Mission Design and Navigation for a 1977–1978 Venus Swingby/Mercury Orbiter

SCOTT K. ASNIN* AND DARRELL G. ROOS*
Martin Marietta Corporation, Denver, Colo.

A mission opportunity to orbit the planet Mercury, employing a gravity assist swingby at Venus and chemical propulsion for the Mercury orbit insertion, has been identified for launch in June 1977 and arrival in March 1978. Characteristics of the interplanetary trajectory are analyzed and midcourse delta velocity (ΔV) requirements determined. The paper also examines constraints which influence the design of the orbiter phase of the mission, proposes an orbit selection rationale, and defines a strategy for mapping the Mercury surface. Over an optimized 15 day launch window, launch energy (C_3) ranges from 42 to 48 km^2/sec^2 with hyperbolic excess velocity at Mercury between 7.0 and 7.1 km/sec. Navigation analysis has sized midcourse ΔV at 227 m/sec. With Titan IIIE/Centaur as the launch vehicle and a single stage solid propulsion system for Mercury orbit insertion (290 sec specific impulse and 0.93 mass fraction), a useful payload weight of 300 kg can be inserted into an orbit of 0.8 eccentricity.

Nomenclature

a.u. = astronomical unit
C_3 = launch energy
e = eccentricity
E_I = Earth insertion
h_p = periapsis altitude
M_E = Mercury encounter
M/C = midcourse correction
MOI = Mercury orbit insertion
MVM = Mariner Venus/Mercury
S/C = spacecraft
V_E = Venus encounter
V_{hp} = hyberbolic excess velocity
ΔV = delta velocity
θ_{AIM} = trajectory aim parameter
μ = mean
$\rho\lambda\lambda$ = longitude correlation
σ = standard deviation
σR_s = spin radius uncertainty
$\sigma\lambda$ = longitude uncertainty
σz = z-height error

Introduction

ALTHOUGH the idea of using gravity assist at an intermediate planet to reach a target planet is no longer novel, the first application of this technique will occur on the Mariner Venus/Mercury flight scheduled for launch this year (1973). Assuming success of the mission, which involves a flyby of both inner planets in 1974, it is reasonable to expect increased scientific interest in a sequel to MVM. On that premise, an opportunity to orbit Mercury during the late 70's and early 80's was sought which did not require such costly items as a Saturn V launch vehicle or the development of Solar Electric Propulsion. Emphasis of the search was directed toward using the gravity assist potential of Venus, as does MVM, to achieve favorable launch and arrival conditions and keep within the capability of Titan class launch vehicles. The result of this investigation was the identification of a unique orientation of the inner planets in 1977, reoccurring approximately on a 40 yr cycle, for which a transfer trajectory could be designed with the potential for a useful science payload in Mercury orbit. This paper presents trajectory and navigation analyses for the interplanetary cruise to Venus and Mercury, discusses the constraints on and rationale for preliminary orbit design of a Mercury orbiter, and proposes a reference orbit which satisfies probablie misson objectives.

Background of the Mission Search

The occurrence of favorable ballistic Venus-Mercury opportunities has been studied in some detail by Manning[1] for the period from 1980 to 2000. No previous work, however, has properly identified the 1977 opportunity presented here. In order to develop a means of isolating potential ballistic trajectories to both Venus and Mercury, an *a priori* model of ideal trajectory and planet geometry was postulated by Hollenbeck.[2] Three crucial considerations were suggested in the development of this idealization.

First, if conventional propulsion systems were to be adequate for orbit insertion, it would be necessary to employ in some manner the gravity assist potential of Venus. Second, considering the orbit eccentricity of Mercury about the sun (0.2056), the relative velocity of a spacecraft approaching Mercury is about 4.5 km/sec less when the planet is encountered at its perihelion than it would be for aphelion encounter (assuming a coplanar transfer). A slower approach velocity directly reduces orbit insertion ΔV requirements, which translates into increased payload in Mercury orbit. Finally, since the orbit plane of Mercury is inclined to the ecliptic by about 7°, selection of a Venus-Mercury orientation which reduces the out-of-plane velocity component at Mercury for an approaching spacecraft is also crucial to minimization of insertion ΔV.

These criteria pointed to a trajectory/planet configuration which 1) encounters Venus at a node of the Venus-Mercury orbit plane intersection to achieve the necessary plane change through gravity assist, and 2) arrives at Mercury at the opposite node, with that node being nearest Mercury's perihelion. This model can be achieved with a Type II transfer (transfer angle greater than 180°) from Earth to Venus. Given this proposed flight profile, a search was then conducted of Earth-Venus-Mercury orientations over a 40-yr time span,

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Index categories: Lunar and Interplanetary Trajectories; Spacecraft Mission Studies and Economics; Spacecraft Navigation, Guidance, and Flight Path Control Systems.

* Senior Engineer, Advanced Planetary Missions.

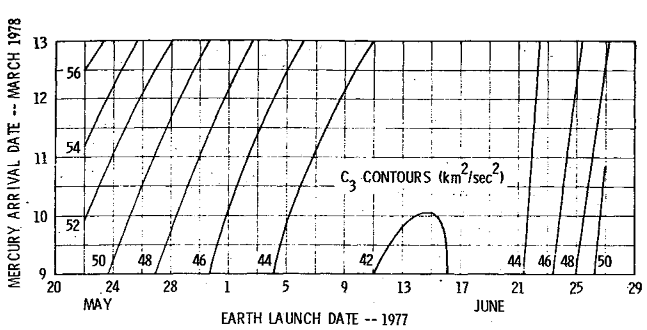

Fig. 1 Launch energy profile.

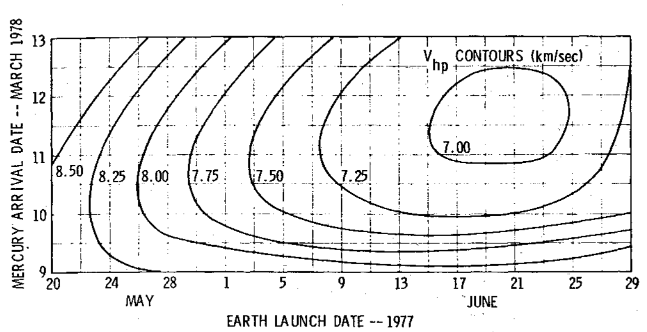

Fig. 2 Hyberbolic excess velocity of Mercury.

from 1960 to 2000, encompassing the set of basically all Earth-Venus positions at the ascending passage (nearest perihelion) of Mercury through the Venus orbit plane. The outcome of that search was the discovery of the promising opportunity for launch in June 1977. Conic trajectory analysis by Van Pelt[3] defined more exactly the launch, swingby, and arrival dates for the mission to Mercury orbit, and an integrated trajectory analysis by the author verified the mission.

Performance

With verification of the 1977 trajectory, launch and arrival windows for this opportunity were searched in a refined mode. The primary trajectory constraints involved maintaining a closest approach radius at Venus greater than 1.1 planet radii (one Venus radius = 6050 km) and keeping the hyperbolic excess velocity low at Mercury to minimize insertion ΔV. Figures 1 and 2 illustrate the launch energy requirements and resulting V_{hp} at Mercury for a month long window at Earth. The V_{hp} contours of Fig. 2 indicate the mission dependence on arriving at Mercury within about a 5 day time span. Even more critical to performance, however, is the timing of Venus swingby. With launch and arrival dates specified, the Venus encounter date is uniquely determined, and its variation over the entire launch window is no more than about one day.

These curves indicate a near-optimal trajectory for launch June 19, 1977 with a C_3 of 43 km^2/sec^2, and Mercury arrival on March 11, 1978 with a V_{hp} of 7.0 km/sec. Venus is

Table 1 Performance assumptions

Launch vehicle	Titan IIIE/Centaur
Launch window	15 days
Midcourse ΔV	230 m/sec
Mercury orbit periapsis	500 km altitude
Mercury orbit eccentricity	0.8
Orbit insertion propulsion	single stage solid
Specific impulse	290 sec
Mass fraction	0.93

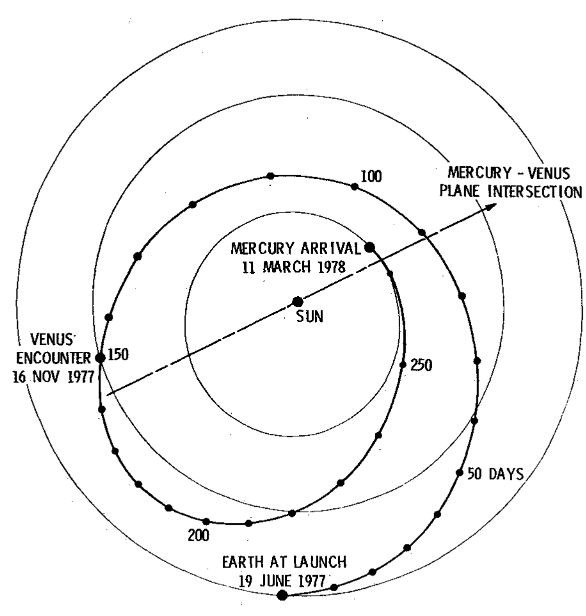

Fig. 3 Heliocentric transfer trajectory.

encountered Nov. 16, 1977 at a closest approach radius of 1.26 planet radii. With the assumptions listed in Table 1 for mission performance, a useful payload weight in orbit of 300 kg can be achieved.

Interplanetary Trajectory Characteristics

A scaled pictorial of the heliocentric trajectory is shown in Fig. 3, illustrating the nature of the two Type II transfers involved. In the figure the heavy dots represent planet positions at encounter, while the smaller dots indicate 10-day flight time intervals. Passage at Venus occurs over the planet's sunlit hemisphere, with the encounter geometry such that the spacecraft remains in line-of-sight with Earth during the flyby.

Use of Type II transfers between the inner planets is necessary for this mission from energy considerations. Several aspects of the resulting trajectory, however, pose problems which influence navigation, spacecraft design, and the scheduling of mission events.

At 250 days into the interplanetary cruise, the sun occults the spacecraft from Earth view. This seriously interferes with tracking at a crucial point in the mission, just prior to the final approach to Mercury. The situation effectively forces the final midcourse to be performed long before encounter, and degrades control and knowledge of orbit insertion conditions. Figure 4 illustrates the relationship of the lines-of-sight to the sun and spacecraft during the cruise phase.

Navigation is degraded further by the zero declination of the spacecraft at 120 and 265 days from launch, shown by the second curve of Fig. 4. Zero declination at 120 days affects

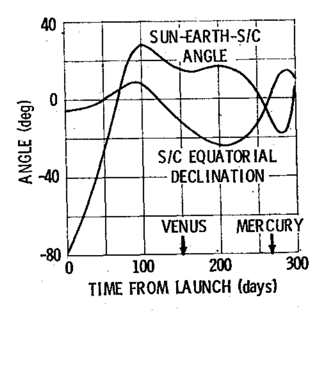

Fig. 4 Spacecraft - sun - Earth geometry during interplanetary cruise.

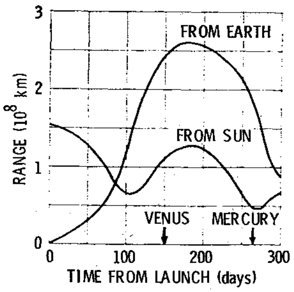

Fig. 5 Spacecraft range from Earth and sun during cruise.

the inital portion of tracking for the critical midcourse corrections at Venus. At 265 days zero declination coincides with orbit insertion and degrades orbit determination for the insertion maneuver and knowledge of the resultant orbit. Use of Quasi-Very-Long-Baseline Interferometry (QVLBI) would improve tracking at these points, but will probably not be available for a 1977 mission.

Range from Earth, another parameter affecting navigation, is shown in Fig. 5. Maximum range for the mission occurs near Venus swingby, where good quality tracking data and communication are most desired. Range of the spacecraft from the sun is also illustrated in Fig. 5. This parameter relates to the amount of thermal protection required for the vehicle. As indicated by the trajectory pictorial, most of the cruise trajectory is within the orbit of Venus, and for the 180 day orbiter phase, heliocentric distance corresponds to that of Mercury.

Although none of these characteristics appear prohibitive to the feasibility of a Mercury orbiter mission, they contribute directly to the sizing of thermal protection, antenna design and location, and the midcourse propellant load necessary for navigation. Quantitative effects on navigation will be discussed in a later section.

Mercury Orbit Selection and Constraints

Science objectives, trajectory limits, thermal constraints, and propulsion system capability all must be considered in establishing a set of feasible Mercury orbits, from which a best compromise is ultimately defined. Table 2 summarizes these objectives and constraints and what they imply to the orbit selection process. The remainder of this section discusses each in detail, and the rationale used to define a reference orbit.

Of primary interest to science is the objective of obtaining quality mapping of the Mercury surface, at a resolution at least comparable to that expected from MVM. This requires a low periapsis altitude. A high apoapsis altitude is desired for magnetic field and charged particle measurements over a

Table 2 Orbit selection objectives and constraints

Objectives	Design
Good mapping resolution	low periapsis altitude
Access to wide latitude range	high inclination
Good resolution at equator	periapsis near equator
Magnetic field measurements	high apoapsis
Minimum insertion ΔV	low periapsis, high e, inject at peri.
Long lifetime	low eccentricity
Constraints	
Locus of periapsis for V_{hp}	high peri. latitude for high inclination
Thermal radiation limits	high altitudes near subsolar point
176 day minimum lifetime	restricts set of orientations for high e

wide range of altitudes and orientations of the vehicle-planet-sun geometry. Both considerations lead to the selection of a highly eccentric orbit, which complements the desire to minimize insertion ΔV and increase payload weight. (Low periapsis altitude and high eccentricity are characterized by high periapsis velocity, which reduces the magnitude of the velocity change required to transfer from the approach hyperbola to the ellipse.) The surface mapping objective also makes desirable access to a wide latitude range under the spacecraft for near-planet coverage. This can be achieved with orbit inclinations between 45° and polar. Access to all longitudes is gained by the planet's 6.1°/day rotation under the orbit. A 176 day mission lifetime provides three complete rotations of the Mercury surface under periapsis, which is necessary to allow a periapsis passage over all longitudes when illuminated. Additionally, a specific aspect of the mapping is to have good resolution over the equator to study Mercury's "hot spots," and this requires a periapsis location at near-equatorial latitudes.

So from science and insertion considerations alone, an ideal Mercury orbit design would have a periapsis altitude of about 500 km and an eccentricity of 0.8, with a 21 hr orbital period. The orbit inclination would be near-polar, with periapsis near the equator, and for proper illumination of the surface during the early part of the mission, periapsis should be about 90° east of the subsolar point, at the evening terminator. In addition, if the orbit orientation were such that orbital perturbations caused periapsis altitude to decay moderately during the course of the mission, areas imaged early could possibly be studied again from lower altitudes at better resolution.

With this ideal in mind, the set of possible insertion conditions was searched for the reference V_{hp} vector and arrival date at Mercury, considering only planar orbit insertion. The results of this search, with θ_{AIM} (angle in the impact plane from the positive T-axis to the aim vector, Fig. 6) as the independent variable, are summarized in Table 3. In Fig. 7, the locus of subperiapsis points on the Mercury surface is pictured, showing one effect of varying θ_{AIM}. Figures 8–11

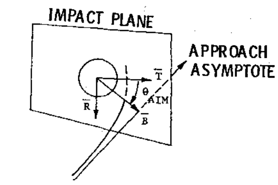

Fig. 6 Geometry of the impact plane.

Table 3 Orbit orientations for varying aimpoint ($h_p = 500$ km, $e = 0.8$)

θ_{AIM}	Orbit inclination	Periapsis latitude	Periapsis illumination[a]	Periapsis altitude history (180 days)
0°	18°	0°	+174°	oscillates 500–800 km
45°	45°	−43°	+160°	decreases to 200 km
90°	88°	−80°	+ 90°	impacts in 100 days
135°	130°	−47°	− 7°	impacts in 150 days
180°	162°	− 5°	− 21°	oscillates 500–160 km
225°	135°	37°	− 39°	increases to 1500 km
245°	116°	53°	− 55°	increases to 1700 km
270°	92°	65°	− 98°	increases to 1800 km
315°	50°	41°	−166°	increases to 1800 km

[a] Periapsis longitude with respect to solar longitude.

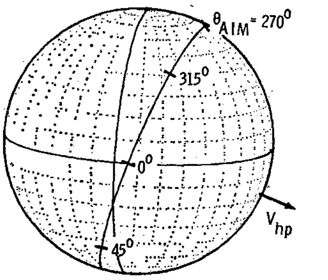

Fig. 7 Locus of subperiapsis at Mercury.

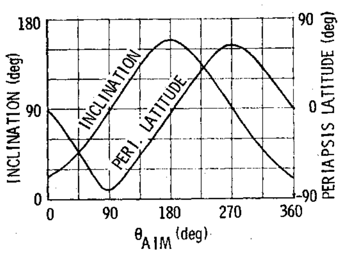

Fig. 8 Inclination and periapsis latitude for varying aimpoint.

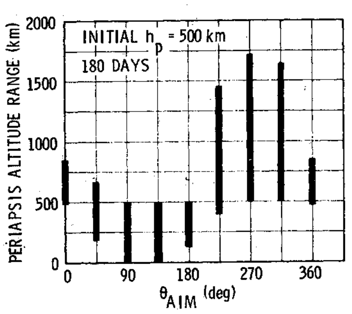

Fig. 9 Periapsis altitude variation over 180 days for varying aimpoint.

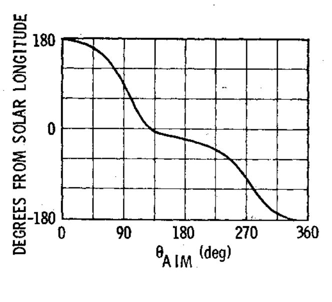

Fig. 10 Initial periapsis-subsolar point relationship for varying aimpoint.

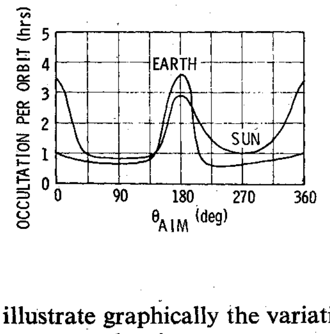

Fig. 11 Maximum occultation times per orbit for varying aimpoint.

illustrate graphically the variation of other orbit design parameters as the aim vector rotates 360° about the planet. These parameters include orbit inclination and periapsis latitude (Fig. 8), periapsis altitude variation[4] for a 180 day orbiter mission (Fig. 9), initial location of periapsis with respect to the solar longitude (Fig. 10), and in-orbit Earth and sun occultation characteristics (Fig. 11).

From both Table 3 and Fig. 8 it is immediately clear that locating periapsis near the equator is incompatible with the desire for a high inclination orbit. Wide latitude coverage, a necessity for a general surface mapper, was assigned greater priority in the orbit selection, and the near-equatorial orbits ($\theta_{AIM} = 0°$ and 180°) were dropped from consideration. Reasonably low altitudes (under 1400 km) are still achieved at the equatorial region, even with periapsis at 50° latitude.

Approach from the south ($\theta_{AIM} = 0°$ to 180°) locates periapsis in the southern hemisphere, and if inclination is kept high, near-planet mapping would be restricted at that area. All of the southern solutions are characterized by a decreasing periapsis altitude history—favorable from a science standpoint. Half of these are unacceptable, however, because they impact within the desired mission lifetime (θ_{AIM} from about 60° to 165°). The orientation for $\theta_{AIM} = 45°$ appears to be a good compromise of various objectives, with an inclination of 45°, southern coverage, and decreasing periapsis altitude. The only disadvantage is the initial periapsis location over the planet's dark side, emerging into sunlight 12 days after insertion.

Northern approaches (θ_{AIM} from 180° to 360°) locate periapsis in the northern hemisphere and are characterized by increasing periapsis altitude histories. None of these solutions result in impact. Approach from the NW (180° to 270°) yields a light side periapsis, while approach from the NE (270° to 260°) gives a dark side periapsis which reaches the sunlit hemisphere in 30 days. A reasonable compromise is suggested by the NW approach ($\theta_{AIM} = 225°$), with an inclination of 45° retrograde, good coverage of the northern hemisphere, and periapsis over the illuminated surface at insertion. Its only disadvantage is an increasing periapsis altitude history.

From the search, then, two types of orbit orientations are selected as promising. One provides northern hemisphere coverage from low altitudes, permits mapping immediately after insertion, and has an increasing periapsis altitude. The other provides southern hemisphere coverage from low altitudes, requires a 12 day wait in orbit before near-planet mapping can begin, but has the science advantage of a decreasing periapsis.

Between these two orbit types, both excellent from science considerations, final selection of the northern mapping mission is dictated by thermal constraints. Due to the proximity of Mercury to the sun, the planet's surface radiates thermal energy outward at an intensity of 5 suns at the subsolar point (one sun being the thermal radiation of the Earth's sun at 1 a.u.). This effect, combined with the direct solar radiation of 11 suns at Mercury distances, tends to bake a spacecraft at low altitudes near subsolar. The most severe thermal situation during the mission exists at the point where the vehicle passes directly over subsolar near periapsis. Since the sun vector lies at the equator, and periapsis is closest to the descending node, this situation occurs twice during the mission when the descending node is close to subsolar. For the southern mapping mission with decreasing periapsis altitude, maximum radiation periods occur at spacecraft altitudes of 1300 and 1100 km over the equator. For the northern periapsis location with increasing periapsis altitude, the maximum thermal periods occur with the vehicle at 1500 and 2000 km over the equator.

Thermal limits for any reasonably sized vehicle eliminate the southern mission, and demand that the northern mission be modified further to gain an additional increase in altitude at the critical times. For the northern mapping mission selected from the initial search, with an initial periapsis of 500 km and located at 37° N lat, thermal constraints are violated while the vehicle passes between periapsis and the descending node during the critical period. The situation is illustrated by Fig. 12, where the contours represent the proximity of the spacecraft to the subsolar point. To avoid thermal problems, the spacecraft altitude must be increased by 450 km during this part of the mission. This could be accomplished with orbit trim maneuvers, or by modifying the initial orbit design

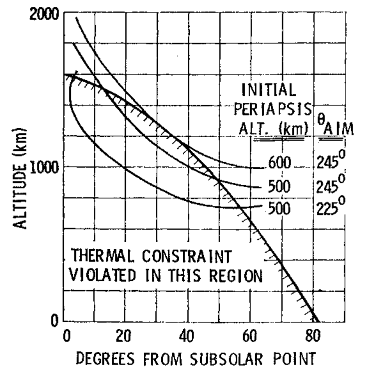

Fig. 12 Thermal constraint on altitude during critical thermal period.

to achieve the altitude increase. Increasing the initial periapsis to 1000 km would resolve thermal problems, as would increasing periapsis latitude (selecting a higher inclination) and moving periapsis farther from the equatorial region. After a number of iterations, a combination of both orbit adjustments was selected. Designing the initial orientation for a θ_{AIM} of 245° results in a periapsis at 52° N lat with a 64° inclination, and this, along with increasing the targeted periapsis altitude to 600 km, satisfies thermal constraints.

The reference orbit defined in Table 4 has been derived from a synthesis of the preceding considerations. A time history over the mission lifetime of periapsis altitude and periapsis location with respect to the illuminated surface is presented in Fig. 13. This design orbit requires a Mercury orbit insertion burn of about 4250 m/sec to transfer from the approach hyperbola at periapsis.

An interesting sidelight to the mission is the existence of a timely acquisition of Mercury by the radio telescope at Arecibo, P.R. The station is operated by Cornell Univ. for

Table 4 Characteristics of the reference orbit

θ_{AIM}	245°
Perpiapsis altitude	600 km
Apoapsis altitude	24880 km
Eccentricity	0.8
Inclination	64° retrograde
Periapsis latitude	53° N
Periapsis illumination[a]	−50°
Time from insertion to darkside	7 days

[a] Periapsis longitude with respect to solar longitudes.

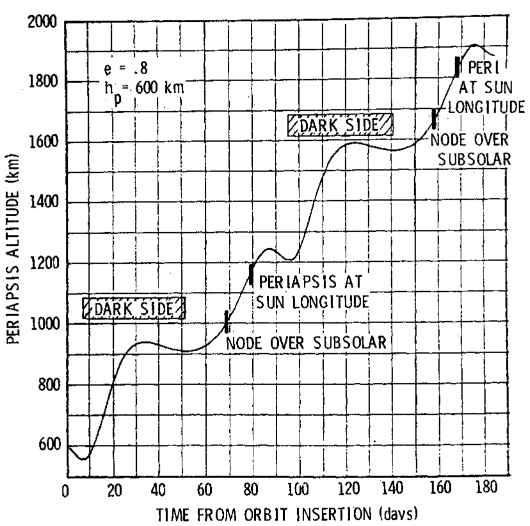

Fig. 13 Periapsis altitude history for the reference orbit.

the National Science Foundation, and although not connected with NASA, could potentially be used to assist the mission. Mercury is acquired by Arecibo four days before the spacecraft reaches the planet and can be viewed for varying periods each day until 206 days after MOI, or for the duration of the proposed orbiter mission.

Surface Coverage Characteristics

Three periods of near-planet surface mapping exist during the 180 day life of the orbiter. These periods represent times during which the spacecraft passes over the illuminated surface at low altitudes. Wide area coverage, with poorer resolution, can of course be gained during almost all of the mission from points on the orbit more distant from the planet.

For the purpose of determining regions of acceptable photographic coverage, two assumptions are made concerning photo imaging of the surface. First, it is assumed that adequate mapping resolution can be achieved from altitudes below 3000 km. Second, surface illumination is assumed adequate for points on the surface between 40° and 90° from the subsolar point. These limits will undoubtedly change as the spacecraft optical system becomes better defined and the albedo characteristics of Mercury are more thoroughly understood. They should, however, reflect reasonable estimates of the imaging capability in orbit.

The first opportunity for near-planet mapping occurs immediately after orbit insertion and lasts about seven days. During this time only a small portion of the illuminated surface can be mapped from low altitudes. This opportunity might be used primarily to verify the performance of near-planet imaging, to calibrate surface resolution achieved, and to observe for any image smear. Toward the end of this phase, wide area coverage from high altitudes could commence as the illuminated surface moves under apoapsis for a 46 day period, allowing coverage of longitudes 34° E, east to 100° W. (It should be noted here that the longitudes indicated are used for reference only.)

A second near-planet mapping opportunity occurs about 38 days after MOI when periapsis begins to approach the sunlit side from the east. For a period of 56 days the illuminated Mercury surface is under the orbit periapsis with the spacecraft at altitudes between 900 and 2000 km. (Periapsis has increased to 900 km by the start of this phase primarily as the result of solar perturbations on the orbit.) This opportunity should be considered the first prime period for near-planet mapping. The timing of this phase allows digestion of the wide area mapping data received earlier and real-time selection of scientifically interesting areas to be mapped in greater detail from low altitude. Areas accessible during the period are illustrated in Fig. 14.

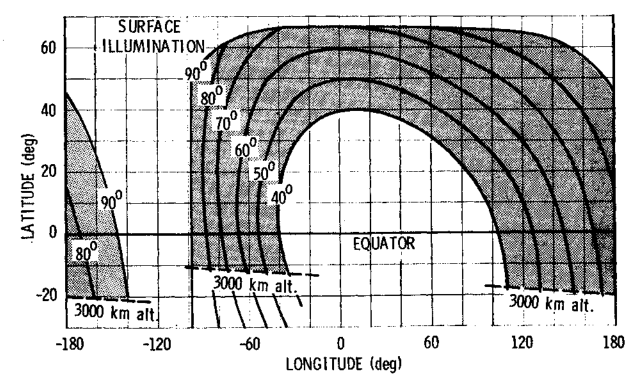

Fig. 14 Surface coverage for second near-planet mapping opportunity.

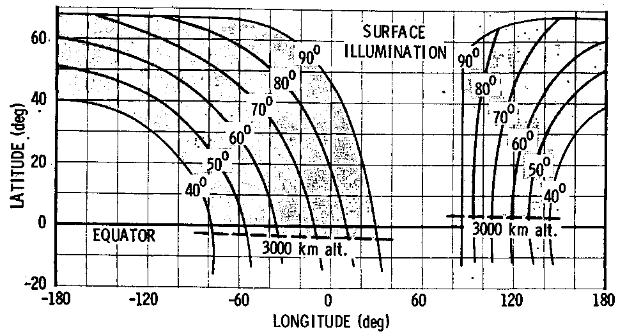

Fig. 15 Surface coverage for third near-planet mapping opportunity.

At about 130 days into the mission the third and final near-planet mapping opportunity begins. Just prior to this time, the illuminated surface from 157° W long east to 80° E long moves under apoapsis and can be mapped in the wide area mode from high altitudes. This period combined with the first period of wide area mapping represents the potential of 100% coverage of the Mercury surface at high altitude resolution. Coverage for the third near-planet mapping opportunity is presented in Fig. 15. As this mapping phase ends, 183 days after MOI, so does the design life of the orbiter. Most of the northern hemisphere of Mercury has been accessible for high-resolution mapping, and the entire surface has potentially been mapped from higher altitudes.

In conclusion, the reference orbit design provides the science/imaging systems with continuous periods of surface mapping, alternating between near-planet detailed coverage of select surface features to wide area surface mapping of the Mercury surface. Most of the northern hemisphere, including about 67% of the equatorial region, is available at good illumination for detailed study, and all of the planet is accessible for area mapping.

Navigation Analysis

Navigation for the 1977 Venus Mercury orbiter mission is dominated by a combination of unfavorable geometrical conditions which characterize the cruise trajectory and its relationship primarily with the Earth and Venus. Some of these characteristics have already been discussed qualitatively.

Four midcourse corrections have been designed for the flight. The first is executed ten days after launch ($E_I + 10$) and removes injection errors. A second midcourse is required three days prior to Venus encounter ($V_E - 3$) to reduce, by as much as possible, trajectory dispersions existing before the Venus flyby. Following Venus encounter by two days ($V_E + 2$), a large third midcourse must be performed to correct trajectory dispersions which become greatly magnified during the flyby. The final midcourse is scheduled for 30 days before Mercury encounter ($M_E - 30$) to improve trajectory accuracy for the insertion maneuver. This final correction is backed up to a point long before its ideal location to avoid solar interference with the doppler signal.

Assumptions relevant to the navigation analysis are listed in Table 5. All the errors quoted are one-sigma values. Injection errors have been modeled as 3 km spherical in position and 1.732 m/sec spherical in velocity with no correlation. Equivalent station location errors for the Deep Space Network (DSN) stations Goldstone, Madrid, and Canberra are based on a somewhat pessimistic extrapolation of improved tracking capability into the late 1970's. Charged particle calibration is assumed, but not QVLBI. DSN tracking arcs preceding the second and fourth midcourse corrections are designed for 30 days, followed by one day to command and execute the maneuvers. The knowledge error assumed for the start of each of these periods is 1000 km spherical in position and 0.5 m/sec spherical in velocity. For the first midcourse, tracking extends from $E_I + 1$ to $E_I + 9$, and for the post-Venus midcourse, from V_E to $V_E + 1$. (Tracking from $V_E - 2$ to V_E, just before closest approach, was found not to be particularly useful to post-Venus navigation.) Maneuver execution errors are sized at 0.33% proportionality and 0.33° pointing, with ephemeris errors conservatively estimated to be 20 km spherical for Venus and 60 km spherical for Mercury.

Table 5 Navigation error assumptions

Injection errors (r,v)	3 km, 1.732 m/sec
Maneuver execution errors	
Proportionality	0.33%
Pointing	0.33°
Station location errors	
Spin radius, σR_s	0.73 m
Longitude, $\sigma\lambda$	2.04 m
Correlation, $\rho\lambda\lambda$	0.9
Z-height, σz	10 m
Doppler noise	1 mm/sec/min-count
Venus position	20 km
Mercury position	60 km

With this set of assumptions a statistical definition[5] was made of all four midcourse corrections and is presented in Table 6. The ΔV load ($\mu + 3\sigma$) for the four maneuvers was found to be 20.6, 12.2, 186.4, and 7.3 m/sec, respectively, for a total load of 226.5 m/sec. The dominance of the post-Venus midcourse (M/C 3), representing over 80% of the total navigation ΔV requirement, is the most significant aspect of the analysis. Its size is directly the result of trajectory geometry effects on the navigation process, and varies from the more moderate values reported for the MVM mission[6] due to the particularly unfavorable conditions existent for the 1977 trajectory.

In brief, the post-Venus M/C is large because the pre-Venus M/C cannot effectively reduce trajectory dispersions to reasonably low levels, these dispersions become greatly magnified by the close flyby, and the resulting errors must be removed as soon as possible by M/C 3. High error magnification through swingby derives from the relatively low flyby altitude of 1600 km, while the unfavorable geometries are responsible for the inefficiency of pre-Venus orbit determination and the pre-Venus midcourse.[7] Heliocentric knowledge error for the Venus approach is high, with a plateau in the 100–150 km range for z-uncertainty, due to the zero declination and distance of the spacecraft from Earth during DSN tracking before Venus. The situation is aggravated further by the high relative approach velocity of 13 km/sec, which places execution of the pre-Venus midcourse at a point before the perturbative effects of Venus gravity can be incorporated into orbit determination for M/C 2. Moving M/C 2 closer to encounter would increase its size and amplify the effects of its associated execution errors.

In comparison with the more modest navigation requirements of MVM, a mission which has a similar zero declination problem during Venus approach, navigation for the 1977 trajectory must contend with a faster approach velocity (13 km/sec compared with 8 km/sec), a much greater Earth-spacecraft range (1.5 a.u. compared with 0.3 a.u.), and a much

Table 6 Statistical description of midcourse maneuvers

Event	Days from launch	ΔV(m/sec) μ	σ	$\mu + 3\sigma$
M/C 1	10.0	6.90	4.57	20.6
M/C 2	147.6	3.94	2.76	12.2
M/C 3	152.6	62.15	41.40	186.4
M/C 4	235.3	2.26	1.68	7.3

lower flyby altitude (1600 km compared with 4000 km). These conditions translate into considerably poorer orbit determination for the pre-Venus midcourse, and greater error magnification during flyby. Improvements to navigating at zero declination might come from the consideration of, in increasing value, ranging data, optical navigation, and QVLBI. Their application should be investigated further in the event the 1977 Mercury orbiter is allowed to evolve into a real mission.

Conclusions

A ballistic trajectory to Mercury orbit in the 1977–1978 time frame has been found with the potential for a reasonable orbiting science payload. A scientifically productive extended lifetime orbiter mission has been defined in detail for this opportunity. Areas of concern to any further evolution of the mission involve the severe thermal environment with which the orbiter must contend, and the large ΔV requirement for navigation. The general trajectory/performance feasibility of the mission is demonstrated.

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