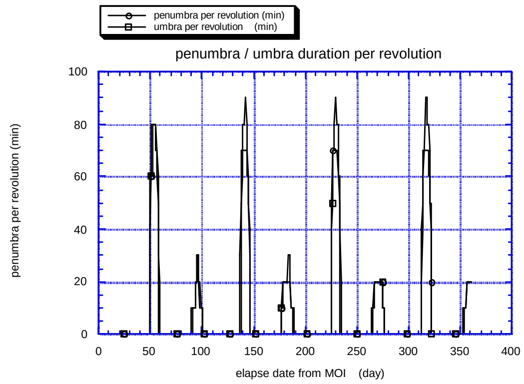

Fig.9: Eclipse Time per Revolution

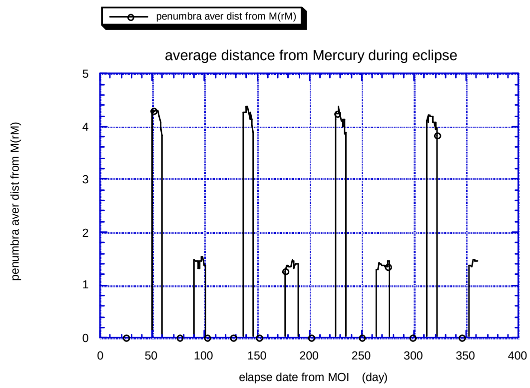

Fig.10: Distance from Mercury at Eclipse

5. Delta-V Estimation

Required trajectory correction maneuvers and attitude control maneuvers are summarized in Table 11 and Table 12 below, including Earth-Mercury interplanetary transfer phase as well as Mercury orbiting phase. Delta-V for three week launch window and correction of the Earth departure error due to launch vehicle injection error are also allotted for the delta-V budget. Delta-V for correction of the deterministic maneuver as well as for swingby navigation are also estimated separately and included in the budget.

The spacecraft design, especially propulsion system design, is based on this delta-V estimation. Most maneuvers are performed by bi-propellant propulsion system (orbit maneuver engine) with Isp of 310 sec, while small delta-Vs and attitude control maneuvers are by mono-propellant Reaction Control System (RCS) with Isp of 180 sec for continuous thrust mode. The total delta-V by OME is 2843 m/s and total delta-V by RCS is 53 m/s. When the spacecraft mass at Earth departure is 1650 kg, the fuel weight is 1071 kg including the margin.

Table 11: Delta-V Budget:
Earth-Mercury Interplanetary Transfer Phase

launch window (3 weeks)	40 m/s
Earth departure error correction	70 m/s
Deep Space Maneuver-1 (deterministic)	255 m/s
Deep Space Maneuver-1 correction	10 m/s
navigation (Venus-1 swingby)	25 m/s
Deep Space Maneuver-2 (deterministic)	0 m/s
Deep Space Maneuver-2 correction	0 m/s
navigation (Venus-2 swingby)	25 m/s
Deep Space Maneuver-3 (deterministic)	252 m/s
Deep Space Maneuver-3 correction	10 m/s
navigation (Mercury-1 swingby)	25 m/s
Deep Space Maneuver-4 (deterministic)	72 m/s
Deep Space Maneuver-4 correction	10 m/s
navigation (Mercury-2 swingby)	25 m/s
Deep Space Maneuver-5 (deterministic)	238 m/s
Deep Space Maneuver-5 correction	10 m/s
navigation (Mercury-3 approach)	25 m/s
attitude control (Interplanetary)	25 m/s

* DSM-1 ~ DSM-5 corresponds to that of the first day of the launch window（8/1/2005).

Table 12: Delta-V Budget:
Mercury Orbiting Phase

Mercury orbit insertion (300km x 30 rM)	1300 m/s
MOI gravity loss	29 m/s
periherm control at apoherm	3 m/s
plane change to polar orbit (->i=90deg)	129 m/s
periherm control at apoherm	23 m/s
apoherm altitude control (30 rM -> 6 rM)	220 m/s
orbit maintenance (periherm control)	50 m/s
attitude control (Mercury orbiter)	25 m/s

* S/C at Earth Departure: 1650 kg (Assuming NASDA’s H-IIA launch vehicle),
* Bi-Propellant engine thrust: 1700 N
* half year mission

Summary

This paper shows the trajectory design strategy of the ISAS ballistic Mercury orbiter mission including multiple swingby interplanetary phase, three-impulse Mercury orbit insertion phase and Mercury orbiting phase. For the multiple swingby trajectory design, new algorithms were developed to decrease high nonlinearity with satisfactory convergence property. The result shows the required delta-V of the mission is around 3,000 m/s including attitude control maneuver.

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