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ABSTRACT

Due to the elliptical shape of Mercury's orbit and the slow planetary spin rate, Mercury has a large surface temperature difference that creates highly variable spacecraft thermal environments that are a function of both planet solar distance and spacecraft orbit plane position. Being able to analytically simulate the severe thermal environments experienced by a spacecraft over the lifetime of a Mercury orbiting mission make it possible to realize a feasible spacecraft thermal design.

The analysis described throughout this paper was used to characterize the temperature response as a function of initial phase angle conditions (α_p) when referenced at Mercury perihelion for a 3-axis stabilized spacecraft. Variables in the analysis include solar distance, argument of periapsis, and α_p . The selected orbit is highly elliptical, with a 720-minute period and a near polar inclination. Analysis results presented in this paper are specific for the spacecraft configuration, orbit geometry and inclination used, but the analytical techniques described can be applied to any spacecraft configuration, orbit geometry or inclination.

INTRODUCTION

Mercury is the only terrestrial planet that has never been explored by a remote orbiting spacecraft. The only man-made space probe to visit Mercury has been Mariner 10. Built and launched in the United States, Mariner 10, a 3-axis stabilized solar powered spacecraft, has provided the only images and scientific exploration of this tiny planet. Using three flybys, Mariner 10 was able to map and measure only about 45% of the planet surface during a one-year period between 1974–1975. The dark side flybys were all at planet aphelion and never near the subsolar point. Mariner 10 was designed and tested to withstand a solar-only 5 Earth solar constant (ESC) environment, ignoring to the intense omni-directional heat radiated from Mercury's surface on the Sun-lit side.

Due to severe mass restrictions and extremely harsh thermal environments, a Mercury orbiting spacecraft poses many engineering and operational challenges.

MESSENGER (MErcury Surface, Space, ENvironment, Geochemistry, and Ranging) is a 3-axis stabilized solar powered spacecraft using a high-performance all chemical propulsion system fully integrated into an all graphite-epoxy structure. The mission design uses a ballistic trajectory with multiple Venus and Mercury gravity assists. MESSENGER will eventually orbit Mercury for one Earth year (or four Mercurian years) returning a wealth of scientific data and complete planet coverage, something not accomplished by Mariner 10. MESSENGER has been proposed to NASA by The Johns Hopkins University Applied Physics Laboratory, and was recently down selected as one of five Discovery mission finalists. Mission cost and launch vehicle choices are very constrained under NASA Discovery guidelines. The largest acceptable launch vehicle for a NASA Discovery mission is a Boeing Delta II 7925H-9.5 ($M_0 = 1066$ kg). Driven by the 2700 meter per second mission Δv requirements, over one half of the launch mass is allocated to propellant. It quickly becomes apparent that due to the high Δv nature of this mission, the spacecraft mass allocated to useful payload is limited. Spacecraft mass must be used with great discretion since the main purpose of the mission is to get maximum science return and not orbit a thermal design or propulsion system at Mercury. The thermal analysis presented in this paper shows the dependence of spacecraft temperatures to the relationship between the orbit plane ascending node and the subsolar point when referenced at Mercury perihelion. Utilizing this relationship during the mission design will ensure relatively benign planetary heating environments when in orbit at Mercury. By taking advantage of the mission design and corresponding phase angle geometry, the spacecraft thermal design can be made simple and robust without unnecessary mass or risk. Since it is impossible to intuitively define the worst-case thermal conditions experienced by an orbiting spacecraft, a complete spacecraft thermal design can only be realized through a comprehensive thermal analysis involving the variable planet and solar environments. For MESSENGER, the worst-case thermal conditions occur at 3 different point of the orbit (Fig. 1), demonstrating the need for a comprehensive analysis. [1]

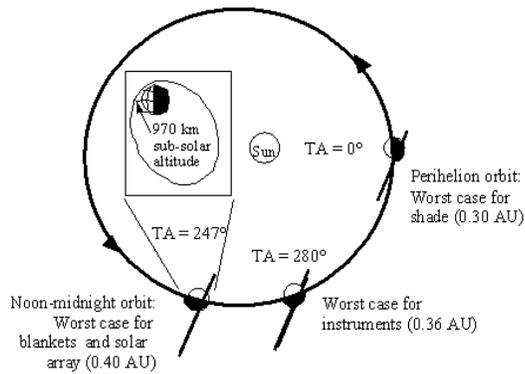


Figure 1. Orbit at Mercury showing worst-case thermal conditions for the thermal shade, blankets solar array, and instruments.

SPACECRAFT CONFIGURATION

In order to accomplish a Mercury orbiting mission, a typical spacecraft will take one of two basic configurations; 3-axis stabilized with some sort of Sun shade and articulating solar panels or spin with body mounted solar cells and the Sun vector maintained perpendicular to the spin axis. Science requirements and thermal considerations weigh heavy in the decision on which configuration to choose. From day one, the MESSENGER configuration has been 3-axis stabilized due mainly to science requirements. It was decided that the spacecraft thermal control could be accomplished by properly choosing an α_p angle that best balances the required spacecraft Δv and orbiting thermal environments when at Mercury.

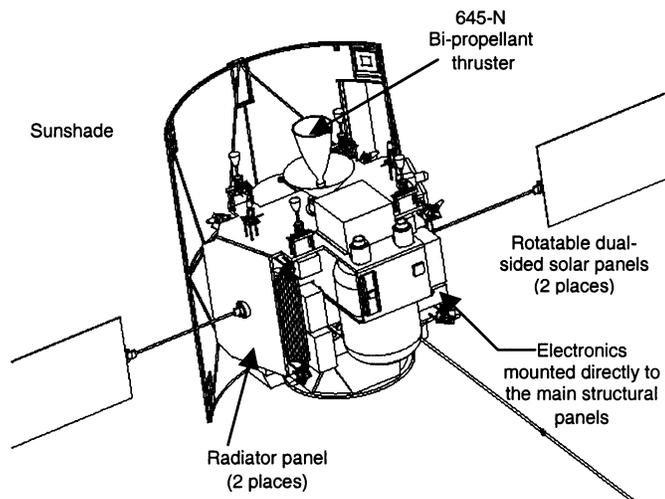


Figure 2. MESSENGER spacecraft

Figure 2 shows a three-dimensional view of the MESSENGER spacecraft concept, which was used in the thermal analysis. Operationally, the Sunshade always maintains a Sun pointing attitude during cruise and orbiting phases of the mission. Rotatable dual-sided solar arrays extend beyond the umbra created by the Sunshade. The arrays are rotated so that the maximum panel temperature is maintained at 150°C. Each face of the array is designed to produce ample spacecraft power

during inner and outer cruise and the orbiting phase of the mission. Spacecraft radiators are located on the two large side panels, which also support the solar array drive motors. The thermal analysis assumed a 250-watt constant internal spacecraft heat load due to the electronics, which are mounted to the vertical panels making up the core structure. The instruments, all of which are behind the Sunshade, are mounted on the top and bottom decks of the spacecraft. Instruments mounted to the bottom deck are always viewing the planet when in orbit. The instruments on the top deck are mostly making Sun and particle measurements. The propulsion system is a high specific impulse (I_{sp}), dual-mode system with a minimal number of tanks and associated plumbing and is fully integrated with the structure. Δv maneuvers are designed so the engines can be operated while keeping the Sunshade pointed at the Sun.

MERCURY'S ORBIT

In order to understand the thermal analysis presented in this paper, one must first understand the orbital interactions with planetary and solar positions. Mercury's highly elliptical 88 day heliocentric orbit causes the solar constant to vary between 4.7 ESC (planet aphelion/0.46 AU) and 11.1 ESC (planet perihelion/0.30 AU). Also, Mercury rotates once every 59 days causing the planet surface temperature to reach steady temperature conditions.

Because of low planetary albedo, Mercury's surface absorbs between 75–94% of the incident solar radiation, creating extremely harsh infrared (IR) thermal environments. The maximum surface temperature at the sub-solar point when at Mercury perihelion is 430°C and reduces to 297°C at aphelion while surface temperature distribution and corresponding re-radiated heat drops off as a function of the cosine law from the sub-solar point to the dawn-dusk terminator. When the spacecraft is in the vicinity of the terminator or is in eclipse over the dark side, the planet's surface temperature is -180°C, which is independent of solar distance and drives the thermal environments. The planet surface temperature distribution and corresponding re-radiated IR flux distribution as seen by the spacecraft is a function of both solar distance and orbit plane position from the sub-solar point. [2]

Establishing an analytical relationship between the sub-solar point and the orbit plane ascending node would be the first step in generalizing a solution routine to find the most optimum injection position for an orbiting spacecraft. This general relationship between the sub-solar point and the orbit plane ascending node can be used to characterize the spacecraft temperatures as a function of initial injection position which, if chosen properly, will allow for reduced thermal risk during the orbital phase of the mission.

Figure 3 defines the relationship α_p , which is the measure of phase angle between the sub-solar point, the center of Mercury and the ascending node of the space-

craft orbit when at Mercury perihelion. Unlike the Earth, once a spacecraft is placed into orbit at Mercury, the orbit plane stays inertial with no appreciable rotation of the ascending node or line of apsides.

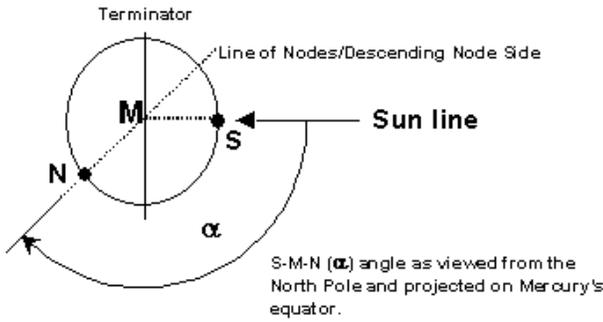


Figure 3. The sub-solar point (S) to Mercury (M) to ascending node (N) angle (α) α_p denotes the initial angle between the sub-solar point and the ascending node when at Mercury Perihelion. α_p also establishes the specific value of the α angle at any planet true anomaly position during a Mercury year.

Figure 4 illustrates how, over one Mercury year, the Sun will in effect complete a 360° revolution around the orbit plane, illuminating different portions of the orbit plane with a solar intensity corresponding to the planet solar distance. Phase angle (α) changes and solar distance variation are related to Mercury's true anomaly position; a true anomaly of zero is defined at Mercury perihelion (0.30 AU/11.1 ESC) and a true anomaly of 180° is defined at Mercury aphelion (0.46 AU/4.7 ESC). As Mercury orbits the Sun and moves from perihelion to the true anomaly position of 90°, the α angle increases according to the relationship:

$$\alpha = [\alpha_p + \text{planet true anomaly}]$$

Where α is the local phase angle and α_p is the initial condition defined at perihelion and is constant for a one complete revolution of Mercury around the Sun.

For the example, shown in Figure 4, with the initial condition of α_p equal to 90°, the local α angle is equal to 180° when the planet true anomaly position (PTA) is equal to 90°. As Mercury orbits the Sun, the α angle for any Mercury solar distance can be calculated by knowing α_p and the planet's true anomaly. The most non-optimum thermal orbit for a 3-axis stabilized spacecraft is when the sub-solar point and the orbit periapsis are co-linear at Mercury perihelion.

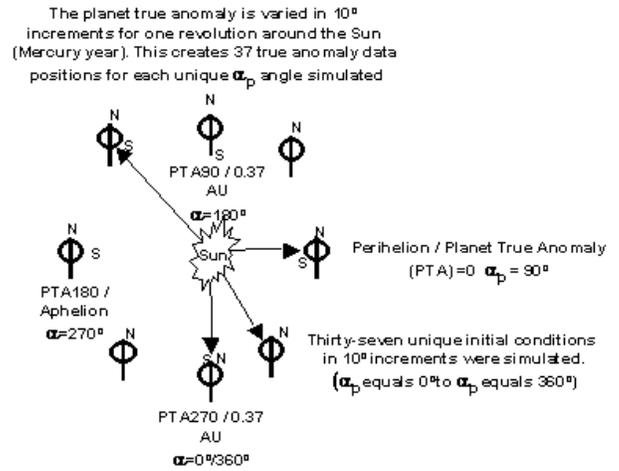


Figure 4. The spacecraft orbit plan remains inertially fixed during a Mercury year. Note the α_p angle is 90°.

Assuming an elliptical spacecraft orbit having an argument of periapsis equal to 180°, α_p would have to be equal to 180° for this condition to exist. The selected α_p equal to 180° angle make this phase angle position non-optimum for two reasons: (1) the thermal heating experienced by the spacecraft from Mercury is the absolute worst case because the orbit periapsis lies over the sub-solar point at perihelion and (2) the spacecraft experiences the longest eclipse period which causes the widest range of thermal cycling for spacecraft components. But if for instance, the same spacecraft orbit had α_p equal to 90°, as illustrated by Figure 4, instead of α_p equal to 180°, then the orbit periapsis *will never* be coincident with the sub-solar point at Mercury perihelion and the spacecraft will not experience the absolute worst case heating and thermal cycling conditions. Therefore when making assumptions during the thermal design process of a Mercury orbiting spacecraft, do not randomly assume the worst case thermal condition will exist some time during the orbiting mission. By assuming the absolute worst case thermal environment when the spacecraft can never experience the absolute worst case thermal environment increases the potential for adding mass, operational constraints and or other protective schemes which probably add little value to the overall thermal control of the spacecraft for a given α_p .

SPACECRAFT ORBIT GEOMETRY

The spacecraft orbit geometry used for the orbital heating calculations is the same used for the MESSENGER mission and is illustrated by Figure 5. The MESSENGER orbit geometry used Ω equal to 120°. The three other Ω conditions analyzed in conjunction with the MESSENGER Ω condition sweep a 90° sector in 30° increments of

argument of periapsis, allowing for temperature comparisons to be made between the different argument of periapsis angles. The highly elliptical and nearly polar MESSENGER orbit tightly balances power, thermal, science, propulsion, and mission ground operations requirements. The inclination, 80° , gives certain thermal biases based on the α_p angle chosen. For example the thermal analysis showed slightly warmer spacecraft conditions when the α_p angle was 90° as compared to 270° . Both of these initial conditions represent dawn-dusk orbits, except that the 90° case has the orbit tilted 10° on the Sun side of the North Pole at perihelion while the 270° is tilted 10° toward the dark side at perihelion.

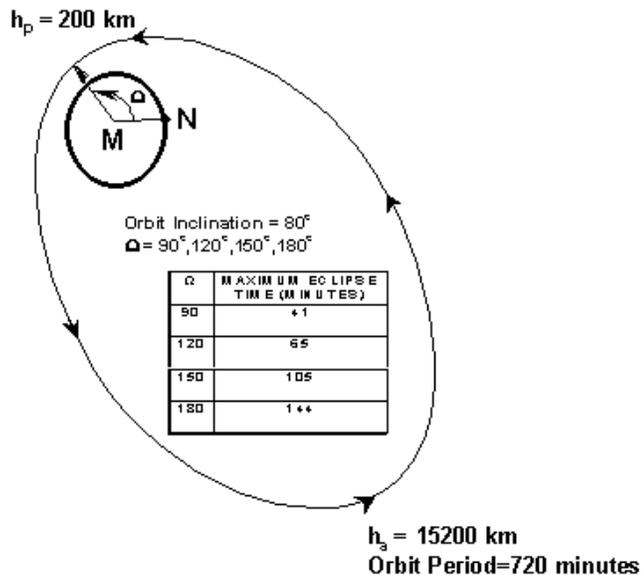


Figure 5. The spacecraft orbit geometry used in the analysis. Ω defines the four arguments of periapsis conditions simulated. The MESSENGER mission uses $\Omega = 120^\circ$.

Because of the highly elliptical shape of the spacecraft orbit (eccentricity = 0.74), orbit average heating from the planet is very misleading. Thermal analysis shows almost no difference in predicted orbit average temperatures for the same spacecraft surfaces at very different solar distances when being heated by the planet. This is because the orbital heating from Mercury is integrated and averaged over the 720-minute orbit period. The time the spacecraft views the planet, during which the orbit altitude range has an effect on the thermal response of a viewing surface, is less than 40 minutes per orbit. When integrated and averaged, the planet thermal effect is almost negligible. Therefore transient analysis must be used to truly characterize the thermal response of the spacecraft and identify potential thermal design drivers. During most of the orbit the spacecraft never gets direct heating from Mercury. Consequently, there is a very large thermal recovery period associated with this orbit, which allows the spacecraft to return to a benign initial state before being re-heated during planet viewing. If this orbit is properly phased at perihelion, it will be shown that a

Mercury orbiting mission can be accomplished with very low thermal risk.

THERMAL ANALYSIS DESCRIPTION

The object of the thermal analysis was to reduce the complicated interactions between the local phase angle (α) and the planet solar distance into an easy to understand relationship which represents the worst case spacecraft temperatures as a function of initial phase angle referenced at Mercury perihelion (α_p). This allowed for the determination of the optimum thermal α_p angle referenced at Mercury perihelion. As previously discussed the angular variation during a Mercury year between the sub-solar point and spacecraft ascending node changes mathematically as a function of Mercury's true anomaly position. Since Mercury's solar distance can be directly derived from its true anomaly position, the solar constant and planet surface temperature can also be varied as a function of planet true anomaly.

With the initial α_p angle equal to zero, steady state and transient thermal calculations were performed over one Mercury year, spacing the calculations at 10° planet true anomaly increments and varying the α angle, the solar constant and the Mercury sub-solar emissive power as a function of planet true anomaly. Thirty-seven planet true anomaly positions were used to simulate one Mercury revolution around the Sun with the associated spacecraft heating for a given α_p angle (see Figure 4). The thermal model, while keeping track of all prior calculations, indexed the previous α_p angle by 10° and another Mercury year of analysis was performed. This process was repeated in 10° increments of the previous α_p until the final α_p was equal to 360° and totaled 37 unique initial phase angles (α_p) and corresponding Mercury revolutions around the Sun. At the end of each Mercury year which was associated with a unique α_p , all of the model maximum and minimum temperatures for that Mercury year were saved and stored as a function of planet true anomaly position. Also, upon completion of a Mercury year at a given α_p , the maximum and minimum spacecraft temperature and heating rates which occurred during the Mercury year at the given α_p were saved as a function of α_p . In effect, the 37 maximum and the 37 minimum temperatures and heating rates which were saved as a function of planet true anomaly have been reduced to one data value representing the Mercury year maximum and minimum for temperatures and heating rates at a given α_p . This was done for each of the four orbits listed in Figure 5.

The spacecraft's worst hot and cold case temperature response at a specific orbit argument of periapsis (Ω) can now be determined as a function of α_p . The relationship between the overall maximum spacecraft temperature and the initial phase angle at Mercury perihelion for a given Ω angle summarizes the effects of orbit position at perihelion and the resultant spacecraft temperature

response. This enables the design team to choose an acceptable range of initial phase angles at perihelion which will balance thermal control, mass, power, Δv and science.

In order to accomplish such a numerically intensive task, specially designed software was used to systematically simplify this potentially daunting analysis. The software allows the effects of varying solar and planetary heating, orbit plane phasing (α), solar array rotation, and surface optical property degradation to be readily synthesized by SINDA. The spacecraft TRASYS model was designed to calculate spacecraft interactions with the Sun and Mercury independent of solar distance (or planet true anomaly position). Thirty-seven spacecraft-to-planet sub-solar point positions spaced at 10° angular increments were run for each of the argument of periapsis orbits simulated. Each of these positions represents a unique α_p angle and can be phased in SINDA as a function of planet true anomaly. The resolution of orbit position phasing is ten angular degrees requiring the SINDA model to move planet true anomaly in exactly the same ten angular degree increments for the proper matching of orbit plane position (α) and planet solar distance (true anomaly) during the course of a Mercury year. As a result, the database of 37 orbit positions was used for each of the 37 α_p angles simulated, and this was done for each of the four arguments of periapsis conditions modeled. By removing the association between spacecraft orbit phase angle (α) and planet solar distance, the number of unique orbital heating data files needed to accomplish the same level of detail in the thermal analysis was reduced from 1369 (or 37^2) to 37. The thermal software allowed the generalization of the orbital heating analysis and greatly simplified this very complex problem.

The solar arrays, by design, are articulated for temperature control. A database of rotated solar array positions at each of the 37 orbit positions had to be generated to simulate orbital heating of the solar arrays due to the Sun and planet. Thirty-nine solar array tilt positions were modeled at each of the 37 orbit phase positions, totaling to 1443 orbital heating files for each of the four argument of periapsis positions. A tilt angle of zero represents the array fully projected to the Sun and a tilt angle of 90° has the array edge to the Sun. The increments in rotational spacing between tilt positions varied from 5° to 1° with the 1° increments occurring at the highest tilt positions where sensitivity between temperature and tilt angle are the strongest. If the solar array TRASYS model was specific with regard to orbit plane position and solar distance, the total number of heating rate files would be 53391 to accomplish the same level of detail in the analysis for each of the argument of periapsis positions simulated. It is evident that the thermal software adds enough generality to enable unprecedented flexibility with the SINDA analysis that greatly simplifies the amount of case specific orbital modeling needed with TRASYS. A more com-

plete discussion of the thermal software used for this analysis can be obtained in Ercol and Krein, 1997 [3].

THERMAL ANALYSIS RESULTS

The thermal analysis results presented will take the form of maximum temperature plotted against α_p . These plots summarize the maximum predicted spacecraft transient temperatures during one Mercury year as a function of α_p for each Ω angle. Optimizing across the parameters of α_p and Ω angle is the most efficient way to achieve the lowest mass thermal control system while minimizing spacecraft fuel and achieving the highest science return. The MESSENGER mission does just this.

MESSENGER is injected into a near-terminator orbit at the Mercury true anomaly position of 337° with the sub-solar point, Mercury ascending node angle (α) equal to 271.5° . Solving for α_p via equation (1) yields α_p equal to -65.5° measured in a counterclockwise sense from the sub-solar point or 294.5° using the clockwise angular convention described by Figure 3. During the life of the mission the orbit plane will precess approximately 5° in a clockwise manner around the equator ending with α_p equal to 299.5° .

It is evident from the analysis data presented in Figures 6 and 7 that the spacecraft orbit plane is approaching the optimum thermal phase angle as α_p slowly increases from 294.5° to 299.5° . Keep in mind that the resolution of the phase angle increments is 10° and that α_p equal to 294.5 was not exactly analyzed. Therefore α_p equal to 290° is the closest model approximation to the actual injection position and will represent with enough accuracy the expected worst hot case temperature extremes for the MESSENGER spacecraft.

Figure 6 represents the maximum predicted transient temperature of the outer surface of unobstructed planet-facing multi-layer insulation (MLI). As illustrated, the overall maximum MLI temperature achieves approximately 410°C and occurs at the very non-optimum thermal condition where the sub-solar point and orbit periapsis are coincident at planet perihelion (both α_p and Ω are equal to 180°).

Referring to Figure 6, with α_p equal to 290° and Ω equal to 120° , the maximum predicted planet facing MLI surface temperature during one Mercury year is approximately 275°C . This temperature is 125°C cooler than the maximum temperature predicted during the non-optimum thermal conditions created when both α_p and Ω are equal to 180° . Figure 7 represents a summary of the maximum predicted internal spacecraft temperatures as a function of α_p . With 250 watts of internal heat dissipation, the maximum spacecraft temperature when simulating the MESSENGER mission and the non-optimum thermal condition (both α_p and Ω are equal to 180°) are 40°C and 97°C respectively.

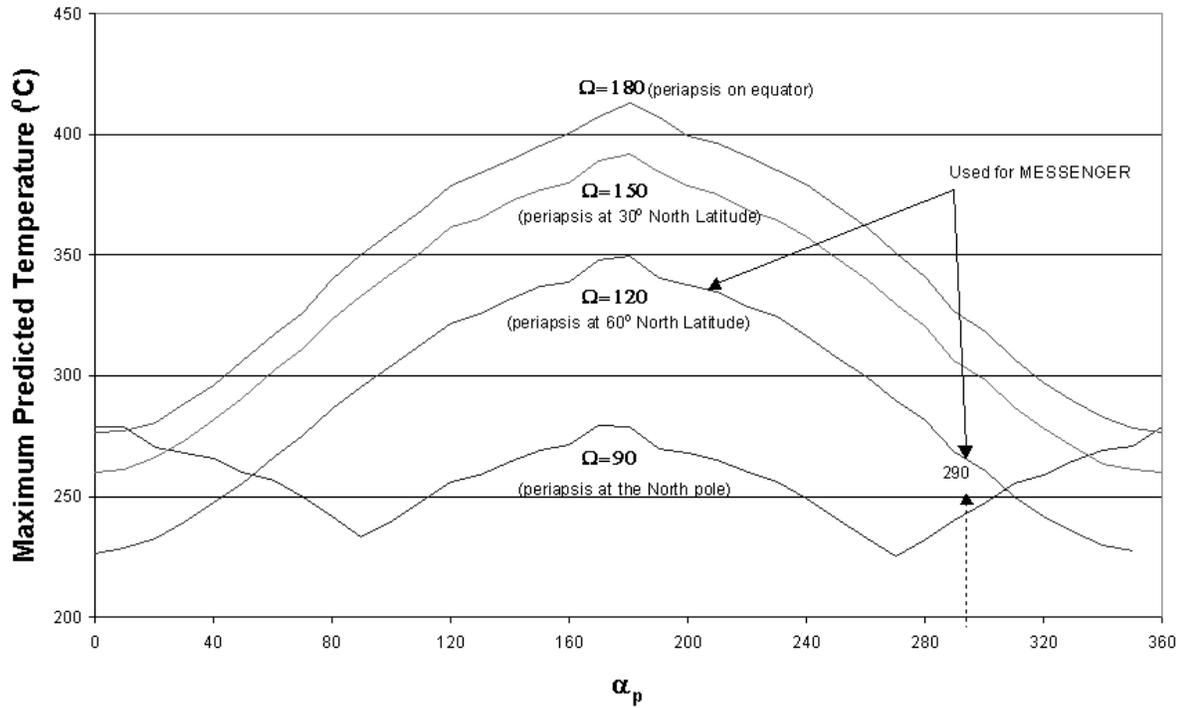


Figure 6. Summary of maximum external MLI transient response.

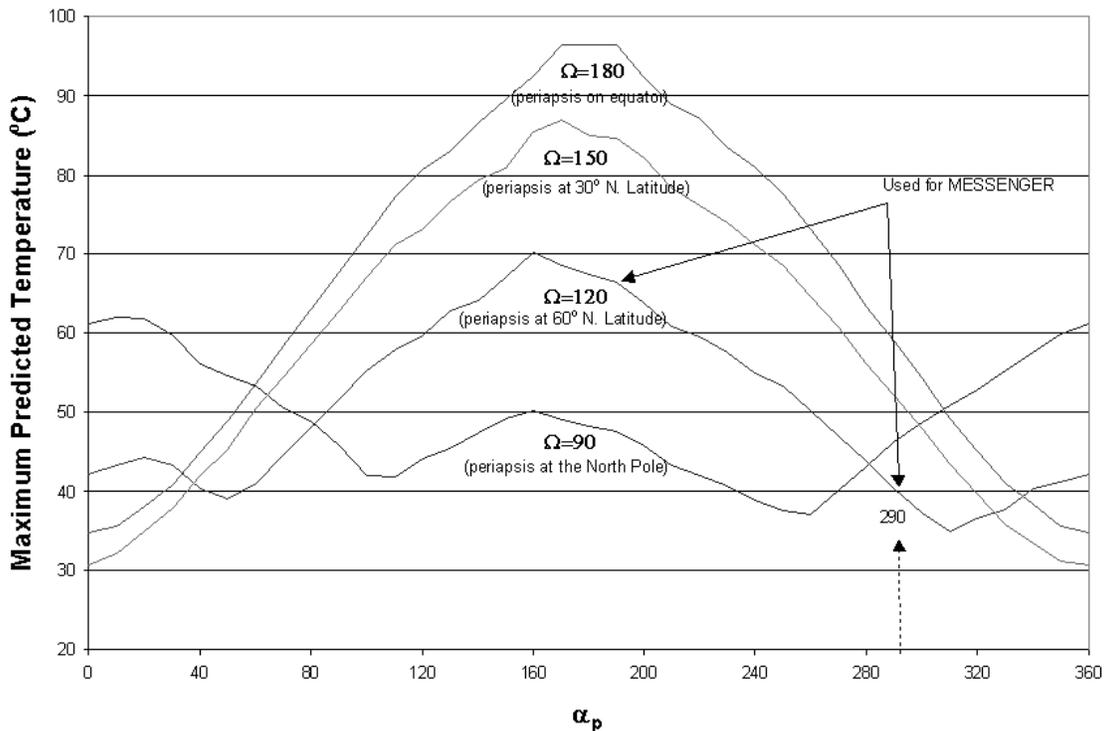


Figure 7. Summary of maximum internal spacecraft transient response.

Referencing Figure 8, a peak internal temperature of 40°C for the MESSENGER mission occurs at the planet true anomaly position of 280° (0.36 AU/6.9 ESC), which corresponds exactly with the maximum temperature listed in Figure 7 when using the MESSENGER conditions. A peak internal temperature of 97°C for the non-

optimum thermal case occurs at planet perihelion and matches exactly with the results presented in Figure 6 for the given condition. Note that the peak internal temperature for the MESSENGER mission does not occur during the maximum eclipse orbit when the orbit periapsis is closest to the sub-solar point (true anomaly of 250°); nor

does it occur at perihelion where the solar and planetary thermal environments are at a maximum. The peak internal temperature for MESSENGER occurs when the planet true anomaly is equal to 280° . Obviously, it is very hard to estimate without the benefit of a good thermal model where during the Mercury year the worst case hot temperature for any given component will occur.

Figure 8 further illustrates Figures 6 and 7 represent the summary of peak temperatures predicted over a Mercury year at specific α_p and Ω angles. Each temperature corresponding to a specific α_p in Figures 6 and 7 represents the maximum expected temperature during one Mercury year for that specific α_p angle and the desired Ω angle. Once the α_p angle is picked for a mission, any thermal issues uncovered by reviewing the appropriate temperature data in formats similar to Figures 6 or 7 can be reviewed in higher detail. From the standpoint of spacecraft thermal design, a more complete understanding of temperature trending for specific α_p and Ω conditions and different spacecraft components can be achieved by reviewing the data in formats similar to that depicted by Figure 8. This data format depicts spacecraft temperatures as a function of planet true anomaly position and illustrates where during the mission to expect the worst hot and cold case temperature performance. This knowledge can help the science and engineering teams correct any spacecraft temperature related problems by either making hardware modifications or by adjusting spacecraft operational requirements.

The thermal engineer can also examine the transient temperature response for components such as the solar arrays and electronics as a function of orbit time at the true anomaly position in question. All the temperature and heat data generated by the SINDA model as a function of orbit time exists in the transient results file created at each analyzed planet true anomaly position for the specific α_p and Ω conditions. This method of data analy-

sis is an effective technique in uncovering thermal problems associated with any spacecraft component. Upon inspection of a component's temperature response as a function of true anomaly, the transient data associated with the true anomaly position in question can be analyzed and all thermal issues for that component can be resolved.

Figure 9 shows the spacecraft MLI temperature response over one Mercury year plotted as a function of planet true anomaly. The peak temperature results from Figure 9 can be compared to ones shown in Figure 6 for the MESSENGER mission and the non-optimum thermal condition.

Figure 10 shows the transient temperature response for the solar arrays at the true anomaly positions that give maximum spacecraft eclipses. The MESSENGER mission solar array temperature response is compared to the solar array temperature response during non-optimum thermal condition (both α_p and Ω are equal to 180°). The maximum eclipse orbit for the MESSENGER mission occurs at the planet true anomaly position of 250° (0.39 AU/5.5 ESC) while the non-optimum thermal orbit occurs at Mercury perihelion (0.30 AU/11.1 ESC). Each of the MESSENGER solar arrays is dual sided to help maximize power watt density and minimize panel temperature over all mission solar conditions. Side one has solar cells mixed with Optical Solar Reflector's (OSR) and is designed to be used during the cruise phase, when solar distances are <0.5 AU, and exclusively during the orbiting phase. Side two is fully packed with solar cells and is used when the solar distance is >0.5 AU. Since this paper deals only with the orbiting phase of a Mercury mission, side one shall be defined as "the front side" and side two as "the backside". The data relates the front side average temperature to the time relative to orbit periapsis. During all sunlit portions of the orbit, the thermal model attempts to maintain the solar array temperature at 150°C by rotating the arrays as necessary. As the arrays

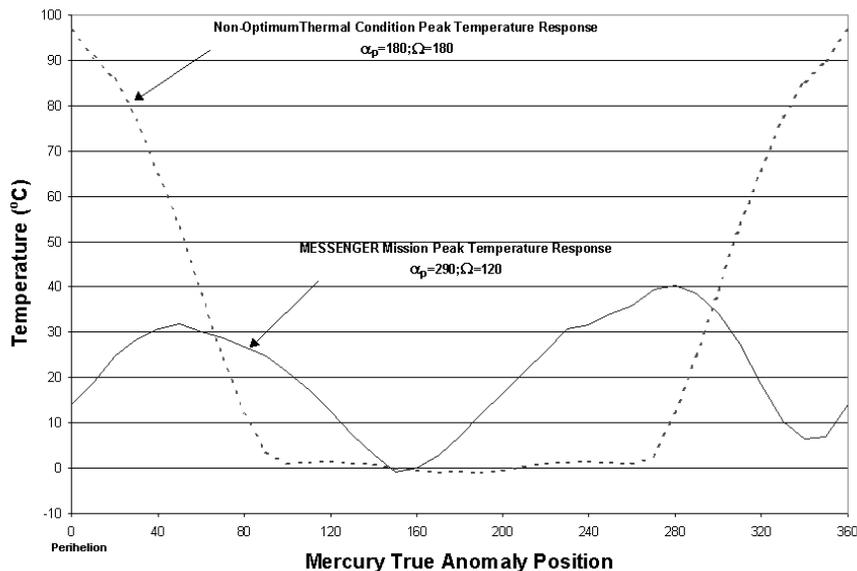


Figure 8. Spacecraft maximum internal bulk temperature over one Mercury year.

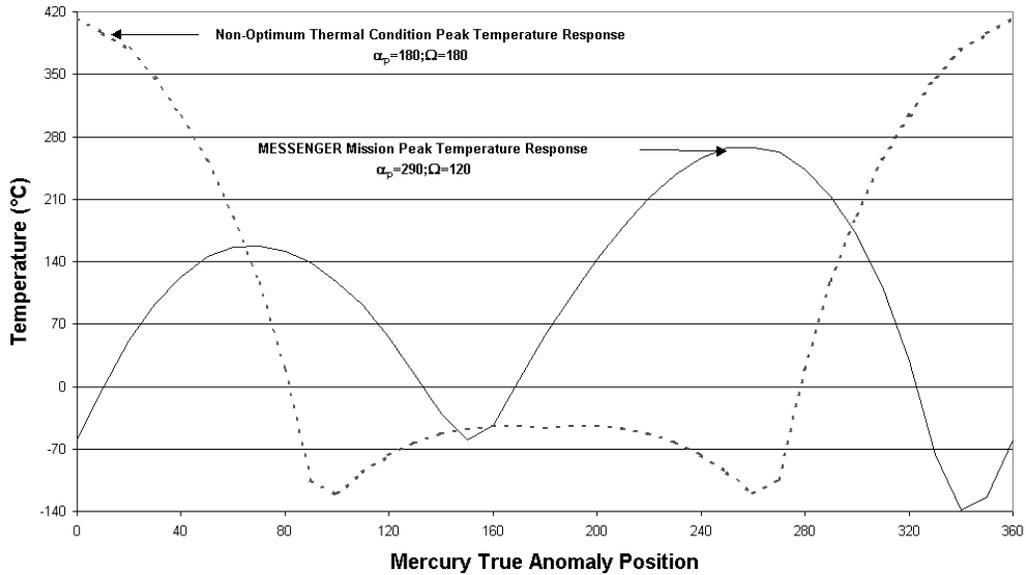


Figure 9. Spacecraft maximum external MLI temperature over one Mercury year.

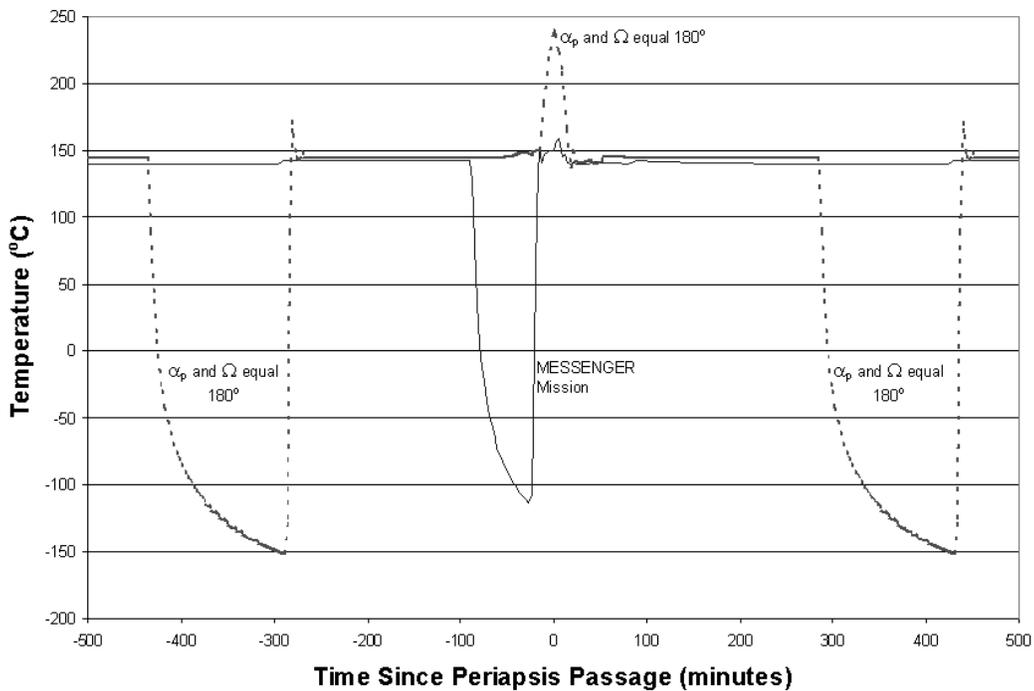


Figure 10. Transient solar array/OSR side temperature response during maximum eclipse orbits.

approach orbit periapsis and cross the sub-solar point, the thermal model rotates the arrays edge on to the Sun to reduce the solar input while both front and back sides receive IR and albedo heating from the planet. It is evident from the analysis that thermally controlling the arrays is achievable for the MESSENGER mission and impossible for the non-optimum condition. For the non-optimum condition with the array edge on to the Sun, the peak front side temperature exceeds 240°C. While in the 144-minute eclipse, the solar array temperature

approaches the coldest extreme of -150°C, causing the solar arrays to be thermal cycled over a temperature range of 400°C.

Figure 11 shows the transient orbital heating as applied to the front and backsides of the solar arrays during the maximum eclipse orbit for the non-optimum thermal condition. After exiting the 144-minute eclipse, the thermal model control algorithm tries to achieve a front side temperature of 150°C by rotating the solar array into a solar normal position. As the front side temperature overshoots

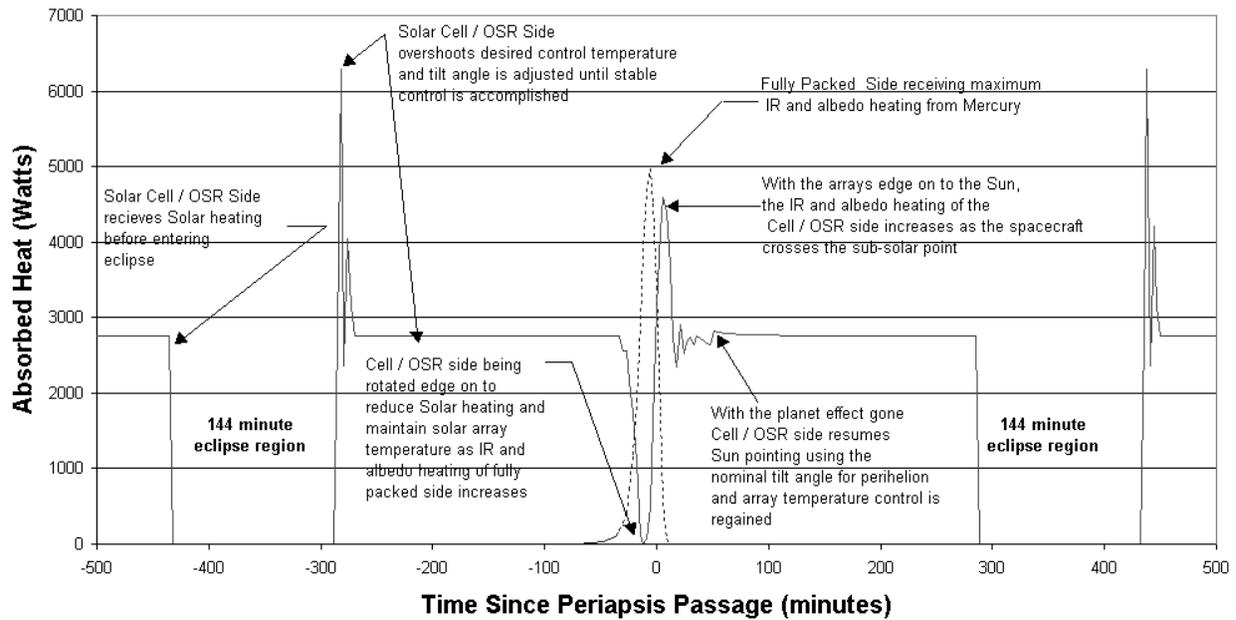


Figure 11. Transient solar array OSR side orbital heating during sub-solar crossing orbit with α_p and Ω equal to 180° .

the desired control point, the algorithm readjusts the tilt angle until stable temperature control is achieved. As the solar array begins to pass over the sunlit side of the planet, the back side of the solar array starts to receive IR and albedo heating. During this time the thermal control algorithm proportionally increases the tilt angle to reduce the solar loading on the front side while the back side heat loads, due to planet proximity, is increasing. The array is finally tilted edge on to the Sun. Soon after the array is fully tilted, the front side begins to receive planetary IR and albedo heating and the back side heating begins to reduce. Planetary heating of the front side continues as the solar array passes the sub-solar point and moves into the Southern Hemisphere. After about 40 minutes of front and back side planetary heating, the control algorithm is able to reposition the solar array at the nominal tilt angle for perihelion conditions ($\sim 65^\circ$) and maintain this angle until the spacecraft exits the 144-minute eclipse. Then the cycle is repeated. The heating rate results depicted represents the non-optimum thermal condition and not the MESSENGER mission, but the same control technique logic is used during the MESSENGER mission solar array analysis.

CONCLUSION

From the thermal analysis presented, it is apparent that the optimum injection point for a Mercury orbiting mission can be chosen so that spacecraft temperature control is very manageable. The MESSENGER mission is an example of how optimizing the spacecraft injection point yields a low mass and low risk spacecraft design. Because of the near-optimum initial phase angle chosen

at Mercury perihelion, MESSENGER spacecraft components such as the solar arrays and electronics experience lower operating temperatures and a benign thermal cycling range, which translates into higher reliability and prolonged mission life. When the MESSENGER mission thermal analysis is compared to the non-optimum thermal case, it is obvious that a poor choice of α_p and Ω can lead to spacecraft orbiting environments that can be very punishing. As illustrated the non-optimum thermal case represents a thermal engineer's nightmare. Passive spacecraft-bus thermal control is unachievable. Articulating solar arrays will not maintain temperature when close to the sub-solar point around perihelion. External spacecraft surfaces can get extremely hot. The MESSENGER choice of α_p and Ω eliminates all of the concerns associated with the non-optimum thermal case and provides a low thermal risk orbiting mission.

In summary, the analysis described throughout this paper have provided a comprehensive comparison of all potential thermal environments for the given orbit geometry and orbit inclination. This potentially daunting analysis was greatly simplified by the thermal software and the integrated SINDA and TRASYS modeling approach. All of the analysis techniques described are general and can be used for any orbit geometry and spacecraft attitude. Results of the MESSENGER mission analysis has helped to ensure an optimized spacecraft system design by communicating design drivers in clear and unambiguous formats to engineers, scientists and mission designers which will lead to an efficient low mass and low risk spacecraft design. Mercury Awaits.

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ACRONYM LIST

ESC	Earth Solar Constants
ICES	International Conference on Environmental Systems
IR	infrared
MESSENGER	MErcury Surface, Space, ENvironment, Geochemistry, and Ranging
MLI	multi-layer insulation (MLI)
OSR	Optical Solar Reflector's (OSR)
PTA	Planet True Anomaly
SAE	Society of Automotive Engineers
SINDA	Systems Improved Numerical Differencing Analyzer
TRASYS	Thermal Radiation Analyzer System