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STUDY OF BALLISTIC MODE MERCURY ORBITER MISSIONS

Volume I

SUMMARY REPORT

July 1973

Prepared Under Contract No. NAS2-7268

by

MARTIN MARIETTA CORPORATION
Denver, Colorado 80201

for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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Program Manager

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FOREWORD

This report summarizes the scope, approach, and major results for Contract NAS 2-7268, Study of Ballistic Mode Mercury Orbiter Missions. Detailed documentation is contained in two companion volumes: "Ballistic Mode Mercury Orbiter Mission Opportunity Handbook," NASA CR-2298, and "Study of Ballistic Mode Mercury Orbiter Missions, Volume II, Technical Report," NASA CR-114618.

Four specific mission opportunities were studied, corresponding to launches in 1977, 1980, 1985, and 1988. Results of investigations of alternate flight techniques to enhance mission performance of these opportunities, as well as to generate new opportunities, are also reported.

The primary conclusions drawn from this six-month study effort are:

1) Ballistic mode Mercury orbiter missions offer adequate performance for effective follow-up of the MVM'73 science findings and an orderly program of advanced Mercury exploration.

2) The existing and programmed technology base is adequate for implementation of Mercury orbiter spacecraft design.

3) When the pending MVM flyby has been accomplished and the results analyzed, the data base will be adequate to support detailed orbiter spacecraft design efforts.

Martin Marietta Corporation wishes to acknowledge the contribution of Ames Research Center personnel who suggested the investigation of multiple Venus swingby for improved performance potential.

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BACKGROUND

The Mariner Venus/Mercury spacecraft, scheduled to flyby Mercury March 30, 1974, represents the first and only planned U.S. mission to this planet. A strategy for follow-up of the MVM findings has not been defined.

An orbiter spacecraft is the next logical step in an orderly program of continued exploration. This difficult mission has been the subject of extensive analysis in recent years, and definitive planning information has begun to materialize.

The engineering and scientific challenges involved in designing a spacecraft for sustained operation in the unique Mercury environment have been overshadowed by the lack of available means for placing an adequate mass in Mercury orbit. Early investigations of the direct ballistic flight mode were discouraging due to the large launch energy requirements. Later analyses of the Venus gravity-assist flight technique, while reducing the Earth departure energy, still resulted in high arrival speeds at Mercury indicating a requirement for a Saturn V class launch vehicle to provide enough performance for a useful mission. Accordingly, recent effort has been oriented to the use of solar electric propulsion as a solution for the performance requirements. Mission prospects have thus seemed contingent on new technology developments in a climate of funding austerity.

More thorough analysis of the ballistic flight mode utilizing Venus gravity-assist has resulted in identification of timely, high-performance mission opportunities compatible with programmed launch vehicles and conventional spacecraft propulsion technologies. Further definition of these mission opportunities and preliminary assessments of associated science return and technology requirements formed the basis for the study contract summarized in this document.

PERFORMANCE POTENTIAL OF BALLISTIC FLIGHT MODE

Figure 1 presents a summary of the characteristics and performance potential for four mission opportunities which formed the baseline scope for this study contract. Shown in the time context of the MVM flyby, these mission options constitute a basis for planning an evolutionary program of advanced Mercury exploration.

The 1977 mission opportunity offers adequate performance to support a useful orbiter vehicle and would represent near-ideal time phasing with MVM for incorporation of flight experience and follow-up of science findings. However, due to the funding situation, implementation of this option is unlikely, even if MVM results in increased scientific impetus.

A more likely candidate for an initial orbiter mission is the 1980 opportunity, which offers sufficient performance for a well-instrumented spacecraft of either the Pioneer or Mariner class. This mission will be referenced for many of the illustrative discussions to follow.

Availability of a larger launch vehicle such as Shuttle/Centaur is a prerequisite for the 1985 mission opportunity shown on Figure 1. The performance advantage over Titan IIIE is expected to be about 70% and would correspond to an orbited mass of about 300 kg.

The 1988 mission opportunity would support a second generation orbiter if required for furtherance of science objectives. Alternatively, a small lander may be within the 800-kg capability of a Shuttle/Centaur or equivalent.

An adjunct of the basic study was concerned with investigation of alternate flight techniques. Specifically, performance improvement potential was evaluated for midcourse propulsive maneuvers and multiple Venus swingbys. Summary results of these analyses are presented on Figure 2.

Small maneuvers (200 to 400 mps) applied near perihelion of the Earth-Venus transfer trajectory were found to compensate for the relatively poor planetary alignments characterizing the 1977 and 1985 mission opportunities. Executing the maneuvers with a low performance auxiliary propulsion system appropriate to navigation requirements resulted in approximately doubling the allowable orbited weight for the 1985 mission. In combination with a Shuttle/Centaur launch vehicle, a net orbiter mass of over 600 kg could be accommodated.

Initial investigations of multiple Venus swingby potential showed that this flight technique was not directly applicable to any of the baseline mission opportunities. However, renewed search for opportunities consistent with multiple Venus encounters resulted in identification of two timely high-performance missions. The 1983 mission has the disadvantages of three Venus encounters and a 31-month flight duration. Compensating considerations, in addition to high performance, include compatibility with the Titan IIIE launch vehicle and timing appropriate for a backup to the 1980 opportunity.

The 1988 double Venus swingby mission appears superior in all respects to the baseline 1988 opportunity. In addition to improved performance, flight time is reduced by about 3 months. Of particular interest is the potential of this opportunity with a Shuttle/Centaur or equivalent to orbit about 1000 kg. This level of performance should be adequate to support a modest lander.

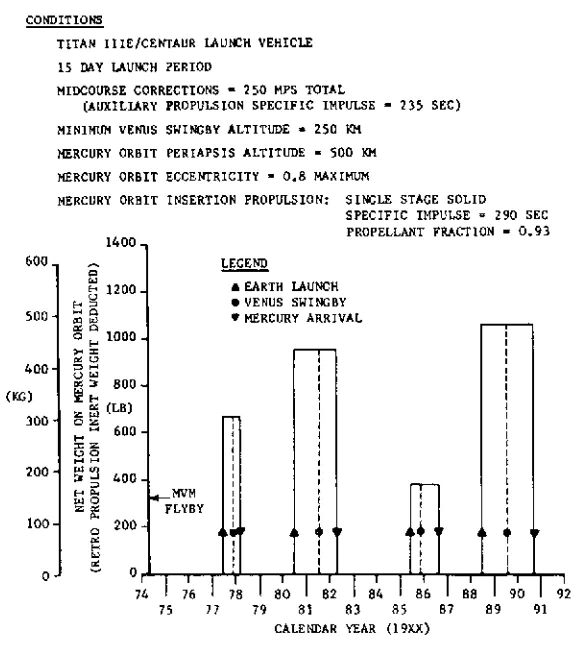

FIGURE 1. BASELINE MERCURY ORBITER OPPORTUNITIES

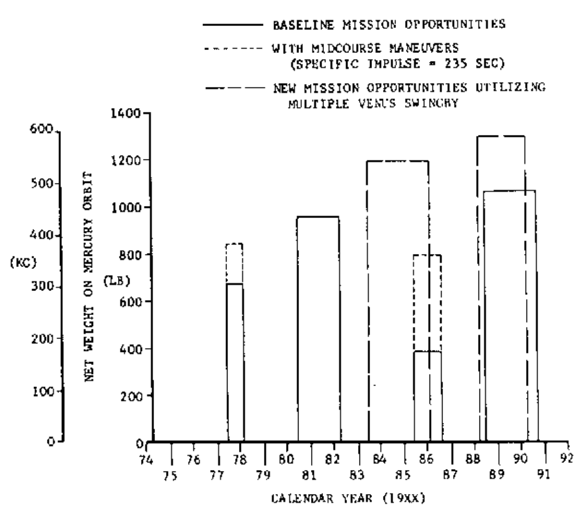

FIGURE 2. POTENTIAL OF ALTERNATE FLIGHT TECHNIQUES

All of the mission opportunities presented on Figure 1 are predicated on the basic flight geometry shown on Figure 3. A primary requirement for high performance potential involves achieving Mercury encounter near Mercury perihelion and near the intersection of the Venus and Mercury orbit planes. Secondarily, Venus swingby must also be accomplished near the intersection of the orbit planes. Meeting these conditions will produce a near-tangential encounter with Mercury and minimize the relative approach velocity. The corresponding Earth to Venus trajectory must be Type II to achieve Venus encounter while ascending from the Sun.

The minimum flight time for the basic flight profile is about 9 months. This condition applies to the 1977 opportunity and repeats at intervals of about 20 years. All of the other mission opportunities identified require use of extra solar revolutions of the spacecraft to accommodate planet phasing. For example, the 1980 mission depicted on Figure 3 involves one extra spacecraft revolution prior to Venus swingby and a second extra revolution after Venus swingby and prior to Mercury encounter. The resultant total flight time of about 22 months is typical of the high-performance Mercury orbiter missions identified.

An illustration of the performance requirements resulting from the Venus gravity-assist ballistic flight mode is presented on Figure 4. These data for the 1980 mission are representative of near-ideal alignments of Earth, Venus, and Mercury which produce high performance by high utilization of Venus gravity benefits (as reflected in the low Venus swingby altitudes involved). The effects of constraining Venus altitude are also shown on the figure.

The launch energy requirements for the 1980 mission ($C_3 \sim 35$ km^2/sec^2) are much less than for direct ballistic transfer from Earth to Mercury ($C_3 \sim 90$ km^2/sec^2). Also, the approach velocity at Mercury is substantially reduced (by about 1 km/sec). Both of these performance benefits are provided by the Venus gravity field and result in useful ballistic missions at the expense of increased flight time and complexity.

Also shown on Figure 4 are the performance improvements associated with modest powered swingby maneuvers. In practice, the benefits of a 100-mps velocity maneuver at Venus (a reduction in Mercury approach velocity of about 200 MPS) can be realized for a net cost of about 26 mps. This results from simultaneous execution of the powered swingby maneuver and the required post-Venus navigation midcourse maneuver (estimated to be 207 mps). The foregoing method was employed to calculate the performance capabilities presented on Figures 1 and 2 for the 1980 mission opportunity.

A characteristic of these 3-planet missions not evident from the optimized data of Figure 4 is the tight timing required for the Venus swingby and Mercury encounter events. An Earth launch period of 15 days appropriate to ground checkout and launch operations corresponds to less than one day variation in Venus swingby date and slightly more than one day span in Mercury encounter date. The trajectory for a specific Earth launch date is, of course, constrained even more. Feasibility of navigation for these demanding flight sequences has been confirmed by detailed analyses.

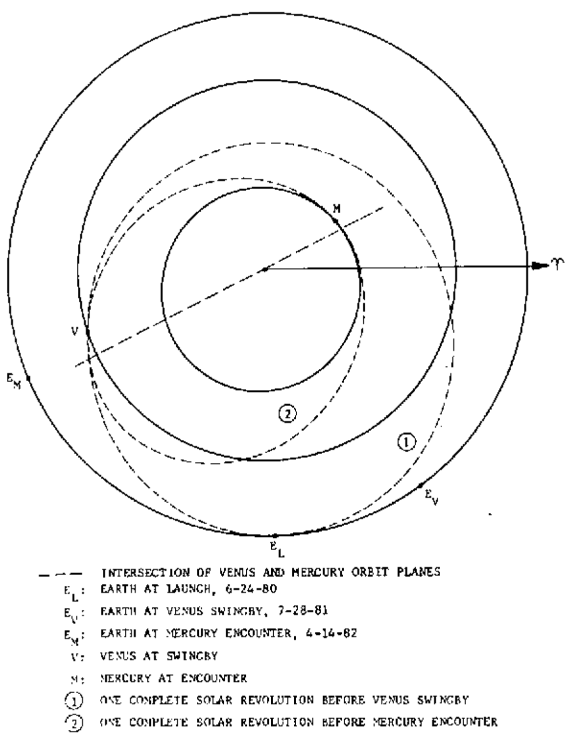

FIGURE 3 TYPICAL HELIOCENTRIC FLIGHT PROFILE

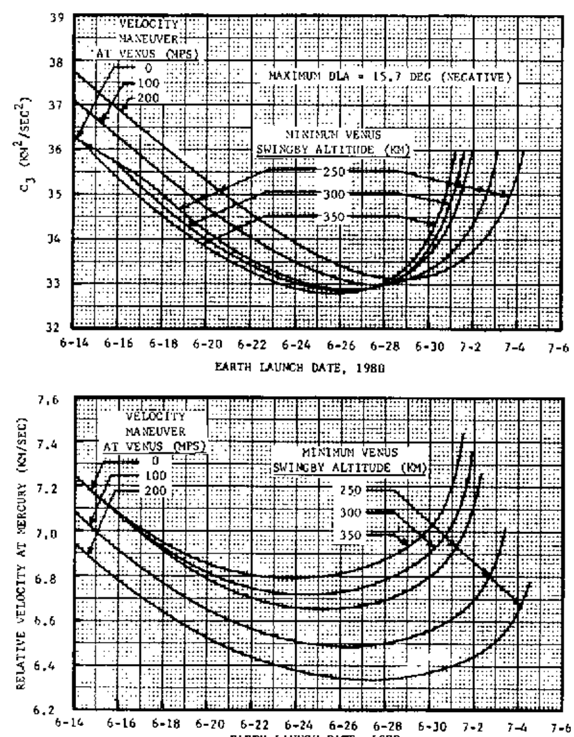

FIGURE 4 TYPICAL PERFORMANCE PARAMETERS

ORBIT SELECTION CONSIDERATIONS

Parametric performance analyses have been conducted to permit general interpretation of the mission opportunity data. Figure 5 presents results in terms of the retro-velocity increment for a representative orbit insertion propulsion system. Data corresponding to the Titan IIIE/Centaur launch vehicle is depicted for all mission opportunities and to the Shuttle/Centaur launch vehicle for opportunities compatible with projected availability.

Orbit eccentricities above about 0.925 are expected to be unstable in the presence of solar influences. A value of about 0.8 is judged the practical upper limit and has been employed for performance comparisons and analyses of on-orbit operations. Lower eccentricities are possible for the better opportunities.

A low periapsis altitude (e.g., 500 km) is beneficial to performance and compatible with navigation capabilities. However, other constraints such as thermal environment and long term orbit stability may necessitate use of higher values. Orbit insertion at an initial periapsis altitude of 1000 km would increase retro-velocity requirements (for 0.8 eccentricity) by about 125 mps. Figure 5 shows the decrease in net orbited spacecraft weight to be typically about 6%. Alternatively, an initial 500-km periapsis could be raised to 1000 km with a post-insertion apoapsis maneuver of about 30 mps, corresponding to auxiliary propulsion system propellant expenditure of about 1.3% of the initial orbited weight. Either of these methods represents a practical means of relieving spacecraft design and operational problems.

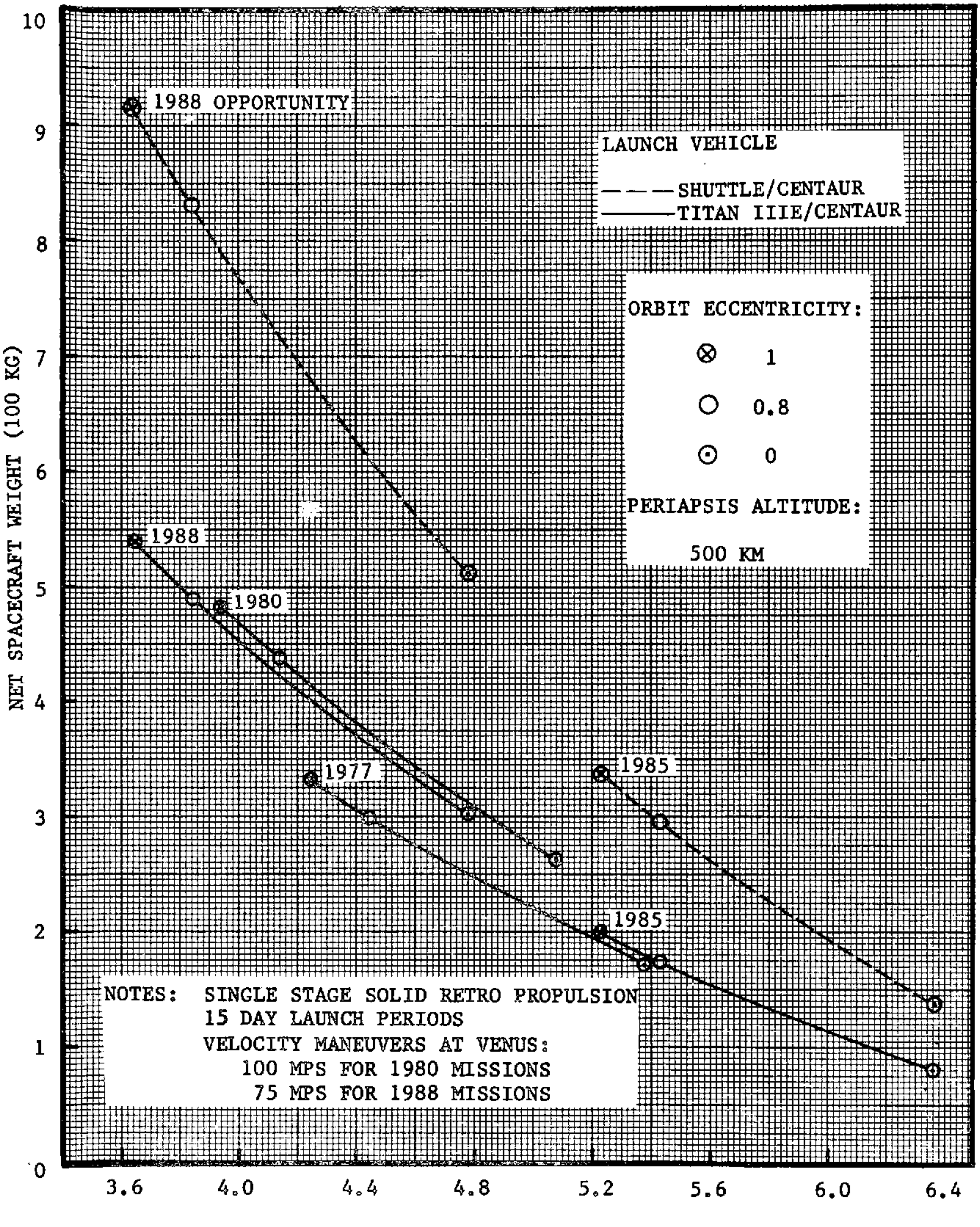

FIGURE 5. SPACECRAFT WEIGHT OPTIONS

The thermal environment at Mercury as modelled for this study is shown on Figure 6. Solar energy reflected and re-radiated from the planet surface produces a severe thermal environment for orbiter spacecraft. The direct solar flux at Mercury varies from 10.6 Earth solar constants (suns) at perihelion to 4.6 suns at aphelion. These values compare to the MVM design criteria of 6 suns (night-side flyby near aphelion) and the Helios solar probe objective of 16 suns. Current technology is therefore expected to be adequate for the direct solar input experienced by a Mercury orbiter.

Additional IR input from the Mercury surface could substantially increase the design requirements for an orbiter spacecraft. For example, Figure 6 shows that the Mercury environment at a point 500 km directly over the subsolar point when Mercury is at perihelion corresponds to 22.5 equivalent thermal suns. Such conditions in combination with the direct solar flux are beyond the state of the art in many subsystem areas such as thermal control and solar cell power generation.

Fortunately, the extremes of the Mercury environment need not be experienced to satisfy many science objectives. The options for alleviating the thermal requirement include: targeting the orbital periapsis such that when periapsis is on the subsolar meridian, Mercury is near aphelion; avoiding low latitude positioning of periapsis; increasing periapsis altitude (initially or after orbit insertion as previously discussed); or a combination of these techniques. Additional thermal considerations include the effects of solar perturbations. As discussed more fully later, the long term effects on orbit periapsis altitude can result either in surface impact or steady increase at a rate of more than 200 km per month.

Thermal considerations represent the primary design challenge for a Mercury orbiter spacecraft. IR flux from two constantly changing directions and conflicts of preferred spacecraft attitude among thermal control, power generation, science instrument viewing, and earth communications are requirements which need detailed tradeoff analyses. At this point, however, it is judged that a useful Mercury orbiter mission can be implemented with the current thermal technology.

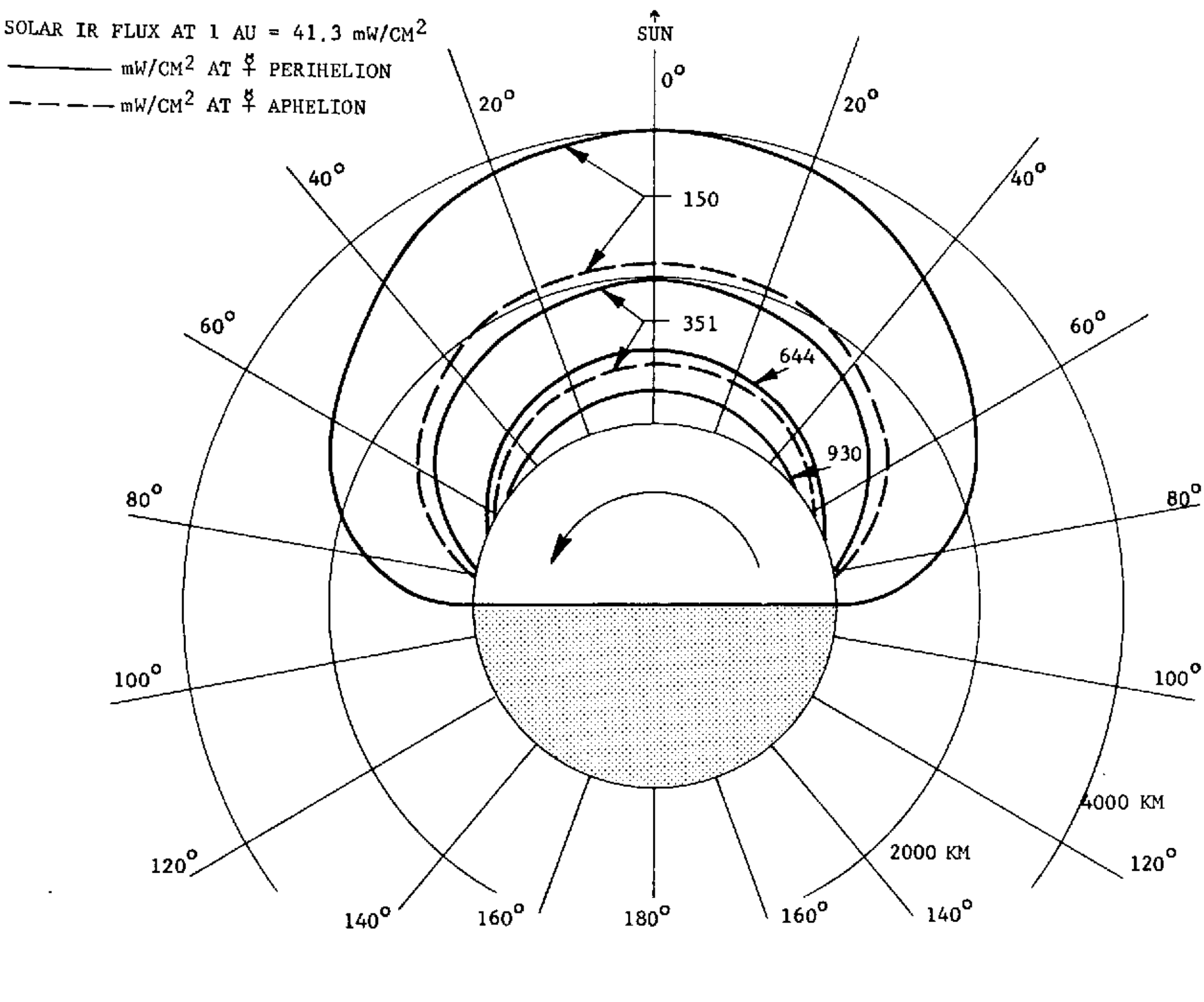

FIGURE 6. THERMAL RADIATION FROM MERCURY

Several orbit parameters pertinent to selection of an orbit orientation are dependent on targeting of the Mercury approach velocity vector. For convenience, Figure 7 is presented to define the targeting parameter, Θ_{AIM}, and allow correlation of the targeting options to typical constraints on orbit periapsis locations and typical combinations of orbit inclination and initial terminator geometry.

All planetary orbiters are subject to the perturbing effects of solar incluences. For Mercury missions, these effects are pronounced and represent a significant orbit orientation consideration. Figure 8 illustrates, for the spectrum of Θ_{AIM} options, the short and long term behavior of periapsis altitude for the 1980 mission opportunity. The initial orbit depicted (0.8 eccentricity, 500-km periapsis altitude) is basically stable. However, the solar gravity field produces significant fluctuations in orbit eccentricity without modifying orbit period (i.e., there is no change in the major axis of the orbit).

The selection of an orbit geometry to satisfy science objectives and to meet realistic design criteria involves many considerations. In general, the entire spectrum of targeting options may be of interest. For those aiming cases exhibiting undesirable natural behavior (such as impacting the surface, too low on the dayside, or too high on the night-side), modest allocations of orbited weight to orbit-adjust propellant could retain the targeting option. Maneuver requirements to adjust periapsis altitude at apoapsis average about 6 mps per 100 km. This technique could be employed early in the mission to provide more initial altitude clearance to accommodate the long term effects depicted on Figure 8. Alternatively, a series of maneuvers could be programmed to tailor the orbit periapsis appropriate to the developing circumstances.

Data are provided in the study contract technical report to parametrically evaluate the mission design and operations considerations. Since many of the primary factors are unique to Mercury missions, complete tradeoffs will be complex and aided by the findings and experience of the MVM '73 flyby.

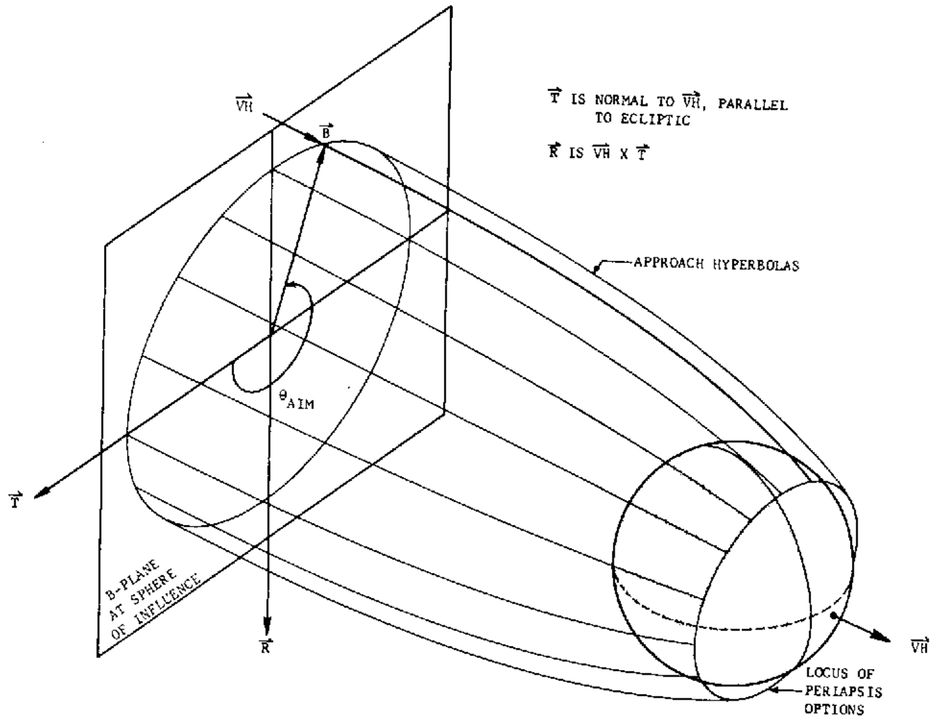

FIGURE 7. MERCURY APPROACH TARGETING OPTIONS

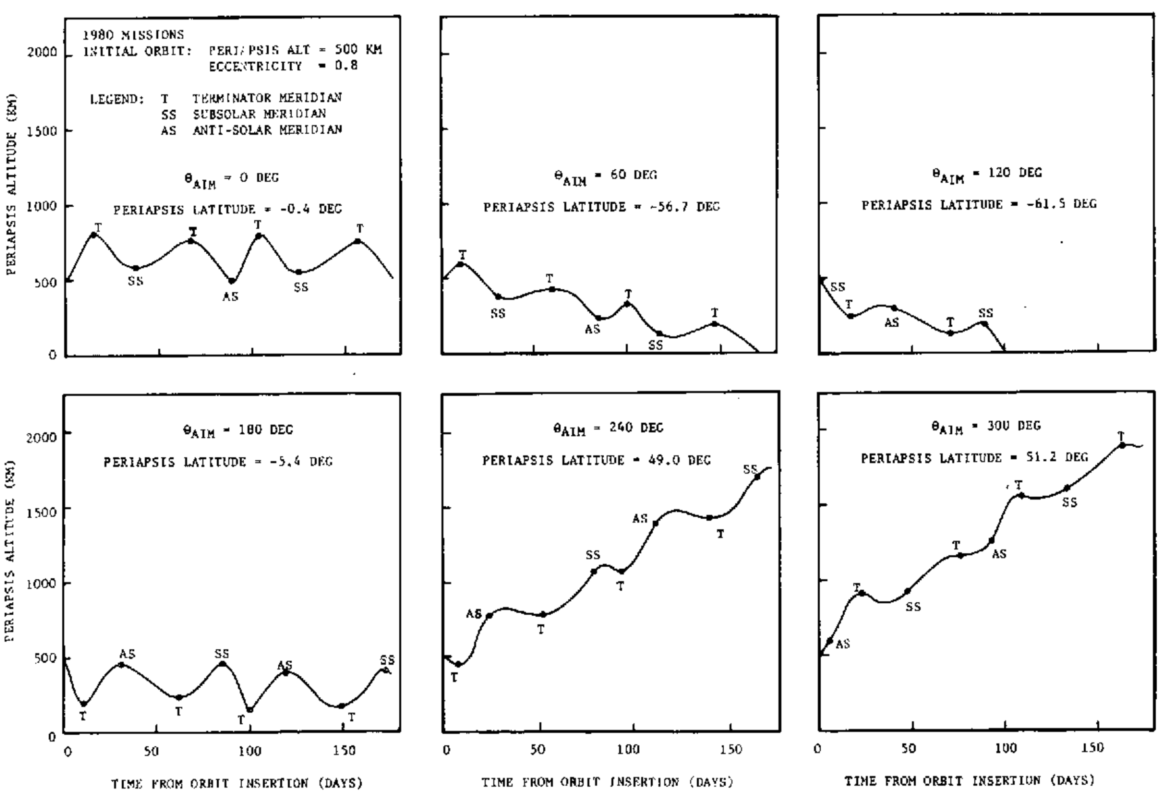

FIGURE 8. TYPICAL PERIAPSIS TIME HISTORIES

SCIENCE EXPERIMENTS

The basic science objectives for an orbiter mission differ from those for a flyby primarily in degree. However, an essential consideration for instrumentation and operation of an orbiter spacecraft involves incorporation of the flight experience and science findings of any prior flyby missions.

In the case of Mercury, the current state of knowledge has dictated a mission profile and instrumentation for the MVM mission heavily oriented to Earth and Solar occultations. Positive findings in the areas of atmosphere, intrinsic magnetic field, etc. would generate strong forcing functions for orbit selection and instrumentation priorities for a follow-up orbiter mission. Conversely, negative or inconclusive results would probably shift the emphasis to global mapping of surface properties.

Prior to analysis of the MVM data, any postulation of instrument complement or best orbit geometry is necessarily speculative. Table I presents a summary of candidate instruments considered during this study contract. Some of the objectives shown are mutually exclusive while many others exhibit a degree of conflict. Final resolution will involve complex tradeoffs even when MVM results are available and will be further affected by weight and cost considerations.

Net orbited weight available for spacecraft systems and science instrumentation is maximum for low periapsis altitude and large orbit eccentricity. Such orbits afford mapping instruments with opportunities for global perspective from high altitude and limited coverage with improved resolution from low altitude. Also, evaluation of planet and solar wind interactions is consistent with eccentric orbits which permit occultations over a range of altitudes.

Instruments such as the γ-ray spectrometer are ineffective from high altitude. However, it is judged that an eccentric orbit with low periapsis altitude, which affords intermittant periods of operation for such instruments, represents a practical compromise for a balanced program of science investigation. Accordingly, preliminary assessments of science return were predicated on this type of orbit. The secondary consideration of orbit geometry options controlled by approach targeting is more dependent on MVM findings and was, therefore, treated parametrically.

Isolation of Mercury gravity harmonics is judged inconsistent with the eccentric orbits deemed appropriate for other science objectives. Consequently, a subsatellite devoted to this function was studied to fully assess the science potential of orbiter missions. Also, science measurements unobtainable from orbit may be accomplished by a modest lander vehicle. These latter two subjects are addressed in later discussions.

TABLE 1 ORBITER SCIENCE SUMMARY

	SCIENCE OBJECTIVES				
	PRIMARY (MERCURY CHARACTERISTICS)				SECONDARY
SCIENCE EXPERIMENTS	SIZE & SHAPE	INTERNAL PROPERTIES	SURFACE PROPERTIES	ATMOSPHERE PROPERTIES	SOLAR ENVIRONMENT
*Imaging	X	X	X		X
Radar Altimeter	X				
*Radio Occultation	X			X	X
Transponder (Gravity Harmonics)		X			
X-Ray Spectrometer			X		X
γ-Ray Spectrometer			X		X
IR Spectrometer			X	X	X
*Magnetometer		X	X	X	X
UV Spectrometer			X	X	X
IR Limb Scanner				X	
*IR Radiometer			X		
*Plasma Probe		X	X	X	X
Photometer			X	X	X
*Charged Particle Telescope					X
Meteoroid Detector					X
Plasma Wave Sensor					X
Neutron Monitor					X

* MVM '73 Experiments

X denotes that experiment contributes to science objective.

A unique characteristic of Mercury affecting mission planning is illustrated on Figure 9. The indicated planet position at spacecraft arrival time is typical of ballistic mode missions. The night-side hemisphere at arrival time is employed as a surface landmark for demonstration of the 3/2 spin-orbit coupling.

Successive positions of Mercury at 1/4 sideareal year time intervals (22 days) display the basic effects of orbit eccentricity. Correlation with the constant rate of planet rotation is shown by the progress of the arrival-time terminator. As indicated on the figure, near-cancellation of the two motions through perihelion results in essentially no change of viewing opportunities for about the first three weeks of orbiter operation. Later, near aphelion, the view changes rapidly and has covered the entire globe after a complete solar revolution of Mercury. One more Mercury sidereal year is required to return to the geometry at encounter and complete the illumination phase angle cycle. Therefore, the initial period of spacecraft operations offers the opportunity for global mapping and prompt follow-up with selective high resolution imaging while waiting for the view to change sufficiently for resumption of routine mapping.

For reference, the illuminated hemisphere viewed by MVM is approximately half visible at orbiter spacecraft arrival and the remaining initial view is essentially new. In the cases of 1980, 1983, and 1985 launch, this previously viewed portion of the planet is adjacent to the morning terminator and will be entirely visible about 55 days after orbit insertion. For 1977 and 1988 missions, geometries differ by one half of the spin-orbit cycle with the result that the MVM view is adjacent to the evening terminator and receding into darkness. These facts are relevant to development of mapping strategies, orbit selection rationale, and mission duration objectives.

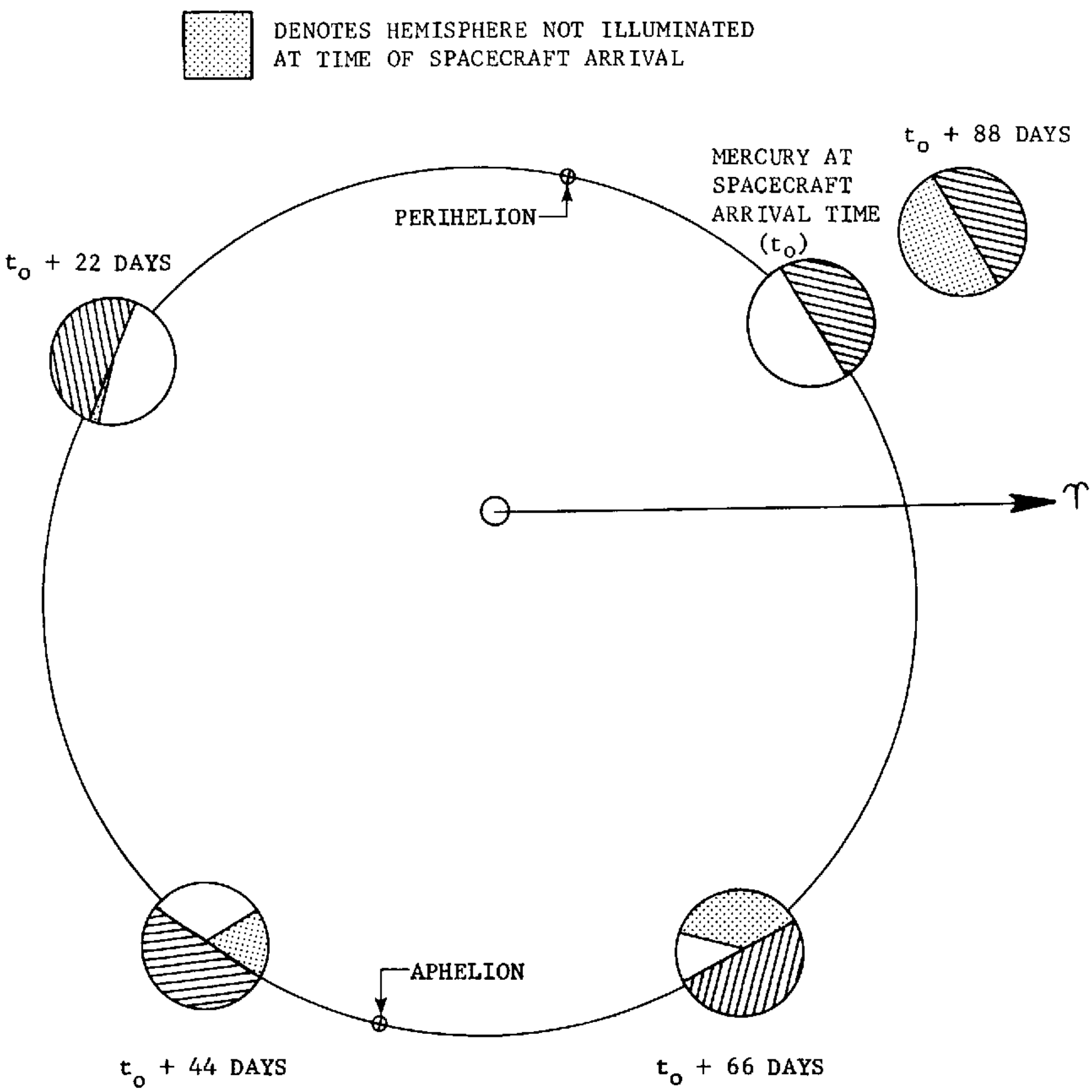

FIGURE 9. TYPICAL MERCURY ILLUMINATION HISTORY

A major science objective for advanced Mercury missions involves determination of the gravity field harmonics and asymetries. This information would permit analysis of such parameters as density distributions, and spin-orbit coupling mechanics.

Orbit selection for a single Mercury orbiter spacecraft involves a number of considerations consistent with high eccentricity and conservative periapsis altitude. This type of orbit is expected to preclude significant gravity measurements, especially in view of the solar influences perturbing such orbits. A subsatellite deployed to a lower orbit from the main orbiter spacecraft offers prospects of obtaining useful gravity measurements with modest investment of orbiter payload capabilities. This concept, depicted schematically in Figure 10, is recommended for consideration for an initial Mercury Orbiter mission. Valid assessment is contingent on detailed analyses of the orbit determination problem.

Subsatellite deployment requirements have been addressed parametrically. Retro-velocity increments for subsatellite deployment are on the order of 1 km/sec for typical initial orbits and no deflection of the orbit plane. Options include subsatellites with close approach to the planet surface. For example, if the orbiter periapsis has been lowered to 100 km (implying night-side location for thermal considerations), the subsatellite can be deployed with a conservative apoapsis (e.g., 1500 km) for a retro maneuver of about 740 mps. Alternatively, a high value of orbiter periapsis (e.g., 1500 km, appropriate to day-side location) would require a subsatellite retro of 1025 mps to achieve 100-km periapsis on the planet night-side. Design considerations include the sensitivity of altitude to maneuver execution accuracy. For example, the latter case would impact the surface if overperformed by 25 mps.

A method of sizing a simple subsatellite retro propulsion system (e.g., a solid rocket motor) for a range of deployment conditions would involve sizing for the largest maneuver anticipated and, for actual conditions at deployment, pointing out-of-plane to achieve the desired degree of propulsion loss. Larger intentional plane deflections up to 45 degrees may be of interest if evaluation of the gravity field requires multiple subsatellites with differing orbits.

The weight of a subsatellite equipped with no functional instruments except cooperative tracking aids would depend on the method of tracking. Orbit determination from Earth may be feasible with low power output if Arecibo could be employed. (Aricebo accessibility is excellent for the first four months of 1980 mission orbit operations). Alternatively, the subsatellite could be tracked from the main orbiter spacecraft if the compound orbit determination problem can be solved with sufficient accuracy.

Representative weights for subsatellites deployed with spin-stabilized solid retro motors have been calculated. For example, in-plane deployment on a 500-km circular orbit, which requires a retro maneuver of about 940 mps, corresponds to an initial weight 160% of the actual subsatellite net weight (estimated to be 10 kg). Equivalent values for a 45-degree plane deflection are 2600 mps and 400%.

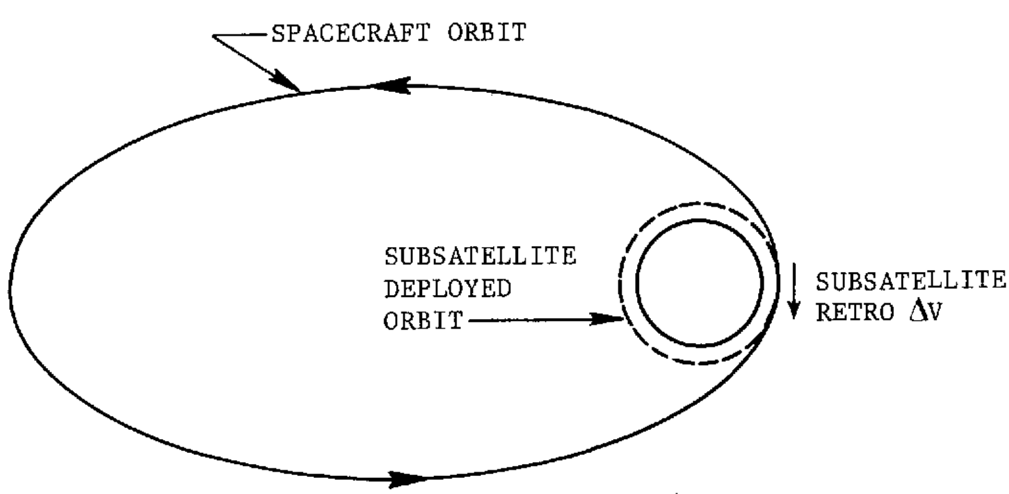

FIGURE 10. ORBITER/SUBSATELLITE GEOMETRY

Preliminary investigations have been conducted to evaluate the prospects of a ballistic mode mission accommodating a modest Mercury lander. It was assumed that a lander would be deployed from Mercury orbit and supported by an orbiter spacecraft for initial attitude reference, communication relay, etc. Further, it was judged that a minimum lander would be constrained by thermal considerations to the planet night-side (permitting up to 3 months of operation) or, possibly, near-polar latitudes if the planet equatorial plane is confirmed to be parallel to the planet orbit plane.

Figure 11 illustrates a landing technique which was assessed for general performance requirements and sensitivity to maneuver execution errors. The landing sequence is predicated on simple thrust attitude techniques. Horizontal attitude for velocity maneuvers to achieve vertical free-fall conditions would be established by the orbiter spacecraft and retained by the spin-stabilized lander vehicle through maneuver execution. Vertical attitude reference for arrest of free-fall velocity would require planet-oriented sensors such as doppler radar. Magnitude errors in execution of the apoapsis maneuver produce a residual component of horizontal velocity which will be multiplied by a factor of about 11 at the time of surface approach. However, the vertical component of surface approach velocity is insensitive to both magnitude and direction errors of the apoapsis maneuver.

The horizontal component of orbiter velocity varies only about 25 mps for true anomalies of $\pm$ 10 degrees from apoapsis. Accordingly, propulsion system sizing could easily accommodate landing site selection over a 20 degree surface range while maintaining vertical lander descent. For the technique of zeroing the horizontal velocity component only, the remaining vertical component of velocity produces about the same surface contact velocity and landing time for all descent paths.

Lander sizing calculations have been performed for a final landed weight of 50 kg. This value has not been confirmed by detailed design, but represents a preliminary estimate for a vehicle with the following design characteristics.

1) 5 kg of science instrumentation

2) Planet night-side operation

3) 5-watt SNAP power supply (Earth communications via orbiter spacecraft)

4) Shock absorbing pads to accommodate vertical contact velocities to about 25 mps

5) Self-righting roll cage to accommodate horizontal contact velocities to about 50 mps.

6) Hydrazine monopropellant propulsion for de-orbit and terminal descent; solid rocket motor for main retro propulsion.

7) Terminal descent velocity budget of 100 mps.

For the configuration described, about 400 kg of orbited weight must be allocated to the lander. An additional 25 kg or so would be required for lander support systems such as a spin table, thermal protection in orbit, etc. Relating these values to the performance capabilities presented on Figures 1 and 2 for the high performance 1988 mission opportunities indicates a probable requirement for the Shuttle/Centaur class launch vehicle.

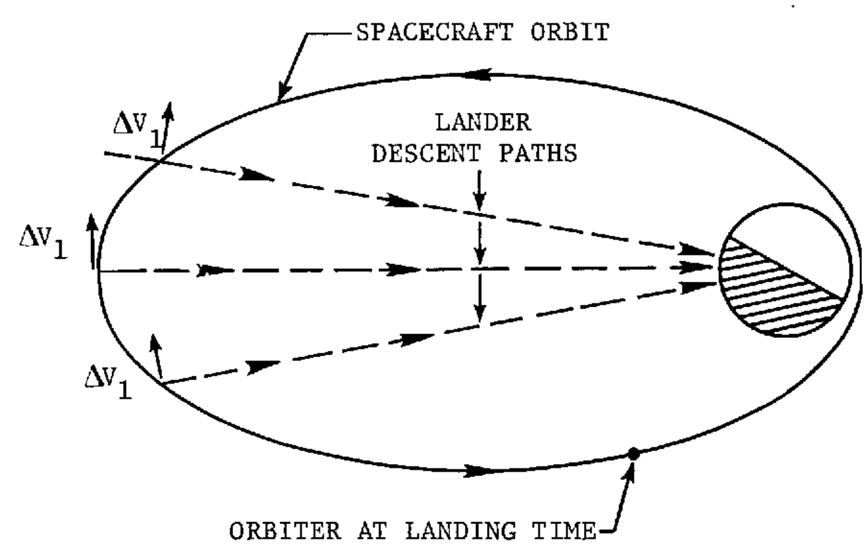

FIGURE 11. REPRESENTATIVE LANDING TECHNIQUE

TECHNOLOGY REQUIREMENTS

The study scope did not include spacecraft design effort. However, technology assessments were conducted for major subsystems appropriate to spin-stabilized and 3-axis-stabilized spacecraft designs.

Table 2 summarizes the assessed criticality of vehicle systems with respect to the current state of the art. Items designated "Medium Criticality" are not deemed necessary for a useful orbiter spacecraft but may offer operational advantages.

The primary design constraint for a Mercury orbiter spacecraft involves the thermal environment and associated interactions with orbit geometry options and science return. As indicated by Table 2, available technology with moderate extrapolation is judged adequate for spacecraft thermal control and for systems necessarily exposed to the thermal flux, such as power generation. Assessment of internal equipment is predicated on maintenance of compatible thermal conditions.

A solid rocket motor for the orbit insertion maneuver was judged compatible with the cruise phase thermal environment and the thrust requirement for prompt deceleration. Evaluation of alternative liquid propulsion systems would require detailed design effort and was beyond the scope of this study contract.

TABLE 2. TECHNOLOGY ASSESSMENT SUMMARY

VEHICLE SYSTEM	NON-CRITICAL (STATE-OF-THE-ART ADEQUATE)	LOW CRITICALITY (MODEST EXTRAPOLATION OF STATE-OF-THE-ART)	MEDIUM CRITICALITY (NO FLIGHT EXPERIENCE)
Thermal Control System			
Thermal Coatings		X	
Solar Reflectors	X		
Thermal Insulation		X	
Thermal Louvers		X	
Heat Shield			X
Phase Change Material			X
Power Generator System			
Solar Cell Array		X	
Thermoelectric			X
Imaging System			
Frame		X	
Spin Scan			X
Attitude Control System			
Spin Stabilized	X		
Three Axis Stabilized	X		
Attitude Control Propulsion			
Liquid	X		
Gas	X		
Auxiliary Propulsion System			
Monopropellant	X		
Bipropellant	X		
Retro Propulsion System			
Solid		X	
Telecommunication System	X		