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# Mission analysis for the Mars 2007 opportunity

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# NAVAL POSTGRADUATE SCHOOL
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# THESIS

**MISSION ANALYSIS FOR THE MARS 2007 OPPORTUNITY**

by

Stephen B. Zike

December 1998

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**ABSTRACT *(maximum 200 words)*** In 2007, NASA will launch an orbiter and a lander to Mars in support of science and exploration goals. NASA's Jet Propulsion Laboratory is responsible for the mission design. A trajectory analysis is necessary to ensure that the most cost-effective interplanetary transfer is implemented. This thesis presents a comprehensive analysis of all possible type 1, 2, 3, and 4 Earth-Mars trajectories with reasonable launch energy requirements as well as possible return trajectories to Earth for the case of a sample return. Launch periods were determined using the JPL programs MIDAS and CATO. The corresponding C3 requirements for each trajectory were then utilized to obtain the performance capabilities for the Delta II series, Atlas II series, and Ariane 4/5 launch vehicles. The injected mass derived from the performance data was subsequently used as the spacecraft design point. The goal of this analysis was to identify the trajectory type and orbiter capture scheme that produced the maximum post-capture orbiter mass. The advantages and disadvantages of propulsive capture, aerocapture, and aerobraking are addressed for numerous launch scenarios in which the orbiter and lander are either launched on separate launch vehicles or on a single launch vehicle. This comparison was successful in demonstrating the impact of the orbiter capture scheme on the selection of the optimal trajectories.

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# MISSION ANALYSIS FOR THE MARS 2007 OPPORTUNITY

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# ABSTRACT

In 2007, NASA will launch an orbiter and a lander to Mars in support of science and exploration goals. The NASA's Jet Propulsion Laboratory is responsible for the mission design. A trajectory analysis is necessary to ensure that the most cost-effective interplanetary transfer is implemented. This thesis presents a comprehensive analysis of all possible type 1, 2, 3, and 4 Earth-Mars trajectories with reasonable launch energy requirements as well as possible return trajectories to Earth for the case of a sample return. Launch periods were determined using the JPL programs MIDAS and CATO. The corresponding C3 requirements for each trajectory were then utilized to obtain the performance capabilities for the Delta II series, Atlas II series, and Ariane 4/5 launch vehicles. The injected mass derived from the performance data was subsequently used as the spacecraft design point. The goal of this analysis was to identify the trajectory type and orbiter capture scheme that produced the maximum post-capture orbiter mass. The advantages and disadvantages of propulsive capture, aerocapture, and aerobraking were addressed for numerous launch scenarios in which the orbiter and lander are either launched on separate launch vehicles or on a single launch vehicle. This comparison was successful in demonstrating the impact of the orbiter capture scheme on the selection of the optimal trajectories.

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# I. INTRODUCTION

## A. BACKGROUND

The Mars Surveyor Program continues to launch spacecraft to the red planet on a regular basis in support of science and exploration goals. The Jet Propulsion Laboratory (JPL), under contract with the National Aeronautics and Space Administration (NASA) is responsible for establishing the mission objectives and developing the appropriate mission design. Each mission is specifically designed to either further our understanding of Mars and space or to demonstrate an emerging technology that will ultimately make it feasible to send a manned mission to Mars. The NASA Human Exploration and Development of Space (HEDS) program is currently projecting a manned mission in the 2014 timeframe. The current technologies under development that would enable such a mission include precision landing and ascent propulsion, in-situ propellant production, planetary protection and exobiology, vehicle survivability, miniaturization and integration of instruments into robotic vehicles, and the flight-qualification of science instruments. Additionally, as Mars and Earth shared similar conditions billions of years ago, a comparison of the two planets might enable scientists to gain a better understanding into Earth's history and possibly its future.

In the summer of 1996, NASA/JPL successfully sent a spacecraft to Mars for the first time in 20 years as Pathfinder and Sojourner landed safely on its surface. The subsequent mission in the Mars Surveyor Program was the Mars Global Surveyor (MGS). The MGS orbiter is currently establishing a mapping orbit about Mars in an attempt to gather data on the surface, atmospheric, and magnetic properties of the planet. Scientists anticipate this data will aid in the planning of future missions. The Mars

Surveyor 98 mission will launch both an orbiter and a lander at the end of this year to collect and return science data using both in-situ and remote sensors. Mars 98 represents the next generation of spacecraft being sent to Mars. The orbiter will assume a circular mapping/communications orbit in support of the lander prior to conducting various environmental experiments. The lander will land in proximity to the South pole and will be equipped with cameras, a robotic arm, and various instruments to measure the Martian soil composition. Additionally, microprobes will be piggy-backed on the lander and released prior to landing. These microprobes will impact the surface penetrating it by as much as 18 inches in an attempt to detect water ice.

Opportunities for sending spacecraft to Mars only occur approximately every two years due to the current launch vehicle limitations. Therefore, the follow-on mission to Mars 98 will occur in early 2001. Although originally planned for the demonstration of several firsts (aerocapture and precision landing), these technologies have been delayed to a future yet-to-be-determined mission. The current mission is comprised of an orbiter and a lander utilizing as much of the Mars 98 hardware design as possible. The lander however, will also be equipped with a HEDS payload to demonstrate the feasibility of propellant production and oxygen generation.

The following Mars mission will be a technology demonstration in 2003. This mission will consist of a lander with an enlarged rover similar in design to the one to be used in 2005, which will demonstrate the enabling technologies necessary for the successful completion of a sample return mission in 2005. The 2005 sample return mission will consist of an orbiter/return vehicle and a lander/rover/ascent vehicle. The rover will collect samples, return to the ascent vehicle, and launch for a rendezvous with the orbiter before injecting back to Earth.

## B. MARS 2007

The baseline for the 2007 opportunity consists of an orbiter and a lander/rover configuration launched on separate launch vehicles. However, as this is a preliminary analysis to be used as a starting point for the mission design, a single launch vehicle for both orbiter and lander was considered as well as the possibility of a sample return. The goal of mission planning is to effectively minimize the cost to the program while reducing the risk. This includes identifying all possible Earth-Mars and Mars-Earth trajectory options for the mission. Although there is a substantial amount of data presented, the focus of the analysis is on the process of mission design. The trajectory analysis parameters are presented such that the applicability of each to mission design is clearly defined. A description of each JPL program utilized is provided with an emphasis on the importance of the analysis results rather than how to specifically use the programs. Determination of the most suitable launch periods for the orbiter and the lander is an intensive process that deals with a series of trades between the trajectory parameters. These trades are identified throughout the analysis and used to form an intuition regarding similar trajectories. The geometry of the spacecraft trajectory is important in the determination of pointing angles in the case of the solar arrays and communications antennas. Ranges from the spacecraft to Earth and the Sun help determine the communications link budget as well as the size of the solar array required.

The analysis of the trajectories deals with a wide range of topics with the ultimate goal of identifying a set of trajectories that maximize the post-capture mass of the orbiter.

Beginning with a launch vehicle selection, an initial mass can be determined using the performance capabilities given the launch energy requirement associated with each trajectory. Upon arrival, the orbiter must enter into orbit about Mars by using one of three capture schemes - propulsive capture, aerocapture, and aerobraking. Each capture method is unique and impacts the post-capture mass differently. The specific effect is driven by the parameters associated with each trajectory.

A propulsive capture is accomplished by conducting a maneuver at the periapsis of the incoming hyperbolic trajectory and subsequently circularizing the orbit with a second maneuver. Aerobraking also uses a propulsive maneuver at the periapsis, but only to capture into a highly elliptical orbit. The orbiter will then lower the periapsis of the orbit into the upper crust of the Martian atmosphere allowing the drag force generated to remove energy from the orbit and consequently cause a decay in the apoapsis until the apoapsis is near the final orbit altitude. At that time, the orbit is circularized using propulsive means. The aerobraking method can require many days to achieve the final orbit, as it is a function of the period of the initial orbit and the density of the atmosphere at the penetration altitude. Aerocapture makes a direct entry into the atmosphere thus removing the required energy in one maneuver. Upon exit from the atmosphere, only a small amount of propellant is required to then circularize the final orbit.

The post-capture orbiter masses are calculated for various scenarios using two launch vehicles as well as one launch vehicle. Determination of this mass depends on a series of trades with respect to the propellant mass required, the impact of the trajectory analysis parameters on the mission, and the estimated mass for the thermal protection system (TPS). Comparisons between capture schemes and trajectory types are then conducted providing the mission designer with sufficient data to determine the most cost-effective trajectory type and capture scheme combination.

# II. TRAJECTORY ANALYSIS PARAMETERS

## A. INTRODUCTION

An interplanetary trajectory to Mars is defined by at least a four-body problem involving the Earth, the Sun, the spacecraft, and Mars. Although it is more suitable to use an N-body numerical solution, this problem can be closely approximated by the use of patched conics. The patched conic method allows for simplification of the N-body problem into a series of two-body problems which when pieced together, provide amazingly accurate results. When one body (central) is much more massive than the other, the gravitational center is taken to be the center of the central body. The secondary body then moves in Keplerian fashion about the central body such that the central body is one focus of the conic (ellipse, parabola, hyperbola). The trajectory is normally divided into three phases. The first consists of the departure phase in which the two bodies of interest are the origin planet (Earth in this case) and the spacecraft. The trajectory is a hyperbola with the Earth at one focus. All other bodies are neglected. The second phase is the cruise phase in which the sun and the spacecraft are the two bodies of interest and the trajectory is now an ellipse about the sun. The third and last phase is the arrival phase in which the target planet (Mars in this case) and the spacecraft are the bodies of interest and the trajectory is another hyperbola using Mars as the focus.

Each celestial body is assumed to have a somewhat arbitrary sphere of influence (SOI), generally defined by the radius of the body and the ratio of masses between the body and the sun raised to some power (generally understood to be 0.4). Within the SOI of a celestial body, only that body and spacecraft are considered in the equations of motion for the trajectory. These SOI boundaries then become the points at which the three trajectories are "patched" together, thus forming a "patched conic." In preliminary analysis, this method is suitable as a first look. However, numerical integration methods are required in order to optimize the trajectories.

The first step in interplanetary trajectory analysis is to gain an appreciation and understanding of the trajectory parameters. There are many possible trajectories between Earth and Mars for a given period of time. What determines whether one is better than another lies in the analysis of each of the trajectory parameters. Understanding of these applicable parameters provides insight and intuition during the preliminary planning stages of the mission design and enables the best trajectory to be chosen for the mission. Typical parameters used for trajectory analysis include launch energy (C3), declination of the launch asymptote (DLA or $\delta_\infty$), right ascension of the launch asymptote (RLA or $\alpha_\infty$), velocity of the arrival asymptote (VHP or $V_{\infty_A}$), declination of the arrival asymptote (DAP or $\delta_{\infty_A}$), and right ascension of the arrival asymptote (RAP or $\alpha_{\infty_A}$).

## B. LAUNCH ENERGY ($C3$)

The launch energy or C3, defined as the square of the hyperbolic escape velocity, is used as the initial criterion in trajectory analysis. Each trajectory has an associated C3 requirement necessary to inject a spacecraft, so only those trajectories where the C3 is near the minimum are of practical interest. Locating these minimum energy opportunities is critical in determining a suitable launch period. These opportunities occur regularly at intervals defined by the synodic period of Earth and Mars. For Earth and Mars, this synodic period is approximately 778 days, but because the orbits are not coplanar or circular, this period will vary somewhat from opportunity to opportunity. A launch period can then be established (generally 20-30 days) with the nominal launch date occurring at the minimum launch energy.

The relationship between C3, injected mass, and the launch vehicle are closely coupled. As the C3 varies over the launch period, the corresponding injected mass also varies for a given launch vehicle. From day to day within this launch period, launch vehicle performance must equal or exceed the departure energy requirement for the specified spacecraft mass. Therefore, the relationship between C3 and injected mass becomes a primary driver for the spacecraft design. In some cases for a fixed spacecraft design, a lower C3 can lead to the use of a smaller and less expensive launch vehicle. This is especially important given the current climate of fiscal constraints.

## C. DECLINATION OF THE DEPARTURE ASYMPTOTE $(\delta_\infty)$

Generally, Mars missions are flown from Cape Canaveral. However, Mars 96 was launched out of the Baikonur Cosmodrome and the Mars 2001 orbiter will be launched from Vandenburg while future Mars missions may also include an Ariane 5 launch vehicle out of the Kourou Island launch facility. When solving the launch problem, the latitude of the launch site and the launch azimuth (as measured from the North Pole) determine the minimum inclination for the initial parking orbit. The following equation (Sergeyevsky, 1983, p.11) shows this relationship:

$$\cos i = \cos\Phi_L \sin\Sigma_L \quad (2.1)$$

$\Sigma_L$ = launch azimuth
$\Phi_L$ = launch latitude

For example, from Cape Canaveral (approximately 28.5 degrees latitude), the lowest achievable inclination for an initial parking orbit would be 28.5 degrees for a due East launch azimuth ( $\Sigma_L$ = 90). This becomes important when considering the declination of the outgoing hyperbolic asymptote. If the inclination of the parking orbit is greater than the declination of the asymptote, there exists two opportunities for injection. If the inclination and the declination are equivalent, only one opportunity exists for injection. Lastly, for the case where the inclination of the initial parking orbit is less than the declination of the outbound asymptote, there are no opportunities for injection that exist without a plane change first. Figures 2.1 and 2.2 depict the geometric relationships in the launch problem while the mathematical proof that follows demonstrates this principle (Ross, 1998):

| | |
|---|---|
| i = | inclination of the orbit plane with the equator |
| $\Theta$ = | angle between line of nodes and the launch asymptote $\overline{LA}$ measured in the equatorial plane |
| $\delta$ = | angle between launch asymptote and equator |

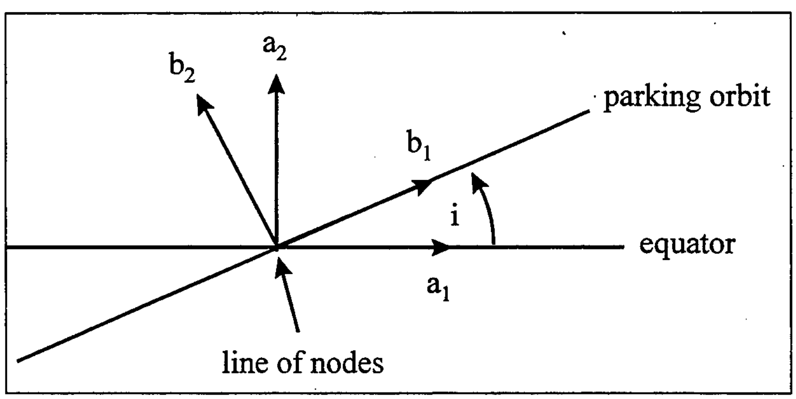


Figure 2.1: Equatorial View – Parking Orbit with respect to the Equator

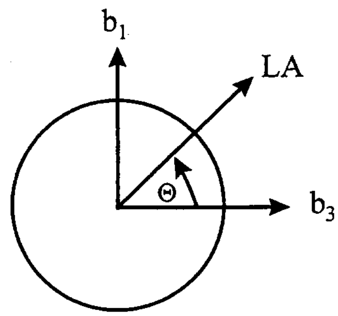


Figure 2.2: Polar View - Launch Azimuth with respect to the Parking Orbit

$$\overline{LA} = \cos\Theta\hat{b}_3 + \sin\Theta\hat{b}_1$$

$$\hat{a}_2 = \sin i\hat{b}_1 + \cos i\hat{b}_2$$

$$\overline{LA}\cdot\hat{a}_2 = \sin i \sin\Theta$$

but the angle between $\overline{LA}$ and $\hat{a}_2 = 90 - \delta_\infty$, so the dot product equals

$$\overline{LA}\cdot\hat{a}_2 = \left|\overline{LA}\right|\left|\hat{a}_2\right| cos\left(90 - \delta_\infty\right) = sin\,\delta_\infty$$

$$\therefore sin\,\delta_\infty = sin\,i\,sin\,\Theta$$

If $i > \delta_\infty$ assuming that $0^\circ \le i \le 90^\circ$ and the inclination of the initial orbit is greater than the declination of the outbound asymptote, $sin\,i > sin\,\delta_\infty$, then

$$sin\,\Theta = \frac{sin\,\delta_\infty}{sin\,i} \Rightarrow sin\,\Theta < 1$$ and therefore two opportunities for injection.

In the case where the departure asymptote is greater than the latitude of the launch facility, two solutions can be considered. First, provided that the launch azimuth constraints are not violated, a launch azimuth other than due East could increase the inclination of the initial parking orbit. However, the more the launch azimuth varies from a due East launch, the less the rotation of the Earth contributes to the velocity. For example, at the equator, a launch vehicle launching due East will benefit from an additional 0.465 km/sec using the standard formula $v = \omega \times r$. The second solution is to conduct a plane change once the spacecraft is established in the initial parking orbit inclination. This option is extremely costly in terms of propellant and is generally not a consideration.

## D. RIGHT ASCENSION OF THE DEPARTURE ASYMPTOTE $(\alpha_\infty)$

The right ascension of the departure asymptote as measured eastward from the vernal equinox, can be used in conjunction with $\delta_\infty$ to determine the daily launch windows. In general, the range of acceptable launch azimuths from Cape Canaveral is 70-115° (Sergeyevsky, 1983, p10). This is due to safety considerations given for overflight of land during the launch vehicle ascent. This range of azimuths coupled with the declination of the outbound asymptote, help determine when and how long each launch window opportunity occurs during the day. As the declination and the right ascension of the departure asymptote change each day with the different trajectories, the duration of these daily launch windows will also change. Assuming that the inclination of the parking orbit is greater than the declination of the departure asymptote, there are two daily launch windows each day. A sample calculation for daily launch windows is given below:

| | |
|---|---|
| Launch Date: | 11/26/06 |
| Declination of Outbound Asymptote: | $\delta_\infty = 21.3^\circ$ |
| Right Ascension of Outbound Asymptote: | $\alpha_\infty = 193.9^\circ$ |
| Launch Latitude: | $\Phi_L = 28.5^\circ$ |

Launch Longitude: $\lambda_L = 259.5^\circ$ E

Launch Azimuth Range: $\Sigma_{L_1} = 95^\circ$ to $\Sigma_{L_2} = 103^\circ$

GMT = Local Time + 4 hours

From (Sergeyevsky, 1983, p.9),

$$\text{cotan}\Sigma_L = \frac{\cos\Phi_L \tan\delta_\infty - \sin\Phi_L \cos(\alpha_\infty - \alpha_L)}{\sin(\alpha_\infty - \alpha_L)} \tag{2.2}$$

where $\alpha_L$ is the right ascension of the launch site. Substituting appropriate values and solving for $\alpha_L$ numerically yields the following results. Note that for each value of launch azimuth, there are two solutions for $\alpha_L$. For this example,

$$\alpha_{L_1} = 159.3^\circ \text{ , } 249.3^\circ$$
$$\alpha_{L_2} = 170.0^\circ \text{ , } 269.4^\circ$$

A relative launch time, $t_{RLT}$ with respect to a 24 hour period, can then be determined in sidereal time using the following equation yielding one value for each $\alpha_L$ and thereby defining the two daily launch windows (Sergeyevsky, 1983, p.10):

$$t_{RLT}' = 24.0 - \frac{\alpha_\infty - \alpha_L}{15.0} \text{ (hrs)} \tag{2.3}$$

$$\begin{array}{lcc} t_{RLT_{1a}} = 21.7\ hrs & & \text{Relative Launch Period 1} \\ t_{RLT_{1b}} = 3.7\ hrs & & 3.7 - 5.0\ hrs \\ t_{RLT_{2a}} = 22.4\ hrs & \rightarrow & \text{Relative Launch Period 2} \\ t_{RLT_{2b}} = 5.0\ hrs & & 21.7 - 22.4\ hrs \end{array}$$

The sidereal times can then be converted to GMT times (Sergeyevsky, 1983, p.10):

$$t_L = \frac{t_{RLT}(15.0)}{\omega_{earth}} + \frac{\alpha_\infty - GHA_{date} - \lambda_L(EAST)}{\omega_{earth}} \tag{2.4}$$

where

$$GHA_{date} = 100.075 + 0.9856123008(d_{50}) \tag{2.5}$$

$d_{50}$ = launch date in terms of full integer days elapsed since January 1, 1950

$$\omega_{earth} = 15.041067179 \frac{\deg}{hr} \tag{2.6}$$

Solving for GMT and converting to local time in one step produces the daily launch windows in local Florida time:

<u>Local Launch Windows (Eastern Standard Time)</u>

(1) 09h 02m 19s - 09h 44m 12s

(2) 15h 05m 11s - 16h 22m 59s

Although these windows appear relatively short in duration, recall that this example only uses an 8° range for launch azimuths. Increasing the range of allowable launch azimuths will in fact result in much longer daily launch windows on the order of several hours in some cases. It should also be noted that some launch vehicle contractors issue launch azimuth constraints dependent upon their ability to conduct mission operations with ground stations after launch. Additionally, some launch vehicles are unable to update launch azimuth targeting data while on the launch pad. The consequence is a single, pre-determined launch azimuth for that mission resulting in an instantaneous launch time. For cases when a particular launch time is desired, the problem shown above can be worked in reverse to determine the launch azimuths necessary to meet those launch times. However, range constraints still apply so desired launch times may not be possible if the launch azimuths are outside the acceptable range for the launch site.

## E. VELOCITY OF THE ARRIVAL ASYMPTOTE $\left(V_{\infty_A}\right)$

The velocity of the arrival asymptote $\left(V_{\infty_A}\right)$ is perhaps the most important parameter upon arrival at Mars for both the orbiter and the lander. As the orbiter arrives on the incoming hyperbolic trajectory, it has an associated velocity relative to Mars. In order for the orbiter to capture into orbit about the planet rather than continuing on the hyperbolic trajectory, the spacecraft must conduct a maneuver called a Mars Orbit Insertion (MOI) maneuver. The two types of orbiter capture schemes that employ such a maneuver are termed propulsive capture and aerobraking. In general, for a strictly propulsive capture, the orbiter is targeted to a periapsis altitude of 250 km. Although the actual duration of the MOI burn could be on the order of many minutes, it is treated as a tangential impulse for purposes of preliminary analysis. The purpose of the MOI maneuver during propulsive capture is to place the orbiter into an orbit such that the initial apoapsis following the MOI maneuver is roughly the final mission altitude for the orbiter (400 km in the orbit mapping case). The magnitude of $V_{\infty_A}$ determines the amount of $\Delta V$ in the following equation that is required for the MOI maneuver (Sergeyevsky, 1983, p.25):

$$\Delta V = \sqrt{V_{\infty_A}^{\ 2} + \frac{2\mu_p}{r_p}} - \sqrt{\frac{2\mu_p r_a}{r_p\left(r_a + r_p\right)}} \qquad (2.7)$$

$r_p$ = Periapsis radius
$r_a$ = Apoapsis radius
$\mu_p$ = Mars gravitational constant

Once the MOI is completed, the spacecraft will conduct a second tangential burn (again assumed impulsive) in Hohmann-type fashion at the 400 km apoapsis thus circularizing the orbit.

$$\Delta V = \sqrt{\frac{\mu_p}{r_f}} - \sqrt{\frac{2\mu_p}{r_a} - \frac{\mu_p}{a}} \qquad (2.8)$$

a = Semi-major axis

$r_f$ = Final orbit radius

The $\Delta V$ from equation (2.8), when added to the MOI $\Delta V$, represents the total requirement for orbit insertion (Vallado, 1997, p.290). The propellant mass required can then be determined using the rocket equation in the following form (Brown, 1992, p.52):

$$m_p = m_i\left(1 - e^{\frac{-\Delta v}{g I_{sp}}}\right) \qquad (2.9)$$

$g = 9.806\ km/s^2$
$I_{sp}$ = Specific Impulse
$m_i$ = mass prior to maneuver

In most cases, due to the large $\Delta V$ associated with the MOI, a bi-propellant propulsion system and associated $I_{sp}$ will be used.

Aerobraking for the orbiter differs from the purely propulsive capture in that the spacecraft orbit is not immediately circularized using a two-burn Hohmann-type transfer. The spacecraft however, still conducts a burn at periapsis much like the propulsive case, but the magnitude of the burn is such that the resultant orbit is highly elliptical. The amount of $\Delta V$ required for the MOI maneuver for this case can be related to the period of the initial capture orbit (Sergeyevsky, 1983, p.25).

$$\Delta V = \sqrt{V_{\infty_A}^{\ 2} + \frac{2\mu_p}{r_p}} - \sqrt{\frac{2\mu_p}{r_p} - \sqrt[3]{\left(\frac{2\mu_p \pi}{P}\right)^2}} \qquad (2.10)$$

P = Period of the Capture Orbit (sec)

Aerobraking begins when the orbiter conducts a series of burns at successive apoapsis passes effectively lowering periapsis from the entry altitude of 250 km to somewhere just inside the upper reaches of the Martian atmosphere (approximately 110-115 km). The atmospheric drag on the spacecraft serves to reduce the energy of the orbit and thus the apoapsis altitude on each successive pass through the atmosphere. The density of the atmosphere at periapsis, the orbital velocity and ballistic coefficient of the spacecraft determine how much energy is removed each pass and therefore how quickly the orbit is reduced. Over a period of time, the apoapsis lowers to the point where only a small series of $\Delta V$ corrections are required to walk the periapsis altitude out for a fully circularized orbit. This method has been employed twice in the past with Magellan about Venus and currently with Mars Global Surveyor and is seen as an excellent alternative to propulsive capture. This is especially true when the $V_{\infty_A}$ is large enough to preclude a propulsive capture. Aerobraking requires little propellant other than an occasional trim maneuver or periapsis raise maneuver to account for the unexpected changes in the Martian atmospheric density (known to only 35% accuracy). The obvious savings in propellant mass is offset by the increased aerobraking time to circularize the orbit.

Aerocapture is the third capture scheme considered for the orbiter. In this capture scheme, the orbiter dives deep into the Martian atmosphere using the atmospheric drag on the spacecraft to remove a large percentage of the energy from the arrival trajectory. As the orbiter climbs back out of the atmosphere after aerocapture, only a small maneuver is required to circularize the orbit. The savings in propellant are weighed against the additional thermal protection system mass necessary to protect to the spacecraft inside the heat shield and back shell. The greater the magnitude of $V_{\infty_A}$, the higher the heating rates will be and thus the higher the requirement for the mass of the thermal protection system. A lander faces similar circumstances during Martian atmospheric entry.

## F. DECLINATION OF THE ARRIVAL ASYMPTOTE $\left(\delta_{\infty_A}\right)$

The B-plane is defined as the plane at the arrival planet that is perpendicular to the $V_{\infty_A}$ vector. It is useful to define the aim point for the trajectory in terms of a B-plane reference. Figure 2.3 shows a representative B-plane. The orthogonal reference frame is defined by the T axis (intersection of the B-plane with the equatorial plane), the S axis (parallel to incoming asymptote passing through the center of the planet), and the R axis which completes the coordinate frame such that $\hat{R} = \hat{S} \times \hat{T}$. The aim point for the incoming $V_{\infty_A}$ vector is defined by the B vector. The B vector can be described by a magnitude equal to the semi-minor axis of the incoming hyperbola and the B-plane angle $(\Theta)$ which is defined in the B-plane and measured clockwise from the T axis.

The inclination of the initial entry orbit can be determined using the B-plane angle and the $\delta_{\infty_A}$ (Sergeyevsky, 1983, p.20):

$$\cos i_{PEQ} = \cos\Theta_{PEQ} \cos\delta_{\infty_{PEQ}} \qquad (2.11)$$

PEQ = referenced to the planet equator

Equation 2.11 shows that the minimum orbiter inclination possible is equal to the declination of the arrival asymptote. This minimum inclination however, can only be achieved for a B-plane angle of 0°. Realistically, the orbiter in '07 will most likely operate in a sun-synchronous orbit meaning that the chosen inclination must ensure the precession of the ascending node matches the rotation rate of Mars about the sun. This results in an inclination of 92.92°. For example, a $\delta_{\infty_A}$ of $-30°$ would yield a B-plane angle of

$\pm 93.33°$.

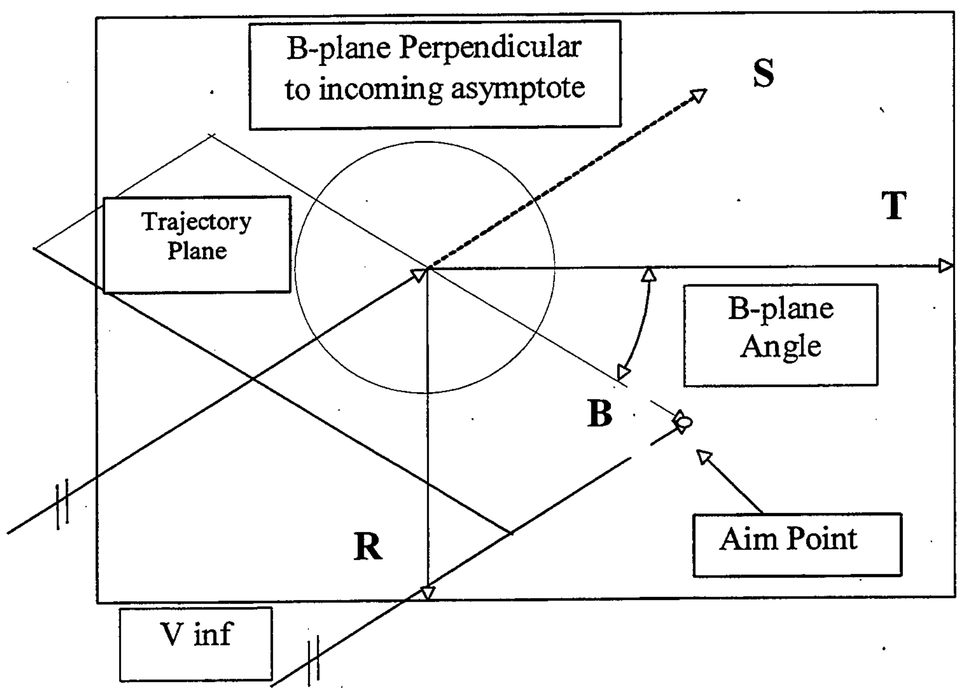


Figure 2.3: Mars B-Plane

The $\delta_{\infty_A}$ also is important to the lander. In the preliminary mission planning stages, the ballistic landing region for the lander can be determined using simple geometry and orbital mechanics. Figure 2.4 depicts the geometry associated with the ballistic lander problem followed by the calculation of the ballistic landing region.

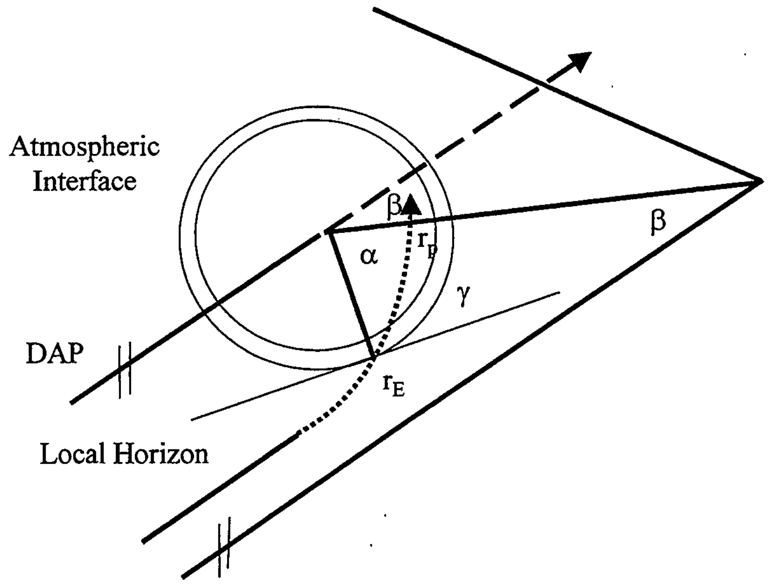


Figure 2.4: Ballistic Landing Region

The angles α and β must be determined to calculate the minimum and maximum latitudes achievable by the lander. The angle of the asymptote, β, is related to the eccentricity by (Brown, 1992, p.27),

$$\boxed{\cos\beta = \frac{1}{e}} \tag{2.12}$$

Beginning with the energy equation, the eccentricity is found in the following manner:

$$E = \frac{\mu}{2a} = \frac{V_\infty^{\;2}}{2} \tag{2.13}$$

Solving for a:

$$a = \frac{\mu}{V_\infty^{\;2}} \tag{2.14}$$

From the trajectory equation, the periapsis radius is,

$$r_p = a(e-1) \tag{2.15}$$

In order to satisfy the desired flight path angle ($\gamma$) at the entry radius ($r_E$), a fictitious periapsis radius, $r_p$, can be determined for the hyperbolic trajectory from the following equation (Sergeyevsky, 1983, p.24):

$$r_p = \left(\frac{\mu_p}{V_\infty^2}\right)\left[-1+\sqrt{1+\left(\frac{V_\infty^2}{\mu_p}\right)r_E\cos^2\gamma_E\left(2+\left(\frac{V_\infty^2}{\mu_p}\right)r_E\right)}\right] \qquad (2.17)$$

For Earth-Mars trajectories, the entry radius at Mars is generally understood to be the atmospheric interface (125 km altitude). Substituting for the semi-major axis and the fictitious periapsis leads to a solution for eccentricity from which $\beta$ can then be found:

$$e = 1+\frac{r_p V_\infty^2}{\mu} \qquad (2.16)$$

The derivation for $\alpha$ is a bit more involved but follows below (Mase, 1998). From Figure 2.4, the incoming velocity vector can be decomposed into a radial and a tangential component as shown below:

$$\bar{v} = \dot{\bar{r}} = \dot{r} + r\dot{\alpha} = V_r + V_\alpha \qquad (2.18)$$

The angular momentum vector can also be determined using the standard definition:

$$\bar{h} = \bar{r}\times\bar{v} = \bar{r}\times(\dot{r}+r\dot{\alpha}) = r^2\dot{\alpha} \qquad (2.19)$$

From the trajectory equation (Vallado, 1996, p. 111),

$$h^2 = \mu r_E(1+e\cos\alpha) \qquad (2.20)$$

Using substitution from Eq. (2.18), (2.19), and (2.20), the velocity components of the arrival velocity vector can be rewritten into the following forms:

$$V\cos\gamma = V_\alpha = \frac{h}{r_E} = \frac{\mu}{h}(1+e\cos\alpha) \qquad (2.21)$$

$$V\sin\gamma = V_r = \frac{\partial r}{\partial t} = \frac{\partial r}{\partial \alpha}\cdot\frac{\partial \alpha}{\partial t} \tag{2.22}$$

The chain rule yields an equation for $V_r$.

$$\frac{\partial \alpha}{\partial t} = \frac{h}{r_E^{\,2}} \quad \text{and} \quad \frac{\partial r}{\partial \alpha} = \frac{\partial}{\partial \alpha}\left(\frac{h^2}{\mu}\cdot\frac{1}{(1+e\cos\alpha)}\right) \tag{2.23}$$

$$V_r = \frac{h}{r_E^{\,2}}\left(\frac{h^2}{\mu}\cdot\frac{e\sin\alpha}{(1+e\cos\alpha)^2}\right) \tag{2.24}$$

Simplifying the expression through substitution puts $V_r$ in a more useful form.

$$(1+e\cos\alpha)^2 = \frac{h^4}{\mu^2 r_E^{\,2}} \tag{2.25}$$

$$V_r = \frac{\mu e\sin\alpha}{h} = V\sin\gamma \tag{2.26}$$

The flight path angle is now

$$\tan\gamma = \frac{V\sin\gamma}{V\cos\gamma} = \frac{\dfrac{\mu e\sin\alpha}{h}}{\dfrac{\mu}{h}(1+e\cos\alpha)} = \frac{e\sin\alpha}{1+e\cos\alpha} \tag{2.27}$$

Rearranging equation (2.21) and (2.26):

$$e\cos\alpha = \frac{hV\cos\gamma}{\mu} - 1 \tag{2.28}$$

$$e\sin\alpha = \frac{hV\sin\gamma}{\mu} \tag{2.29}$$

Eq. (2.28) and (2.29) can now be used to solve for tan α as shown below:

$$\tan\alpha = \frac{e\sin\alpha}{e\cos\alpha} = \frac{\dfrac{hV}{\mu}\sin\gamma}{\dfrac{hV}{\mu}\cos\gamma - 1} \tag{2.30}$$

Remembering that

$$h = r_E V \cos\gamma \tag{2.31}$$

results in the final expression for tan α as a function of the entry radius, the velocity at entry, and the flight path angle.

$$\boxed{\tan\alpha = \dfrac{\dfrac{r_E V^2}{\mu}\sin\gamma\cos\gamma}{\dfrac{r_E V^2}{\mu}\cos^2\gamma - 1}} \tag{2.32}$$

where

$$V^2 = {V_{\infty_A}}^2 + \frac{2\mu}{r_E} \tag{2.33}$$

Therefore, from geometry, the ballistic landing region for the lander is:

$$\text{Landing Region} = \delta_{\infty_A} \pm (\alpha + \beta) \tag{2.34}$$

## G. RIGHT ASCENSION OF ARRIVAL ASYMPTOTE $\left(\alpha_{\infty_a}\right)$

The right ascension of the arrival asymptote is important in determining the Local Mean Solar Time (LMST) for the orbiter during its orbit. In general, the requirement for the LMST of the orbiter is driven by the science objectives for the mission. The orbiter will generally capture into a sun-synchronous 6pm-6am orbit (descending / ascending nodes). This is usually done to accomplish various science objectives that require specific lighting conditions as well as to optimize conditions for solar array charging and communications with the lander. The following is an example of how the LMST is determined for a given arrival asymptote.

| | |
|---|---|
| Departure / Arrival date: | 061126 / 090123 |
| Declination of the Arrival Asymptote, $\delta_{\infty_A}$: | -28.8° |

Right Ascension of the Arrival Asymptote, $\alpha_{\infty_A}$: 289.3°

The Local True Solar Time (LTST) is defined by:

$$LTST = \frac{\left(\alpha_p - \alpha_{TS}\right)24}{360} + 12 \qquad (2.35)$$

$\alpha_p$ = Right Ascension of the descending node

$\alpha_{TS}$ = Right ascension of the true sun

The Local Mean Solar Time is defined by:

$$LMST = \Delta\alpha = \alpha_p - \alpha_{FMS} \qquad (2.36)$$

$$\alpha_{FMS} = 121.783 + 0.524041(\Delta T) \qquad (2.37)$$

$\Delta T$ = Number of integer days past 08/01/93

Using the interactive Fortran software routine QUICK from JPL enabled the determination of the right ascension of the true sun at the specified arrival date. In this case, $\alpha_{TS} = 195.2^\circ$. From this, the LTST can be calculated as shown below:

$$LTST = \frac{(289.3 - 195.2)24}{260} + 12 = 18h\ 16m\ 23s$$

To find Local Mean Solar Time, $\alpha_{FMS}$ was calculated to be 205.1°.

$$\Delta\alpha = \alpha_p - \alpha_{FMS} \rightarrow \Delta\alpha = 289.3^\circ - 205.1^\circ = 84.2^\circ$$

The positive value for $\Delta\alpha$ indicates the equivalent time after noon. Therefore,

$$LMST = 17h\ 36m\ 0s$$

assuming that the burn at periapsis is tangential and not out of plane of the arrival hyperbolic trajectory.

# III. JPL MISSION ANALYSIS SOFTWARE (MAS)

## A. INTRODUCTION

Preliminary trajectory analysis can be accomplished using many of the FORTRAN compatible subroutines available as part of the Multi-Mission Analysis Software Library (MASL) at JPL. The output from these subroutines provides data in a format conducive to preliminary analysis of interplanetary mission design. The subroutines provide the user with a useful and sometimes graphical representation that more clearly summarizes the data. For this analysis, five different JPL subroutines were used (Contour, QUICK, MIDAS, CATO, Kplot). The operation and application of each program will be described briefly in the following sections. In addition, the utility of each program in the mission design process is demonstrated through several examples.

The subroutines currently run in a Unix based operating system resident on the JPL network. Access to program use is limited to remote log-in from either a Unix workstation or a PC using Telnet. File transfer can be accomplished with a File Transfer Protocol (FTP) application.

## B. CONTOUR

### 1. Background

This subroutine enables the mission designer to graphically depict trajectory analysis parameters in launch date vs. arrival date space. For a given set of launch dates and arrival dates, these contours will provide data for all possible ballistic transfer trajectories between two bodies of interest. Before using Contour, the nominal launch and arrival dates for a mission should be defined. As previously discussed, only those periods surrounding the minimum energy trajectories are of interest. To better define where those trajectories occur, it is necessary to discuss the idea of a Hohmann transfer.

The Hohmann transfer represents the minimum energy trajectory between Earth and Mars. Recall that the frequency of occurrence for the Earth-Mars geometry in order to achieve a Hohmann transfer is equal to the synodic period (approximately 778 days) between the two bodies (Chobotov, 1991, p.300). For a Hohmann transfer to be feasible, an inner departure planet must lead an outer arrival planet by $\beta_{12}$ as defined by (Prussing, 1993, p. 116):

$$\beta_{12} = \pi\left[1-\left[\frac{1+R}{2R}\right]^{\frac{3}{2}}\right] \tag{3.1}$$

$$\text{where} \quad R = \frac{r_2}{r_1}$$

$r_1$ = Inner planet distance from the sun (in Astronomical Units)

$r_2$ = Outer planet distance from the sun (in Astronomical Units)

Assuming the orbits of Earth and Mars are near circular, the time for a Hohmann transfer can be determined using the standard equation (Prussing, 1993, p.104),

$$t_H = \pi\sqrt{\frac{a^3}{\mu}} \tag{3.2}$$

a = Semi-major axis of transfer ellipse

$t_H$ = Half the period of the ellipse

Determining this Mars lead angle and the time of occurrence can be accomplished using an astrodynamic analytical program that can propagate the orbits of Earth and Mars until the Hohmann transfer geometry is obtained. Once a rough date for the lead angle is determined, $t_H$ can be added to it to determine the nominal arrival date. Although this method will clearly provide a nominal launch and arrival date, JPL has already determined periods of minimum energy transfers for the 2002 – 2020 timeframe (Matousek and Sergeyevsky, 1998, p.4-5).

For preliminary mission design, a C3 of 10 $km^2/s^2$ is a good rule of thumb for Earth to Mars missions. Advances in lightweight or multi-use structural materials, component miniaturization, or improvements in propulsion system technology could enable launches of C3 > 10 $km^2/s^2$ as spacecraft mass decreases. Launch energies greater than 10 $km^2/s^2$ are certainly achievable today with larger launch vehicles, but again, fiscal constraints do not always provide for this alternative. Coupled with the pressure to conduct as much science as possible on each mission, the odds of decreasing the spacecraft mass even with technological advances are slight. In any event, the JPL analysis included all possible ballistic trajectories with launch energies of 25 $km^2/s^2$ or less to account for future uncertainties. Table 3.1 shows the applicable JPL analysis data for the nominal 2007 Earth-Mars opportunity (Matousek and Sergeyevsky, 1998).

Table 3.1: Earth-Mars Ballistic Trajectories for 2007

| Type | C3 (km^2/s^2) | DLA (deg) | V inf at Arr (km/s) | DAP (deg) | Launch Date (mm/dd/yy) | Arrival Date (mm/dd/yy) | Flight Time (days) |
|---|---|---|---|---|---|---|---|
| 1 | 18.8 | 49.3 | 3.9 | -25.6 | 9/23/07 | 4/19/08 | 209 |
| 2 | 12.7 | 17.9 | 2.8 | 14.3 | 9/22/07 | 9/26/08 | 370 |
| 3+ | 9.0 | -24.3 | 5.5 | -2.2 | 1/11/07 | 12/18/08 | 707 |
| 4- | 8.7 | -0.1 | 6.0 | -28.1 | 2/15/07 | 6/11/09 | 847 |

The minimum energy trajectories determined in the JPL study became the starting point for this analysis.

Interplanetary trajectories are classified by types in terms of the amount of true anomaly swept out by the heliocentric trajectory during the transfer. Table 3.2 shows the classifications of trajectory types. For the type 3 and 4 trajectories, there are two

Table 3.2: Trajectory Types

| Trajectory Type | True Anomaly |
|---|---|
| 1 | $0 \leq \Theta < 180$ |
| 2 | $180 < \Theta \leq 360$ |
| 3- / 3+ | $360 \leq \Theta < 540$ |
| 4- / 4+ | $540 < \Theta \leq 720$ |

solutions to the Lambert problem. The shortest flight time solution for each of these types is symbolized by a minus sign (e.g., 3- or 4-) while the longer flight time trajectories are designated with a plus sign (e.g., 3+ or 4+).

Once the minimum energy transfer dates are defined for each trajectory type, Contour can be used to provide a summary of launch energies for all trajectories over the entire period of selected launch and arrival dates. The Contour code enables the user to distinguish the trajectory type by simply assigning a value to the variable **rn** that is equal to the number of complete revolutions about the central body made by the trajectory. For example, a type 1 or type 2 would require **rn** be set to zero indicating that the trajectory transfer arc does not exceed 360°. Figure 3.1 shows an example of the different C3 contours for all possible Earth-Mars Type 1-2 trajectories for the 2007 opportunity. Note that there are two distinct regions on the contour separated by what is termed the "ridge." For this example, the lower right portion of the contour constitutes the Type 1 trajectory while the upper left region represents the Type 2 trajectory contours. Table 3.2 indicates that the point of separation between the Type 1 and Type 2 occurs at the 180° true anomaly point. The ridge for this contour is that point. It is also important to note

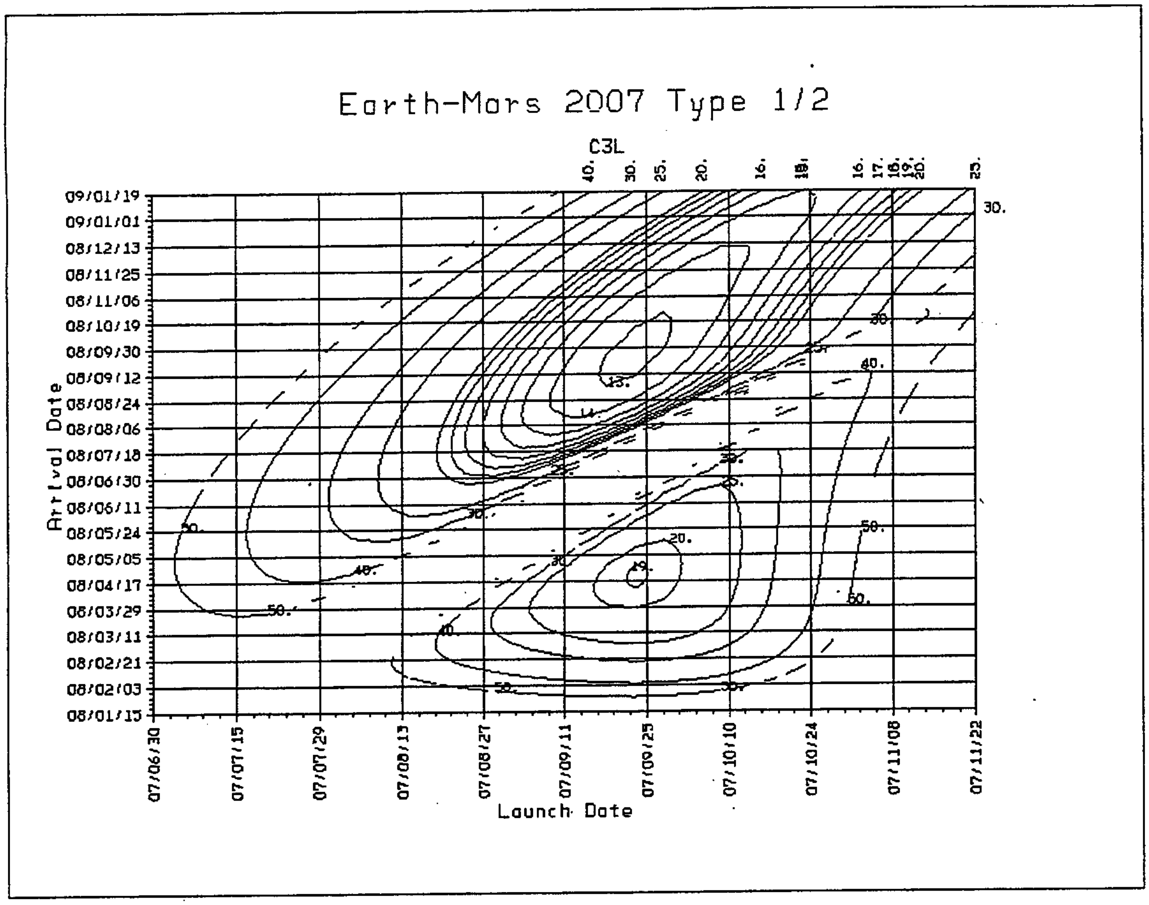


Figure 3.1: Earth-Mars 2007 Type 1 / 2 Trajectory (C3)

another significant difference between a Type 1 and Type 2 trajectory. Type 1 trajectories sweep out a smaller amount of true anomaly during transfer than the type 2 trajectories so the flight time is generally shorter. This can be verified by simply looking at the launch and arrival dates for the Type 1 and Type 2 trajectories using the contour plot.

### 2. Contour Inputs

Using a RDNAM type input file similar to FORTRAN NAMELIST files, the trajectory analysis parameter of choice can be plotted against the departure and arrival period as shown in the previous example. An added feature allows more than one parameter to be mapped over the same launch and arrival periods to help the mission designer note parameter dependencies as well as trends. Contour supports the six primary trajectory analysis parameters discussed in Chapter II while other parameters supported by contour are shown in Table 3.3 below (Schlaifer):

Table 3.3: Contour Parameters

| **Contour Type** | **Description** |
|---|---|
| PHL / PHA | Launch or Arrival Solar Phase Angle |
| RCL / RCA | Launch or Arrival Communications Range to Earth |
| O(X)L / O(X)A* | Launch or Arrival Heliocentric Orbital Elements |
| ZAL / ZAA | Angle between Departure $V_{\infty_A}$ Vector and Sun-Earth Vector / Angle between the Arrival $V_{\infty_A}$ vector and Arrival Planet-Sun Vector |

* X equals 1-6 depending on the orbital element desired

An example of the declination of the departure asymptote plotted over the previous contour of launch energy is shown in Figure 3.2.

The contour subroutine is initiated with the following command line,

**contour <filename.inp**

which allows contour to read data in from a previously generated file. Upon run completion, the **plot-ps –k** command creates a postscript output file named plot-ps-file. To view the file in a PC environment, it can be imported via FTP. Prior to this, however, the plot-ps-file should be renamed using a .ps extension so that the viewing program will recognize the file type. A suitable postscript viewer (Ghostscript in this case) can then be used to view the output file. If the output is acceptable, the Device Independent Bitmap created in Ghostscript can then be imported into any of the Microsoft applications using the "Paste Special – Device Independent Bitmap" command.

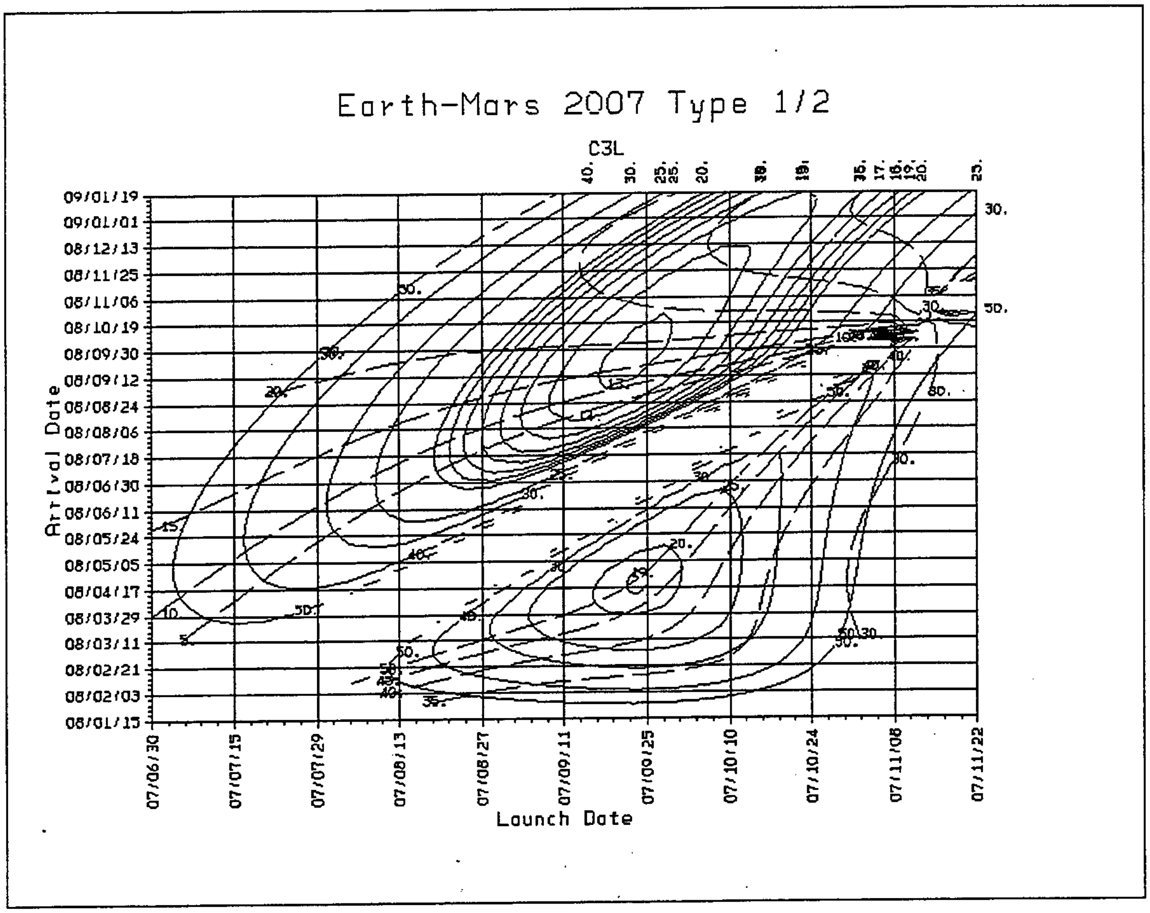


Figure 3.2: Earth-Mars 2007 Type 1 / 2 Trajectory (C3, DLA)

## C. QUICK

### 1. Description

This subroutine provides the capability to perform scalar, vector, and matrix arithmetic, propagate conic orbits, and determine planetary and satellite coordinates within a Fortran-like environment. The subroutine is interactive thus allowing the user to execute commands without first compiling a program. QUICK is initiated by simply typing the command **quick** at the command prompt. QUICK has a number of commands that may be used to set background conditions or conditionally control the execution of other commands (Schlaifer, p16). QUICK also contains many built-in math and astrodynamic functions that are referred to by the function name followed by appropriate arguments within parenthesis. The format is similar to that used in FORTRAN, however, slight modifications exist. The user's guide should be consulted for formats for expressions, operators, commands, and functions.

### 2. Applications

QUICK can be a valuable tool in the initial mission planning stages. As an example, QUICK was used to verify the nominal dates of the JPL ballistic trajectories calculated for the 2007 opportunity as shown in Table 3.1. From Table 3.1, the nominal Earth to Mars 2007 Type 1 trajectory had a minimum C3 of 18.8 $km^2/s^2$. Figure 3.1 depicts the Type 1 contours for C3 below the "ridge". Upon closer inspection, an estimate for the minimum C3 launch/arrival date can be determined by finding the lowest C3 contour on the plot and gauging the center of that contour. For the type 1 trajectory, the launch date is estimated to be 09/23/07 (mm/dd/yy) while the corresponding arrival date is 04/19/08. The following QUICK code in Figure 3.3 demonstrates the procedure that can be used to verify the results in Table 3.1. Note that the launch energy value obtained using QUICK does in fact equal the minimum C3 provided in the JPL analysis.

Another common application of QUICK is orbit "fitting" given a set of input parameters. By defining the central body, the position of the departure planet at launch, the position of the target planet at arrival, and the time of flight based on those dates, QUICK can calculate an optimized orbit. The format of the output state can be specified using an IORB Flag reference as summarized in the user's manual (Schlaifer, p.47-48). The code in Figure 3.4 fits an orbit for the Earth to Mars 2007 Type 1 trajectory.

```
>ip=(3,4)                                          ; Earth=3, Mars=4
IP     = 3.0000000000000    4.0000000000000
>datin=(070923,080419)                             ; Launch/Arrival Dates
DATIN  = 70923.000000000   80419.000000000
>jda=c3min(ip,datin,0,1)                           ; Minimize Launch C3

Input central body number (ISIP): 0                ; Select Central Body
Sun
EPHOPN opened /usr/local/ephem/de403s.bsp          ; Call to Current Ephemeris File

JDA    = 2454366.8677065   2454576.6403913         ; Output Dates for Minimum C3
>date(jda)                                         ; Output Dates for Minimum C3
DATE   = 70923.084929845   80420.032209809
>Sv0=orbvel(0)                                     ; Calculate Initial Orbit Velocity
SV0    = 0.37750194853685    33.479208637714    1.9333340994602
>ev0=plvel(jda(1),3)                               ; Calculate Initial Earth Velocity
EV0    = -0.40896797134736   29.676633804569    -538.2431654262E-06
>vinf=(absv(sv0-ev0))                              ; V∞ – departureV∞
VINF   = 4.3379687105974
>c3=vinf**2                                        ; Launch Energy = V∞^2
C3     = 18.817972534122
```

Figure 3.3: Verification of JPL Ballistic Trajectory Analysis using QUICK

It should be noted that QUICK treats celestial bodies as point masses and therefore, the ballistic trajectories calculated using orbfit are measured from the center of Earth to the center of Mars.

```
>ldate=070923                                       ; Launch Date
 LDATE   =  70923.000000000
>adate=080419                                       ; Arrival Date
 ADATE   =  80419.000000000
>jdldate=date(ldate)                                ; Launch Julian Date
 JDLDATE =  2454366.5000000
>jdadate=date(adate)                                ; Arrival Julian Date
 JDADATE =  2454575.5000000
>tfl=jdadate-jdldate                                ; Total Flight Time (days)
TFL      =  209.00000000000
>time=tfl*86400                                     ; Total Flight Time (sec)
 TIME    =  18057600.000000
>cplann(jdldate,0)                                  ; Select Central Body
CPLANN  =   Sun
>earthpos=bodpos(jdldate,3,0)                       ; Earth Position at Launch
EPHOPN opened /usr/local/ephem/de403s.bsp           ; Current Ephemeris File Opened
EARTHPOS    =  150134078.39271   -1327336.8734704   -117.17288617848
>Marspos=bodpos(jdadate,4,0)                        ; Mars Position at Arrival
MARSPOS     = -205224447.45181   140414329.69073    7982035.9869378
>orbit=orbfit(earthpos,marspos,time,0,'')           ; QUICK calculated orbit
ORBIT   =  COMPLETE
>orbprt(81,0)                                       ; Print Classical Orbital Elements
                                                    ; IORB flag =81
Classical Orbital Elements
Sun centered, Earth ecliptic and equinox of J2000
Semi-major Axis         206590326.516398         km
Eccentricity            0.273330400252939
Inclination             3.29611242987477         degree
Node Angle             -0.505762940264790        degree
Arg of Periapsis        -1.86379853419440        degree
Mean Anomaly            1.02278770831965         degree
ORBPRT  =              206590326.51640   0.27333040025294   3.2961124298748
                       -0.50576294026479 -1.8637985341944   1.02287877083196
```

Figure 3.4: Earth-Mars 2007 Type 4- Orbit Fit using QUICK

## D. MISSION DESIGN AND ANALYSIS SOFTWARE (MIDAS)

### 1. Description

The MIssion Design and Analysis Software (MIDAS) program from JPL is a patched conic interplanetary trajectory optimization program that is defaulted to minimize the total weighted mission $\Delta V$. Although not a necessary option for Earth-Mars trajectory analysis, MIDAS does provide the capability to add or delete deep space maneuvers and gravity assist maneuvers as necessary in the optimization process. Also, The concept of patched conics was addressed in Chapter II. Although MIDAS is a patched conic optimization program, the method utilized in this program does not use celestial spheres of influence to distinguish the different conics in the overall "patched" conic. Instead, when MIDAS optimizes for the minimum total weighted mission $\Delta V$, the optimal trajectory is determined by summing the magnitudes of the $\Delta V$ maneuvers required to 1) inject onto the interplanetary trajectory from a nominal parking orbit about Earth and 2) capture into a final circular orbit of a specified altitude about Mars. Therefore, the conics that are "patched" together simply consist of the initial parking orbit about Earth, the heliocentric elliptical transfer orbit, and the final circular orbit about Mars (atmospheric interface altitude).

MIDAS is designed to be a general-purpose ballistic trajectory optimization tool that can accommodate various types of trajectories. The program employs an unconstrained gradient search algorithm requiring partial derivatives to optimize the trajectory. With a gradient search such as this, it is not uncommon for the trajectory to converge to a local minimum solution that is not the desired solution. In such cases, it is necessary to update the initial estimates for the independent variables and rerun the program. (Sauer, 1991, p.4)

MIDAS requires a RDNAM type input file to execute. In addition to a basic set of input requirements, additional capability exists to constrain particular mission parameters (e.g., declination of the departure asymptote, parking orbit inclination, launch energy, etc.) thus allowing more user control over the optimization process. The user's guide contains all of the possible input variables. MIDAS also provides the user with the ability to perform parameter studies on any of the independent variables. For example, guide contains all of the possible input variables. MIDAS also provides the user with the ability to perform parameter studies on any of the independent variables. For example, MIDAS could be given a range of launch and arrival dates. Given other input parameters, the program could then search those dates for the optimal trajectory based on total weighted mission $\Delta V$.

For this analysis, MIDAS was used as a first look at possible trajectories. However, the Earth-Mars trajectories are fairly simplistic when comparing them to other missions such as the Pluto Express, so many of the capabilities available in MIDAS were not utilized. MIDAS was also restricted to purely ballistic trajectories to reduce any additional complexity that would be required by the Guidance, Navigation and Control Subsystem.

**2. Applications**

MIDAS can create a user specified list in the input file that identifies variables to be included in a tabular output file (**.lst** file). The trajectory analysis parameters addressed in Chapter II were chosen as the output parameters for the MIDAS runs. This data can then be exported (manually) to a spreadsheet program like Excel and graphed as required. To execute MIDAS, the following command line is necessary:

**> midas filename.inp -seo**

The switches in the execution command enable selection of various output file formats. The user's guide provides a comprehensive list of these switches and their associated formats. For this analysis, three switches were utilized. The **–s** switch represents a detailed trajectory output file (**.out** files) while the **–e** represents a condensed version of the output file (**.sav** files) with some additional data for perihelion and aphelion. These output files contain the state vectors and orbital elements for all of the calculated trajectories presented in various formats and referenced to several different coordinate frames. This data can be utilized as input data to more sophisticated trajectory optimization programs. The **–o** switch however, prohibits MIDAS from including trajectory updates (deep space maneuvers) for the reason discussed earlier. This results in a purely ballistic trajectory.

An example input file for an Earth-Mars 2007 Type 4- trajectory is provided in Figure 3.5. The input file references the Earth parking orbit altitude as well as the altitude for the Mars atmospheric interface. The launch date was fixed while the arrival date was allowed to float between a range of values, thus allowing MIDAS to find an optimal arrival date. Twenty-one output trajectories were calculated for this

```
head='2007 Earth-Mars Type 4- trajectory'            ; 061126
shota='earth'                                        ; departure planet
bulsi='mars'                                         ; arrival planet
jdl=2006,11,26                                       ; launch date
jdate=0                                              ; days from launch date (fixed)
adate=775                                            ; days from launch date for arrival
alt=185,125                                          ; Earth parking orbit, Mars/ATM I/F
rn=-1                                                ; type 3/4- orbit
choice=1
rp=1.0
varyn='adate'                                        ; parameter study for arrival dates
minyn=780,1,800                                      ; range of arrival dates in study
vlist='+jdate','+adate','c3','dla','rla','vhp','dap','rap'   ; creates .lst file
$
```

Figure 3.5: Sample MIDAS Input File for the Type 4- Trajectory

example, but for demonstration purposes, only an excerpt from the corresponding **.sav** file is shown in Figures 3.6. The MIDAS **.out** files can be accessed via **roemer.jpl.nasa.gov** under path the **~zike/midas_examples/inpfiles**.

```
nt= 16.01  iter= 1   kgo= 0  flags  0  0  0  nm= 0  nmt= 0  ndl=  2   nda=  2
nv(1)= 15      adate= 790.000
veq=  6.0011    grad=  .000000     dvt=  6.0011    dvmt=   .0000   dvpl= 2.3425
tend= 790.000     fty=  2.1629   hca= -1.61

jdate=     .0   2006 11 26  0  0      dvl=  3.6586      c3=  9.6754      dla= 21.
rla= 193.860

re=  6563.1   trp=     16.8    2006 12 12 20  2
rp=   .9786     xmp=  .1344     ymp=   .9692     zmp=   .0128
tra=   261.6   2007  8 14 14 22       ra=  1.4526     xma= -.1995     yma= -1.4387
zma=  -.0191   trp=   506.4    2008  4 15  8 42   rp=   .9786     xmp=  .1344
ymp=   .9692     zmp=   .0128   tra=   751.1    2008 12 16  3  2
ra=  1.4526     xma= -.1995     yma= -1.4387      zma=  -.019

adate=  790.0    2009  1 24  0  0      dva=  2.3425     vhp=  3.0847
dap=-29.190    rap= 289.660      rp=  3397.2
```

Figure 3.6: Sample MIDAS Output File (.sav) for the Type 4- Trajectory

The variables shown above in the **.sav** file are clearly defined in the MIDAS user's guide. The trajectory analysis parameters generated in the **.lst** file are shown in Table 3.4. The nominal launch/arrival date pair in bold text represents the minimum launch energy trajectory for a 11/26/06 launch date. The same process for other launch dates is followed ultimately resulting in one optimal trajectory for each launch date. Although the algorithm employed in MIDAS calculates a patched conic trajectory, MIDAS provides a quick trajectory analysis for a given mission. The more detailed data that corresponds to the optimal trajectory for each launch date can be extracted from the **.sav** and **.out** files and used as inputs to other programs.

Table 3.4: Sample MIDAS **.lst** File for the Type 4- Trajectory

| Launch | Arrival | C3 | DLA | RLA | VHP | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 1/14/09 | 9.819 | 23.5 | 193.8 | 3.122 | -29.0 | 295.9 |
| 11/26/06 | 1/15/09 | 9.790 | 23.4 | 193.8 | 3.112 | -29.0 | 295.2 |
| 11/26/06 | 1/16/09 | 9.765 | 23.2 | 193.7 | 3.104 | -29.1 | 294.6 |
| 11/26/06 | 1/17/09 | 9.743 | 23.1 | 193.7 | 3.097 | -29.1 | 293.9 |
| 11/26/06 | 1/18/09 | 9.725 | 23.0 | 193.7 | 3.092 | -29.1 | 293.3 |
| 11/26/06 | 1/19/09 | 9.709 | 22.8 | 193.7 | 3.088 | -29.1 | 292.7 |
| 11/26/06 | 1/20/09 | 9.697 | 22.7 | 193.7 | 3.085 | -29.2 | 292.1 |
| 11/26/06 | 1/21/09 | 9.687 | 22.5 | 193.7 | 3.083 | -29.2 | 291.4 |
| 11/26/06 | 1/22/09 | 9.680 | 22.3 | 193.8 | 3.082 | -29.2 | 290.8 |
| 11/26/06 | 1/23/09 | 9.676 | 22.1 | 193.8 | 3.083 | -29.2 | 290.2 |
| **11/26/06** | **1/24/09** | **9.675** | **22.0** | **193.9** | **3.085** | **-29.2** | **289.7** |
| 11/26/06 | 1/25/09 | 9.677 | 21.8 | 193.9 | 3.088 | -29.2 | 289.1 |
| 11/26/06 | 1/26/09 | 9.682 | 21.6 | 194.0 | 3.091 | -29.2 | 288.5 |
| 11/26/06 | 1/27/09 | 9.689 | 21.4 | 194.1 | 3.096 | -29.2 | 288.0 |
| 11/26/06 | 1/28/09 | 9.699 | 21.2 | 194.2 | 3.102 | -29.2 | 287.4 |
| 11/26/06 | 1/29/09 | 9.711 | 21.0 | 194.3 | 3.109 | -29.2 | 286.9 |
| 11/26/06 | 1/30/09 | 9.726 | 20.8 | 194.4 | 3.117 | -29.2 | 286.4 |
| 11/26/06 | 1/31/09 | 9.744 | 20.6 | 194.5 | 3.126 | -29.2 | 285.9 |
| 11/26/06 | 2/1/09 | 9.765 | 20.4 | 194.7 | 3.136 | -29.1 | 285.4 |
| 11/26/06 | 2/2/09 | 9.788 | 20.1 | 194.8 | 3.147 | -29.1 | 284.9 |

In addition, Excel can be used to graphically present this preliminary trajectory analysis in a useful format that rapidly builds intuition for the mission designer about the analysis parameters. For example, consider a launch period for a 2007 Earth-Mars mission that extends from 02/05/07 to 02/24/07. The trajectory chosen in Figure 3.7 is the type 4-. For this launch period, the nominal launch date occurs on 02/15/07. As the launch date varies from this nominal date, the C3 increases. This almost "horseshoe" shape C3 vs. launch date graph is typical for all trajectories reinforcing the fact that there is indeed an optimal trajectory.

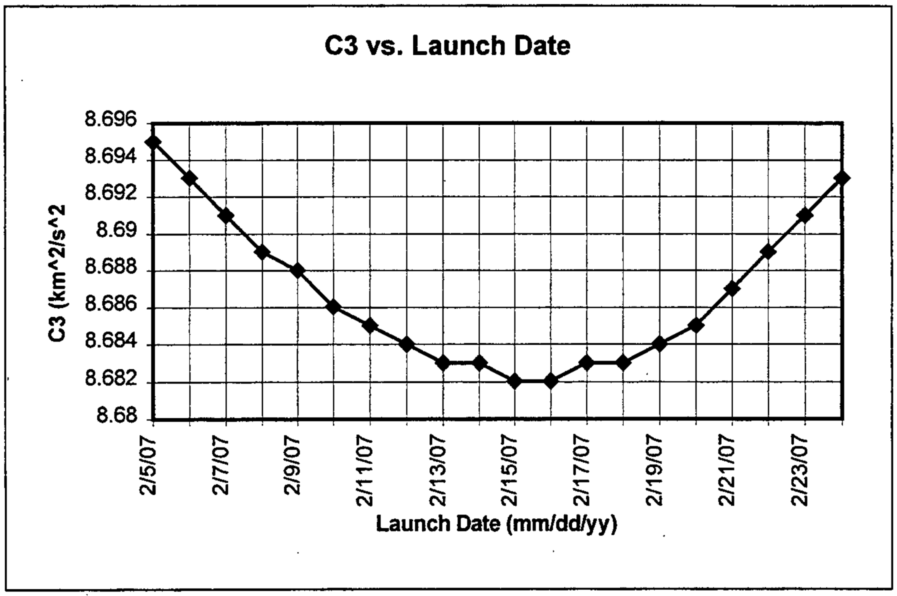


Figure 3.7: Earth-Mars 2007 Type 4- Trajectory (C3 vs. Launch Date)

## E. COMPUTER ALGORITHM FOR TRAJECTORY OPTIMIZATION (CATO)

### 1. Description

CATO is designed to minimize the total deterministic $\Delta V$ required for an interplanetary trajectory subject to various constraints. Preliminary trajectory design such as that achieved with MIDAS serves as input to this program. CATO determines an optimal trajectory consistent with the user-defined set of constraints while meeting certain arrival conditions beginning with a specified state vector (obtained from MIDAS .out files). Trajectory modeling is based on numerical integration of the equations of motion for a point-mass spacecraft. This spacecraft is subject to gravitational accelerations that include the inverse-square acceleration due to the central body, the point-mass gravitational accelerations due to any combination of the sun, planets, and satellites, plus acceleration due to the oblateness of planetary central bodies. Trajectory constraints may include any quantity in the IORB flag reference included on p. E-5 of the

CATO user's manual. The constraints chosen may be either equality or inequality constraints and will be closely related to the mission requirements and science objectives. (Bright, 1996, p.3-1)

CATO employs a parameter optimization algorithm based on a series of linearizations of the highly non-linear N-body problem. The algorithm changes the independent variables including a series of states along the trajectory during successive iterations to reduce the cost function (total $\Delta V$). A CATO trajectory is broken up into a sequence of user-defined trajectory legs with each leg consisting of a control point defined by a seven variable state vector. The first component of the state vector references the epoch while the other six represent the user-selected parameterization of the state at that epoch. CATO allows for the optimization of any of the six parameterized variables within the control state vector. The boundaries between successive legs of the trajectory are defined by breakpoints that only reference the epoch. Breakpoints and control points then alternate along the trajectory with a breakpoint defining both the beginning and the end of the trajectory.

Using the initial state, CATO then begins the optimization process. Each trajectory leg is generated separately and independently. As CATO optimizes the trajectory by connecting the legs together, the final trajectory will generally display some discontinuities between legs in both position and velocity. However, these discontinuities are usually small enough to neglect. All of the CATO trajectories for this analysis converged to solutions having positional discontinuities less than 25 m and velocity discontinuities almost zero. In linearizing this problem, CATO goes through a process called re-weighting the cost function in a loop-within-a-loop structure. Once the re-weighting process converges, the resulting trajectory solution is used to determine a new linearization of the non-linear problem. The re-weighting process is done again on this new linearization until there is only a negligible change from one iteration to the next. The cost function (total $\Delta V$) at this point, has been minimized.

### 2. Applications

The command line to execute CATO is simply: **> cato**

The developmental ephemeris file DE403s.bsp is called to ensure that CATO has access to the most current celestial ephemeris. Input files created using a text editor can be loaded into CATO with the following command: **> file filename**

CATO has an interactive mode that enables data input from the keyboard. This can be initiated by either including the statement INTERACTIVE in the last line of the input file or by simply typing **> int** at the command prompt. Using the interactive mode to run CATO is particularly useful when only a few of the breakpoint and control point parameters require modification.

An example CATO input file is shown in Figure 3.8 for an Earth-Mars 2007 Type 4- trajectory. It is important to set the IORB flag for the control point state vectors such that the output variables are meaningful to the trajectory. In this case, the departure variables corresponding to an IORB flag of 110223 represent hyperbolic quantities that are referenced to an Earth Equator and Equinox of Epoch coordinate frame. The parameters optimized at departure are C3, DLA, and RLA while the epoch, periapsis radius, B-plane angle, and time of periapsis passage are fixed. At arrival, an IORB flag of 23 presents the arrival state vector in an Equatorial and Equinox of Epoch coordinate

```
ODECONTROL
&ode GravBods(1:5)=3,103,4,10,0
/
SPACECRAFT
&spa area=        8
  massDef=        650
  GR=             1.5
/
+BP "Earth Departure"
&bp jdMin=        2454065.0
  jd=             2454065.1
  jdMax=          2454065.2
  djd=            0.001
  fixedJd=        .true.
  cBody=          3
/
+CP "Departure Hyperbola"
&cp bodyU=3, iorbu=110223,
  Umin(0:6)=          2454065.4,   6563.0,   -15,   -600,     0,   -28.5,     0
  U(0:6)=             2454065.5,   6563.1,     0,      0,  9.75,    21.2, 194.0
  Umax(0:6)=          2454065.6,   6563.2,    15,    600,  28.5,      15,   360
  DuFinite(0:6)=           1e-6,      0.1,  1e-4,    2.0,  1e-4,    1e-4,  1e-4
  FixedU(0:6)=                T,        T,     T,      T,     F,       F,     F
/
+BP "Earth-Mars Mvr"
&bp jdMin=        2454460.3
  jd=             2454460.4
  jdMax=          2454460.5
  djd=            0.001
  fixedJd=        .true.
  cBody=          0
/
+CP "Arrival Hyperbola"
&cp bodyU=4, iorbu=23,
  Umin(0:6)=          2454855.1,      249,    -95,  -600,     2,    -90,     0
  U(0:6)=             2454855.2,      250, -92.92,     0,  3.09,  -28.8, 289.3
  Umax(0:6)=          2454855.3,      251,    -90,   600,  6.92,     90,   360
  DuFinite(0:6)=           1e-6,      0.1,   1e-4,   2.0,  1e-4,   1e-4,  1e-4
  FixedU(0:6)=                T,        T,      T,     T,     F,      F,     F
/
+BP "Mars Arrival"
&bp jdMin=        2454855.4
  jd=             2454855.5
  jdMax=          2454855.6
  djd=            0.001
  fixedJd=        .true.
  cBody=          4
/
INTERACTIVE
```

Figure 3.8: CATO Input File for Earth-Mars 2007 Type 4- Trajectory

frame with respect to Mars. The optimized parameters at arrival are $V_{\infty_A}$, DAP, and RAP while the epoch, periapsis radius, B-plane angle, and time of periapsis passage are fixed. Note that the six optimized parameters in this example are the same trajectory analysis parameters discussed in Chapter II.

Consideration should also be given to constraint parameters in the selection of an IORB flag. In the Earth-Mars trajectories, it is important to be able to constrain certain variables. For example, constraining the periapsis altitude at departure and arrival enables the mission designer to establish the parking orbit altitude as well as the entry altitude at Mars. Also, choosing the B-plane angle at Mars is particularly important in the targeting process for the orbiter and the lander. Figure 3.9 depicts the CATO output of the optimization process. This particular excerpt reflects the last iteration of the optimization. Reyes provides a more thorough description of each line in Figure 3.9. Regardless, some of the data warrants discussion here. For an initial guess (input file or keyboard entered), CATO will minimize the trajectory using the re-weighting process described earlier. A proposed step value like the one in the output will be added to the initial guess and CATO will re-weight the trajectory a number of times before converging to a solution. That solution then becomes the new guess and the process repeats again. Due to the complexity of the program, the user maintains control over the linearization process. Each linearization must be initiated by the user with the **>con** command. Determination for convergence rests with the user.

```
Command > con

Breakpoint States (User-Specified Coordinates):
2454065.1000+  -1.68781039261E+04 -1.35578463272E+05  8.17570184199E+04  8.4232
5874300E-01  3.11549525819E+00 -2.07428442601E+00

2454460.1000-   1.69898959887E+08 -6.97732283752E+07 -7.41526378890E+06  5.1442
0802787E+00  2.62219932317E+01  3.00157563529E-01
2454460.1000+   1.69898959884E+08 -6.97732283798E+07 -7.41526379009E+06  5.1442
0802887E+00  2.62219932319E+01  3.00157563808E-01

2454855.5000-  -3.86610314468E+04  9.62394419500E+03 -8.06610745838E+04 -1.1829
2413179E+00  4.11606408822E-01 -2.98382081960E+00

Breakpoint Constraint Discontinuities:
      5.74367176101E-03

 *** MINIMIZING ***

=======================
Re-weighted    1 times
=======================

 Linearized dKs  1.90276709046E-18

 Linearized dCs  3.85405808491E-25
 SUM          3.85405808491E-25

Previous Vars:  9.75487623555E+00  2.13001113046E+01  1.93887598919E+02  3.0865
9215593E+00 -2.88434399546E+01  2.89330899641E+02

Proposed Step:  3.44889566656E-10  1.56286876534E-08 -3.67904152779E-09  1.2827
0022238E-10 -6.38891673841E-09  9.82205443536E-10

Proposed Vars:  9.75487623590E+00  2.13001113203E+01  1.93887598916E+02  3.0865
9215605E+00 -2.88434399610E+01  2.89330899642E+02
```

Figure 3.9: Example CATO Output for Earth-Mars 2007 Type 4- Trajectory

The following criteria should be used to determine whether convergence has occurred:

1) Breakpoint constraint discontinuity (positional error between legs) should be on the order of meters;
2) Linearized dK's indicate how well CATO has linearized the problem (smaller is better);

3) Linearized dC's represent the $\Delta V$ required at each breakpoint (very small for ballistic trajectories);

4) The size of the proposed step in relation to the previous values of the variables indicates how much the trajectory is changing on each iteration;

5) The number of times the program re-weights the trajectory also indicates convergence. If CATO is only re-weighting the solution once, then it is a safe assumption that the solution is close to convergence.

Using the criteria above, it is obvious from the output file that this trajectory has converged. Although the breakpoint discontinuity is approximately 6 meters, this is deemed acceptable at this stage of mission design.

Once the solution has converged, the **>sho** command can be used to view the final state vectors for the trajectory. Figure 3.10 shows the CATO output for the final trajectory in state vector format. The values from this output can then be manually imported into Excel for graphing.

```
Command > sho

================================================================================
BREAKPOINT  : Earth Departure                 Fid JD:  2454065.1000000
                Lower                Value                    Upper        |    { NOMVR}
   Julian Date  2454065.000000       2454065.1000000          2454065.2000000   day

================================================================================
CONTROL POINT: Departure Hyperbola            Fid JD:  2454065.5000000
PARAMETER SET: Hyperbolic Asymptote   equator and equinox of epoch     WRT:  Earth
                            Lower                   Value                 Upper
Julian Date                 2454065.4000000         2454065.5000000       2454065.6000000  day
Periapsis radius            6563.0000000000         6563.1000000000       6563.2000000000  km
B Plane Angle               -15.000000000000        0.0000000000000       15.000000000000  deg
Time wrt Periapsis          -600.00000000000        0.0000000000000       600.00000000000   s
* C3 (energy)               0.0000000000000         9.7548762358998       10.0000000000 km2/s2
* Decl V-infinity           -28.500000000000        21.300111320272       28.500000000000  deg
* Rt Asc V-infinity         0.0000000000000         193.88759891560       360.00000000000  deg
================================================================================
BREAKPOINT  : Earth-Mars Mvr                  Fid JD:  2454460.1000000
                            Lower                   Value                 Upper    | R  F
MAGNITUDE
Julian Date                 2454460.0000000         2454460.1000000       2454460.2000000  day
================================================================================
PARAMETER SET: Hyperbolic Asymptote   equator and equinox of epoch     WRT:  Mars
                            Lower                   Value                 Upper
Julian Date                 2454855.1000000         2454855.2000000       2454855.2000000  day
Periapsis Altitude          249.00000000000         250.00000000000       251.0000000000   km
B Plane Angle               -95.000000000000        -92.920000000000      -90.00000000000 deg
Time wrt Periapsis          -600.00000000000        0.0000000000000       600.00000000000   s
* V-infinity                2.0000000000000         3.0865921560539       6.920000000000  km/s
* Decl V-infinity           -90.000000000000        -28.843439961037      90.000000000000  deg
* Rt Asc V-infinity         0.0000000000000         289.33089964157       360.00000000000  deg
================================================================================
BREAKPOINT  : Mars Arrival                    Fid JD:  2454855.5000000
                Lower                Value                    Upper    | R V  { NO MVR }
Julian Date     2454855.4000000      2454855.5000000          2454855.6000000    day
================================================================================
```

Figure 3.10: CATO Output in State Vector Format for Type 4- Trajectory

## F. KPLOT

This subroutine provides for an extensive plotting capability that produces such plots as interplanetary trajectory views, spacecraft views of planets and satellites, polar views of a satellite tour, and views of the spacecraft orbit at satellite flybys. For preliminary analysis purposes, the interplanetary trajectories generated using Kplot are centered about the Sun and referenced to the Earth's ecliptic plane. Although the obliquity of Mars with respect to the Sun is roughly 25.19°, it differs from Earth's obliquity by only about 1.8° allowing Kplot to display a fairly accurate representation in two dimensions. In addition to displaying the interplanetary trajectory on the plot, the orbits of both Earth and Mars are shown providing valuable information about the geometric relationships between the spacecraft and the bodies of interest.

These geometric relationships are critical in the design process. The spacecraft to Sun distance is important in the sizing of the solar arrays during the interplanetary transfer, while the distance from spacecraft to Earth becomes important for communication data rates. It should be noted that because the trajectories calculated using CATO are primarily ballistic, the trajectory appears as an ellipse for the type 3 and 4 trajectories. The orbits of the planets as well as the spacecraft trajectory are annotated with tick marks that allow the user to determine the position of the planets or the spacecraft at any time during the transfer. Also, the analytical calculator available in the subroutine QUICK can be used to numerically verify the geometry of the bodies. The distances of both spacecraft to Sun and spacecraft to Earth can be calculated and subsequently plotted using Excel.

Kplot is initiated using the command,

**kplot <filename**

The input file is generated in the same way as the previous subroutines discussed using the inputs from the Kplot user's manual as necessary. The state vector used to specify the trajectory can either be generated using QUICK as demonstrated in Figure 3.4 or can come directly from CATO. Upon completion of the run, a postscript file can then be created and imported to a PC environment for use. A sample Kplot input file is shown in Figure 3.11 for an orbiter type 4- trajectory launched on 11/26/06. Although the input is

```
title='Mars 2007 Orbiter Traj '                     @ Title
figure=0                                            @ Turns off figure label
char=0.15,0.15                                      @ Sets size of text
nbody=0                                             @ Chooses Sun as central body for view
view=10,11,0,0                                      @ Establishes Earth Ecliptic viewpoint
fov=5.0e8.,0.,0.                                    @ Range of field of view for plot
bodsel=3*0,2*112,5*0,11111                          @ Settings for how bodies will be plotted
bodies(11)=-670                                     @ Identification for spacecraft
tunits(2)='day'                                     @ Time reference is days
vtime=789                                           @ Trajectory plotted until vtime
ttime=0,789                                         @ Epoch plus ttime equals start/stop
shift=0,0
nsc=1                                               @ Selects reference orbit
iorb=81                                             @ Sets IORB flag for state vector input
orb=181857629.67,0.1949555954,2.368918509$          @ State vector from CATO or QUICK
    63.50595146,18.52525236,-12.32243536
bodtic(3)=200                                       @ Defines tick mark spacing for planets
bodtic(11)=100                                      @ Defines tick mark spacing for spacecraft
scname='O'                                          @ Name assigned to spacecraft
scid=-670                                           @ Spacecraft identification
date=061126.,000000.,061126.,000000.,090123.,000000.   @ Launch/Arrival dates
epoch=061126.,000000.                               @ Reference date
cbody=0,39*0                                        @ Choosing Sun as central body for state
timdat=0                                            @ Turns off time data on plot
axes=7,7                                            @ Defines the size of the plot
;
```

Figure 3.11: Sample Kplot input file for Type 4- Trajectory

extensive, it is fairly simple to modify for the trajectory of choice. The Kplot for this input file is shown in Figure 3.12. The Earth and Mars are clearly marked with blue and red circles respectively to designate the location of those bodies at both the launch date and arrival date. This type 4- trajectory appears as an ellipse as alluded to earlier so the transfer actually completes more than one revolution about the sun before arriving at Mars. From this example, the location of the orbiter with respect to the Sun, Earth, and Mars can easily be seen at any point during the transfer.

Using QUICK, distances from the spacecraft to the Sun and Earth were calculated and plotted over the transfer time interval using this sample trajectory. This graph

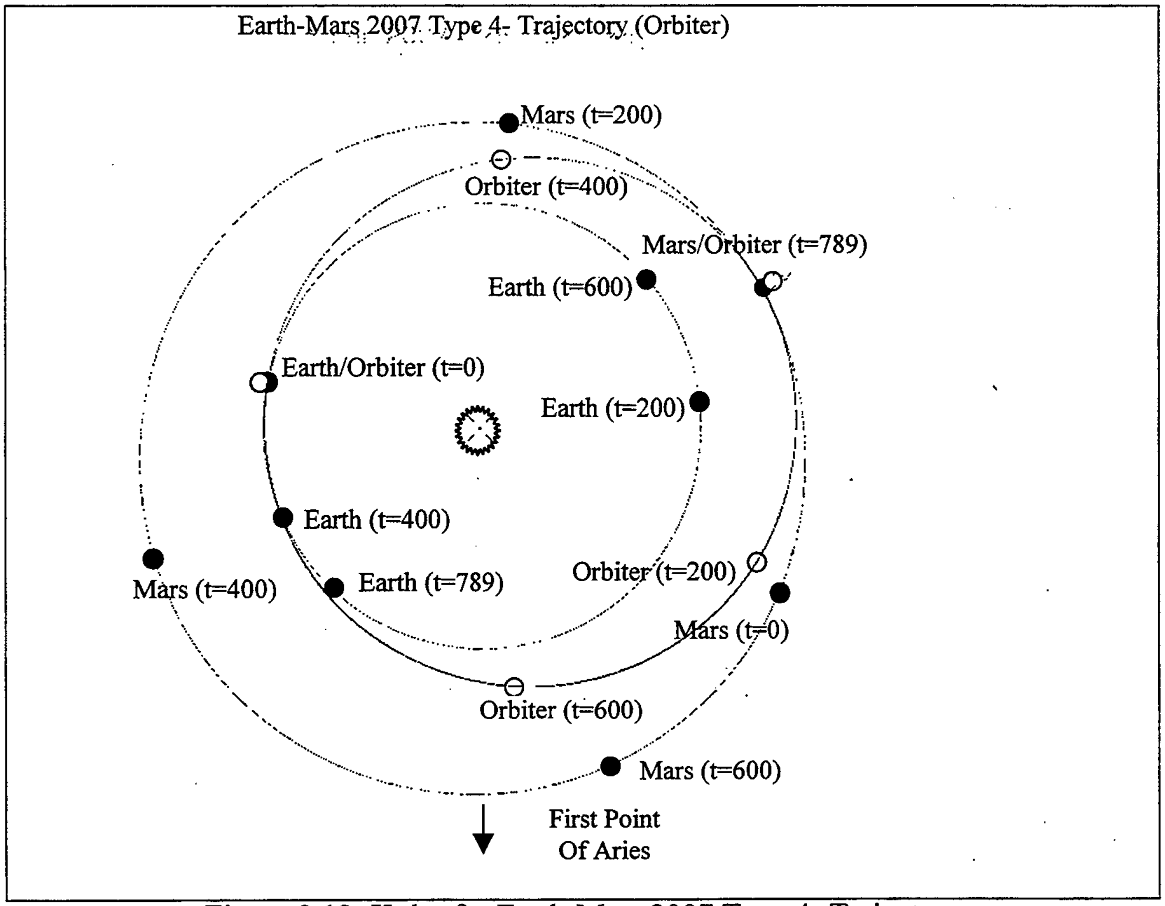

Figure 3.12: Kplot for Earth-Mars 2007 Type 4- Trajectory

provides a numerical reference for the geometric representation shown by Kplot. Figure 3.13 clearly depicts how the distances vary over the transfer interval. Once the orbiter or lander arrives at Mars, the spacecraft distances to the Sun and Earth are treated as equal to the Mars distance from the Sun and Earth (orbiter altitude is negligible). Although these distances do not change drastically from one launch date to another with the launch period, it is necessary to investigate the "worst" case for design purposes.

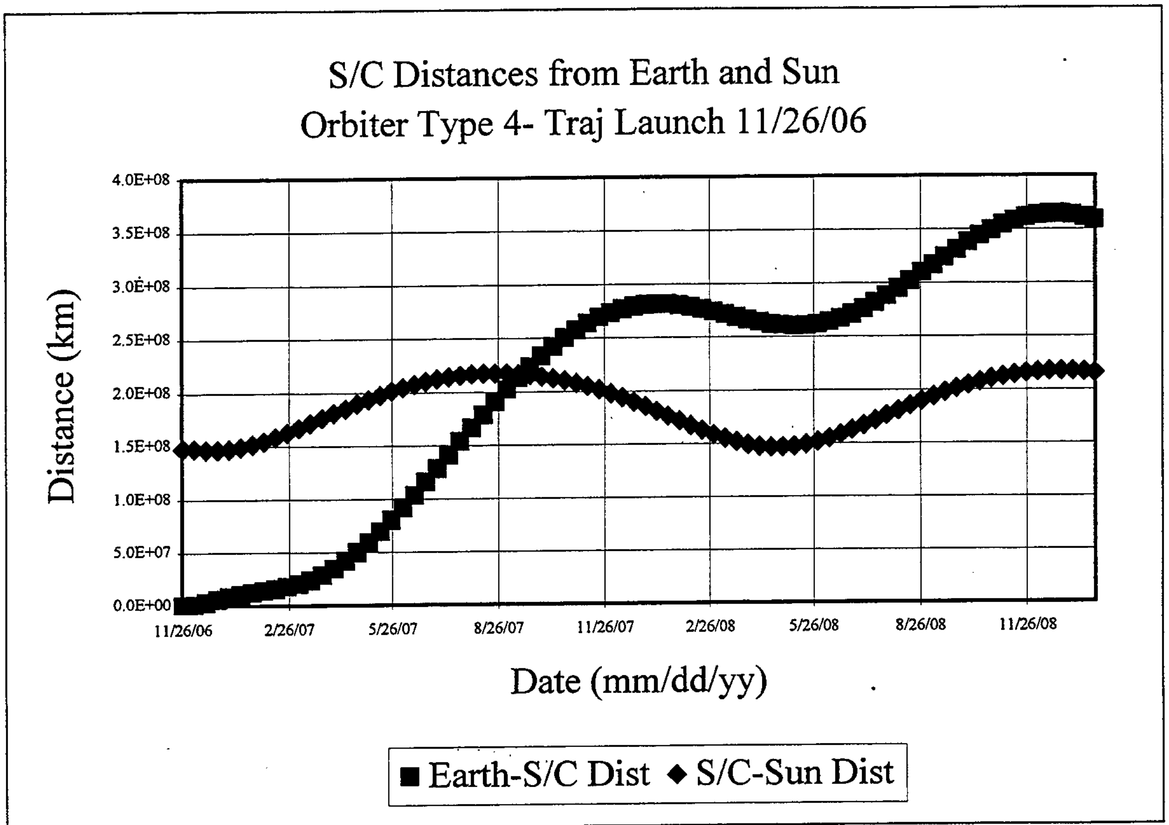


Figure 3.13: QUICK Calculations for Spacecraft Distances to Earth and Sun

# IV. BASELINE TRAJECTORY ANALYSIS

## A. ASSUMPTIONS

The 2007 Mars mission analysis includes several assumptions that more clearly define the scope:

1) In accordance with Mars program policy, it is desirable for the orbiter to be on orbit prior to the arrival of the lander to provide the communications relay function. There are two instances where this may not be necessary; 1) the lander is equipped with a direct Earth link or 2) the Micro-missions program successfully launches and sustains some type of communications architecture about Mars. For purposes of this analysis, the orbiter is assumed to be on orbit prior to the arrival of the lander;
2) The baseline configuration for the Mars 2007 mission consists of an orbiter and lander/rover based on previous designs and launched on two separate launch vehicles. This analysis however, also considers scenarios for a single launch vehicle;
3) The science requirements are assumed to include surface mapping of Mars which would require a nominal 400 km, sun-synchronous orbit for the orbiter;
4) Launch periods are limited to 20 days with a minimum time of 14 days in between launches for two separate launch vehicles;
5) Although current Thermal Protection System (TPS) materials can only withstand around 5.5 km/s for $V_{\infty_A}$, advances in TPS materials are

assumed to allow the $V_{\infty_A}$ to increase to as much as 8.0 km/s (Matousek and Sergeyevsky, 1998, p.11).

## B. TRAJECTORY TYPE SELECTION

Recall that a good rule of thumb for C3 is 10 $\text{km}^2/s^2$. Referring back to Table 3.1 containing the feasible ballistic trajectories for the 2007 opportunity, it appears that only three of the four possible trajectories are reasonable. The type 1 trajectory with a minimum C3 of 18.8 $\text{km}^2/s^2$ appears well beyond launch vehicle capabilities given current spacecraft designs. Additionally, the significantly high declination of the departure asymptote for the type 1 trajectory would result in a launch vehicle performance penalty for reasons discussed previously. Consequently, this trajectory was not considered in the analysis. A quick look at the other parameters for the remaining trajectories indicates that the minimum C3 for the type 2 trajectory is 12.7 $\text{km}^2/s^2$. This relatively high launch energy requirement however, is offset by the magnitude of the corresponding $V_{\infty_A}$. The significantly lower $V_{\infty_A}$ reduces the amount of $\Delta V$ required for the MOI, thus making a reduction in the injected mass at launch due to the higher C3 more acceptable. Therefore, this trajectory could not be discarded, leaving the number of possible trajectories for the '07 opportunity at three.

The next step in the analysis process is to generate the C3 contours associated with each of the remaining three trajectories. The C3 contour for the type 2 trajectory in Figure 4.1 shows a reasonable launch period of 20 days with a maximum launch energy of 13.24 $\text{km}^2/s^2$ during the launch period. As the launch date deviates from the nominal case where the C3 is equal to 12.79 $km^2/s^2$, the launch energy requirement increases as expected. Although the difference between the minimum and maximum launch energy for this 20 day launch period only yields an injected mass differential of about 20 kg (launch vehicle dependent), the C3 rapidly increases beyond this period. Therefore, this trajectory will not accommodate two separate launch periods.

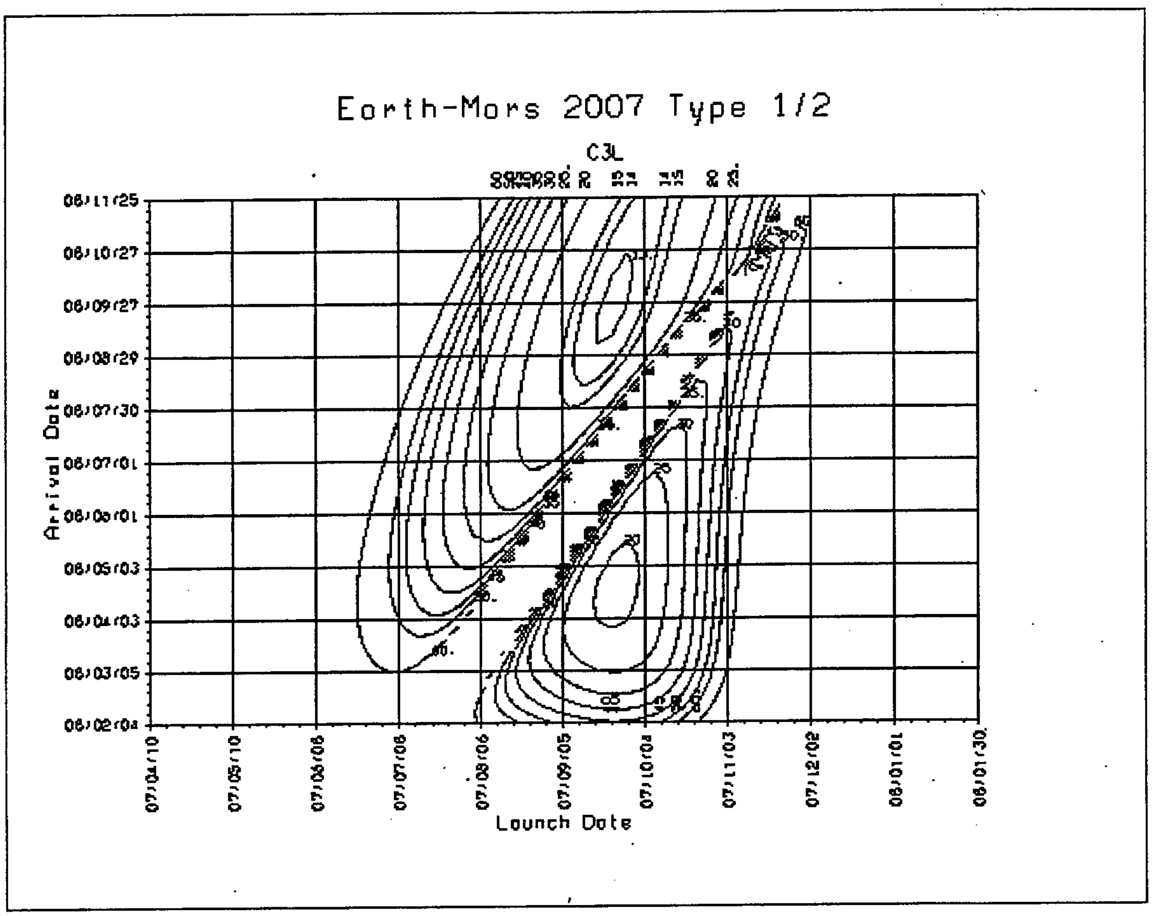


Figure 4.1: Earth-Mars 2007 Type 1 / 2 C3 Contours

As can be seen from Figure 4.2, the region where the type 3+ launch energy is reasonable occurs in close proximity to the "ridge." Recall from Chapter II that the ridge represents near-180° transfer trajectories. As the transfer angle approaches 180°, the transfer arc generally requires a high inclination with respect to the ecliptic in order to achieve the trajectory. Therefore the energy required to rotate the orbit out of the ecliptic is in addition to the energy required for normal trajectory injection. However, the launch energy requirement along the "ridge" is finite but prohibitively large by current standards. It is possible however, to reduce the C3 near the ridge by conducting a "broken-plane" maneuver. This is accomplished by performing a deterministic $\Delta V$ maneuver in the vicinity of the halfway point on the transfer orbit. This would add considerable complexity to the mission design and as a result, the type 3+ trajectory was not considered in this analysis as one of the primary trajectories.

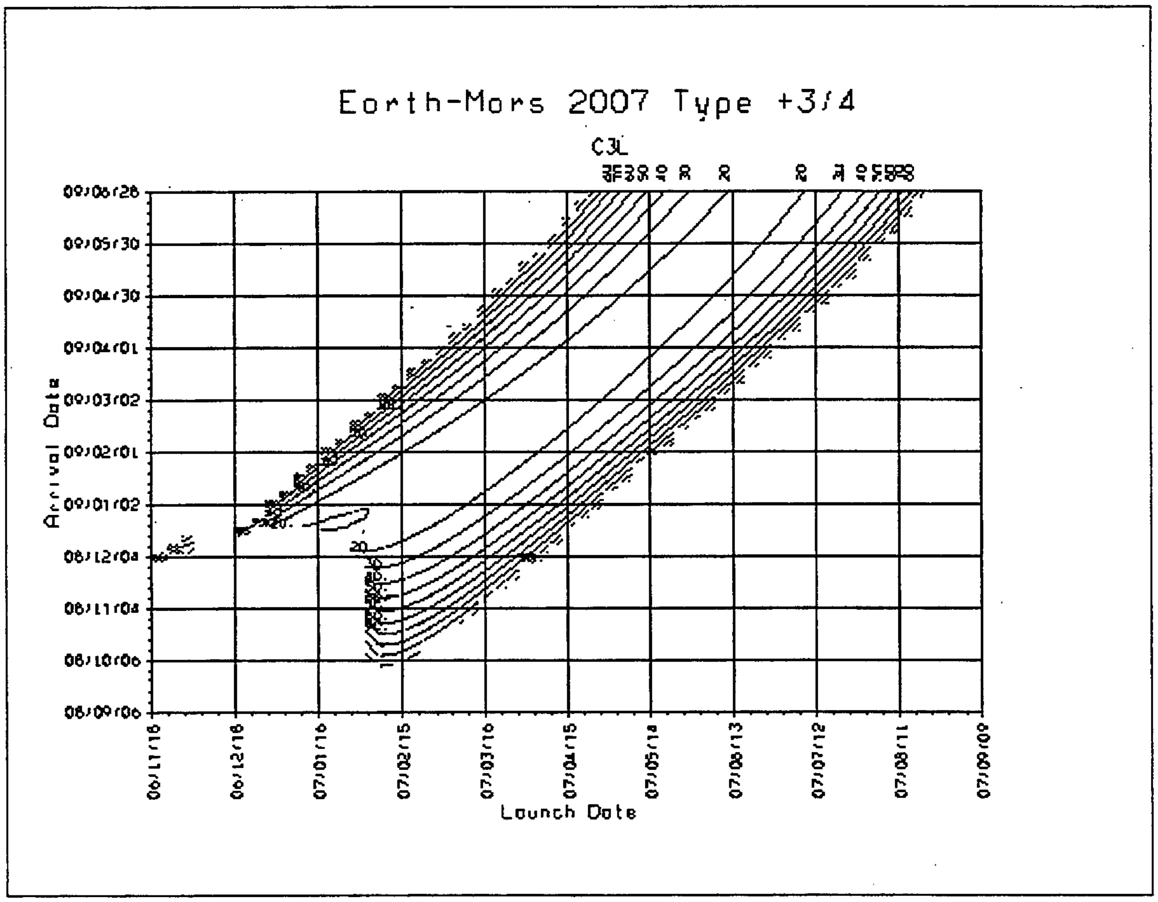


Figure 4.2: Earth-Mars 2007 Type 3+ / 4+ C3 Contours

In Figure 4.3, the type 4- trajectory presents a unique opportunity in that there are over 150 days in which the launch energy is less than 10 $km^2/s^2$. This provides for a large degree of flexibility in choosing the launch periods for the orbiter and the lander. Therefore, from the four possible ballistic trajectories for the 2007 opportunity, only the type 2 and the type 4- were chosen for further analysis.

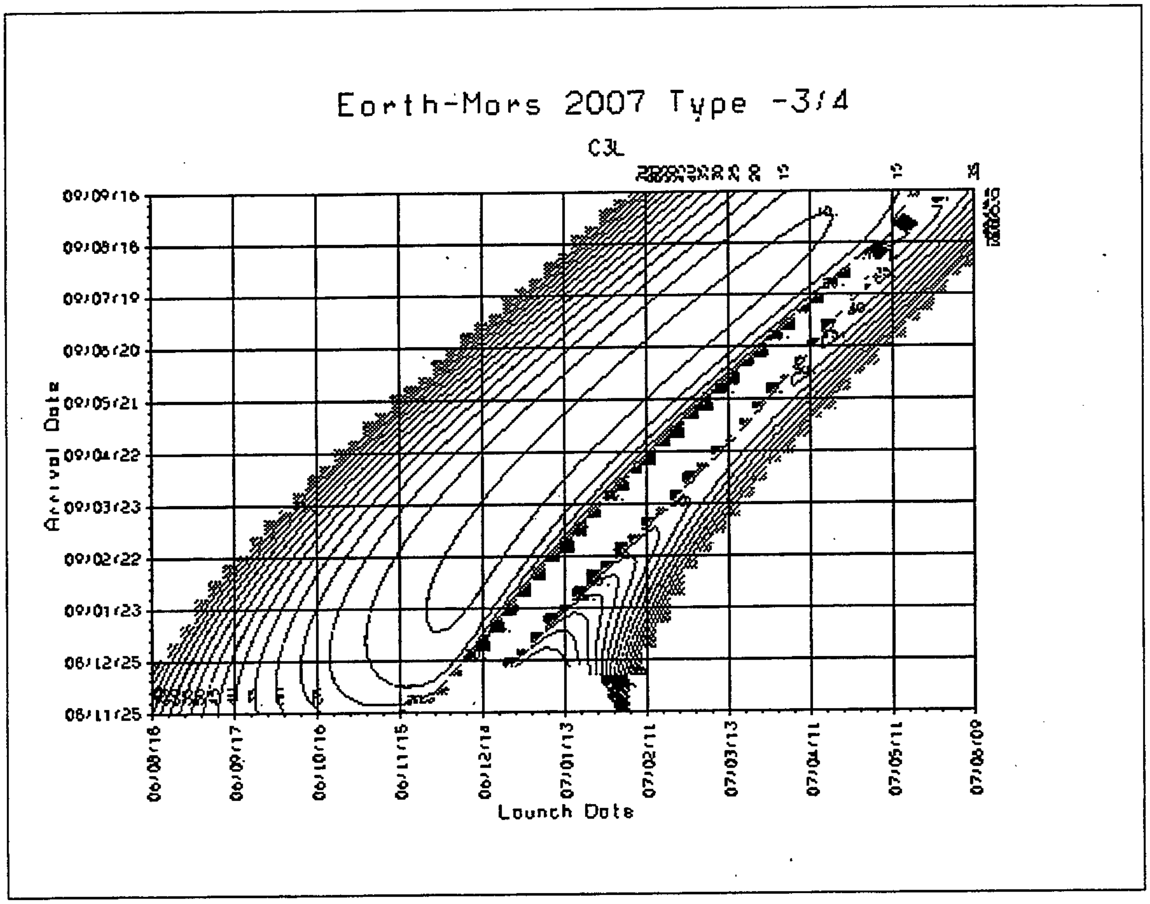


Figure 4.3: Earth-Mars 2007 Type 3- / 4- C3 Contours

## C. ORBITER AND LANDER LAUNCH PERIOD DETERMINATION

The orbiter launch period for a particular trajectory type must be carefully chosen after giving consideration to the various trajectory parameters. The fact that the type 4-

trajectory has a span of more than 150 days with a C3 less than 10 $km^2/s^2$ makes this task even more challenging. Figure 4.4 shows this large region on a C3 contour plot for the type 4- trajectory. Additionally, the contours for $V_{\infty_A}$ are plotted on top to show their relationship to C3 as well as to the launch date. The figure shows that the later the launch date, the larger $V_{\infty_A}$ will be upon arrival at Mars. The significance of $V_{\infty_A}$ was discussed in Chapter 2, but recall that as $V_{\infty_A}$ increases, the amount of propellant required for the MOI maneuver also increases. Also, a higher $V_{\infty_A}$ results in an increase in the TPS mass requirements for the orbiter. Therefore, it appears most cost effective from a mass standpoint for the orbiter to launch as early as possible on the type 4- opportunity.

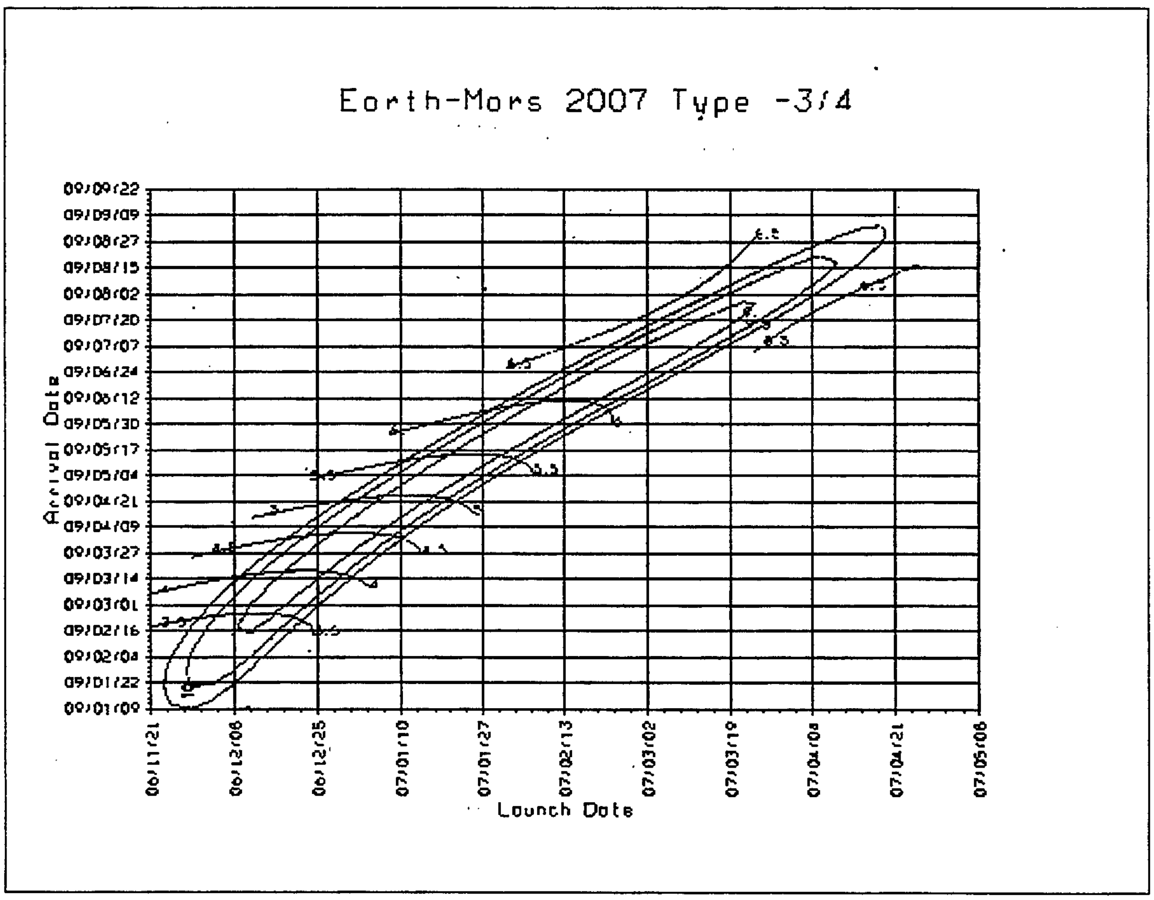


Figure 4.4: Earth-Mars 2007 Type 4- Trajectory (C3, VHA)

The orbiter launch period for the type 2 trajectory is much more straightforward. As alluded to earlier, the type 2 trajectory only accommodates a single 20 day launch period before the launch energy requirements become impractical. Therefore, the orbiter launch period for the type 2 trajectory is centered about the minimum energy launch date.

The launch period for the lander is dependent upon the orbiter launch period. In general, there is a minimum of 10 days required between launches at Cape Canaveral Air Station (CCAS) to accommodate ground launch services. However, as stated in the mission assumptions, a more conservative delay of 14 days between separate launches was used. When the orbiter and lander are both travelling on a type 4- trajectory, the earliest the lander may launch is equal to the latest launch date for the orbiter plus 14 days. This consideration is not a factor however, when the orbiter is travelling on a type 2 trajectory. In that case, the lander on the type 4- trajectory will launch well before the orbiter on the type 2 trajectory, but arrive later at Mars since the type 2 for the orbiter is a much shorter flight time trajectory.

The other constraint on the lander launch period is the orbiter capture scheme. Table 4.1 presents general "rules of thumb" to calculate the earliest lander arrival date given a specific orbiter launch period and capture scheme. For propulsive capture and

Table 4.1: Determination of Lander Arrival Date using Orbiter Capture Schemes

| **Capture Scheme** | |
|---|---|
| Propulsive | Earliest Lander Arrival Date = Latest Orbiter Arrival Date + 7 days |
| Aerocapture | Earliest Lander Arrival Date = Latest Orbiter Arrival Date + 7 days |
| Aerobraking | Earliest Lander Arrival Date = Latest Orbiter Arrival Date + time to Aerobrake |

aerocapture, the orbiter is given an arbitrary 7 days to establish a final orbit. For aerobraking, the time required for the orbiter to aerobrake becomes a consideration as well for the arrival date of the lander. Recall that when aerobraking, the initial capture orbit is attained by conducting a propulsive MOI maneuver. The magnitude of this maneuver is dependent on the period of the initial elliptical orbit. The longer the period of the initial orbit, the longer it will take to aerobrake. Given launch periods for both the orbiter and lander, Table 4.2 provides general "rules of thumb" for determining the minimum and maximum aerobraking time available to the orbiter.

Table 4.2: Determination of Aerobraking Time

| **Aerobraking Time** | |
|---|---|
| Minimum | Aerobraking Time = Earliest Lander Arrival Date - Latest Orbiter Arrival Date |
| Maximum | Aerobraking Time = Latest Lander Arrival Date - Earliest Orbiter Arrival Date |

Using these "rules of thumb" combined with the minimum time delay between launches, the following six scenarios in Table 4.3 were considered for analysis. Note that the launch period for the orbiter remains the same for each of the two trajectory types regardless of the capture scheme. This allows a comparison to be made between the three orbiter capture schemes. When the orbiter utilizes aerobraking for capture, the launch

Table 4.3: Scenarios using Two Launch Vehicles

| Scenario | # LV | Orbiter Traj | Launch Period | Capture | Lander Traj | Launch Period |
|---|---|---|---|---|---|---|
| 1 | 2 | Type 4- | 061126-061215 | Prop | Type 4- | 061229-070117 |
| 2 | 2 | Type 4- | 061126-061215 | AC* | Type 4- | 061229-070117 |
| 3 | 2 | Type 4- | 061126-061215 | AB** | Type 4- | 070115-070315 |
| 4 | 2 | Type 2 | 070913-071002 | Prop | Type 4- | 061126-061215 |
| 5 | 2 | Type 2 | 070913-071002 | AC | Type 4- | 061126-061215 |
| 6 | 2 | Type 2 | 070913-071002 | AB | Type 4- | 061126-070103 |

* AC – Aerocapture ** AB – Aerobraking

period analysis provided for the lander is much longer than the normal 20 days. This allows for flexibility in choosing the period of the initial capture orbit which again, is directly related to the time it will take to complete aerobraking. For all of the trajectory analyses in this thesis, the criterion for comparison between trajectories is the post-capture orbiter mass.

# V. TWO LAUNCH VEHICLES – SCENARIOS 1, 2, 3

## A. SCENARIO 1: LAUNCH PERIODS AND TRAJECTORY ANALYSIS PARAMETERS

The orbiter and lander both travel from Earth to Mars on a type 4- trajectory in this scenario with the orbiter utilizing a propulsive capture scheme to enter into orbit about Mars. Figure 5.1 depicts the launch and arrival periods for both the orbiter and the lander as plotted on a C3 contour. The transfer times vary from 789-809 days for the

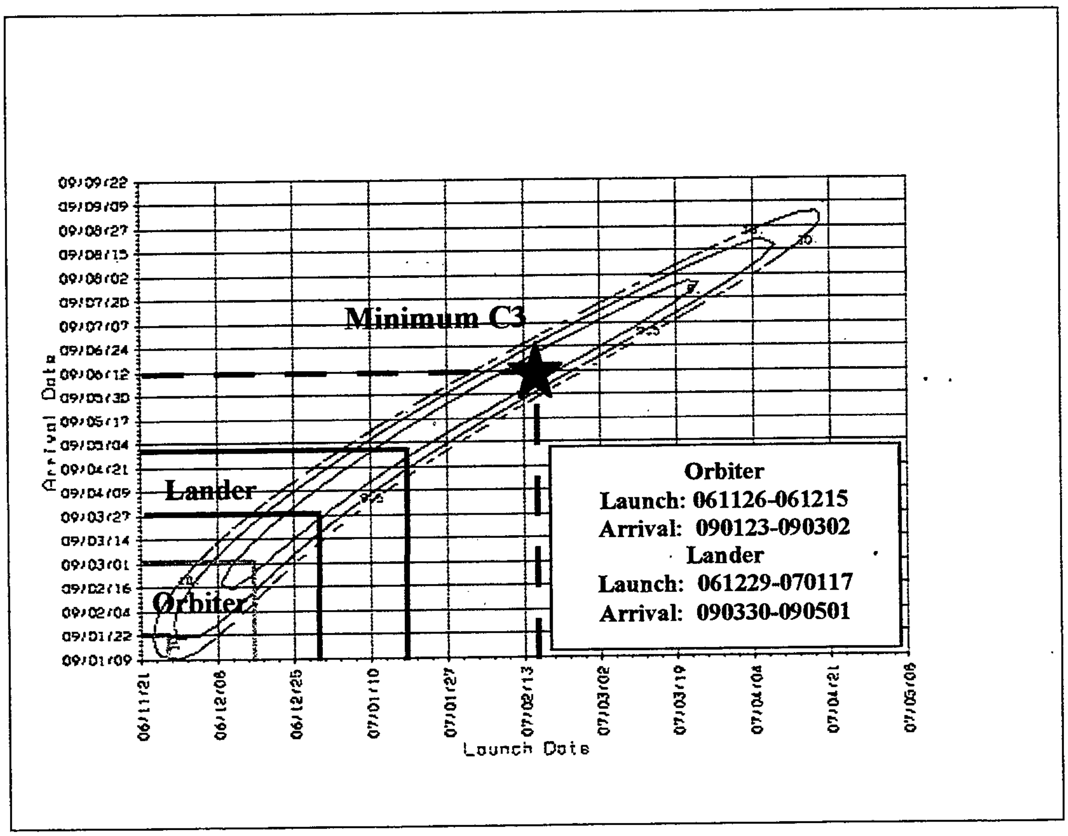


Figure 5.1: Scenario 1 – Orbiter & Lander Launch Periods

orbiter and 822-835 days for the lander depending on the launch date. Note the minimum 20 day separation between the latest orbiter launch date and the earliest lander launch date in accordance with the initial mission assumptions. Also note that the time between latest orbiter arrival and earliest lander arrival (28 days) clearly exceeds the 7 days required for the orbiter to establish a final circular orbit when propulsive capture is used.

Each launch date (orbiter or lander) corresponds to an optimal trajectory as calculated by CATO using the process outlined in Chapter III. The analysis parameters for each trajectory have been included in Appendix A for review. For convenience, plots of C3 and $V_{\infty_A}$ vs. launch date for the orbiter are included in Figures 5.2 and 5.3 while the same plots for the lander are included in Figures 5.4 and 5.5. The remaining parameters of interest ($\delta_\infty, \alpha_\infty, \delta_{\infty_A}, \alpha_{\infty_A}$) are also plotted vs. launch date but included separately in Appendix B. These plots are specific to the launch periods of interest and can be used in the manner described in Chapter II to find the following information:

1) The number of injection opportunities (DLA);
2) The daily launch windows (RLA);
3) The minimum orbiter inclination at Mars (DAP);
4) The ballistic landing region for the lander at Mars (DAP);
5) LMST for the initial orbit (RAP);

Special attention should be given to the C3 plots for this scenario. Recall the decision to use the same trajectory type for both the orbiter and the lander. For this reason, the launch periods are not located around the minimum C3 launch date as shown in Figure 4.5 thus explaining the shape of the orbiter C3 curve. For launch periods that are located about the minimum C3 launch date, the plot manifests itself as a "horseshoe" shape. The lander C3 curve on the other hand, appears to be oscillatory in nature. The scale utilized in presenting this data is such that for a preliminary trajectory analysis, the

C3 over this particular launch period can be considered constant. This is reinforced by the fact that the maximum C3 deviation during the launch period is only about 0.2 $km^2/s^2$ which corresponds to a difference of about 5 kg of injected mass (launch vehicle dependent). Both $V_{\infty_A}$ vs. launch date curves show an increase for later launch dates. This is typical of this type 4- opportunity for the entire 150 launch days. It reasons then that the launches for both orbiter and lander should occur as early as possible in the launch period to minimize $V_{\infty_A}$.

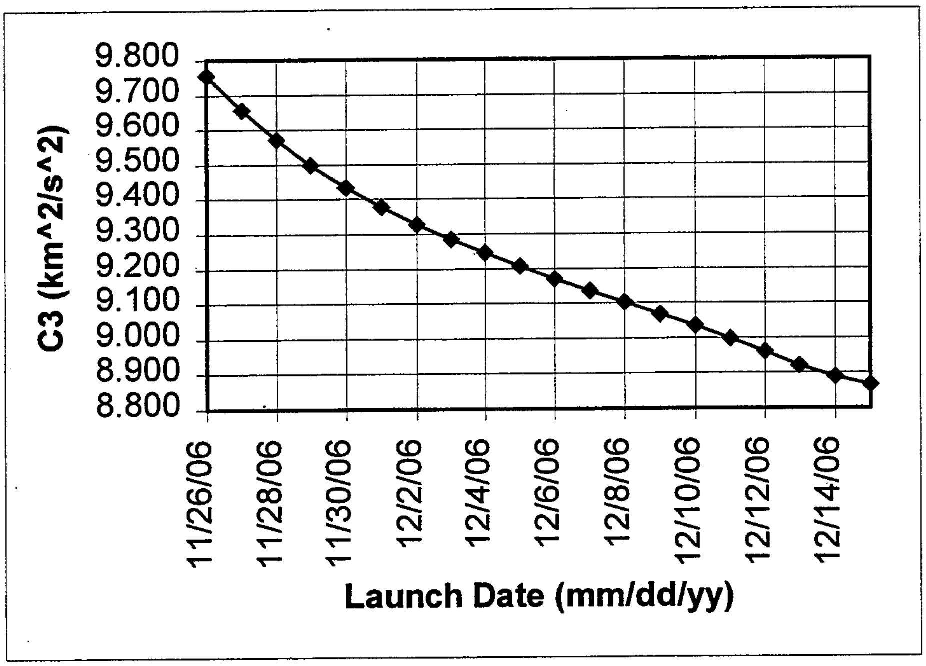


Figure 5.2: Scenario 1 - C3 vs. Launch Date (Orbiter)

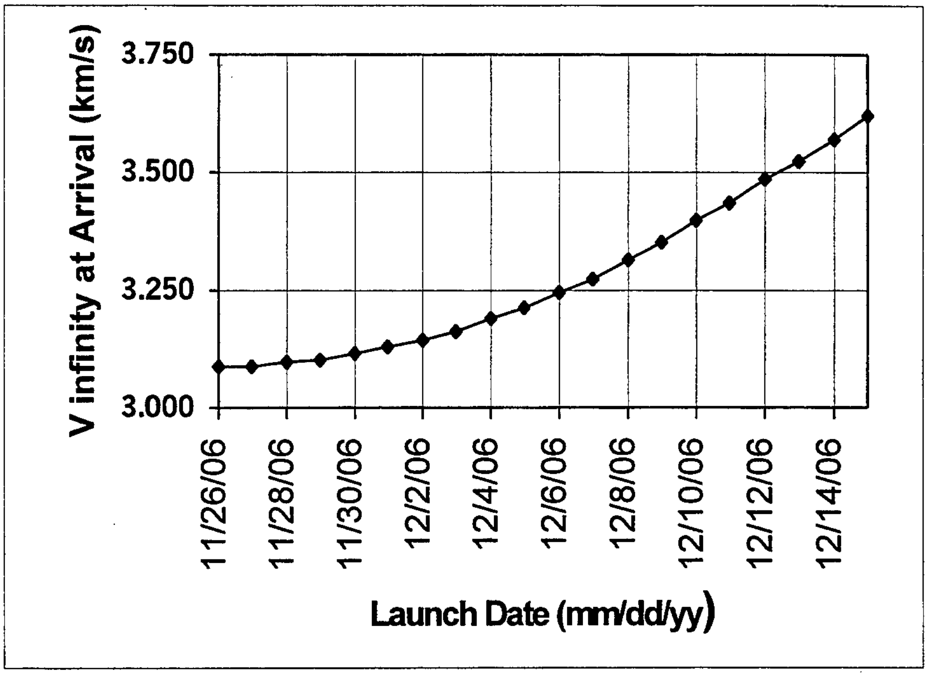


Figure 5.3: Scenario 1 - $V_{\infty_A}$ vs. Launch Date (Orbiter)

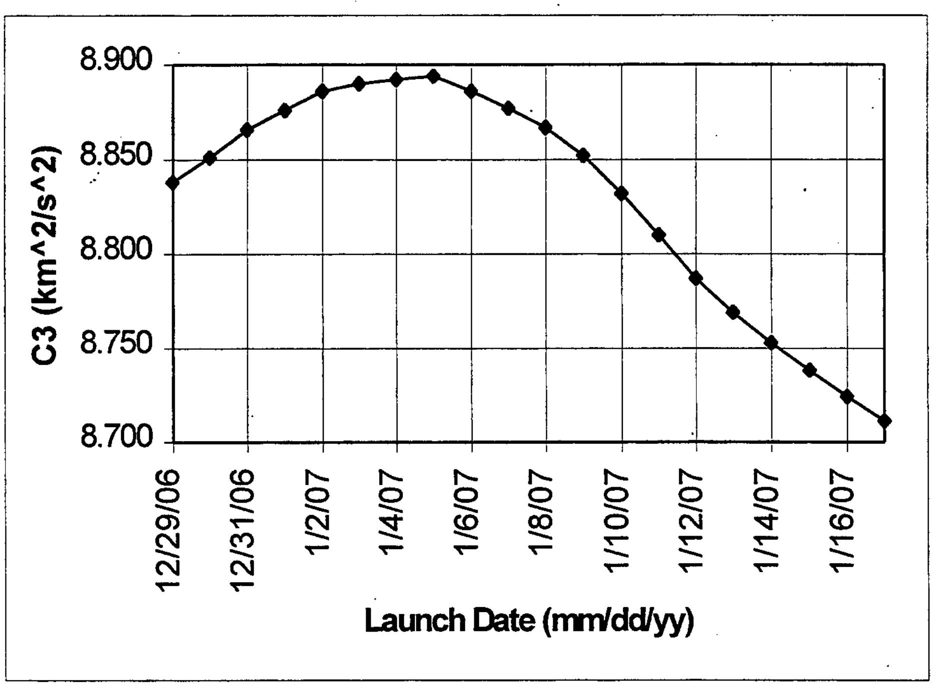


Figure 5.4: Scenario 1 - C3 vs. Launch Date (Lander)

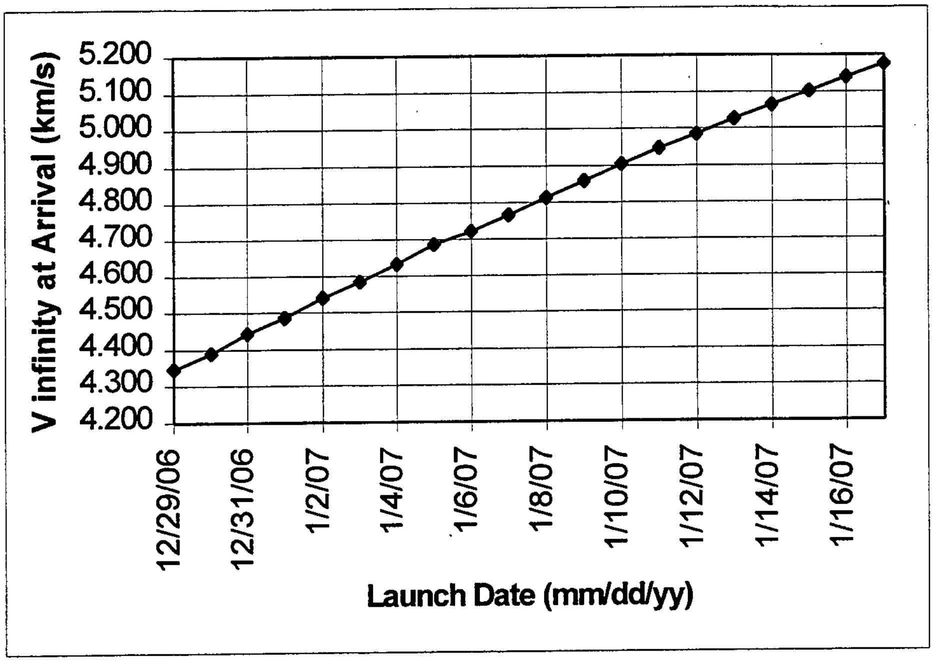


Figure 5.5: Scenario 1 - $V_{\infty_A}$ vs. Launch Date (Lander)

## B. SCENARIO 1: GEOMETRY

The geometric relationship between bodies of interest during interplanetary transfers is important information to mission designers. In particular, the spacecraft-to-Earth and spacecraft-to-Sun distances provide data necessary to size the solar arrays for cruise as well as to calculate the link budgets necessary for communications. Additionally, the Sun-spacecraft-Earth angle provides an important geometric relationship, as spacecraft orientation is generally a compromise between communications (antenna pointing) and solar heating constraints.

Shown in Figure 3.12 is the Kplot representation of the orbiter trajectory for an 11/26/06 launch date. Figure 5.6 shows the lander Kplot for a launch date of 12/29/06. Although each plot is specific to only one launch date within the respective

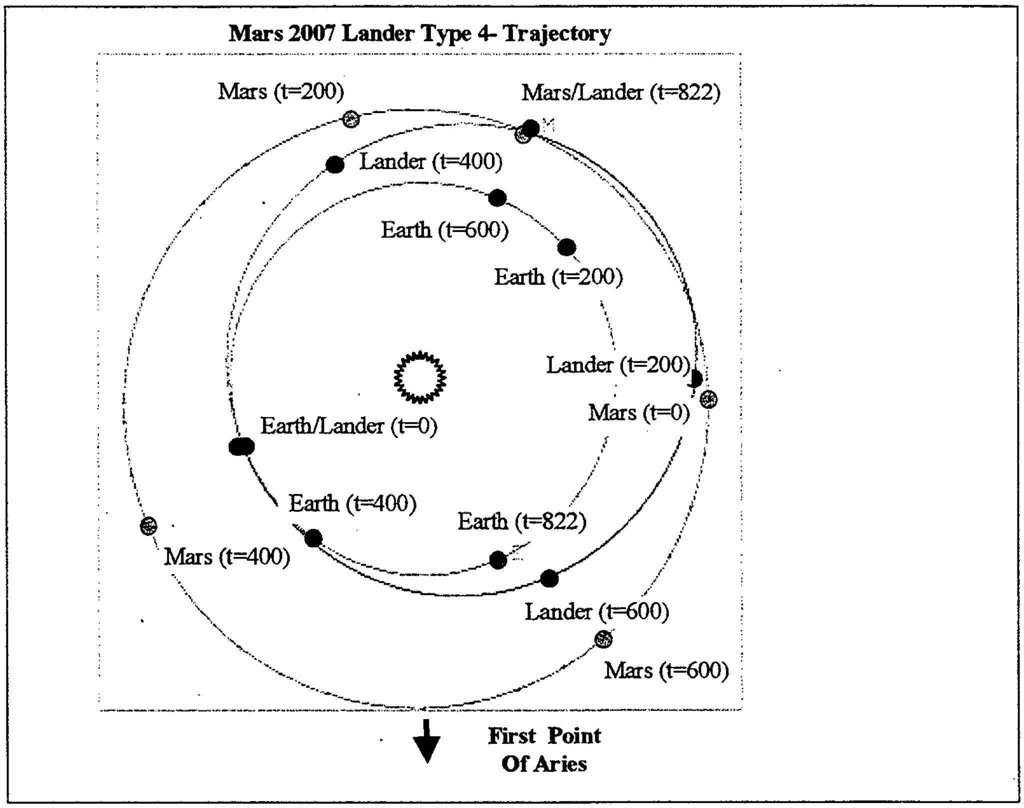


Figure 5.6: Kplot for Lander Type 4- Trajectory (Launch: 061229)

launch period, the changes from the beginning to the end of the launch period are modest due to the ballistic nature of the trajectories. Therefore, only the data for the first launch date is presented here. The Kplot has been annotated at various times (days past launch) with corresponding locations along the trajectory. The orbits of Earth and Mars are also annotated to facilitate an understanding of the geometric relationship between the bodies of interest. Using the astrodynamic functions in QUICK, the spacecraft-to-Earth and spacecraft-to-Sun distances for both orbiter and lander have been calculated. Figures 5.7 and 5.8 present this data for the total transfer time of each trajectory. For completeness, Figures 5.9 and 5.10 depict the Sun-spacecraft-Earth angle during the transfer orbit for the orbiter and lander respectively. These plots were used to verify the geometry shown in the Kplots.

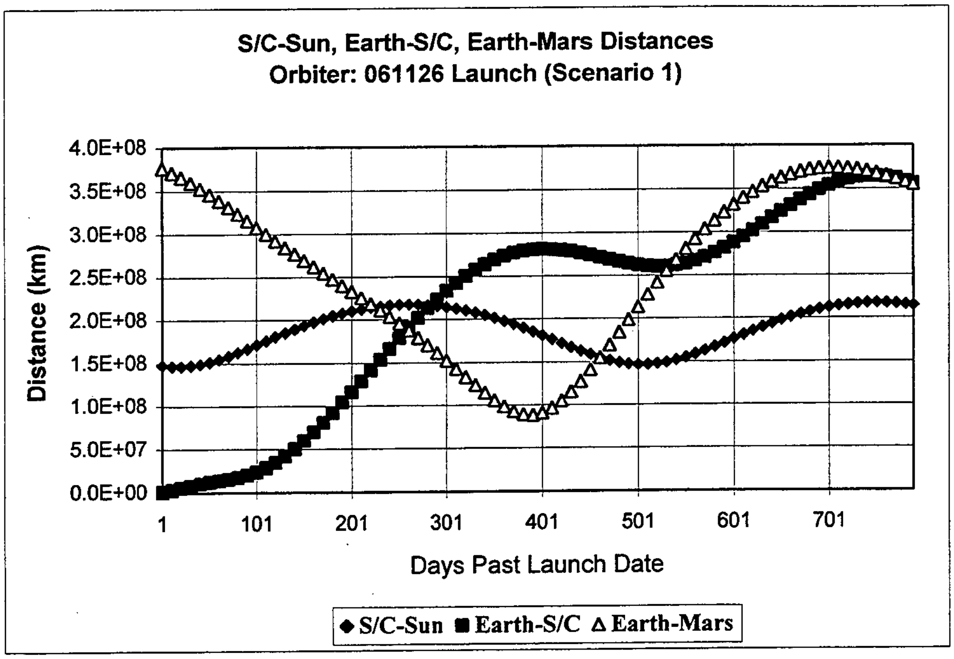


Figure 5.7: Orbiter Distances (061126 launch)

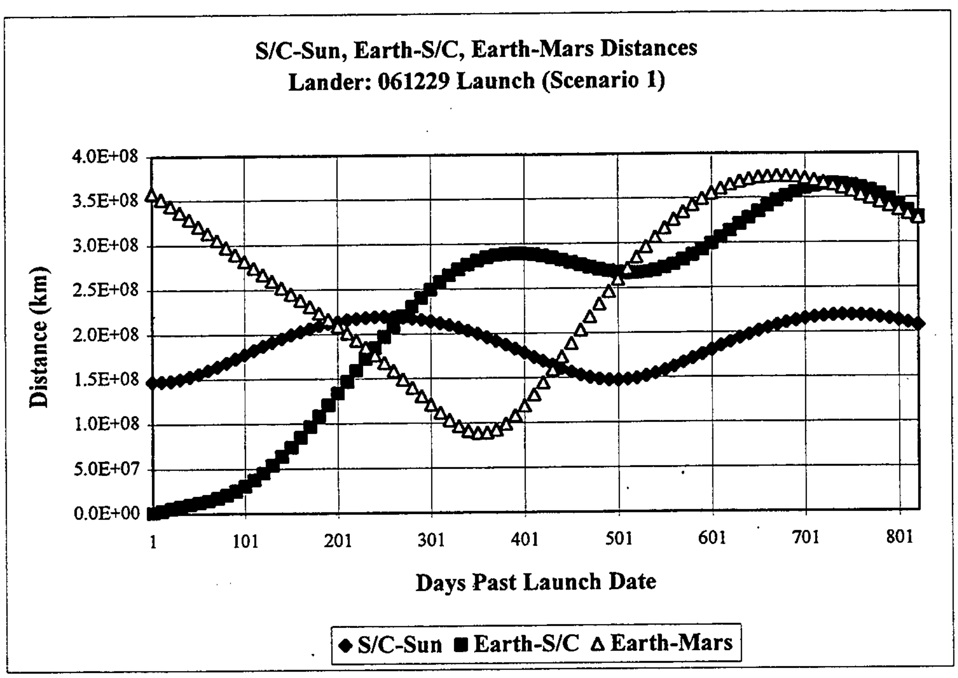


Figure 5.8: Lander Distances (061229 launch)

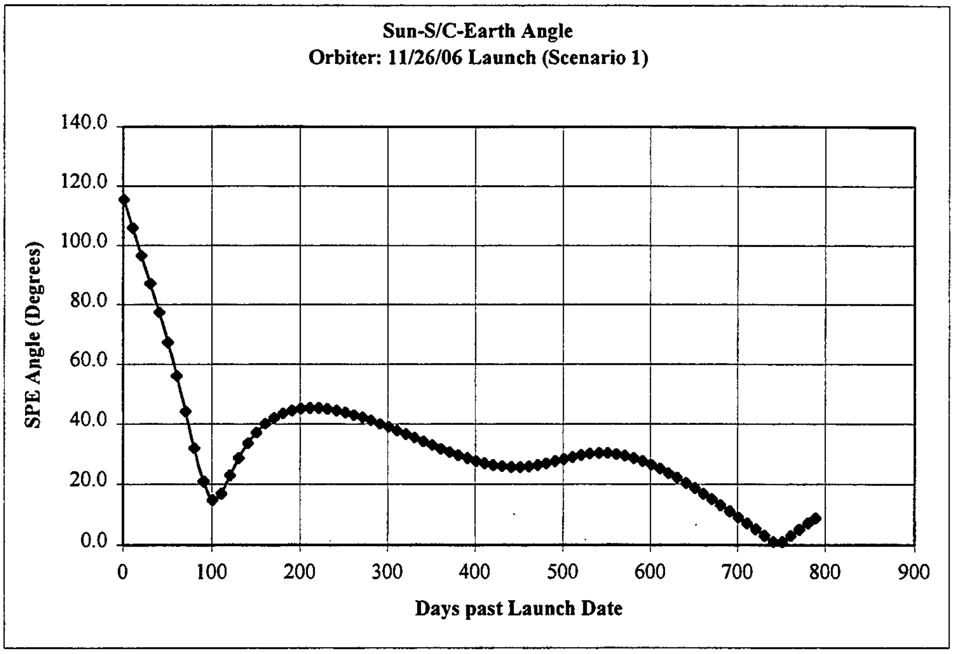


Figure 5.9: Sun-Orbiter-Earth Angle (061126 launch)

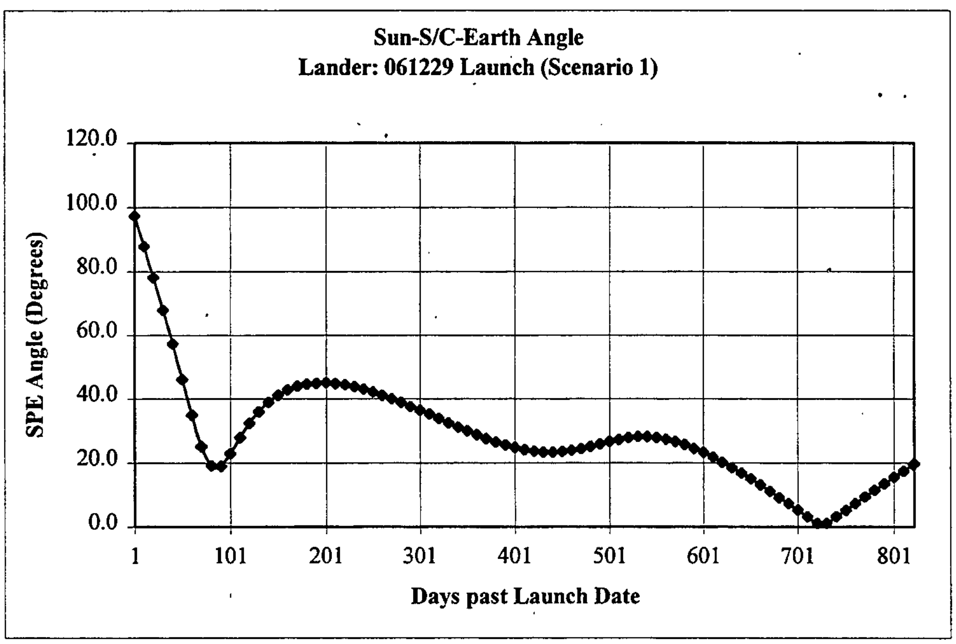


Figure 5.10: Sun-Lander-Earth Angle (061229 launch)

## C. SCENARIO 1: GENERAL ANALYSIS

Given the launch periods of the orbiter and the lander in this scenario, the following analysis is provided. As the lander launch period is dependent on the orbiter launch period, only TPS mass estimates and ballistic landing regions at Mars are calculated for the lander. The analysis for the orbiter includes a calculation of post-capture mass using various launch vehicles. Subsequent comparison with the other two capture schemes will yield the most suitable method. The following assumptions are made for the calculation of post-capture mass for the case of propulsive capture:

1) The $\Delta V$ for the trajectory correction maneuvers (TCMs) is estimated to be no more than 30 $m/s$. The exact magnitude of the these are not considered in this analysis;
2) The orbiter does not include a cruise stage when propulsive capture is primary;
3) The propulsive capture scheme assumes a bi-propellant propulsion system as well as a Hohmann-type maneuver to circularize.

### 1. Orbiter Analysis

In a strict sense, a propulsive capture would consist of a finite burn at MOI to capture into an initial elliptical orbit. This would allow any necessary orbit phasing maneuvers to occur at apoapsis where the velocity is minimum thereby reducing the propellant required. However, for purposes of this analysis, orbital phasing is not considered. Instead, one tangential impulsive burn at periapsis (250 km altitude) and one at apoapsis (equal to the final orbit altitude) will define the $\Delta V$ requirement for the propulsive capture. Equations 2.7 – 2.9 enable the required propellant mass to be calculated for a propulsive capture.

MIDAS was used to determine an optimal arrival date for each launch date in the manner previously described in Chapter 3. Recall that the cost function in MIDAS is the total $\Delta V$ for the trajectory. However, the arrival date and trajectory chosen corresponds to the minimum launch energy trajectory for a given launch date. It should also be noted that the parameters for this minimum C3 trajectory were extracted manually for each launch date from the **.lst** file.

Several Boeing Delta II series launch vehicles were considered for the orbiter. As the design of the spacecraft is not defined at this point, the interplanetary injected mass capability of each launch vehicle was used as a design point. The data for each launch vehicle was obtained from either the Delta II Payload Planners Guide or from previous data compiled at JPL. Performance curves for each launch vehicle from the Payload Planner's Guide or JPL were provided in a C3 vs. injected mass format. Data points from these curves were extracted and curve fit to a sixth order polynomial to describe the injected mass capability for that launch vehicle for all values of C3. Figure 5.11 shows the performance curve for a Delta II 7925 launch vehicle. The resultant curve fit is the following sixth order polynomial:

**Injected Mass** = 1.3652391E-10(**C3**)^6 - 7.1518516E-08(**C3**)^5 + 1.7191205E-05(**C3**)^4 - 2.6543564E-03(**C3**)^3 + 3.0821021E-01(**C3**)^2 - 2.7360060E+01(**C3**) + 1.2982938E+03

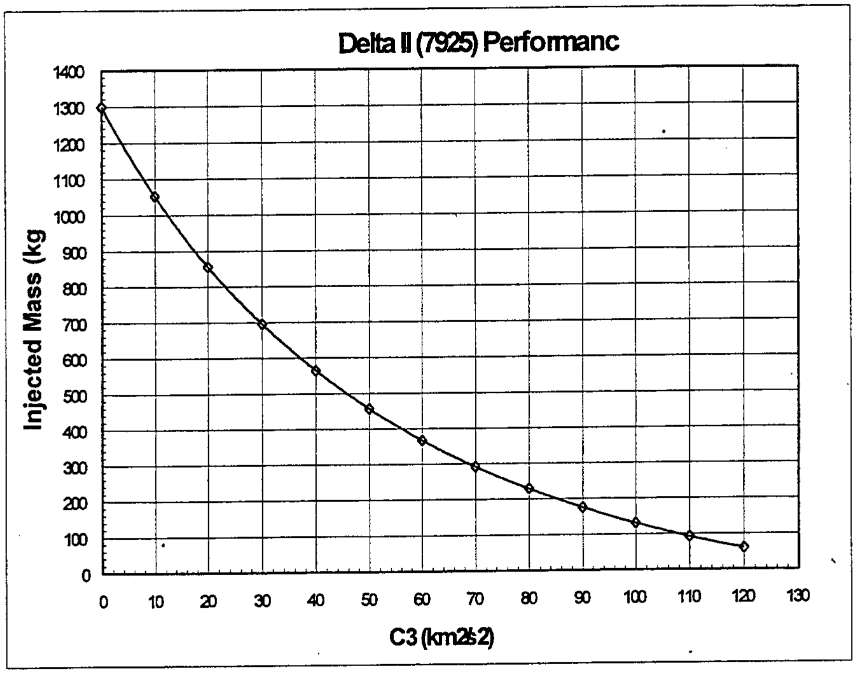


Figure 5.11: Injected Mass vs. C3 (Delta II 7925)

The C3 values for each trajectory as determined by CATO, can now be mapped to an injected mass for each launch vehicle. Current orbiter designs vary greatly. Mars Global Surveyor was launched with an injected mass of 1052 kg while the orbiter for Mars 98 only had a 643 kg injected mass. While the orbiter mass is largely dependent on the mission, a minimum injected mass of 600 kg was used in the launch vehicle analysis. Therefore, it was determined that the performance capability of the Delta II 7325 and 7425 launch vehicles would not support this minimum injected mass. In fact, the Delta II 7325 equipped with a Star 48B upper stage still does not provide the necessary performance. Table 5.1 presents the injected mass capability of several launch vehicles with respect to each launch date in the orbiter launch period. In general, these injected masses demonstrate whether the spacecraft design is within the performance capabilities of a particular launch vehicle. For analysis purposes however, the spacecraft design mass is assumed to be the injected mass for the post-capture mass calculations.

Table 5.1: Injected Mass Capabilities for Various LVs (Scenario 1 - Orbiter)

| Launch | C3 | Delta II 7325 Star 48B (7% margin) Inj Mass (kg) | Delta II 7425 Star 48B (7% margin) Inj Mass (kg) | Delta II 7925 (7% margin) Inj Mass (kg) | Delta II 7925H (10% margin) Inj Mass (kg) |
|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 557.6 | 649.2 | 984.3 | 1100.9 |
| 11/27/06 | 9.656 | 558.7 | 650.6 | 986.4 | 1103.2 |
| 11/28/06 | 9.572 | 559.7 | 651.8 | 988.1 | 1105.2 |
| 11/29/06 | 9.497 | 560.6 | 652.8 | 989.6 | 1106.9 |
| 11/30/06 | 9.433 | 561.4 | 653.7 | 990.9 | 1108.4 |
| 12/1/06 | 9.378 | 562.0 | 654.5 | 992.1 | 1109.6 |
| 12/2/06 | 9.327 | 562.6 | 655.3 | 993.1 | 1110.8 |
| 12/3/06 | 9.284 | 563.1 | 655.9 | 994.0 | 1111.8 |
| 12/4/06 | 9.245 | 563.6 | 656.4 | 994.8 | 1112.7 |
| 12/5/06 | 9.207 | 564.0 | 657.0 | 995.6 | 1113.6 |
| 12/6/06 | 9.171 | 564.5 | 657.5 | 996.4 | 1114.5 |
| 12/7/06 | 9.135 | 564.9 | 658.0 | 997.1 | 1115.3 |
| 12/8/06 | 9.102 | 565.3 | 658.5 | 997.8 | 1116.1 |
| 12/9/06 | 9.068 | 565.7 | 659.0 | 998.5 | 1116.9 |
| 12/10/06 | 9.035 | 566.1 | 659.4 | 999.2 | 1117.6 |
| 12/11/06 | 8.997 | 566.5 | 660.0 | 1000.0 | 1118.5 |
| 12/12/06 | 8.960 | 566.9 | 660.5 | 1000.8 | 1119.4 |
| 12/13/06 | 8.920 | 567.4 | 661.1 | 1001.6 | 1120.3 |
| 12/14/06 | 8.889 | 567.8 | 661.5 | 1002.2 | 1121.0 |
| 12/15/06 | 8.865 | 568.1 | 661.9 | 1002.7 | 1121.6 |

For a propulsive capture, only TCMs require propellant expenditures other than the capture itself. This simplifies the analysis for the case of the propulsive capture. Table 5.2 shows the orbiter post-capture mass calculations for the Delta II 7925 launch vehicle. Again, the rocket equation is used to calculate the propellant mass required for any $\Delta V$ maneuver. The tables for the other launch vehicles are included in Appendix C. It can be seen from this table that the later the launch date, the less post-capture mass remains for the orbiter. Since the C3 during the launch period is decreasing while the

$V_{\infty_A}$ is increasing, it appears that the $V_{\infty_A}$ has more influence on the overall post-capture mass of the orbiter.

Table 5.2: Scenario 1 - Orbiter Post-Capture Mass (Delta II 7925)

| Launch | C3 | V inf (km/s) | Inj Mass (kg) | TCMs (30 m/s) Prop Mass | MOI DV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 984.6 | 9.4 | 2.286 | 504.6 | 470.6 |
| 11/27/06 | 9.656 | 3.090 | 986.4 | 9.4 | 2.288 | 505.8 | 471.2 |
| 11/28/06 | 9.572 | 3.097 | 988.1 | 9.4 | 2.292 | 507.2 | 471.5 |
| 11/29/06 | 9.497 | 3.104 | 989.6 | 9.4 | 2.296 | 508.6 | 471.6 |
| 11/30/06 | 9.433 | 3.116 | 990.9 | 9.4 | 2.302 | 510.2 | 471.3 |
| 12/1/06 | 9.378 | 3.130 | 992.1 | 9.4 | 2.310 | 512.0 | 470.7 |
| 12/2/06 | 9.327 | 3.143 | 993.1 | 9.4 | 2.317 | 513.5 | 470.1 |
| 12/3/06 | 9.284 | 3.163 | 994.0 | 9.5 | 2.328 | 515.6 | 468.9 |
| 12/4/06 | 9.245 | 3.189 | 994.8 | 9.5 | 2.342 | 518.2 | 467.2 |
| 12/5/06 | 9.207 | 3.215 | 995.6 | 9.5 | 2.356 | 520.7 | 465.4 |
| 12/6/06 | 9.171 | 3.246 | 996.4 | 9.5 | 2.373 | 523.7 | 463.3 |
| 12/7/06 | 9.135 | 3.274 | 997.1 | 9.5 | 2.389 | 526.3 | 461.3 |
| 12/8/06 | 9.102 | 3.315 | 997.8 | 9.5 | 2.412 | 530.1 | 458.2 |
| 12/9/06 | 9.068 | 3.354 | 998.5 | 9.5 | 2.434 | 533.7 | 455.3 |
| 12/10/06 | 9.035 | 3.399 | 999.2 | 9.5 | 2.460 | 537.7 | 452.0 |
| 12/11/06 | 8.997 | 3.437 | 1000.0 | 9.5 | 2.481 | 541.3 | 449.2 |
| 12/12/06 | 8.960 | 3.486 | 1000.8 | 9.5 | 2.510 | 545.8 | 445.5 |
| 12/13/06 | 8.920 | 3.522 | 1001.6 | 9.5 | 2.531 | 549.2 | 442.9 |
| 12/14/06 | 8.889 | 3.570 | 1002.2 | 9.5 | 2.559 | 553.5 | 439.1 |
| 12/15/06 | 8.865 | 3.621 | 1002.7 | 9.5 | 2.589 | 558.0 | 435.1 |

### 2. Lander Analysis

Estimating TPS mass requirements is a difficult task. The Integrated Design System (IDS) from NASA Ames Research Center provides an Entry Vehicle simulation that enables the user to determine the maximum convective heat flux ($\dot{Q}_{MAX}$) for entry, given a set of input parameters. The algorithm used in this program is called the Fully Implicit Ablation and Thermal (FIAT) response code (Allen, 1998). The spherical Mars atmospheric model utilized is based on Viking I data. There were several choices for an entry vehicle. It was initially determined that the modified Newtonian biconic shape of the 2003 HEDS demonstrator was the most reasonable model. The dimensions are as follows:

| | |
|---|---|
| Mass: | 600 kg |
| Nose radius: | 0.2664 m |
| Cone interface radius: | 0.483 m |
| Vehicle length: | 3.4506 m |
| Base radius: | 0.6293 m |

One of the inputs controlled by the user is the flight path angle. For a normal descent to the Martian surface, the lander profile will most likely assume a flight path angle of $-12°$ or less. For this simulation however, a flight path angle of $-10.5°$ was chosen such that a worst case TPS mass estimate could be obtained. This shallower flight path angle causes the lander to remain in the Martian atmosphere for a longer period of time thus requiring a thicker ablative material for the TPS. The other input parameter of interest was $V_{entry}$ for the lander at the atmospheric interface. Since the $V_{entry}$ can be mapped directly to $V_{\infty_A}$ using the following equation,

$$V_{\infty_A} = \sqrt{V_{entry} - \frac{2\mu}{r_{entry}}} \qquad (4.1)$$

it was possible to obtain a comparison of the lander $V_{\infty_A}$ to the required TPS mass.

Although the code estimates the TPS mass based on a uniform heat distribution (Allen, 1998), it still serves as a useful baseline for this preliminary analysis. The entry vehicle trajectory is initiated at the atmospheric interface (125 km) using the entered $V_{entry}$. The program will then determine the $\dot{Q}_{MAX}$ for the corresponding trajectory. Once $\dot{Q}_{MAX}$ is determined, a TPS material can be chosen from a menu and a mass estimate can be calculated. It should be noted that because the program assumes uniform heating, the

TPS mass thickness is also uniform. Additionally, the TPS mass estimate only accounts for the heat shield on the spacecraft. The backshell mass was taken from the original Mars 2001 lander (~91 kg) and added to the heat shield mass determined by FIAT to obtain a total aeroshell mass. As many aspects of these calculations include inaccuracies, the TPS mass estimate is increased by 25% as a safety margin.

Figure 5.12 shows a plot of TPS Mass vs. $V_{\infty_A}$ as generated using FIAT data. This data does not reflect the additional mass of the backshell or the 25% safety margin. Note the two different materials on the plot. The SLA-561V appears to be a better choice, but after consulting with NASA Ames Research Center (Allen, 1998), it was determined that the SIRCA-15 was a more suitable material. The SLA-561V material has to be attached to the spacecraft by hand costing both time and money. Also, the material is known to "flake" upon exceeding a maximum thickness. A second-order polynomial was used to curve fit this data to allow for the TPS mass to be estimated for other values of $V_{\infty_A}$.

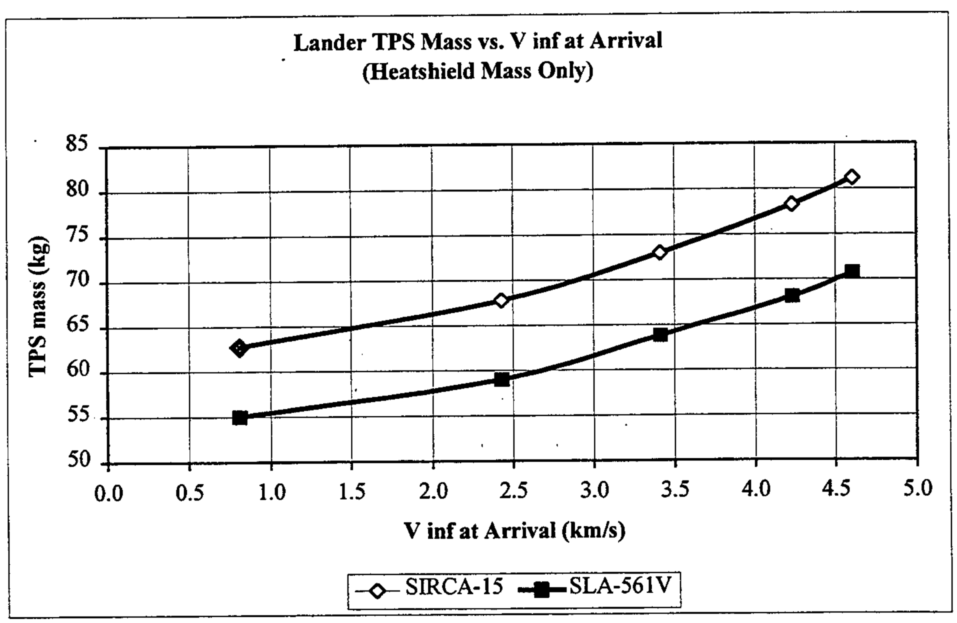

Figure 5.12: Scenario 1 – FIAT TPS Mass (Lander) vs. $V_{\infty_A}$

Using this polynomial, a TPS mass vs. lander launch period plot was generated using the corresponding values of $V_{\infty_A}$ (4.3-5.2 km/s) for each date. Figure 5.13 shows the result after including the backshell mass and the safety margin.

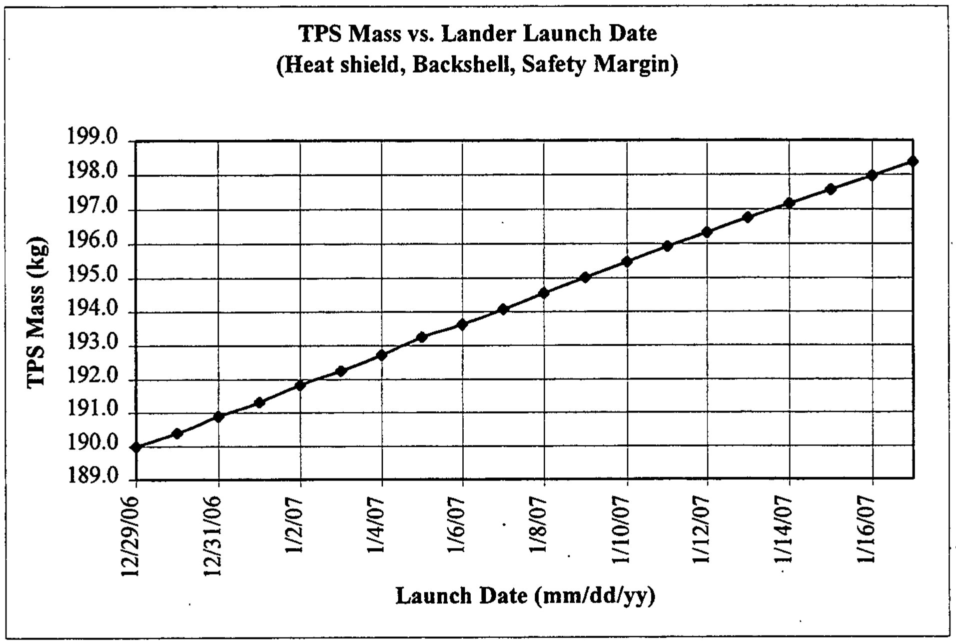


Figure 5.13: Scenario 1 -Lander TPS Mass vs. Launch Date

As the shape of the heat shield model used in FIAT was a biconic, the leading frustum encountered much higher values of $\dot{Q}_{MAX}$ than would be expected from a cone-sphere shaped heat shield like that used on Pathfinder. FIAT was run using a Pathfinder model and although the trajectory was successfully calculated (and thus a $\dot{Q}_{MAX}$), a TPS mass solution could not be obtained. Comparing the $\dot{Q}_{MAX}$ values from the Pathfinder model vs. the $\dot{Q}_{MAX}$ values found using the biconic resulted in $\dot{Q}_{MAX}$ values for the biconic roughly 2.5 times those for the cone-sphere. In addition, the TPS mass was calculated using the area of the heat shield. The area for the Pathfinder model vs. the biconic shaped heat shield is also roughly 2.5. Therefore, the effects cancel and the TPS mass vs. $V_{\infty_A}$ plot can still be used as a gross approximation of the lander TPS mass.

Chapter II addressed the derivation of the ballistic landing region in the DAP section. Figure 5.14 shows a plot of accessible latitudes for the lander given the

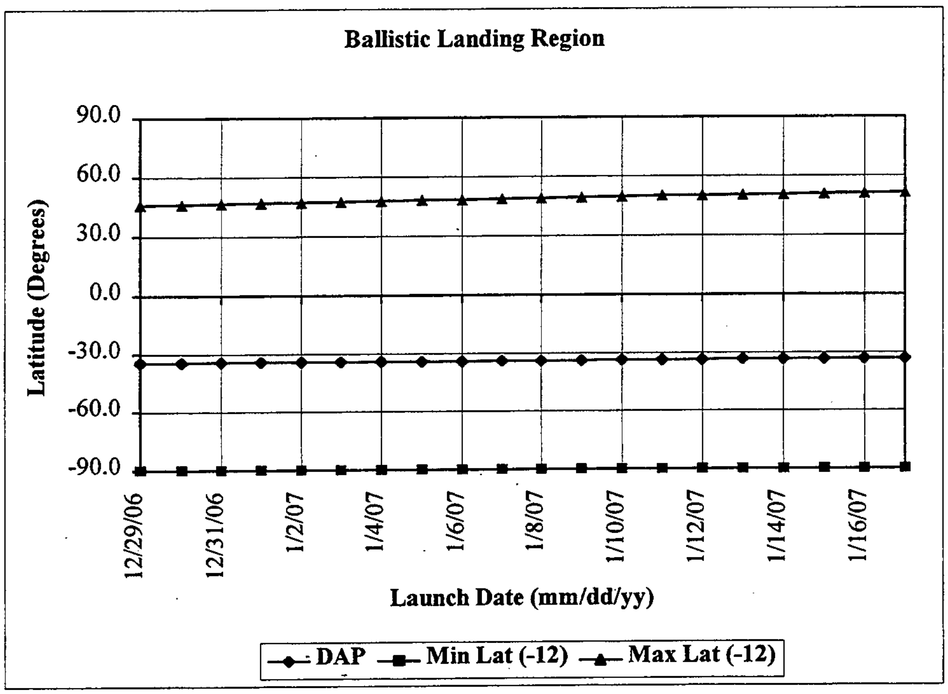


Figure 5.14: Scenario 1 - Ballistic Landing Region

trajectory parameters from CATO. For purposes of this calculation, the flight path angle was chosen to be –12°. The lander will most likely target the equatorial region (30° N - 15° S) to minimize the impact of the seasonal temperatures on the lander design. However, science data from previous missions may facilitate a need to land elsewhere. For this scenario, the lander will arrive at Mars at the beginning of Southern summer.

## D. SCENARIO 2: LAUNCH PERIODS, TRAJECTORY ANALYSIS PARAMETERS, GEOMETRY

This scenario is almost identical to scenario 1 with the exception of the orbiter capture scheme. In this case, the orbiter uses aerocapture to enter into orbit about Mars. The launch periods for both the orbiter and lander remain unchanged as do all of the trajectory parameters and associated geometries. Figures 5.1-5.10 are still applicable and can be referenced for this scenario as needed. However, these plots will not be shown again in this section.

## E. SCENARIO 2: ANALYSIS

The results for the lander analysis in this scenario are also the same as nothing has changed for the lander. However, the orbiter capture scheme in this scenario warrants discussion. Recall that aerocapture uses a single deep maneuver into the atmosphere to reduce the velocity of the incoming trajectory. The atmospheric drag slows the spacecraft to the point that upon exit from the atmosphere, only a modest amount of propellant is required to circularize the orbit. The consequences of using aerocapture are manifested in an increase in the TPS mass required to protect the spacecraft from the large heat increase experienced by the orbiter during the pass through the atmosphere.

Estimating the TPS mass requirement for the orbiter during aerocapture is even more challenging than for the lander because the NASA Ames Integrated Design System was designed for entry vehicles and does not address aerocapture simulation. However, the Aerocapture Simulation (ACAPS) (Leszczynski, 1998) provides a capability to calculate a $\dot{Q}_{MAX}$ value for an incoming trajectory. State vectors from CATO generated type 4- trajectories were used as inputs to the ACAPS simulation. It should be noted that the state vectors obtained from CATO were originally referenced to the Mars equator and equinox of epoch for mission design considerations. The format of the state vector to be used as an ACAPS input however, needs to be referenced to the body equator and meridian of date coordinate frame to account for the rotating atmosphere. Therefore, a coordinate transformation using the IORB reference flags was necessary to format the state vector in the proper frame (IORB flag 42).

As previously stated, ACAPS determines a $\dot{Q}_{MAX}$ value for each trajectory. The ACAPS simulation for this analysis utilized a cone-sphere shaped heat shield with a nose radius of 1 meter and a base radius of 1.325 meters. To use the ACAPS $\dot{Q}_{MAX}$ data with the TPS vs. $\dot{Q}_{MAX}$ plot obtained previously in FIAT from the biconic model, it is necessary to scale the $\dot{Q}_{MAX}$ value and thus the TPS mass by 2.5 as discussed earlier. Note that in this case, there is no scale factor considered for the heat shield areas as ACAPS utilized the cone-sphere. Figure 5.15 shows a plot of TPS mass vs. $\dot{Q}_{MAX}$ as generated from FIAT data. Note that this data is for the TPS heat shield only and does not include the backshell. Estimation of the orbiter TPS mass would also include the backshell, a 25% safety margin, and the 2.5 scale factor for the $\dot{Q}_{MAX}$ ratio. Also note the two different TPS materials. As previously mentioned, the SLA-561V material was determined to not be suitable and will not be included in future plots. The data is then

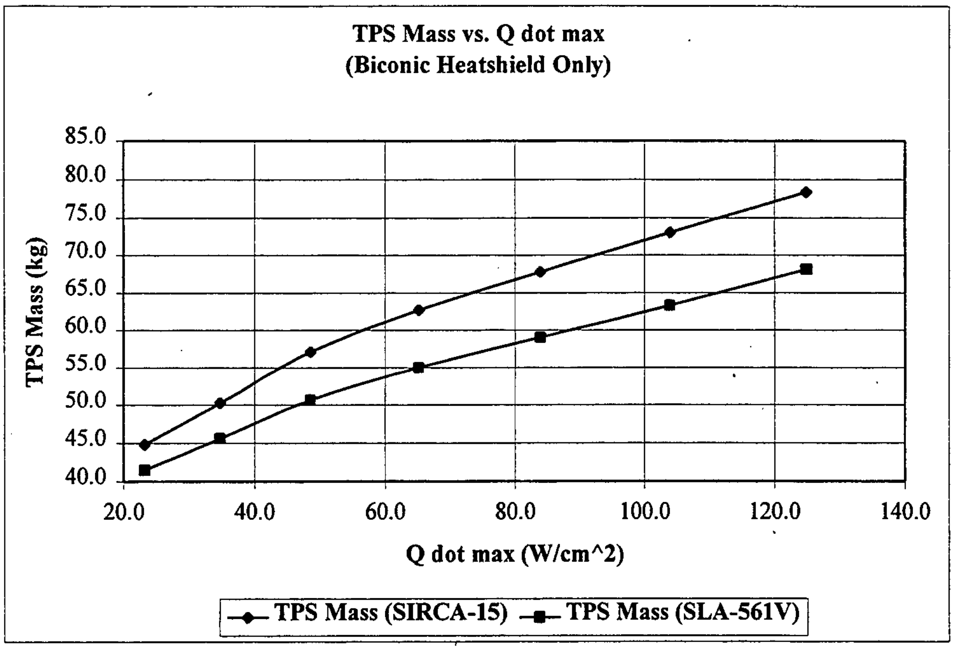


Figure 5.15: FIAT TPS Mass vs. Q dot max

curve fitted and the resulting polynomial is used to determine TPS mass over a wide range of $\dot{Q}_{MAX}$ values. As CATO trajectories reference $V_{\infty_A}$ (ACAPS uses $V_{entry}$), a more useful presentation of the data in the form of TPS mass vs. $V_{\infty_A}$ can be constructed. Figure 5.16 shows these results. For this plot, a 25% safety margin was also added to the FIAT generated data.

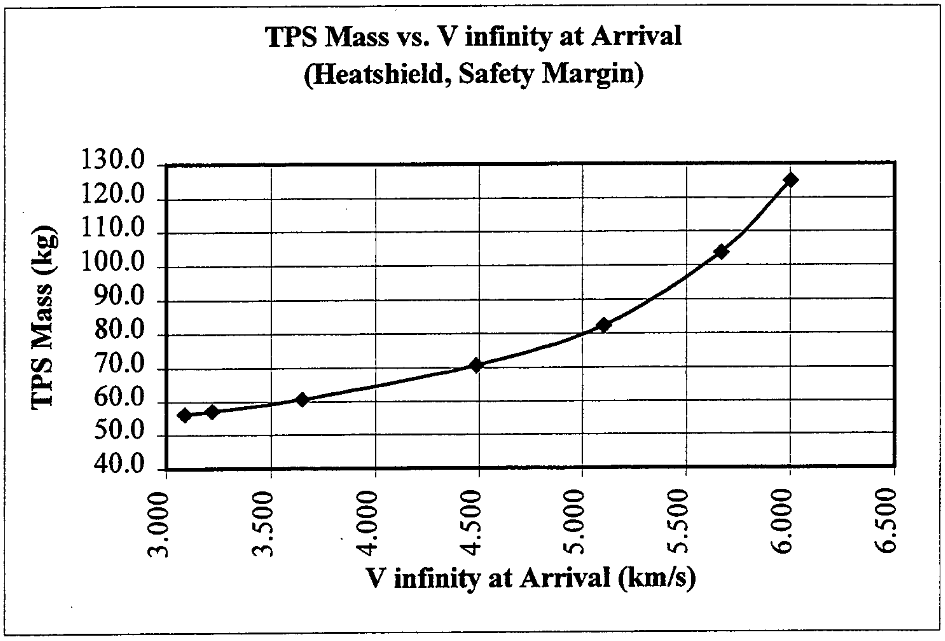


Figure 5.16: TPS Mass vs. $V_{\infty_A}$ (SIRCA-15)

Now that the TPS mass can be estimated for the orbiter during aerocapture, the post-capture orbiter mass for this scenario can be calculated. The assumptions made in calculating the post-capture mass are as follows:

1) The post-aerocapture $\Delta V$ required to circularize the orbit is equal to 130 km/s as adopted from the Mars 2001 orbiter;
2) The orbiter requires a cruise stage with a mass estimated to be 75 kg as taken from the Mars 2001 orbiter design.

The TPS mass for the orbiter used in the post-capture mass calculations includes the backshell mass (63 kg) from the Mars 2001 orbiter in addition to the FIAT generated data shown in Figure 5.16.

Determining whether this method of capture is more advantageous than propulsive capture is governed by comparing the additional TPS mass requirements to the propellant that would have been necessary for a propulsive capture. The criterion for comparison however, is still post-capture orbiter mass. Again for purposes of comparison, the launch vehicle used is a Delta II 7925. Other launch vehicle results are included in Appendix C. Table 5.3 demonstrates that the post-capture mass actually increases slightly during the launch period. This clearly shows that for aerocapture, the increase in $V_{\infty_A}$ during the launch period does not effect the TPS mass requirement as much as it effects the propellant requirement for the propulsive capture.

Table 5.3: Scenario 2 – Orbiter Post-Capture Mass (Delta II 7925)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (30 m/s) (kg) | Cruise Stage (kg) | TPS Mass (kg) | Post AC (kg) | Post AC Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 984.6 | 9.4 | 75 | 203.3 | 697.0 | 28.3 | 668.7 |
| 11/27/06 | 9.657 | 3.090 | 986.4 | 9.4 | 75 | 203.3 | 698.7 | 28.4 | 670.3 |
| 11/28/06 | 9.572 | 3.097 | 988.1 | 9.4 | 75 | 203.4 | 700.3 | 28.4 | 671.8 |
| 11/29/06 | 9.497 | 3.104 | 989.6 | 9.4 | 75 | 203.6 | 701.6 | 28.5 | 673.2 |
| 11/30/06 | 9.434 | 3.116 | 990.9 | 9.4 | 75 | 203.8 | 702.7 | 28.5 | 674.2 |
| 12/1/06 | 9.378 | 3.131 | 992.1 | 9.4 | 75 | 204.0 | 703.6 | 28.6 | 675.1 |
| 12/2/06 | 9.327 | 3.143 | 993.1 | 9.4 | 75 | 204.2 | 704.4 | 28.6 | 675.8 |
| 12/3/06 | 9.284 | 3.163 | 994.0 | 9.5 | 75 | 204.6 | 705.0 | 28.6 | 676.3 |
| 12/4/06 | 9.245 | 3.189 | 994.8 | 9.5 | 75 | 205.1 | 705.3 | 28.6 | 676.7 |
| 12/5/06 | 9.207 | 3.215 | 995.6 | 9.5 | 75 | 205.5 | 705.6 | 28.6 | 677.0 |
| 12/6/06 | 9.171 | 3.246 | 996.4 | 9.5 | 75 | 206.1 | 705.8 | 28.6 | 677.2 |
| 12/7/06 | 9.135 | 3.275 | 997.1 | 9.5 | 75 | 206.7 | 705.9 | 28.6 | 677.3 |
| 12/8/06 | 9.101 | 3.312 | 997.8 | 9.5 | 75 | 207.4 | 705.9 | 28.6 | 677.3 |
| 12/9/06 | 9.068 | 3.354 | 998.5 | 9.5 | 75 | 208.2 | 705.8 | 28.6 | 677.1 |
| 12/10/06 | 9.035 | 3.399 | 999.2 | 9.5 | 75 | 209.2 | 705.5 | 28.6 | 676.9 |
| 12/11/06 | 9.002 | 3.438 | 999.9 | 9.5 | 75 | 210.0 | 705.4 | 28.6 | 676.8 |
| 12/12/06 | 8.960 | 3.487 | 1000.8 | 9.5 | 75 | 211.1 | 705.2 | 28.6 | 676.6 |
| 12/13/06 | 8.920 | 3.522 | 1001.6 | 9.5 | 75 | 211.8 | 705.2 | 28.6 | 676.6 |
| 12/14/06 | 8.889 | 3.570 | 1002.2 | 9.5 | 75 | 212.9 | 704.7 | 28.6 | 676.1 |
| 12/15/06 | 8.876 | 3.647 | 1002.5 | 9.5 | 75 | 214.7 | 703.2 | 28.5 | 674.7 |

## F. SCENARIO 3: LAUNCH PERIODS AND TRAJECTORY ANALYSIS PARAMETERS

The orbiter launch period for this scenario remains unchanged for reasons explained earlier. The aerobraking capture scheme employed by the orbiter in this case creates some uncertainty in the determination of the launch period for the lander. Both spacecraft again travel on the same trajectory, but the time required to aerobrake as shown in Table 4.1 directly effects the earliest lander arrival time. Consequently, the lander launch period analyzed for this scenario is roughly two months long to give consideration to several aerobraking options. Figure 5.17 shows the launch dates of the orbiter and lander as plotted against the C3 contour for the type 4- trajectory. The transfer time for the orbiter is 789-808 days while the transfer times for the lander are

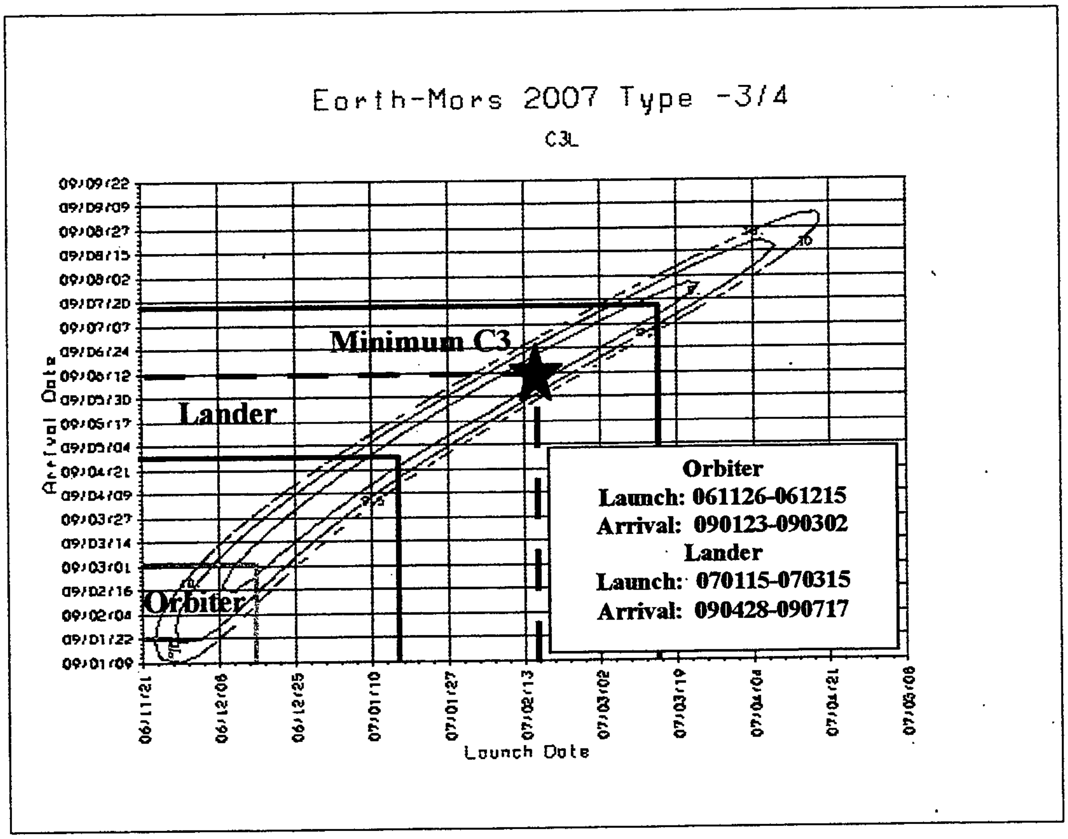


Figure 5.17: Scenario 3 – Orbiter & Lander Launch Periods

834-855 days depending on the launch date. Using Table 4.2, the minimum and maximum times available to the orbiter for aerobraking are 57 and 175 days respectively.

The trajectory parameters for the orbiter are the same as scenarios 1 and 2. However, the trajectory parameters for the lander have changed considerably. Note the C3 and $V_{\infty_A}$ vs. lander launch date plots in Figures 5.18 and 5.19. Although the C3 plot for the lander appears to be quite oscillatory, a closer look at the scale utilized indicates that the C3 can be considered constant over the launch period. The $V_{\infty_A}$ plot shows a

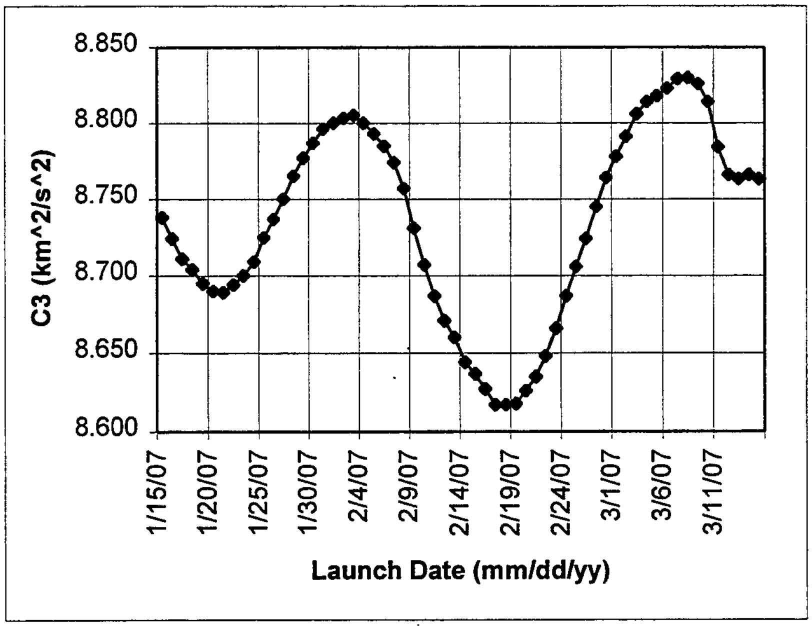


Figure 5.18: Scenario 3 - C3 vs. Launch Date (Lander)

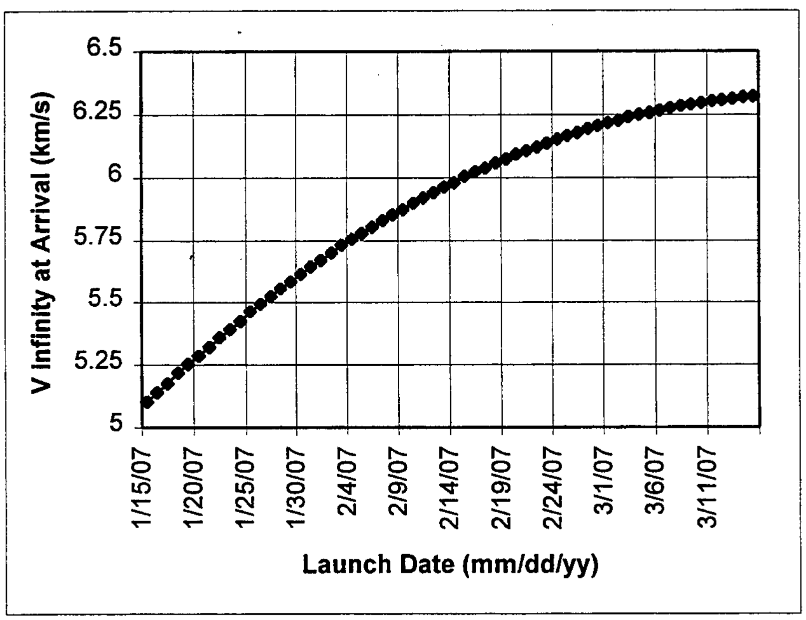


Figure 5.19: Scenario 3 - $V_{\infty_A}$ vs. Launch Date (Lander)

steady increase for later launch dates making it once again appear more desireable to launch the lander as early as possible (given the aerobraking constraints).

## G. SCENARIO 3: GEOMETRY

The geometric relationships between the lander and the other bodies of interest are shown in Figure 5.20. The Kplot representation for the lander in this case corresponds to an 01/15/07 launch date. Also, the spacecraft-Earth and spacecraft-Sun distances as well as the Sun-spacecraft-Earth angle are provided in Figures 5.21 and 5.22 respectively.

**Earth-Mars 2007 Type 4- Trajectory (Lander)**

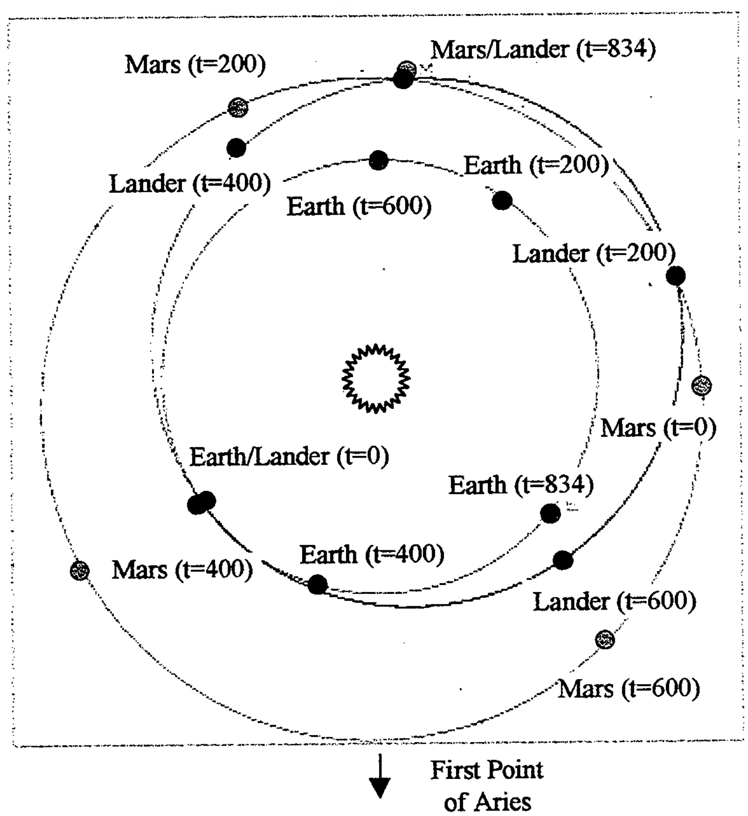


Figure 5.20: Kplot for Lander Type 4- Trajectory (070115 Launch)

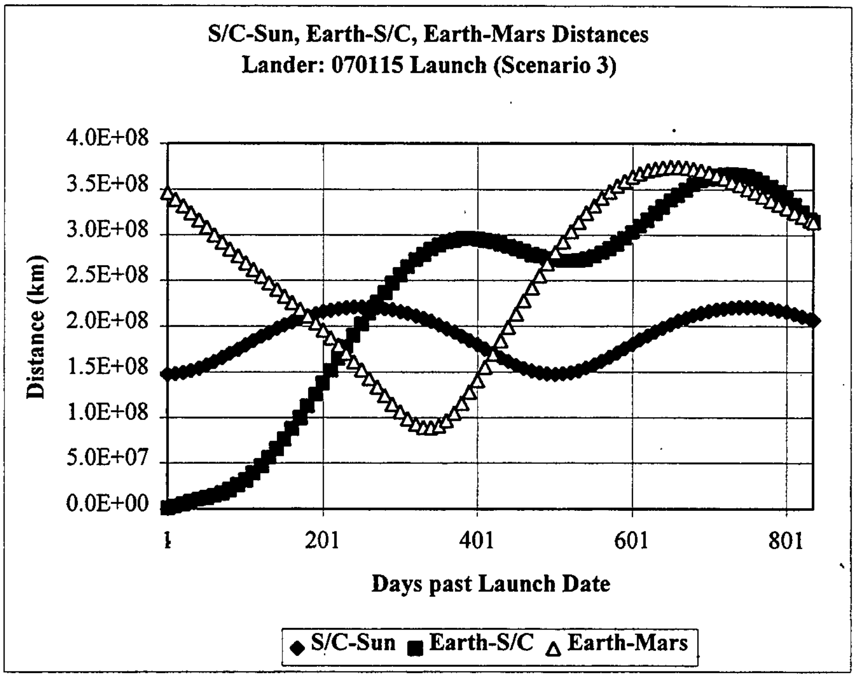


Figure 5.21: Lander Distances (070115 Launch)

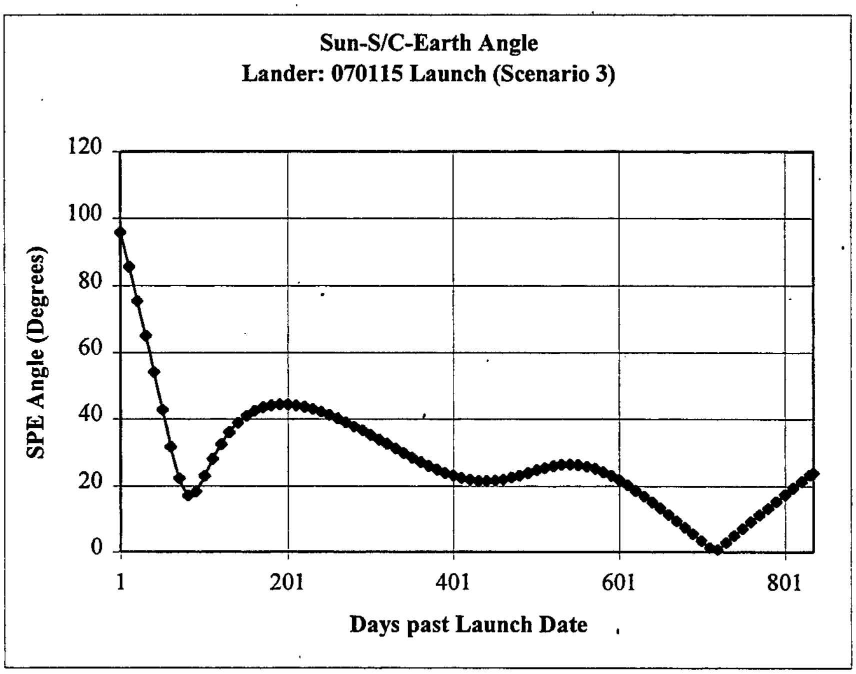


Figure 5.22: Sun-Lander-Earth Angle (070115 Launch)

## H. SCENARIO 3: ORBITER ANALYSIS

Estimation of the time required for aerobraking is important in determining whether this capture scheme is more cost effective for the mission than the other two capture schemes. Using Figure 5.23, simplification of the process to estimate the time is shown in the following derivation (Ross, 1998).

Given a period for the initial capture orbit, the semi-major axis can be determined using the standard equation:

$$P = 2\pi\sqrt{\frac{a^3}{\mu}} \qquad (4.3)$$

a = semi – major axis

$\mu$ = Mars gravitational constant

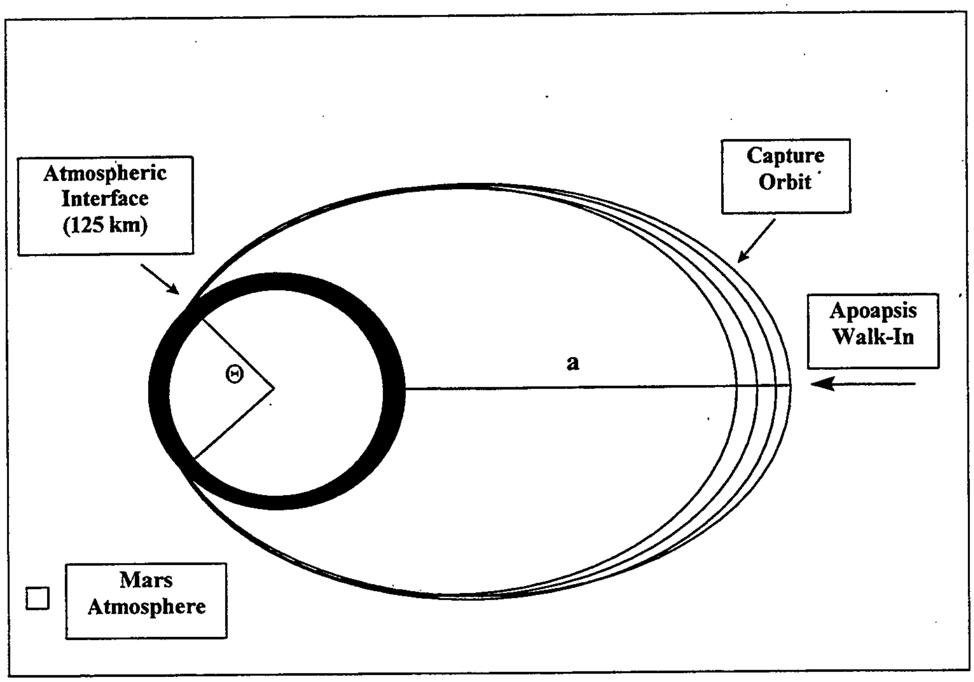


Figure 5.23: Scenario 3 - Aerobraking Diagram

From the trajectory equation, solve for the eccentricity of the initial orbit:

$$r_p = a(1-e) \qquad (4.4)$$

$r_p$ = periapsis radius

As aerobraking is initiated by conducting a series of burns at apoapsis to lower periapsis into the Mars atmosphere, the true anomaly between the atmospheric interface-elliptical orbit intersection point and periapsis can be calculated using a different form of the trajectory equation:

$$r = \frac{a(1-e^2)}{1+e\cos\nu} \qquad (4.5)$$

$$\nu = \cos^{-1}\left(\frac{\frac{a(1-e^2)}{r_E} - 1}{e}\right) \qquad (4.6)$$

$\nu$ = true anomaly
$r_E$ = entry interface radius

The total angle traversed by the spacecraft while within the atmosphere $(\Theta)$ is equal to twice the true anomaly calculated above. The velocity of the spacecraft while in the atmosphere is important in determining the effect of the atmosphere on the energy reduction of the orbit. Assuming that the orbit is symmetrical about periapsis, the velocity of the spacecraft at each interface point is the same. Therefore, the following equation can be used to solve for the velocity at those points:

$$V_1 = V_2 = \sqrt{\frac{2\mu}{r_E} - \frac{\mu}{a}} \qquad (4.7)$$

From the Mean Value Theorem (Finney and Thomas, 1993, p.298),

$$\int_{x_1}^{x_2} f(x)dx = f(x^*)(x_2 - x_1) \qquad (4.8)$$

a function integrated over a definite period can be estimated by simply multiplying the average function by the interval. If the angle $\nu$ is small then the velocity can be considered constant while within the atmosphere. Next the work-energy theorem can be used to determine the work done by the atmospheric drag. This work done is equal to the change in the energy during each aerobraking pass.

$$W = \int_0^{\Theta} D\, ds = \Delta E \qquad (4.9)$$

$$D = \text{Drag Force} = \frac{1}{2}\rho V^2 A C_D \qquad (4.10)$$

$\rho$ = atmospheric density
$A$ = Surface area of spacecraft exposed to drag force
$C_D$ = Coefficient of Drag
m = Spacecraft mass
V = Spacecraft Velocity

Substituting yields the following equation:

$$m\Delta\varepsilon = \frac{1}{2} A C_D \rho(r) \int_0^{\Theta} V^2 ds \qquad (4.11)$$

$\Delta\varepsilon$ = Change in the specific orbital energy

where for purposes of simplification, the altitude of the spacecraft within the atmosphere is considered constant and the atmospheric density used is based on a spherical model. Letting $ds = r\, d\Theta$ and substituting in a constant velocity, V, provides a basic equation whereby the change in the energy during each aerobraking pass can be estimated.

$$m\Delta\varepsilon = \frac{1}{2} A C_D \rho(r) V^2 r_p \Theta \qquad (4.12)$$

The energy equation allows the initial orbit energy to be calculated,

$$\varepsilon = -\frac{\mu}{2a} \tag{4.13}$$

On each aerobraking pass, the MATLAB routine shown in Appendix D, iteratively calculates the change in the energy attributed to the atmosphere. Assuming that the periapsis remains constant throughout the aerobraking phase, this change in energy results in apoapsis decay. Once apoapsis reaches a pre-determined altitude (~2000 km), aerobraking is terminated and the final orbit is circularized propulsively.

The following input parameters are necessary for the MATLAB code to run:

1) An average density of 2.907e-8 $kg/m^3$ was taken from the Mars atmospheric model MARSGRAM for a periapsis altitude of 110 km. The density has been known to vary by three orders of magnitude so the time estimate for completion of aerobraking will not be quite accurate;

2) The coefficient is specific to each spacecraft, but for purposes of analysis here, a value of 2 was assigned;

3) The spacecraft area perpendicular to the drag force was estimated to be 15 $m^2$ based on an area of 17 $m^2$ for MGS and 11 $m^2$ for Mars Surveyor 98.

Figure 5.24 was generated using the output data from MATLAB. The aerobraking time vs. orbiter mass plot enables an aerobraking time to be calculated for each orbiter launch date in the launch period. Note that there are three initial orbit periods in Figure 5.24. This provides greater flexibility in determining the launch date of the lander. The

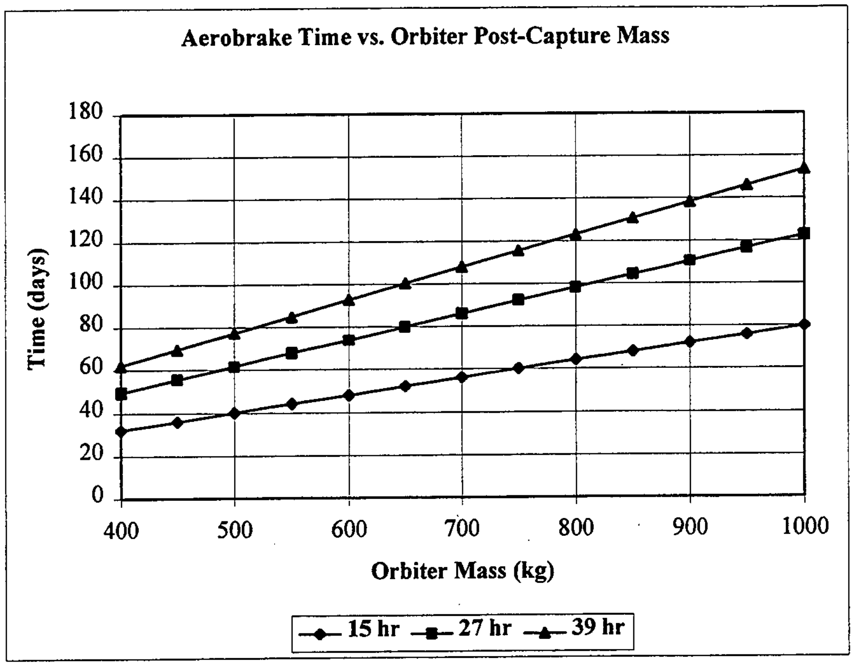


Figure 5.24: Aerobraking Time Estimate

aerobraking times with respect to the initial orbits are shown in Table 5.4. This table

Table 5.4: Scenario 3 - Orbiter Aerobraking Time

| AB Time Launch | 15 hr orbit (days) | 27 hr orbit (days) | 39 hr orbit (days) |
|---|---|---|---|
| 11/26/06 | 53.1 | 84.2 | 107.1 |
| 11/27/06 | 53.2 | 84.4 | 107.3 |
| 11/28/06 | 53.2 | 84.4 | 107.3 |
| 11/29/06 | 53.2 | 84.4 | 107.4 |
| 11/30/06 | 53.2 | 84.4 | 107.3 |
| 12/1/06 | 53.1 | 84.3 | 107.2 |
| 12/2/06 | 53.0 | 84.2 | 107.0 |
| 12/3/06 | 52.9 | 83.9 | 106.8 |
| 12/4/06 | 52.7 | 83.6 | 106.4 |
| 12/5/06 | 52.5 | 83.3 | 106.0 |
| 12/6/06 | 52.3 | 82.9 | 105.5 |
| 12/7/06 | 52.1 | 82.6 | 105.0 |
| 12/8/06 | 51.7 | 82.0 | 104.3 |
| 12/9/06 | 51.4 | 81.5 | 103.7 |
| 12/10/06 | 51.0 | 80.9 | 102.9 |
| 12/11/06 | 50.7 | 80.4 | 102.3 |
| 12/12/06 | 50.3 | 79.8 | 101.5 |
| 12/13/06 | 50.0 | 79.3 | 100.9 |
| 12/14/06 | 49.6 | 78.7 | 100.0 |
| 12/15/06 | 49.1 | 77.9 | 99.1 |

depicts shorter aerobraking times for later launches. If propulsive capture is in fact the most cost effective method to enter into orbit about Mars, the trajectory selection process becomes a series of trades whereby the magnitude of the MOI maneuver, the time to aerobrake, and the trajectory parameters for the lander are considered.

To calculate the post-capture orbiter mass when aerobraking is used, the following assumptions were made:

1) The propellant mass attributed to orbit trim maneuvers was estimated to be 50 kg. This was based on the Mars Global Surveyor mass budget and is equivalent to a $\Delta V$ of 250 m/s for a 650 kg spacecraft;

2) The orbiter does not have a cruise stage when aerobraking;

Post-capture mass calculations are again based on a Delta II 7925 launch vehicle with the results presented in Table 5.5. Note that the decrease in post-capture mass for later launch dates is responsible for the shorter aerobraking times at the end of the launch period. Again, because the MOI maneuver is largely dependent on the period of the initial capture orbit, three initial orbit periods were chosen for comparison purposes. As expected, the results show that capturing into a longer period orbit requires less propellant.

Table 5.5: Scenario 3 - Orbiter Post-Capture Mass (Delta II 7925)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 984.6 | 9.4 | 1.207 | 1.105 | 1.059 |
| 11/27/06 | 9.656 | 3.090 | 986.4 | 9.4 | 1.209 | 1.106 | 1.060 |
| 11/28/06 | 9.572 | 3.097 | 988.1 | 9.4 | 1.213 | 1.110 | 1.064 |
| 11/29/06 | 9.497 | 3.104 | 989.6 | 9.4 | 1.217 | 1.114 | 1.068 |
| 11/30/06 | 9.433 | 3.116 | 990.9 | 9.4 | 1.223 | 1.120 | 1.074 |
| 12/1/06 | 9.378 | 3.130 | 992.1 | 9.4 | 1.231 | 1.128 | 1.082 |
| 12/2/06 | 9.327 | 3.143 | 993.1 | 9.4 | 1.238 | 1.135 | 1.089 |
| 12/3/06 | 9.284 | 3.163 | 994.0 | 9.5 | 1.249 | 1.146 | 1.100 |
| 12/4/06 | 9.245 | 3.189 | 994.8 | 9.5 | 1.263 | 1.160 | 1.114 |
| 12/5/06 | 9.207 | 3.215 | 995.6 | 9.5 | 1.277 | 1.174 | 1.128 |
| 12/6/06 | 9.171 | 3.246 | 996.4 | 9.5 | 1.294 | 1.191 | 1.146 |
| 12/7/06 | 9.135 | 3.274 | 997.1 | 9.5 | 1.310 | 1.207 | 1.161 |
| 12/8/06 | 9.102 | 3.315 | 997.8 | 9.5 | 1.333 | 1.230 | 1.184 |
| 12/9/06 | 9.068 | 3.354 | 998.5 | 9.5 | 1.355 | 1.252 | 1.206 |
| 12/10/06 | 9.035 | 3.399 | 999.2 | 9.5 | 1.381 | 1.278 | 1.232 |
| 12/11/06 | 8.997 | 3.437 | 1000.0 | 9.5 | 1.402 | 1.300 | 1.254 |
| 12/12/06 | 8.960 | 3.486 | 1000.8 | 9.5 | 1.431 | 1.328 | 1.282 |
| 12/13/06 | 8.920 | 3.522 | 1001.6 | 9.5 | 1.452 | 1.349 | 1.303 |
| 12/14/06 | 8.889 | 3.570 | 1002.2 | 9.5 | 1.480 | 1.377 | 1.332 |
| 12/15/06 | 8.865 | 3.621 | 1002.7 | 9.5 | 1.511 | 1.408 | 1.362 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 311.5 | 289.4 | 279.3 | 50 | 663.7 | 685.8 | 695.9 |
| 11/27/06 | 312.4 | 290.3 | 280.2 | 50 | 664.6 | 686.7 | 696.8 |
| 11/28/06 | 313.7 | 291.6 | 281.5 | 50 | 665.0 | 687.1 | 697.2 |
| 11/29/06 | 315.0 | 292.9 | 282.8 | 50 | 665.2 | 687.3 | 697.4 |
| 11/30/06 | 316.8 | 294.7 | 284.6 | 50 | 664.7 | 686.8 | 696.9 |
| 12/1/06 | 318.8 | 296.7 | 286.6 | 50 | 663.9 | 686.0 | 696.1 |
| 12/2/06 | 320.6 | 298.5 | 288.4 | 50 | 663.0 | 685.1 | 695.2 |
| 12/3/06 | 323.2 | 301.2 | 291.1 | 50 | 661.4 | 683.4 | 693.4 |
| 12/4/06 | 326.4 | 304.5 | 294.5 | 50 | 658.9 | 680.8 | 690.8 |
| 12/5/06 | 329.7 | 307.9 | 297.9 | 50 | 656.4 | 678.3 | 688.3 |
| 12/6/06 | 333.6 | 311.8 | 301.9 | 50 | 653.4 | 675.1 | 685.1 |
| 12/7/06 | 337.0 | 315.4 | 305.5 | 50 | 650.6 | 672.2 | 682.1 |
| 12/8/06 | 342.0 | 320.5 | 310.7 | 50 | 646.3 | 667.8 | 677.6 |
| 12/9/06 | 346.8 | 325.4 | 315.6 | 50 | 642.2 | 663.6 | 673.4 |
| 12/10/06 | 352.3 | 331.1 | 321.4 | 50 | 637.4 | 658.6 | 668.3 |
| 12/11/06 | 357.0 | 335.9 | 326.3 | 50 | 633.5 | 654.6 | 664.2 |
| 12/12/06 | 363.0 | 342.1 | 332.5 | 50 | 628.3 | 649.2 | 658.8 |
| 12/13/06 | 367.5 | 346.7 | 337.2 | 50 | 624.6 | 645.4 | 654.9 |
| 12/14/06 | 373.3 | 352.7 | 343.3 | 50 | 619.4 | 640.0 | 649.4 |
| 12/15/06 | 379.5 | 359.0 | 349.7 | 50 | 613.7 | 634.1 | 643.5 |

Now that a post-capture orbiter mass for each capture scheme has been calculated, a comparison can be made to determine which method is most cost effective. Figure 5.25 plots the post-capture orbiter masses for each capture scheme vs. the launch period. It appears that for most of the launch period, aerobraking is the most cost effective capture scheme. Note however, the decrease in the post-capture mass for the cases of propulsive capture and aerobraking near the end of the launch period. The effect of the increasing $V_{\infty_A}$ begins to outweigh the impact of the decreasing C3 during the launch period. The almost constant post-capture mass when using aerocapture however, indicates that the $V_{\infty_A}$ does not have the same effect for later launch dates. In this instance, the effects of

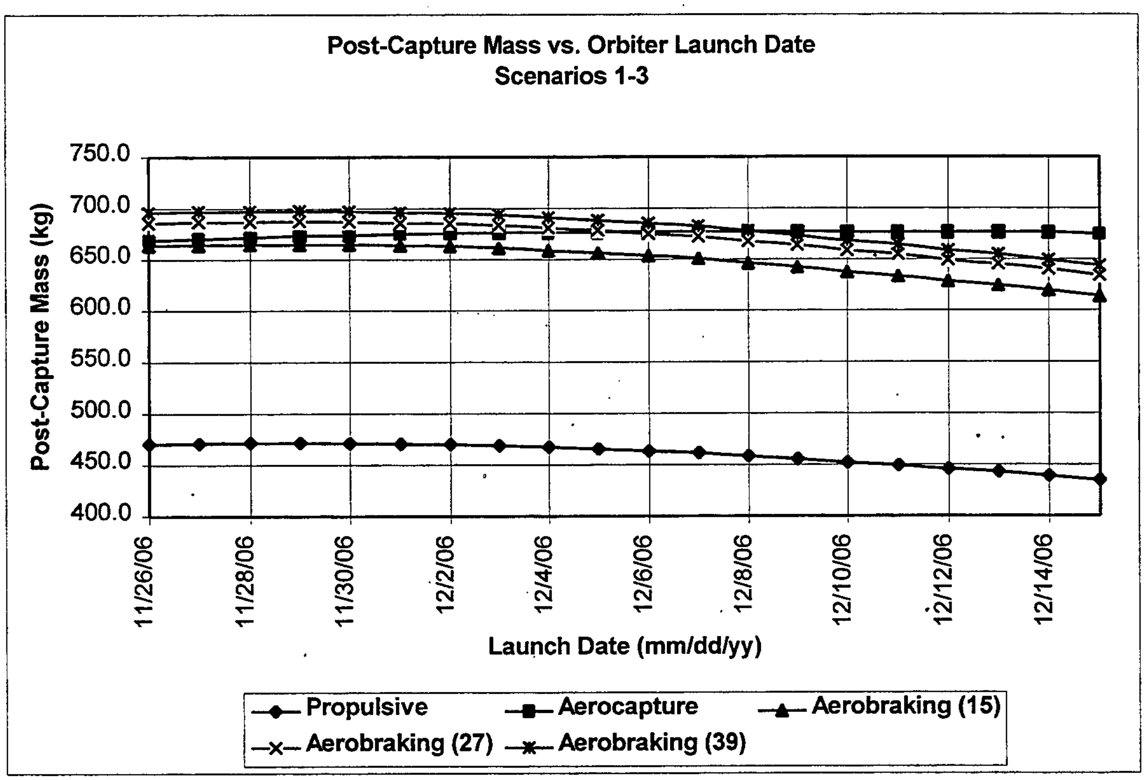


Figure 5.25: Scenarios 1,2,3 – Orbiter Post-Capture Mass

the decreasing C3 and increasing $V_{\infty_A}$ appear to cancel. This result should be intuitive as the orbiter TPS mass estimate varied only a few kilograms during the launch period while the injected mass due to the C3 also only varied a few kilograms.

## I. SCENARIO 3: LANDER ANALYSIS

If aerobraking is determined to be the best capture scheme for the mission, then the lander launch period will be determined by calculating the earliest lander arrival date as suggested in Table 4.1. However, as three different initial orbits are shown in Figure 4.27, it is first necessary to compare them. The criterion chosen for comparison will be the landed mass. Determination of landed mass utilizes the injected mass as the initial lander mass. The propellant required for TCMs, the jettisoned cruise stage, and the jettisoned aeroshell will be subtracted to obtain a landed mass. Prior to these calculations however, a TPS mass estimation will be calculated and a ballistic landing region determined. Using the FIAT data from Figure 4.15, the TPS mass estimation for the lander is shown in Figure 5.26. The Mars '01 backshell and a 25% safety margin is also included. Note that the trend of the TPS mass closely matches the trend of the $V_{\infty_A}$ data

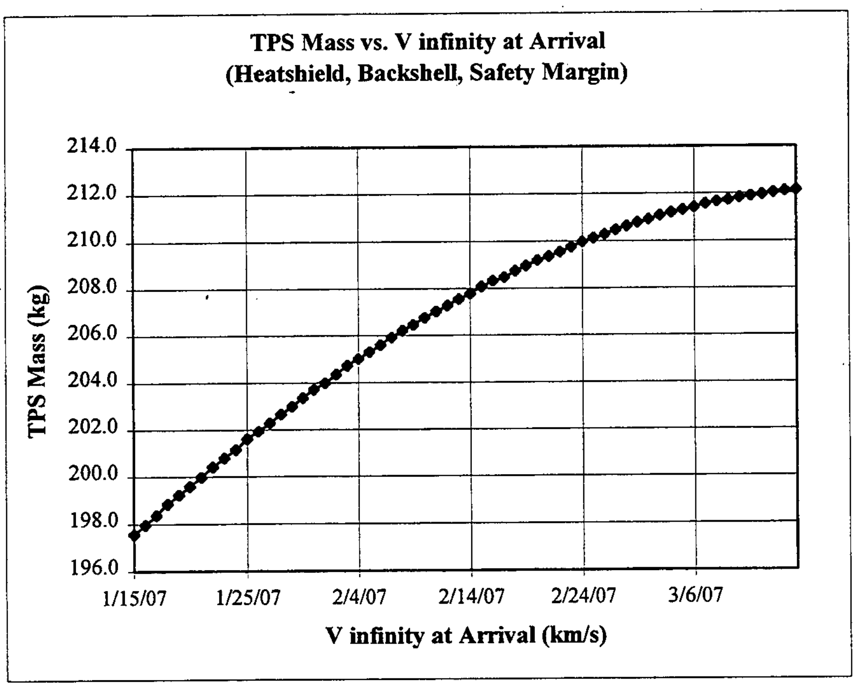


Figure 5.26: Scenario 3 – Lander TPS Mass vs. Launch Date

over this extended launch period. The ballistic landing region as shown in Figure 5.27, will accommodate a landing on most of the Martian surface with the exception of the most northern latitudes.

As the time required for the orbiter to aerobrake from initial orbits of 15, 27, and 39 hours has already been established, the lander launch periods can be determined.

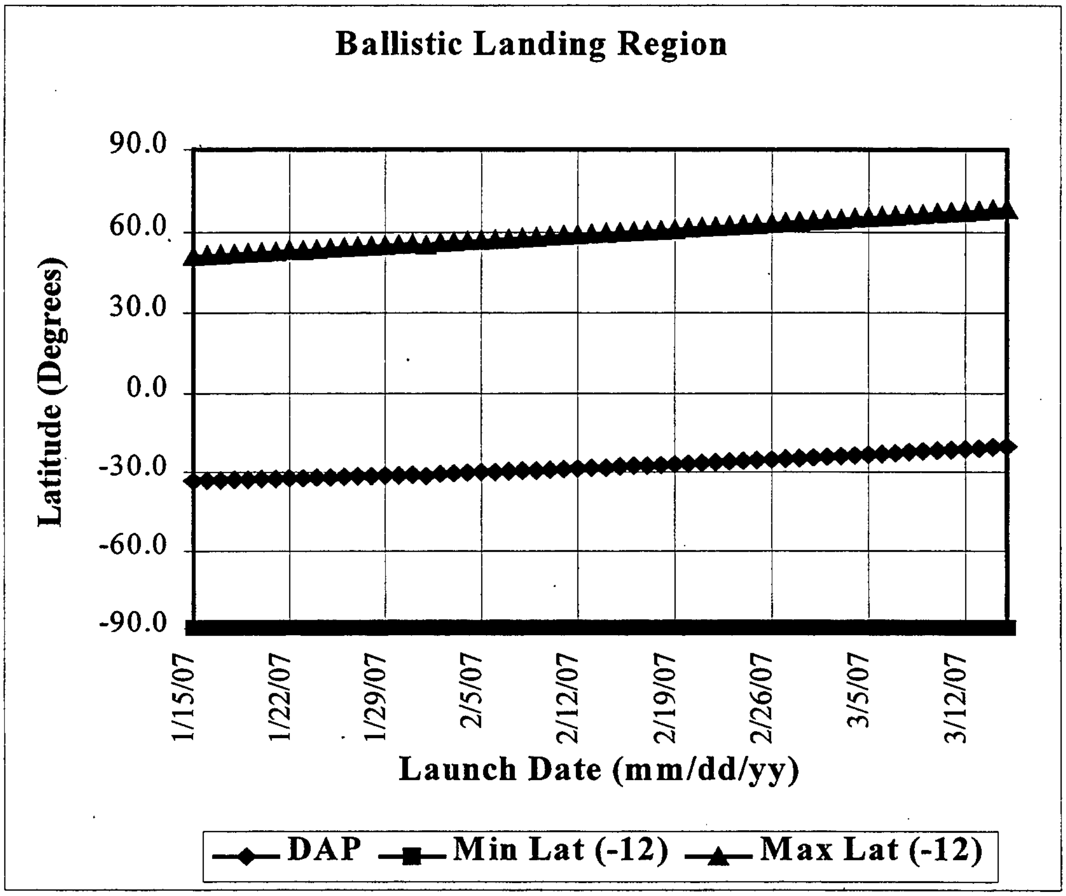


Figure 5.27: Scenario 3 - Ballistic Landing Region

Recall from Table 4.1 that the earliest lander arrival time is equal to the latest orbiter arrival time plus the time required for aerobraking. Table 5.6 shows the calculation of the earliest lander arrival dates for each case. The latest orbiter arrival date which corresponds to the last orbiter launch date in this case is used to determine the earliest lander launch date by referencing the CATO trajectory data for the lander in Appendix A. The three different lander launch periods are shown in Table 5.7. Again, using a Delta II 7925 launch vehicle, the landed mass can be determined for each launch period in the manner previously described. The results are presented in Table 5.8.

Table 5.6: Calculations for Earliest Lander Launch Date

| Orbiter Launch | Arrival | 15 hr orbit (days) | 27 hr orbit (days) | 39 hr orbit (days) | Earliest Lander Launch (15) | (27) | (39) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 1/23/09 | 53 | 84 | 107 | 3/17/09 | 4/17/09 | 5/10/09 |
| 11/27/06 | 1/25/09 | 53 | 84 | 107 | 3/19/09 | 4/19/09 | 5/12/09 |
| 11/28/06 | 1/26/09 | 53 | 84 | 107 | 3/20/09 | 4/20/09 | 5/13/09 |
| 11/29/06 | 1/28/09 | 53 | 84 | 107 | 3/22/09 | 4/22/09 | 5/15/09 |
| 11/30/06 | 1/30/09 | 53 | 84 | 107 | 3/24/09 | 4/24/09 | 5/17/09 |
| 12/1/06 | 1/31/09 | 53 | 84 | 107 | 3/25/09 | 4/25/09 | 5/18/09 |
| 12/2/06 | 2/2/09 | 53 | 84 | 107 | 3/27/09 | 4/27/09 | 5/20/09 |
| 12/3/06 | 2/4/09 | 53 | 84 | 107 | 3/28/09 | 4/28/09 | 5/21/09 |
| 12/4/06 | 2/6/09 | 53 | 84 | 106 | 3/30/09 | 4/30/09 | 5/23/09 |
| 12/5/06 | 2/8/09 | 53 | 83 | 106 | 4/1/09 | 5/2/09 | 5/24/09 |
| 12/6/06 | 2/10/09 | 52 | 83 | 105 | 4/3/09 | 5/3/09 | 5/26/09 |
| 12/7/06 | 2/12/09 | 52 | 83 | 105 | 4/5/09 | 5/5/09 | 5/28/09 |
| 12/8/06 | 2/15/09 | 52 | 82 | 104 | 4/7/09 | 5/8/09 | 5/30/09 |
| 12/9/06 | 2/17/09 | 51 | 82 | 104 | 4/9/09 | 5/9/09 | 5/31/09 |
| 12/10/06 | 2/19/09 | 51 | 81 | 103 | 4/11/09 | 5/10/09 | 6/1/09 |
| 12/11/06 | 2/21/09 | 51 | 80 | 102 | 4/12/09 | 5/12/09 | 6/3/09 |
| 12/12/06 | 2/24/09 | 50 | 80 | 101 | 4/15/09 | 5/14/09 | 6/5/09 |
| 12/13/06 | 2/25/09 | 50 | 79 | 101 | 4/15/09 | 5/15/09 | 6/5/09 |
| 12/14/06 | 2/28/09 | 50 | 79 | 100 | 4/18/09 | 5/17/09 | 6/8/09 |
| 12/15/06 | 3/2/09 | 49 | 78 | 99 | 4/20/09 | 5/18/09 | 6/9/09 |

Table 5.7: Lander Launch Periods

| Lander (15) | | Lander (27) | | Lander (39) | |
|---|---|---|---|---|---|
| Departure | Arrival | Departure | Arrival | Departure | Arrival |
| 1/15/07 | 90428 | 1/29/07 | 90519 | 2/13/07 | 90609 |
| 1/16/07 | 90429 | 1/30/07 | 90520 | 2/14/07 | 90610 |
| 1/17/07 | 90501 | 1/31/07 | 90522 | 2/15/07 | 90611 |
| 1/18/07 | 90503 | 2/1/07 | 90523 | 2/16/07 | 90613 |
| 1/19/07 | 90504 | 2/2/07 | 90524 | 2/17/07 | 90614 |
| 1/20/07 | 90506 | 2/3/07 | 90526 | 2/18/07 | 90615 |
| 1/21/07 | 90507 | 2/4/07 | 90527 | 2/19/07 | 90616 |
| 1/22/07 | 90509 | 2/5/07 | 90529 | 2/20/07 | 90618 |
| 1/23/07 | 90510 | 2/6/07 | 90530 | 2/21/07 | 90619 |
| 1/24/07 | 90511 | 2/7/07 | 90531 | 2/22/07 | 90620 |
| 1/25/07 | 90513 | 2/8/07 | 90602 | 2/23/07 | 90622 |
| 1/26/07 | 90515 | 2/9/07 | 90603 | 2/24/07 | 90623 |
| 1/27/07 | 90516 | 2/10/07 | 90604 | 2/25/07 | 90624 |
| 1/28/07 | 90517 | 2/11/07 | 90606 | 2/26/07 | 90625 |
| 1/29/07 | 90519 | 2/12/07 | 90607 | 2/27/07 | 90627 |
| 1/30/07 | 90520 | 2/13/07 | 90609 | 2/28/07 | 90628 |
| 1/31/07 | 90522 | 2/14/07 | 90610 | 3/1/07 | 90629 |
| 2/1/07 | 90523 | 2/15/07 | 90611 | 3/2/07 | 90701 |
| 2/2/07 | 90524 | 2/16/07 | 90613 | 3/3/07 | 90702 |
| 2/3/07 | 90526 | 2/17/07 | 90614 | 3/4/07 | 90703 |

Table 5.8: Landed Masses

| (15) Landed Mass | (27) Landed Mass | (39) Landed Mass |
|---|---|---|
| 823.8 | 807.3 | 794.3 |
| 822.6 | 806.3 | 793.7 |
| 821.3 | 805.3 | 792.8 |
| 819.8 | 804.4 | 792.1 |
| 818.7 | 803.4 | 791.7 |
| 817.6 | 802.3 | 790.9 |
| 816.4 | 801.5 | 790.3 |
| 815.1 | 800.7 | 789.7 |
| 814.0 | 799.8 | 789.3 |
| 812.9 | 799.0 | 788.7 |
| 811.5 | 798.1 | 788.2 |
| 810.5 | 797.4 | 787.6 |
| 809.5 | 796.5 | 787.1 |
| 808.3 | 795.8 | 786.7 |
| 807.3 | 795.1 | 786.2 |
| 806.3 | 794.3 | 785.7 |
| 805.3 | 793.7 | 785.3 |
| 804.4 | 792.8 | 784.9 |
| 803.4 | 792.1 | 784.5 |
| 802.3 | 791.7 | 784.2 |

As expected, the landed mass for the later lander launch dates is less due to the increased $V_{\infty_A}$ and subsequent increase in TPS mass requirements noted in Figure 5.26. Therefore, the shorter initial capture period for the orbiter results in a larger landed mass for the lander. However, in most instances, the mission objectives will drive these trades between the orbiter capture period and the landed mass.

## VI. TWO LAUNCH VEHICLES - SCENARIOS 4,5,6

### A. SCENARIO 4: LAUNCH PERIODS AND TRAJECTORY ANALYSIS PARAMETERS

The Earth-Mars transfer for the orbiter takes place on a type 2 trajectory in this scenario while the lander travels on a type 4- trajectory. The shorter flight time required of the type 2 trajectory enables the orbiter to launch much later than the lander and still arrive well before the earliest lander arrival. The orbiter employs a propulsive capture scheme in this case to enter into orbit about Mars. Figure 6.1 depicts the launch and arrival periods for the orbiter as plotted against a C3 contour. Figure 6.2 shows a similar

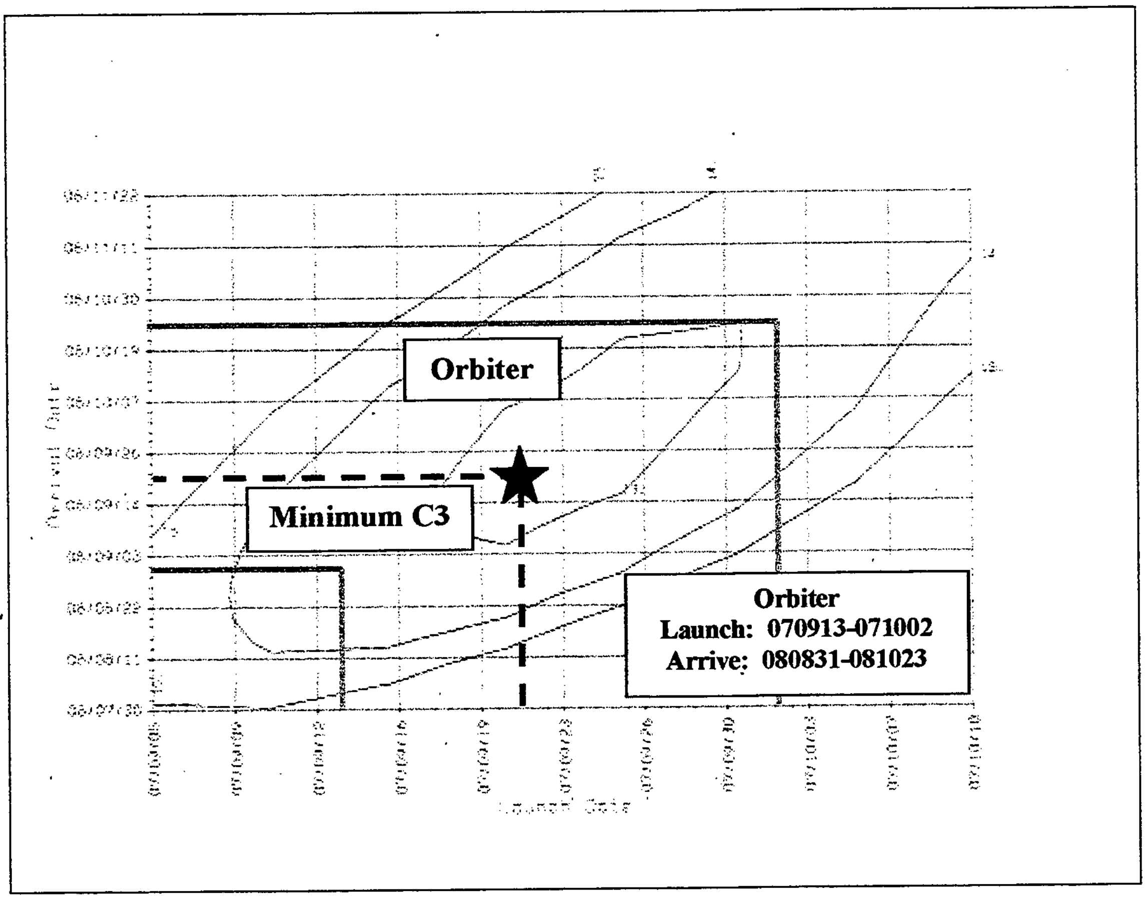


Figure 6.1: Scenario 4 – Orbiter Launch Period for Type 2 Trajectory

plot for the lander. The transfer time varies from 353-387 days for the orbiter and 789-809 days for the lander depending on the launch date. The time difference between the earliest lander arrival date and the latest orbiter arrival date

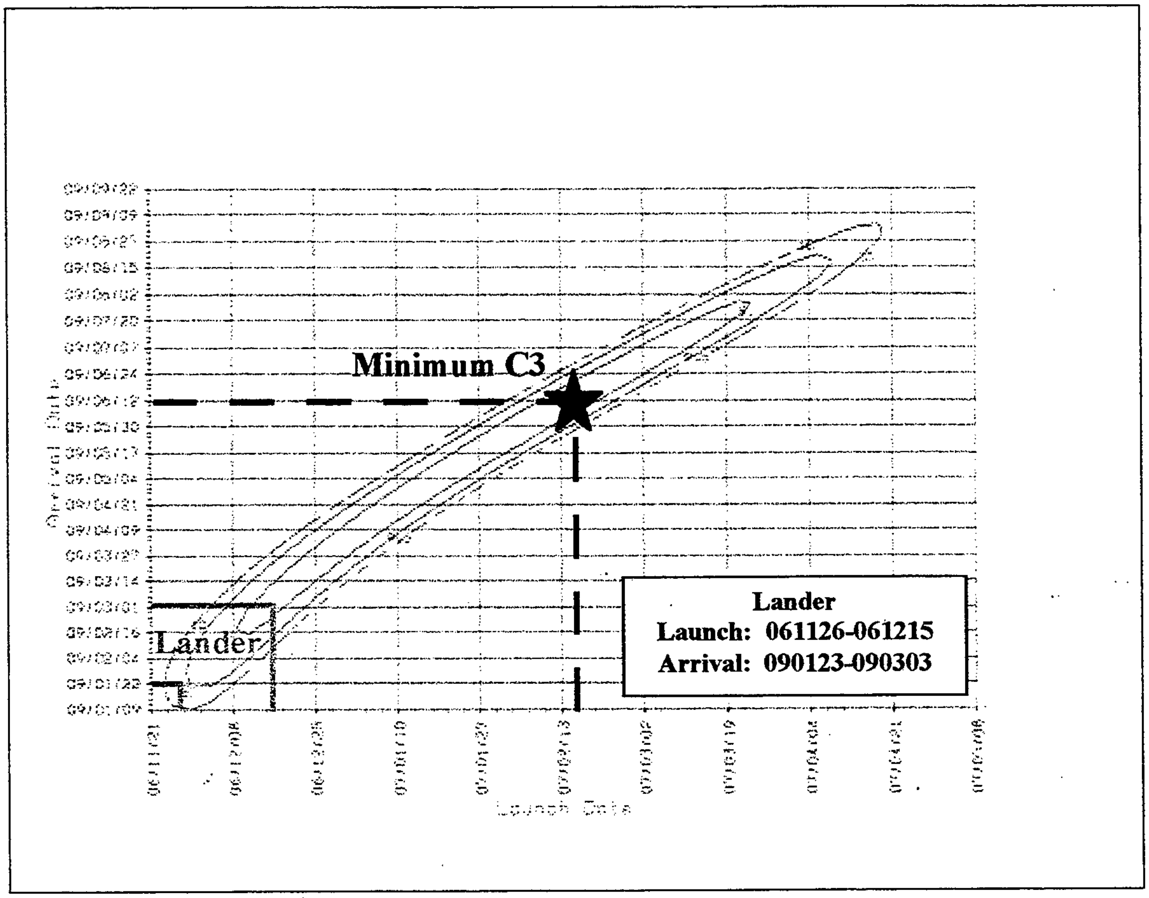


Figure 6.2: Scenario 4 – Lander Launch Period for Type 4- Trajectory

is approximately three months thereby allowing the orbiter more than sufficient time to establish a final orbit after MOI. Note that the orbiter launch period is centered on the minimum C3 launch date for the type 2 trajectory. As the minimum value of C3 is already 12.79 $km^2/s^2$, deviation from that date rapidly produces C3 values that are impractical given current spacecraft designs and launch vehicle capabilities.

Each launch date (orbiter and lander) corresponds to an optimal trajectory as calculated using CATO. The analysis parameters for both the orbiter and lander trajectories are included in Appendix A. The C3 and $V_{\infty_A}$ vs. launch date plots for the orbiter are shown in Figures 6.3 and 6.4. As expected, the C3 values do increase as

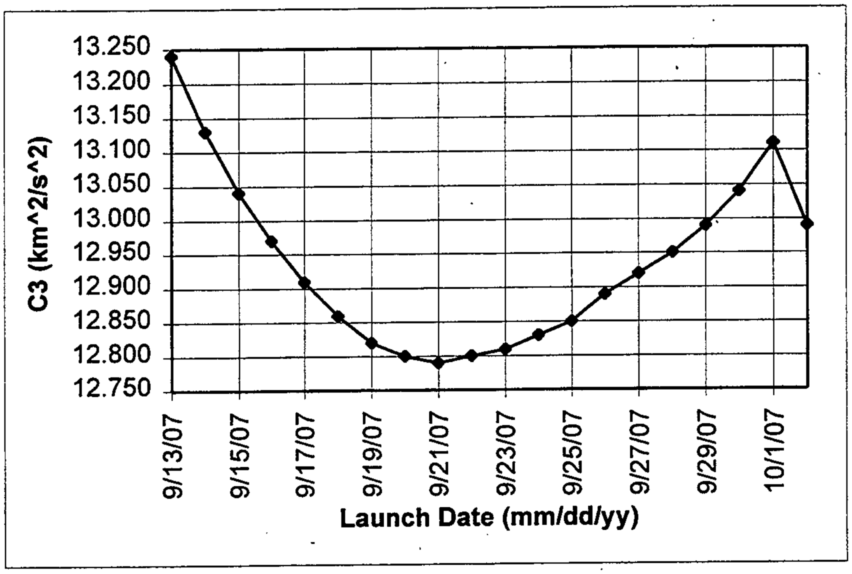


Figure 6.3: Scenario 4 - C3 vs. Launch Date (Orbiter)

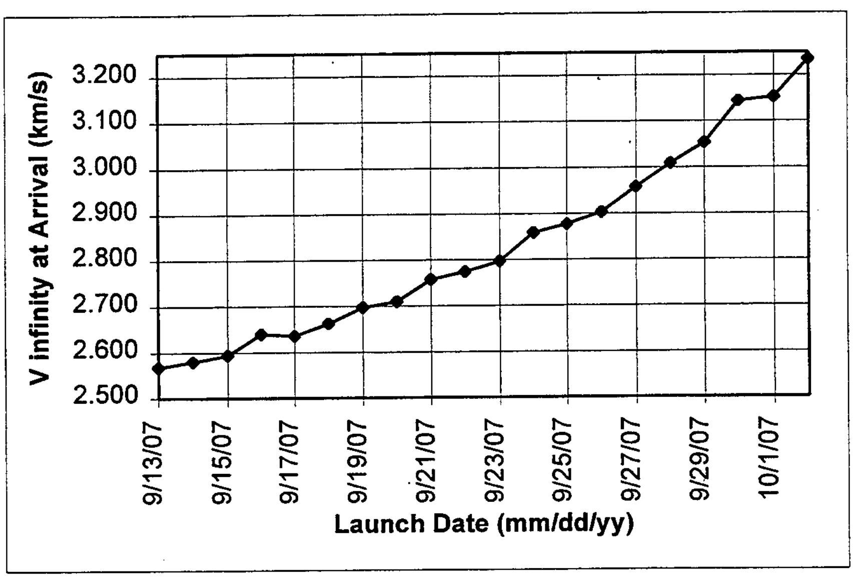


Figure 6.4: Scenario 4 - $V_{\infty_A}$ vs. Launch Date (Orbiter)

the launch date moves away from the minimum energy date. Note that the 10/02/07 launch date presents an inexplicable anomaly in the CATO data. It is also important to note that the higher launch energy requirement for this trajectory effectively decreases the injected mass capability of the launch vehicle. The $V_{\infty_A}$ however, slowly increases during the launch period. As alluded to earlier, the low values of $V_{\infty_A}$ for this type 2 trajectory significantly reduces the propellant requirement for MOI thereby offsetting the decrease in injected mass capability. As this launch period is the

same as that for the orbiter in scenarios 1, 2, and 3, it is expected that the trajectory parameters would be almost identical. The trajectory parameters for the lander are indeed the same exhibiting an increase in C3 and a decrease in $V_{\infty_A}$ during the launch period as previously shown in Figures 5.2 and 5.3. The remaining trajectory analysis parameters as plotted vs. launch date are included in Appendix B.

## B. SCENARIO 4: GEOMETRY

As this is the first scenario to use the type 2 trajectory, a Kplot for the initial launch date of the orbiter launch period is provided in Figure 6.5. The bodies of interest are annotated at various locations with time corresponding to days past launch. Note that the true anomaly about the Sun as traversed by the orbiter is clearly between 180° and 360° thus confirming this characteristic of a type 2 trajectory. As before, the orbiter-to-Earth and orbiter-to-Sun distances have been calculated and are presented in Figure 6.6.

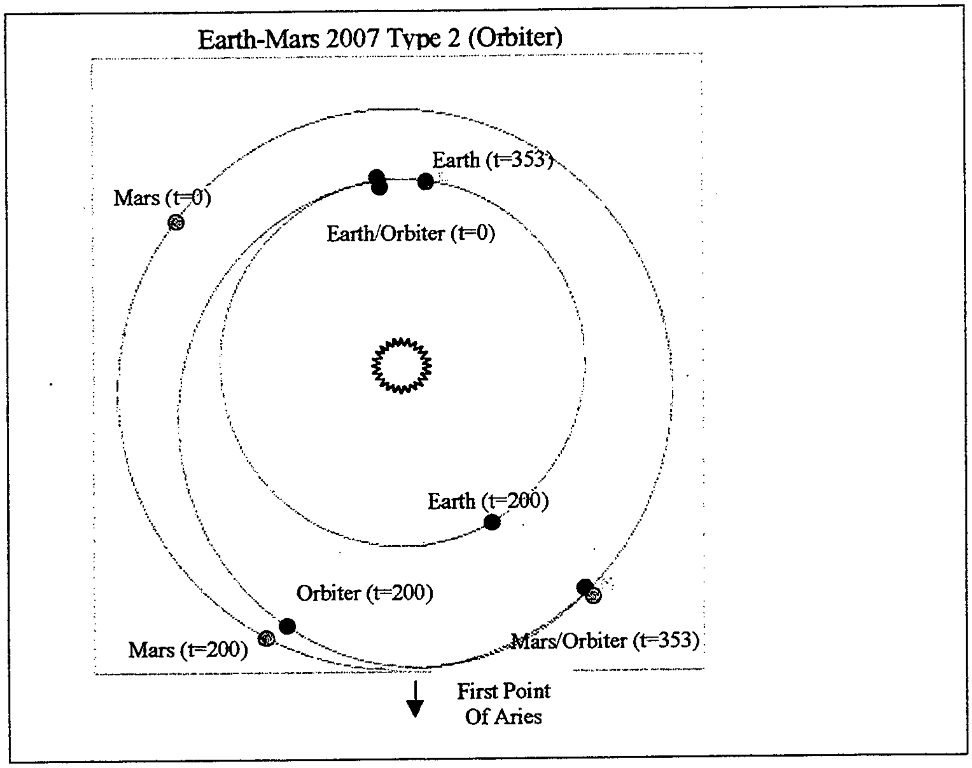


Figure 6.5: Kplot for Orbiter Type 2 Trajectory (070913 launch)

In addition the Sun-orbiter-Earth angle during the transfer trajectory has been calculated in Figure 6.7. The geometric relationship between the lander and the other bodies of interest remain the same as those previously determined for the orbiter in scenarios 1, 2, and 3 and can be seen in Figure 3.12. Therefore, they are not presented again here.

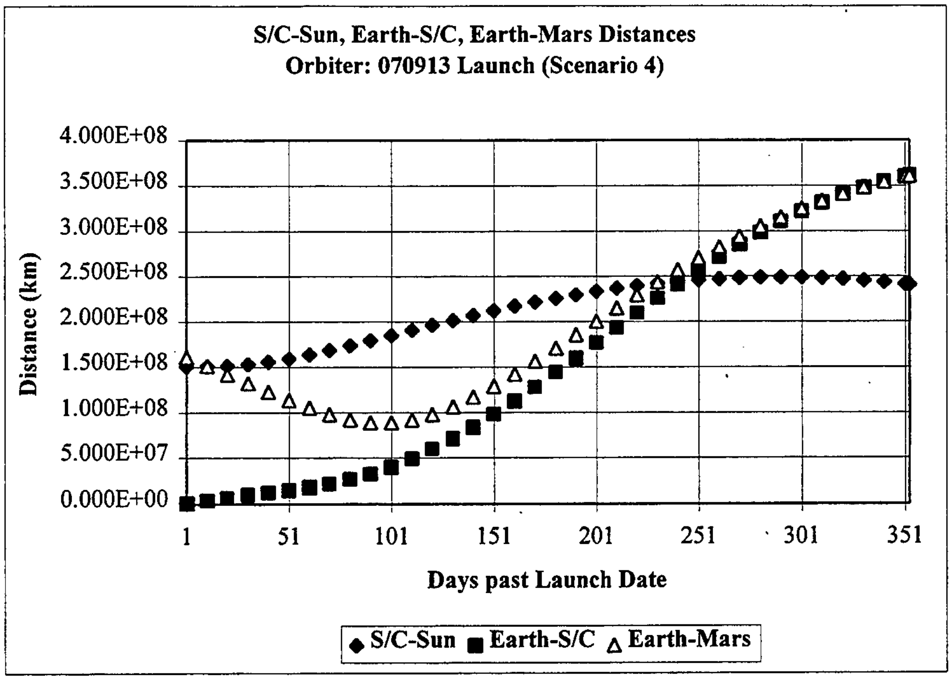


Figure 6.6: Orbiter Distances (070913 launch)

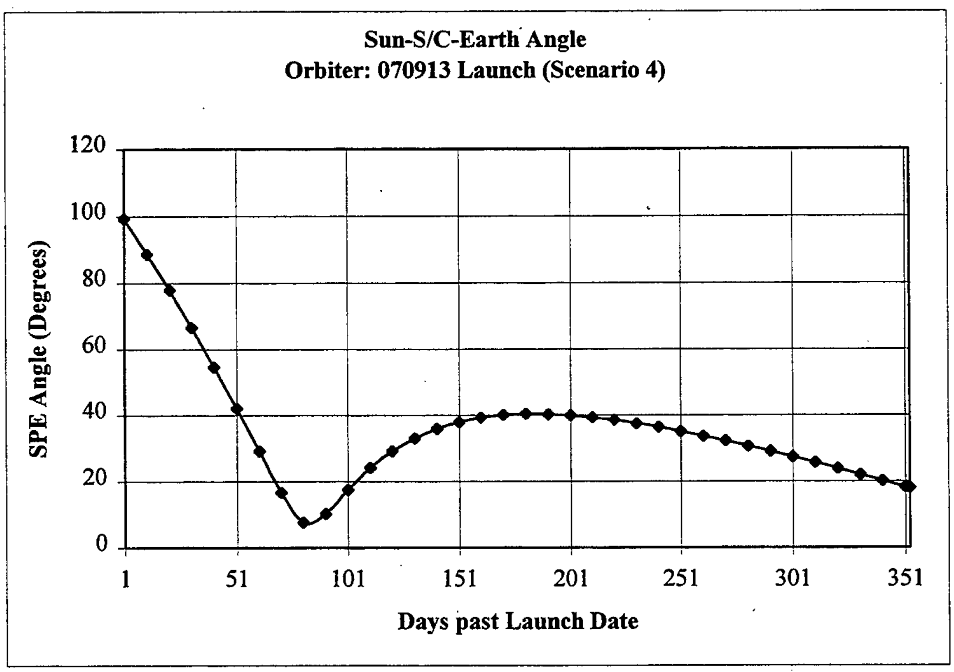


Figure 6.7: Sun- Spacecraft-Earth Angle (070913 launch)

## C. SCENARIO 4 - ORBITER ANALYSIS

The required launch energy for each orbiter trajectory as determined by CATO is used to calculate the injected mass capability for each launch vehicle. The orbiter uses a propulsive capture scheme for this scenario, so only the propellant required for TCMs and the MOI needs to be calculated. For comparison purposes, the Delta II 7925 launch vehicle was chosen again for these calculations. Using the rocket equation, the orbiter post-capture mass can be determined as shown in Table 6.1. The results confirm the counteracting effect of the C3 and $V_{\infty_A}$ values.

Table 6.1: Scenario 4 - Orbiter Post-Capture Mass (Delta II 7925)

| Launch | C3 | V inf (km/s) | Inj Mass (kg) | TCMs (30 m/s) Prop Mass | MOI ΔV (km/s) | Prop Mass (kg) | Post MOI Mass (kg) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 915.5 | 8.7 | 2.026 | 431.3 | 475.5 |
| 9/14/07 | 13.130 | 2.579 | 917.6 | 8.7 | 2.031 | 433.1 | 475.8 |
| 9/15/07 | 13.040 | 2.594 | 919.3 | 8.7 | 2.038 | 434.9 | 475.6 |
| 9/16/07 | 12.970 | 2.640 | 920.7 | 8.8 | 2.060 | 438.9 | 473.0 |
| 9/17/07 | 12.910 | 2.635 | 921.8 | 8.8 | 2.057 | 439.1 | 474.0 |
| 9/18/07 | 12.860 | 2.662 | 922.8 | 8.8 | 2.070 | 441.5 | 472.5 |
| 9/19/07 | 12.820 | 2.697 | 923.5 | 8.8 | 2.087 | 444.4 | 470.4 |
| 9/20/07 | 12.800 | 2.710 | 923.9 | 8.8 | 2.093 | 445.5 | 469.6 |
| 9/21/07 | 12.790 | 2.758 | 924.1 | 8.8 | 2.117 | 449.1 | 466.2 |
| 9/22/07 | 12.800 | 2.774 | 923.9 | 8.8 | 2.125 | 450.2 | 464.9 |
| 9/23/07 | 12.810 | 2.797 | 923.7 | 8.8 | 2.136 | 451.8 | 463.2 |
| 9/24/07 | 12.830 | 2.858 | 923.3 | 8.8 | 2.167 | 456.1 | 458.5 |
| 9/25/07 | 12.850 | 2.876 | 923.0 | 8.8 | 2.176 | 457.2 | 456.9 |
| 9/26/07 | 12.890 | 2.902 | 922.2 | 8.8 | 2.189 | 458.8 | 454.6 |
| 9/27/07 | 12.920 | 2.957 | 921.6 | 8.8 | 2.218 | 462.6 | 450.2 |
| 9/28/07 | 12.950 | 3.008 | 921.0 | 8.8 | 2.244 | 466.1 | 446.2 |
| 9/29/07 | 12.990 | 3.053 | 920.3 | 8.8 | 2.268 | 469.1 | 442.4 |
| 9/30/07 | 13.040 | 3.143 | 919.3 | 8.7 | 2.317 | 475.4 | 435.2 |
| 10/1/07 | 13.110 | 3.151 | 918.0 | 8.7 | 2.321 | 475.3 | 434.0 |
| 10/2/07 | 12.990 | 3.235 | 920.3 | 8.8 | 2.367 | 482.8 | 428.7 |

## D. SCENARIO 4: LANDER ANALYSIS

The lander analysis for this scenario consists of a TPS mass estimate and the determination of the ballistic landing region. Using the TPS Mass vs. $V_{\infty_A}$ plot in Figure 5.12, a TPS mass can be determined for the lander for each launch date in the launch period. Figure 6.8 shows the TPS mass estimate which includes the heatshield, backshell and 25% safety margin.

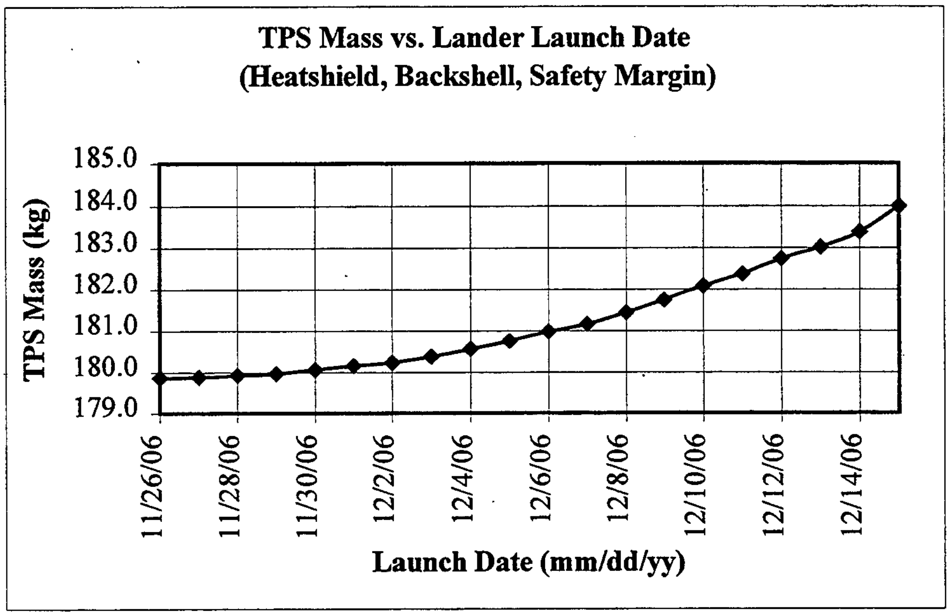


Figure 6.8: Scenario 4 - Lander TPS Mass vs. Launch Date

The ballistic landing region for the lander is shown in Figure 6.9. The flight path angle was again chosen to be -12°. For this scenario, the lander will arrive at Mars in the middle of southern spring.

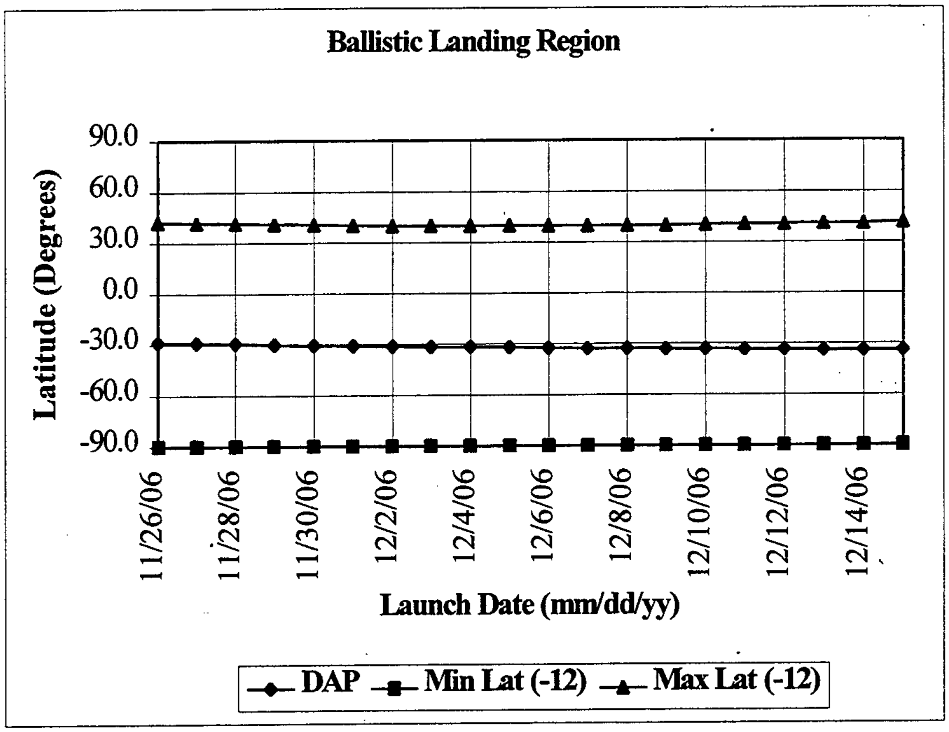


Figure 6.9: Scenario 4 - Ballistic Landing Region

## E. SCENARIO 5: LAUNCH PERIODS AND TRAJECTORY ANALYSIS PARAMETERS

As in scenario 4, the Earth-Mars transfer for the orbiter takes place on a type 2 trajectory while the lander travels on a type 4- trajectory. The orbiter however, uses aerocapture to enter into orbit about Mars in this case. The launch periods for both the orbiter and lander remain unchanged as do all of the trajectory parameters and associated geometries. Therefore, Figures 6.1-6.7 are still applicable and can be referenced for this scenario as needed.

## F. SCENARIO 5: ANALYSIS

The TPS mass estimation and ballistic landing region calculations for the lander in this scenario are identical to scenario 4. Results are previously shown in Figures 6.8

and 6.9, so they will not be displayed again here. As aerocapture is utilized by the orbiter in this case, an orbiter TPS mass estimate must be determined to calculate a post-capture orbiter mass. Using the method previously described in scenario 2, Figure 6.10 depicts the TPS mass vs. launch date data for the orbiter in this scenario.

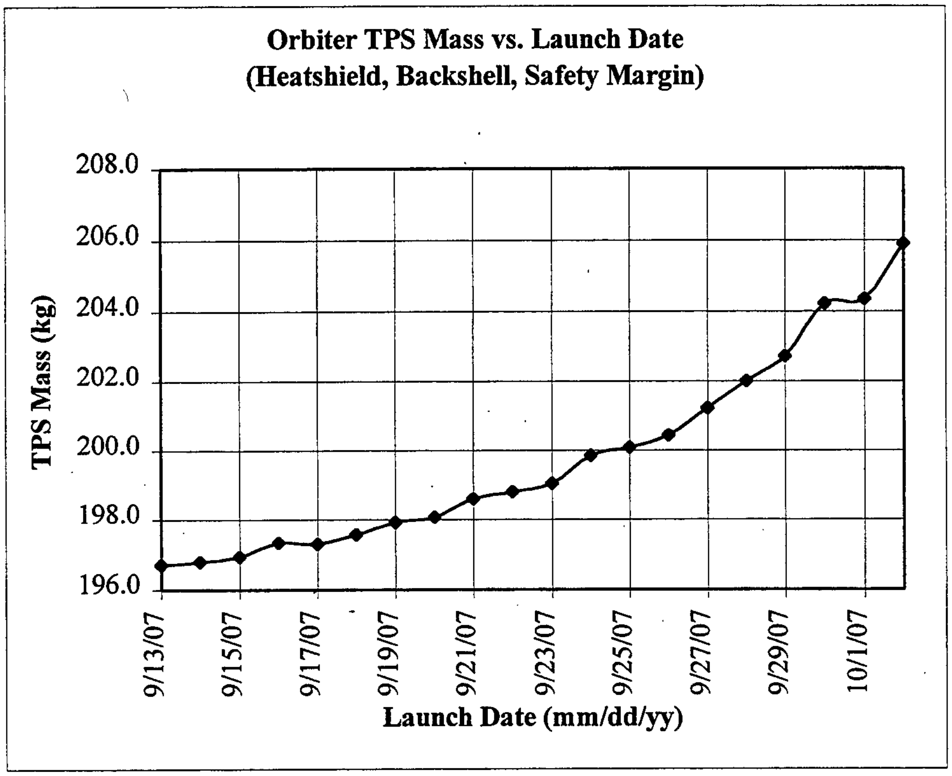


Figure 6.10: Scenario 5 - Orbiter TPS Mass vs. Launch Date

The trend of the TPS mass estimate is as expected given that the $V_{\infty_A}$ increases during the launch period. Calculations for the post-capture orbiter mass are completed using the Delta II 7925 launch vehicle with the results shown in Table 6.2. This data provides an interesting observation regarding aerocapture. The trend of the post-capture orbiter mass during the launch period is closely related to the C3 requirements. For example, in this scenario, the maximum post-capture mass corresponds to the minimum C3 requirement during the launch period. Recall from scenario 2 that this was also the case.

Table 6.2: Scenario 5 – Orbiter Post-Capture Mass (Delta II 7925)

| | | V inf at Arr | Inj Mass | TCMs (30 m/s) | Cruise Stage | TPS Mass | Post AC Mass | Post AC Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|---|
| Launch | C3 | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 9/13/07 | 13.240 | 2.567 | 915.5 | 8.7 | 75 | 196.7 | 635.1 | 25.8 | 609.3 |
| 9/14/07 | 13.130 | 2.578 | 917.6 | 8.7 | 75 | 196.8 | 637.1 | 25.9 | 611.2 |
| 9/15/07 | 13.040 | 2.594 | 919.3 | 8.7 | 75 | 196.9 | 638.6 | 25.9 | 612.7 |
| 9/16/07 | 12.960 | 2.637 | 920.7 | 8.8 | 75 | 197.3 | 639.6 | 26.0 | 613.6 |
| 9/17/07 | 12.910 | 2.634 | 921.8 | 8.8 | 75 | 197.3 | 640.7 | 26.0 | 614.7 |
| 9/18/07 | 12.860 | 2.662 | 922.8 | 8.8 | 75 | 197.6 | 641.4 | 26.0 | 615.4 |
| 9/19/07 | 12.820 | 2.696 | 923.5 | 8.8 | 75 | 197.9 | 641.8 | 26.0 | 615.8 |
| 9/20/07 | 12.800 | 2.709 | 923.9 | 8.8 | 75 | 198.1 | 642.1 | 26.1 | 616.0 |
| 9/21/07 | 12.790 | 2.757 | 924.1 | 8.8 | 75 | 198.6 | 641.7 | 26.0 | 615.7 |
| 9/22/07 | 12.800 | 2.774 | 923.9 | 8.8 | 75 | 198.8 | 641.3 | 26.0 | 615.3 |
| 9/23/07 | 12.810 | 2.796 | 923.7 | 8.8 | 75 | 199.1 | 640.9 | 26.0 | 614.9 |
| 9/24/07 | 12.830 | 2.858 | 923.3 | 8.8 | 75 | 199.8 | 639.7 | 26.0 | 613.7 |
| 9/25/07 | 12.850 | 2.876 | 923.0 | 8.8 | 75 | 200.1 | 639.1 | 25.9 | 613.2 |
| 9/26/07 | 12.890 | 2.901 | 922.2 | 8.8 | 75 | 200.4 | 638.0 | 25.9 | 612.1 |
| 9/27/07 | 12.920 | 2.956 | 921.6 | 8.8 | 75 | 201.2 | 636.6 | 25.8 | 610.8 |
| 9/28/07 | 12.950 | 3.008 | 921.0 | 8.8 | 75 | 202.0 | 635.3 | 25.8 | 609.5 |
| 9/29/07 | 12.990 | 3.053 | 920.3 | 8.8 | 75 | 202.7 | 633.8 | 25.7 | 608.1 |
| 9/30/07 | 13.040 | 3.142 | 919.3 | 8.7 | 75 | 204.2 | 631.4 | 25.6 | 605.7 |
| 10/1/07 | 13.110 | 3.150 | 918.0 | 8.7 | 75 | 204.4 | 629.9 | 25.6 | 604.3 |
| 10/2/07 | 12.990 | 3.235 | 920.3 | 8.8 | 75 | 205.9 | 630.6 | 25.6 | 605.0 |

## G. SCENARIO 6 – LAUNCH PERIODS, TRAJECTORY ANALYSIS PARAMETERS, GEOMETRY

The orbiter launch period remains unchanged from scenarios 4 and 5 for purposes of comparison. The lander launch period however, has been extended to provide additionally flexibility in the selection of the initial capture orbit for the orbiter. The possible lander launch dates are shown in Figure 6.11. The transfer time for the lander ranges from 789-826 days depending on the launch date. Using table 4.2, the minimum and maximum times available to the orbiter for aerobraking are 92 and 220 days respectively. The orbiter travelling on the shorter flight time type 2 trajectory provides for a much longer minimum and maximum aerobraking time than was the case for scenario 3.

The trajectory parameters for the orbiter are also unchanged while those for the lander now include the additional launch dates for the extended launch period. The C3 and $V_{\infty_A}$ vs. lander launch date plots are shown in Figures 6.12 and 6.13. Note that the

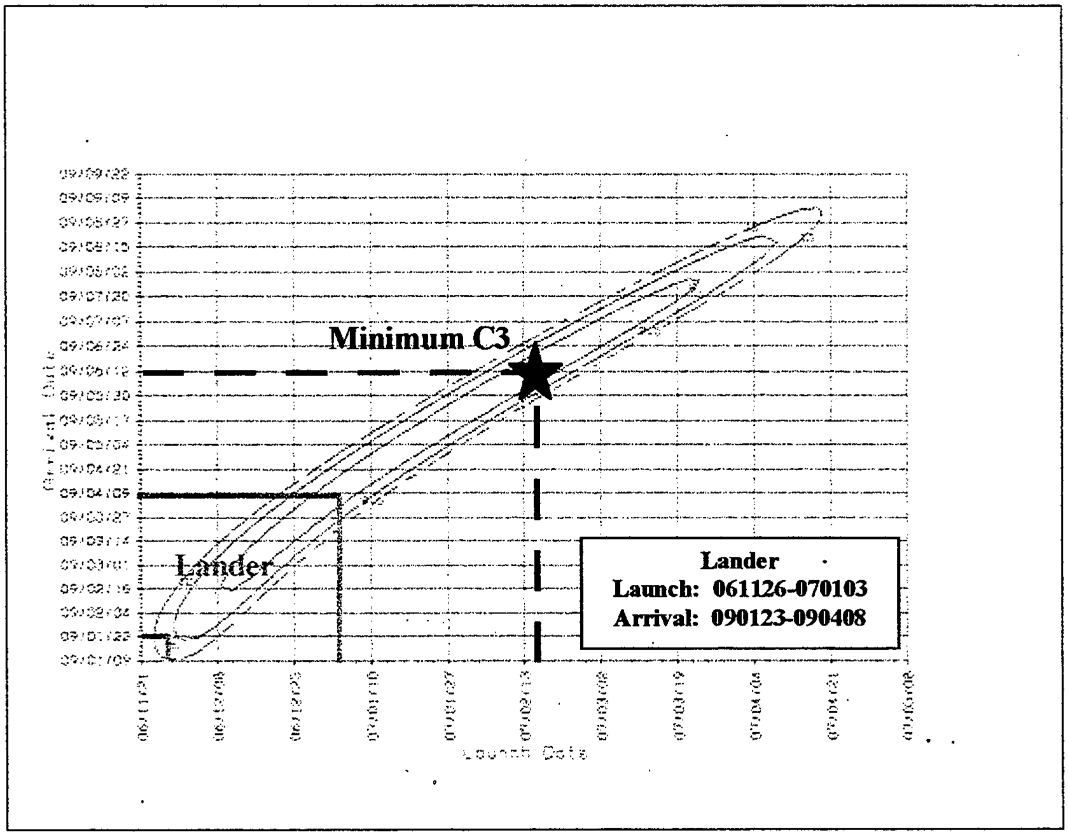


Figure 6.11: Scenario 6 - Lander Launch Period

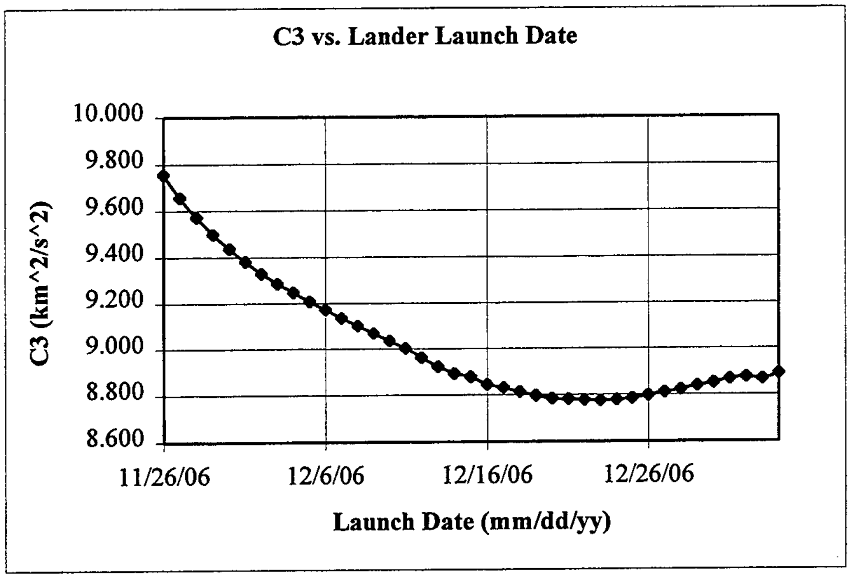


Figure 6.12: Scenario 6 - C3 vs. Launch Date (Lander)

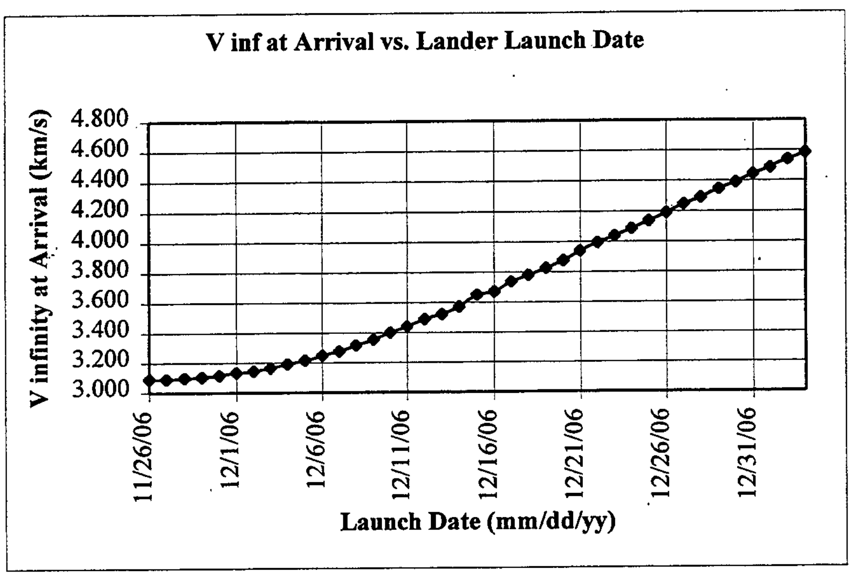


Figure 6.13: Scenario 6 - $V_{\infty_A}$ vs. Launch Date (Lander)

C3 requirements for later lander launch dates become almost constant while the values for $V_{\infty_A}$ steadily increase. This indicates that an earlier launch date could provide a better opportunity. Other trajectory parameters plotted vs. launch date are included in Appendix B.

The geometry for the orbiter in this scenario has been previously shown in Figure 6.5. Again the Kplot only represents the first launch date in the launch period. The geometry for the lander corresponding to the first launch date in the launch period can also been viewed in Figure 3.12 as it is the same geometry as that for the orbiter in scenarios 1, 2, and 3. For this scenario however, Figure 6.14 is provided to show the geometry for the last launch date of the extended lander launch period. The spacecraft-

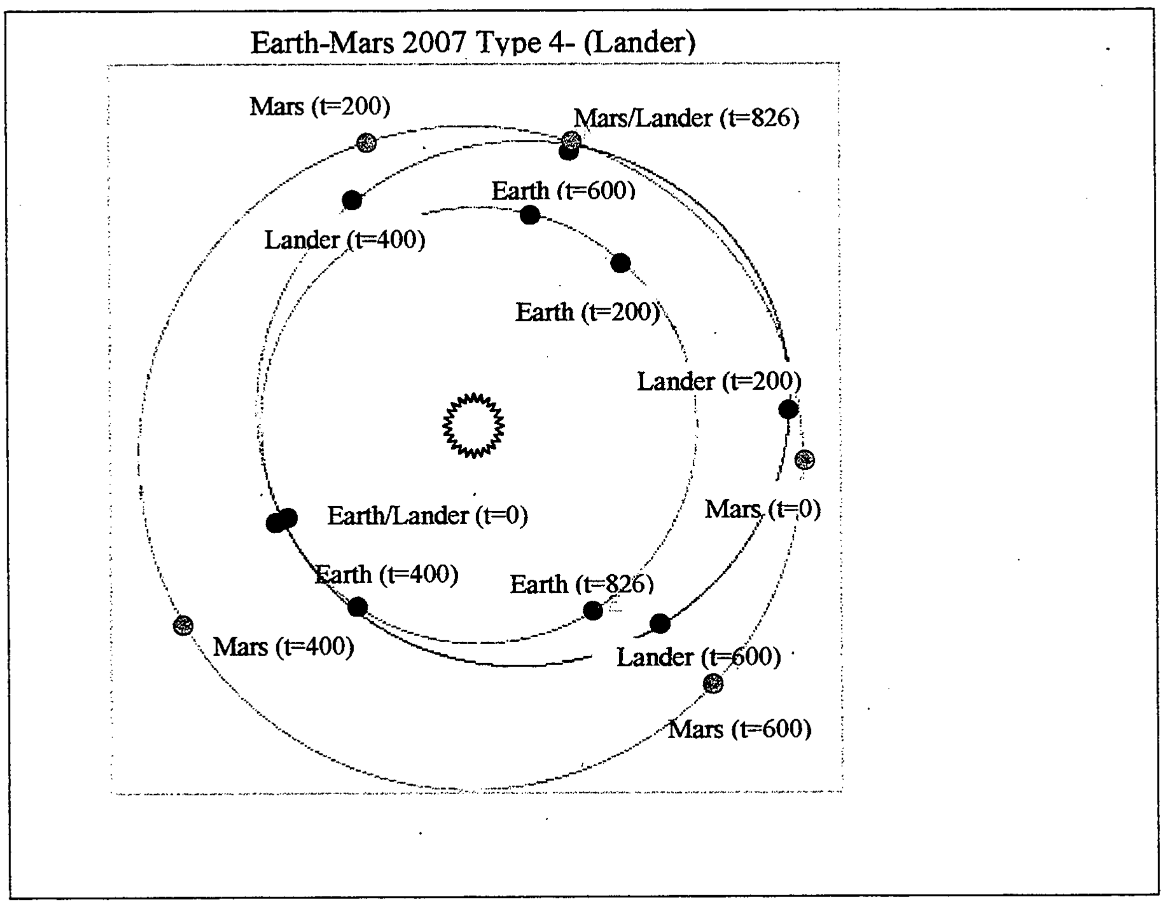


Figure 6.14: Kplot for Lander (070103 launch)

to-Earth and spacecraft-to-Sun distances as well as the Sun-Earth-spacecraft angle for the first launch date in the launch period are shown previously in Figures 5.7 and 5.9 respectively.

## H. SCENARIO 6: ORBITER ANALYSIS

The calculation of post-capture orbiter mass is again based on the Delta II 7925 launch vehicle. The results are presented in Table 6.3. The 15, 27, and 39 hour initial capture orbits were also used again for purposes of comparison. Again two observations can be made. First, the propellant requirement for the MOI into a longer period initial orbit is reduced. Second, the overall trend of the post-capture orbiter mass is dependent on the trend of the $V_{\infty_A}$. The $V_{\infty_A}$ steadily increases during throughout the launch period thus reducing the post-capture orbiter mass for later launch dates. Recall that the C3 vs. launch date in this scenario is shaped like a horseshoe with the minimum C3 case in the center of the launch period. The difference in injected mass between the minimum C3 launch date and the maximum C3 launch date is less than 10 kilograms thus reinforcing the fact that for aerobraking, it is the $V_{\infty_A}$ that drives the post-capture orbiter mass.

Table 6.3: Scenario 6 - Orbiter Post-Capture Mass

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 915.5 | 8.7 | 0.947 | 0.844 | 0.798 |
| 9/14/07 | 13.130 | 2.579 | 917.6 | 8.7 | 0.952 | 0.849 | 0.803 |
| 9/15/07 | 13.040 | 2.594 | 919.3 | 8.7 | 0.959 | 0.856 | 0.810 |
| 9/16/07 | 12.970 | 2.640 | 920.7 | 8.8 | 0.981 | 0.878 | 0.832 |
| 9/17/07 | 12.910 | 2.635 | 921.8 | 8.8 | 0.978 | 0.876 | 0.830 |
| 9/18/07 | 12.860 | 2.662 | 922.8 | 8.8 | 0.991 | 0.888 | 0.843 |
| 9/19/07 | 12.820 | 2.697 | 923.5 | 8.8 | 1.008 | 0.905 | 0.860 |
| 9/20/07 | 12.800 | 2.710 | 923.9 | 8.8 | 1.014 | 0.912 | 0.866 |
| 9/21/07 | 12.790 | 2.758 | 924.1 | 8.8 | 1.038 | 0.935 | 0.889 |
| 9/22/07 | 12.800 | 2.774 | 923.9 | 8.8 | 1.046 | 0.943 | 0.897 |
| 9/23/07 | 12.810 | 2.797 | 923.7 | 8.8 | 1.057 | 0.955 | 0.909 |
| 9/24/07 | 12.830 | 2.858 | 923.3 | 8.8 | 1.088 | 0.985 | 0.939 |
| 9/25/07 | 12.850 | 2.876 | 923.0 | 8.8 | 1.097 | 0.994 | 0.949 |
| 9/26/07 | 12.890 | 2.902 | 922.2 | 8.8 | 1.110 | 1.008 | 0.962 |
| 9/27/07 | 12.920 | 2.957 | 921.6 | 8.8 | 1.139 | 1.036 | 0.990 |
| 9/28/07 | 12.950 | 3.008 | 921.0 | 8.8 | 1.165 | 1.063 | 1.017 |
| 9/29/07 | 12.990 | 3.053 | 920.3 | 8.8 | 1.189 | 1.087 | 1.041 |
| 9/30/07 | 13.040 | 3.143 | 919.3 | 8.7 | 1.238 | 1.135 | 1.089 |
| 10/1/07 | 13.110 | 3.151 | 918.0 | 8.7 | 1.242 | 1.139 | 1.093 |
| 10/2/07 | 12.990 | 3.235 | 920.3 | 8.8 | 1.288 | 1.185 | 1.140 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 236.2 | 213.8 | 203.6 | 50 | 670.6 | 693.0 | 703.2 |
| 9/14/07 | 237.8 | 215.5 | 205.3 | 50 | 671.1 | 693.4 | 703.6 |
| 9/15/07 | 239.8 | 217.4 | 207.2 | 50 | 670.8 | 693.1 | 703.3 |
| 9/16/07 | 244.8 | 222.5 | 212.4 | 50 | 667.1 | 689.3 | 699.5 |
| 9/17/07 | 244.6 | 222.3 | 212.1 | 50 | 668.5 | 690.7 | 700.9 |
| 9/18/07 | 247.6 | 225.4 | 215.2 | 50 | 666.4 | 688.6 | 698.7 |
| 9/19/07 | 251.3 | 229.3 | 219.2 | 50 | 663.4 | 685.5 | 695.6 |
| 9/20/07 | 252.8 | 230.7 | 220.7 | 50 | 662.3 | 684.4 | 694.5 |
| 9/21/07 | 257.8 | 235.9 | 225.9 | 50 | 657.5 | 679.4 | 689.4 |
| 9/22/07 | 259.4 | 237.6 | 227.6 | 50 | 655.7 | 677.6 | 687.5 |
| 9/23/07 | 261.7 | 240.0 | 230.0 | 50 | 653.2 | 675.0 | 684.9 |
| 9/24/07 | 268.0 | 246.4 | 236.6 | 50 | 646.6 | 668.1 | 677.9 |
| 9/25/07 | 269.7 | 248.3 | 238.5 | 50 | 644.4 | 665.9 | 675.7 |
| 9/26/07 | 272.2 | 250.9 | 241.1 | 50 | 641.2 | 662.5 | 672.3 |
| 9/27/07 | 277.8 | 256.7 | 247.0 | 50 | 635.0 | 656.2 | 665.8 |
| 9/28/07 | 283.0 | 262.1 | 252.5 | 50 | 629.2 | 650.2 | 659.8 |
| 9/29/07 | 287.5 | 266.8 | 257.3 | 50 | 624.0 | 644.7 | 654.2 |
| 9/30/07 | 296.8 | 276.3 | 267.0 | 50 | 613.8 | 634.2 | 643.6 |
| 10/1/07 | 297.2 | 276.8 | 267.5 | 50 | 612.0 | 632.4 | 641.7 |
| 10/2/07 | 306.9 | 286.8 | 277.6 | 50 | 604.6 | 624.8 | 633.9 |

The post-capture orbiter mass has now been determined for scenarios 4, 5, and 6 allowing the determination of the most cost effective capture scheme. Figure 6.15 shows the post-capture orbiter masses for the three capture schemes plotted vs. orbiter launch

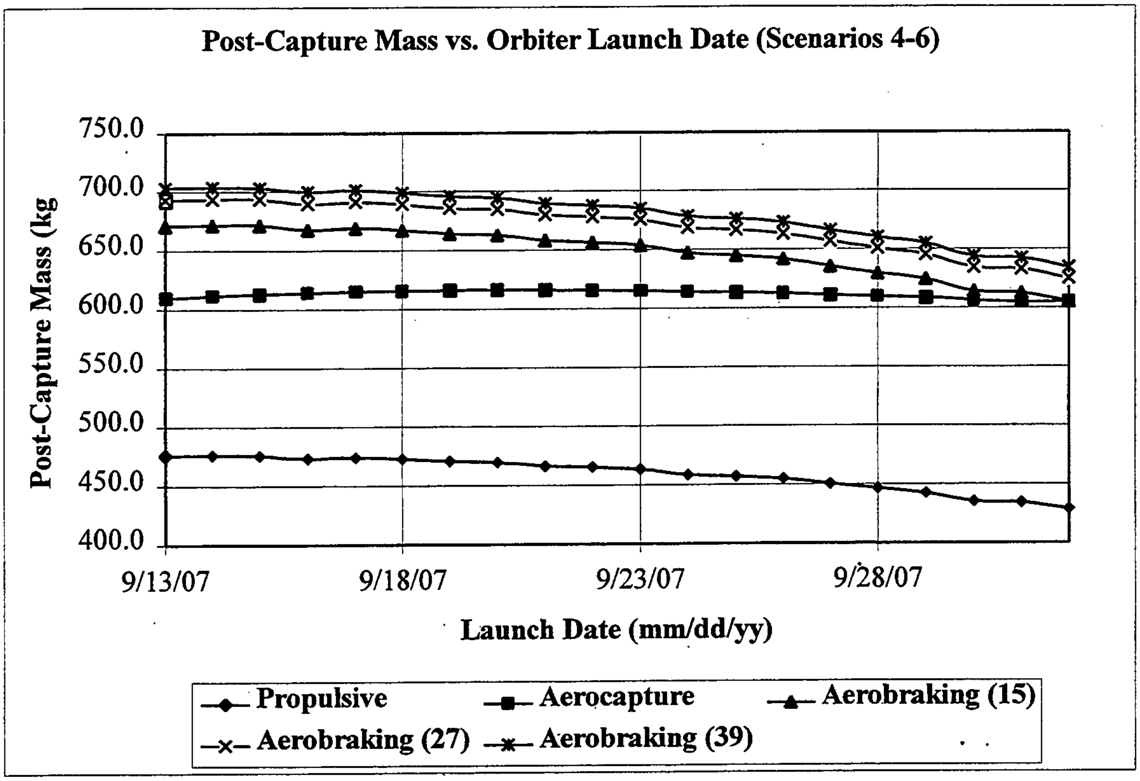


Figure 6.15: Scenarios 4,5,6 – Orbiter Post-Capture Mass

date. The results show that aerobraking is the most cost effective capture scheme. Again, at the end of the launch period, the decrease in post-capture mass for aerobraking and propulsive capture can be attributed to the effect of the increasing $V_{\infty_A}$ which leads to a corresponding increase in the $\Delta V$ required at MOI. Also, note the post-capture mass for aerocapture is roughly constant during the launch period. As the changes in C3 during the launch period are relatively small once again, and the effect of the increasing $V_{\infty_A}$ on TPS mass is also small, the overall impact on the post-capture mass is slight.

## I. SCENARIO 6: LANDER ANALYSIS

As aerobraking was determined to be the best capture scheme for the first half of the orbiter launch period, the corresponding lander launch periods can be determined using the method from scenario 3. Selection of the appropriate capture orbit depends on the post-capture orbiter mass necessary to accomplish the mission as well as the impact that the time it takes to aerobrake will have on the landed mass. The same three initial capture orbits were used to compare the landed mass for this scenario. Landed mass determination is of course dependent on the lander TPS mass in addition to the cruise stage mass and propellant required for any TCMs. A Delta II 7925 was used once again as the launch vehicle. Using the FIAT data from Figure 4.15, the TPS mass estimation for the lander over the extended launch period is shown in Figure 6.16. Again, a 91 kg backshell as well as a 25% safety margin are included in the estimation. Note that the TPS mass increases for later launch dates. This is as expected considering the increase in $V_{\infty_A}$ during the launch period.

Table 6.3 displays the post-capture orbiter mass corresponding to the three initial capture orbits. To calculate the landed mass for each case, the lander launch periods must first be determined. Figure 5.24 provides a set of aerobraking time vs. orbiter mass curves that can be used to estimate the aerobraking time associated with each initial

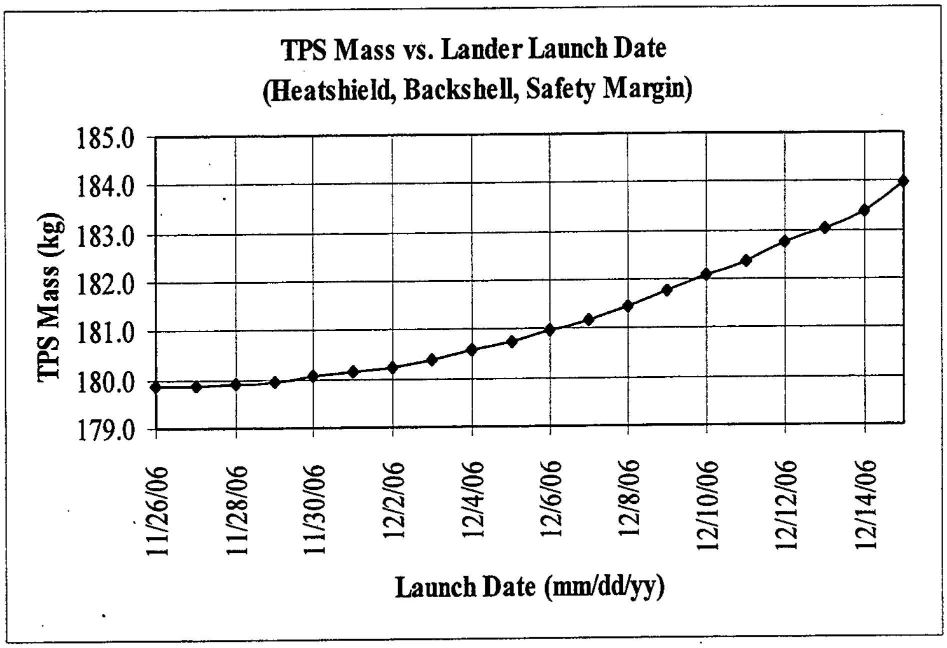


Figure 6.16: Scenario 6 - TPS Mass vs. Launch Date

capture orbit. By adding these aerobraking estimates to the arrival dates for the orbiter, an earliest lander launch date can be found. Table 6.4 shows the times required for aerobraking and the calculation of the earliest possible lander arrival dates. The worst case lander arrival date for each capture orbit can then be used to determine a 20 day lander launch period from CATO trajectory data. The worst case lander arrival for this scenario coincides with the latest orbiter arrival even though the time to aerobrake is decreasing for later launch dates. This is attributable to the reduction in post-capture mass associated with later launch dates as seen in Figure 6.15. Recall that since the orbiter travels on a type 2 trajectory in this scenario, the minimum time available to the orbiter for aerobraking is 92 days. As this is already significant, the lander launch period was determined to begin at the earliest date in the type 4- opportunity. Although the

Table 6.4: Scenario 6 – Calculations for Earliest Lander Launch Dates

| Orbiter Launch | Arrival | 15 hr orbit (days) | 27 hr orbit (days) | 39 hr orbit (days) | Earliest Lander Launch (15) | (27) | (39) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 8/31/08 | 54 | 85 | 108 | 10/23/08 | 11/24/08 | 12/17/08 |
| 9/14/07 | 9/2/08 | 54 | 85 | 108 | 10/25/08 | 11/26/08 | 12/19/08 |
| 9/15/07 | 9/4/08 | 54 | 85 | 108 | 10/27/08 | 11/28/08 | 12/21/08 |
| 9/16/07 | 9/8/08 | 53 | 85 | 108 | 10/31/08 | 12/1/08 | 12/24/08 |
| 9/17/07 | 9/8/08 | 53 | 85 | 108 | 10/31/08 | 12/1/08 | 12/24/08 |
| 9/18/07 | 9/11/08 | 53 | 85 | 108 | 11/3/08 | 12/4/08 | 12/27/08 |
| 9/19/07 | 9/14/08 | 53 | 84 | 107 | 11/6/08 | 12/7/08 | 12/30/08 |
| 9/20/07 | 9/15/08 | 53 | 84 | 107 | 11/6/08 | 12/8/08 | 12/30/08 |
| 9/21/07 | 9/20/08 | 53 | 83 | 106 | 11/11/08 | 12/12/08 | 1/4/09 |
| 9/22/07 | 9/21/08 | 52 | 83 | 106 | 11/12/08 | 12/13/08 | 1/4/09 |
| 9/23/07 | 9/23/08 | 52 | 83 | 105 | 11/14/08 | 12/14/08 | 1/6/09 |
| 9/24/07 | 9/28/08 | 52 | 82 | 104 | 11/18/08 | 12/19/08 | 1/10/09 |
| 9/25/07 | 9/29/08 | 52 | 82 | 104 | 11/19/08 | 12/19/08 | 1/11/09 |
| 9/26/07 | 10/1/08 | 51 | 81 | 104 | 11/21/08 | 12/21/08 | 1/12/09 |
| 9/27/07 | 10/5/08 | 51 | 81 | 103 | 11/24/08 | 12/24/08 | 1/15/09 |
| 9/28/07 | 10/12/08 | 50 | 80 | 102 | 12/1/08 | 12/30/08 | 1/21/09 |
| 9/29/07 | 10/12/08 | 50 | 79 | 101 | 11/30/08 | 12/30/08 | 1/20/09 |
| 9/30/07 | 10/18/08 | 49 | 78 | 99 | 12/6/08 | 1/3/09 | 1/25/09 |
| 10/1/07 | 10/18/08 | 49 | 78 | 99 | 12/5/08 | 1/3/09 | 1/24/09 |
| 10/2/07 | 10/23/08 | 48 | 77 | 98 | 12/10/08 | 1/7/09 | 1/28/09 |

lander could launch earlier and still enable the orbiter sufficient time to aerobrake, the launch energy requirements for the lander would then become impractical. Therefore, as shown in Table 6.5, all three lander launch periods are virtually identical with a small variation in the launch period corresponding to the 39 hour initial capture orbit. The corresponding landed masses can now be calculated. The results are shown in Table 6.6.

Table 6.5: Scenario 6 - Lander Launch Periods

| Lander (15) | | Lander (27) | | Lander (39) | |
|---|---|---|---|---|---|
| Departure | Arrival | Departure | Arrival | Departure | Arrival |
| 11/26/06 | 1/23/09 | 11/26/06 | 1/23/09 | 11/29/06 | 1/28/09 |
| 11/27/06 | 1/25/09 | 11/27/06 | 1/25/09 | 11/30/06 | 1/30/09 |
| 11/28/06 | 1/26/09 | 11/28/06 | 1/26/09 | 12/1/06 | 1/31/09 |
| 11/29/06 | 1/28/09 | 11/29/06 | 1/28/09 | 12/2/06 | 2/2/09 |
| 11/30/06 | 1/30/09 | 11/30/06 | 1/30/09 | 12/3/06 | 2/4/09 |
| 12/1/06 | 1/31/09 | 12/1/06 | 1/31/09 | 12/4/06 | 2/6/09 |
| 12/2/06 | 2/2/09 | 12/2/06 | 2/2/09 | 12/5/06 | 2/8/09 |
| 12/3/06 | 2/4/09 | 12/3/06 | 2/4/09 | 12/6/06 | 2/10/09 |
| 12/4/06 | 2/6/09 | 12/4/06 | 2/6/09 | 12/7/06 | 2/12/09 |
| 12/5/06 | 2/8/09 | 12/5/06 | 2/8/09 | 12/8/06 | 2/15/09 |
| 12/6/06 | 2/10/09 | 12/6/06 | 2/10/09 | 12/9/06 | 2/17/09 |
| 12/7/06 | 2/12/09 | 12/7/06 | 2/12/09 | 12/10/06 | 2/19/09 |
| 12/8/06 | 2/15/09 | 12/8/06 | 2/15/09 | 12/11/06 | 2/21/09 |
| 12/9/06 | 2/17/09 | 12/9/06 | 2/17/09 | 12/12/06 | 2/24/09 |
| 12/10/06 | 2/19/09 | 12/10/06 | 2/19/09 | 12/13/06 | 2/25/09 |
| 12/11/06 | 2/21/09 | 12/11/06 | 2/21/09 | 12/14/06 | 2/28/09 |
| 12/12/06 | 2/24/09 | 12/12/06 | 2/24/09 | 12/15/06 | 3/3/09 |
| 12/13/06 | 2/25/09 | 12/13/06 | 2/25/09 | 12/16/06 | 3/4/09 |
| 12/14/06 | 2/28/09 | 12/14/06 | 2/28/09 | 12/17/06 | 3/7/09 |
| 12/15/06 | 3/3/09 | 12/15/06 | 3/3/09 | 12/18/06 | 3/9/09 |

Table 6.6: Scenario 6 - Landed Masses

| (15) Landed Mass | (27) Landed Mass | (39) Landed Mass |
|---|---|---|
| 720.1 | 720.1 | 725.2 |
| 722.1 | 722.1 | 726.5 |
| 723.8 | 723.8 | 727.5 |
| 725.2 | 725.2 | 728.4 |
| 726.5 | 726.5 | 729.2 |
| 727.5 | 727.5 | 729.8 |
| 728.4 | 728.4 | 730.4 |
| 729.2 | 729.2 | 730.9 |
| 729.8 | 729.8 | 731.4 |
| 730.4 | 730.4 | 731.9 |
| 730.9 | 730.9 | 732.2 |
| 731.4 | 731.4 | 732.6 |
| 731.9 | 731.9 | 733.0 |
| 732.2 | 732.2 | 733.5 |
| 732.6 | 732.6 | 734.0 |
| 733.0 | 733.0 | 734.3 |
| 733.5 | 733.5 | 733.9 |
| 734.0 | 734.0 | 734.4 |
| 734.3 | 734.3 | 734.3 |
| 733.9 | 733.9 | 734.3 |

Based on the previous discussion, the results are not surprising. As the lander launch periods do not vary considerably, neither do the landed masses which means that the orbiter can capture into a longer period orbit to preserve post-capture mass without effecting the landed mass of the lander.

Finally, the ballistic landing region for the lander in this scenario is shown in Figure 6.17. As expected, the lander can target most of the Martian surface for landing with the exception of the northern-most latitudes.

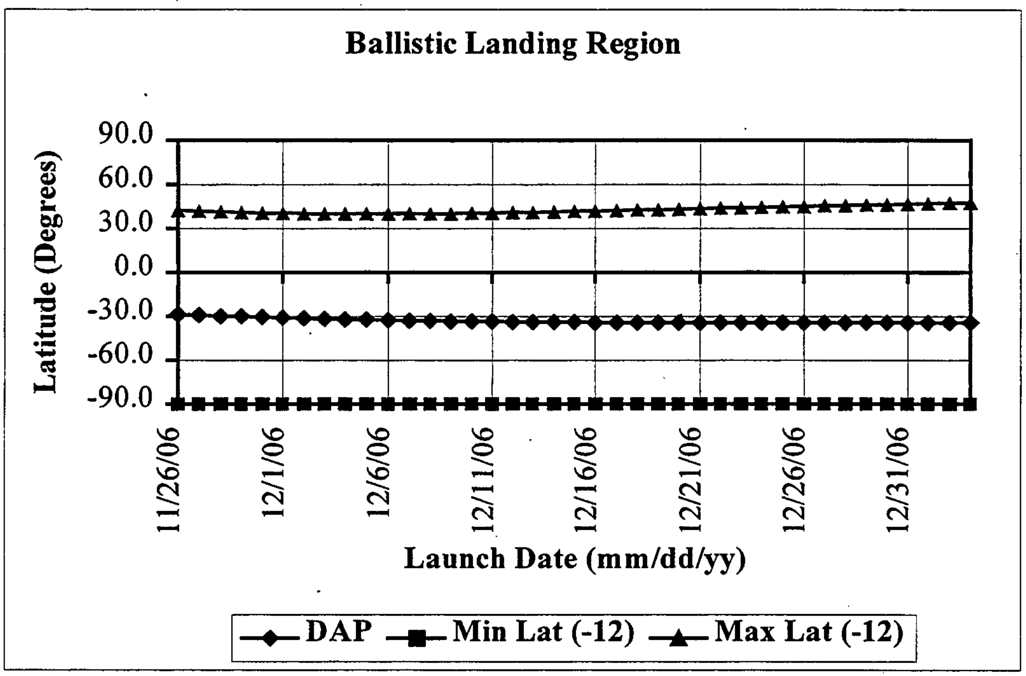


Figure 6.17: Scenario 6 - Ballistic Landing Region

## J. COMPARISON OF SCENARIOS 1-3 AND SCENARIOS 4-6

Scenarios 1-3 and 4-6 were purposely analyzed separately. The criterion used to distinguish between the capture schemes was post-capture orbiter mass given either the type 2 or type 4- trajectory. It is important however, to also compare across the different trajectories types for those instances when the mission might dictate the capture scheme.

Therefore, Figures 6.18-6.20 show comparisons between the trajectories for each capture scheme. For the propulsive capture case, the difference in post-capture masses is negligible between trajectory types indicating the lower C3 requirements for a type 4- trajectory are offset by the smaller values of $V_{\infty_A}$ for the type 2 trajectory.

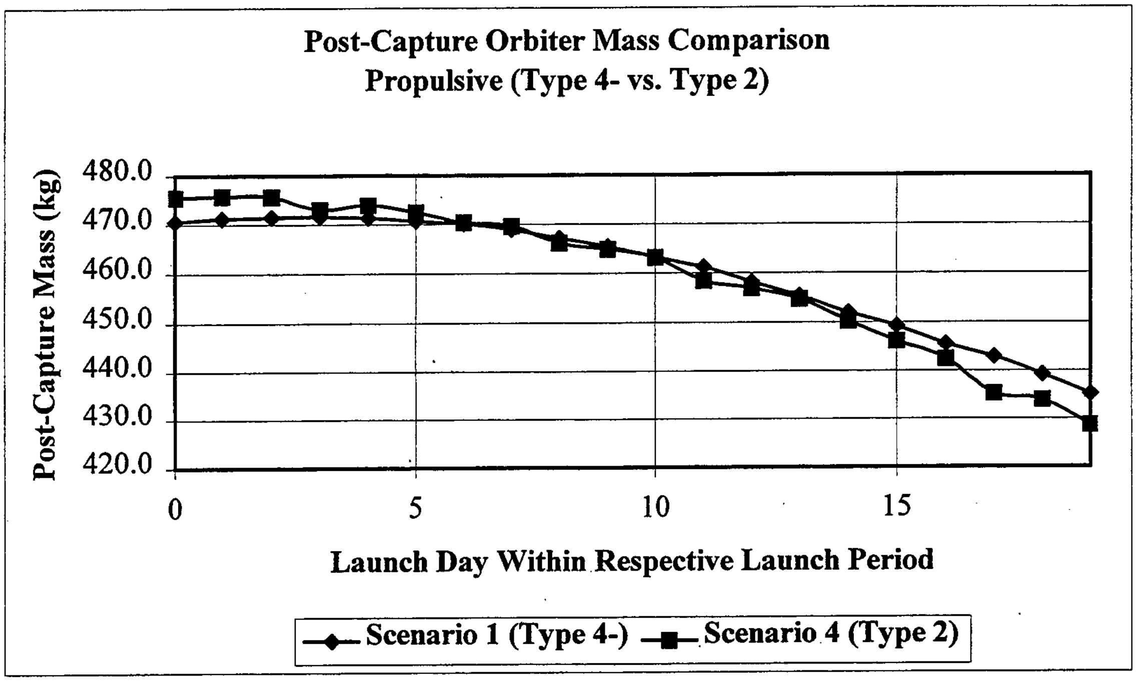


Figure 6.18: Type 4- vs. Type 2 Propulsive Capture Comparison

Aerocapture however, shows a different story. The post-capture mass difference in this case averages approximately 60 kg throughout the respective launch periods. The driver for aerocapture is the C3 requirement as the difference in $V_{\infty_A}$ for the two trajectory types has only a small impact on the TPS mass. Therefore, lower values of C3 for the type 4- trajectory result in a larger post-capture orbiter mass. The results for the aerobraking case are similar to the propulsive case as the driver is the magnitude of the MOI maneuver. The comparison of aerobraking between trajectory types only corresponds to the 27 hour initial capture orbit as the data for the other two initial capture orbits reflect similar results.

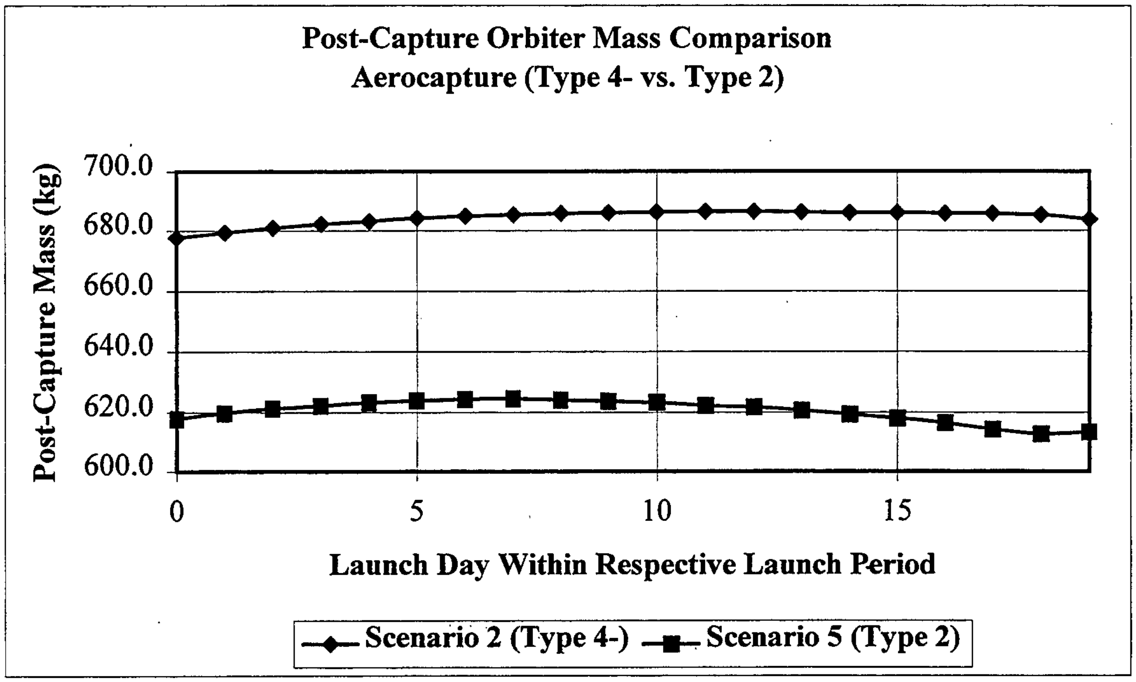


Figure 6.19: Type 4- vs. Type 2 Aerocapture Comparison

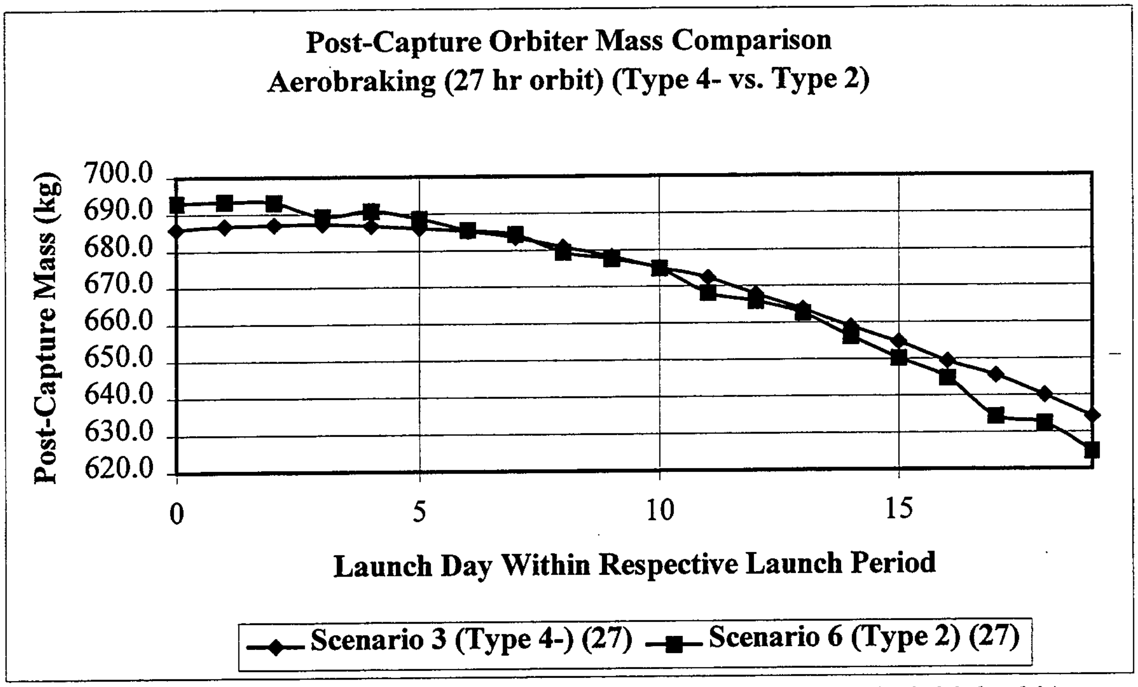


Figure 6.20: Type 4- vs. Type 2 Aerobraking Comparison (27 hr initial orbit)

# VII. ONE LAUNCH VEHICLE

## A. BACKGROUND

Determination of the most cost-effective mission design not only depends on the trajectory chosen, but also on the selection of the launch vehicle. Although the use of separate launch vehicles is the baseline for this mission, it is important to consider the possibility of utilizing a single launch vehicle for both the orbiter and the lander. Analysis of this configuration requires the following assumptions:

1) The orbiter and lander initial masses will each consist of half of the injected mass capability of the launch vehicle;
2) A separate cruise stage will be utilized for the orbiter/lander. The orbiter is not utilized as the cruise stage as it may be difficult to target the lander and still be possible for the orbiter to accomplish the MOI maneuver;
3) The orbiter and lander will remain intact as one unit until some pre-determined separation time in close proximity to Mars;
4) The Mars Deflection Maneuver (MDM) after orbiter-lander separation is estimated to be 30 m/s;
5) The lander either possesses a direct-Earth link or an orbiting communications architecture is in place to provide communications relay;
6) Packaging of the orbiter and lander in the launch vehicle fairing for an aerocapture case is not prohibitive;

The first step in this process is to determine the launch periods for the mission. Again, only the type 2 and type 4- trajectories will be considered. The launch period for the type 2 opportunity is fairly straightforward as the launch energy requirements noted previously preclude a launch period other than one centered on the minimum energy launch date. Therefore, the type 2 trajectory launch period will be identical to the orbiter launch period used in scenarios 4-6. Although the type 4- opportunity provides over 150 launch dates with a C3 less than 10 $km^2/s^2$, analysis for the utilization of two launch vehicles clearly indicates that earlier launch dates are more beneficial for both the orbiter and the lander in terms of post-capture mass and landed mass respectively. Therefore, the type 4- trajectory launch period for the single launch vehicle case corresponds to the orbiter launch period in scenarios 1-3. As the requirement for the orbiter to provide a communications relay is no longer a constraint in this case, the orbiter capture schemes do not impact the lander. Table 7.1 summarizes the launch periods for the single launch vehicle case.

Table 7.1: Scenarios using One Launch Vehicle

| Scenario | # LV | Trajectory | Launch Period | Arrival Period | Total Flight Time | Capture Scheme |
|---|---|---|---|---|---|---|
| 7 | 1 | 4- | 061126-061215 | 090123-090303 | 789-822 | Prop, AB, AC |
| 8 | 1 | 2 | 070913-071002 | 080831-081023 | 353-387 | Prop, AB, AC |

The launch vehicles considered for this analysis include the Delta III (2-stage and 3-stage), the Atlas II family, and the Ariane 4 and 5. The Delta III performance data was obtained from the Delta III Payload Planner's Guide. The Atlas II performance data was obtained from JPL sources and verified using the Atlas Launch System Mission Planner's Guide. The Ariane 4 and 5 performance data was extracted from Reyes' thesis and applied to the specific scenario here.

## B. SCENARIO 7: ANALYSIS

The trajectory analysis parameters and geometry for this scenario are the same as those previously presented for the orbiter in scenario 1. The lander TPS mass requirement and ballistic landing region have previously been calculated in scenario 5 for this same launch period. Therefore, that data will not be shown again here. The calculation of the post-capture orbiter mass however, is still necessary for purposes of comparison. Determination of the post-capture orbiter mass begins with the injected mass capability for the launch vehicles as summarized in Table 7.2. Although the performance capabilities of the launch vehicles vary greatly, a Delta III (2-stage) launch vehicle was chosen for the single launch vehicle analysis. The results for the other launch vehicles are included in Appendix C. It is worth noting that the trajectory targeting is based on the landing requirements for the lander. As a result, a Mars Deflection Maneuver (MDM) for the orbiter is necessary upon orbiter-lander separation to prevent the orbiter from also entering the Martian atmosphere. The magnitude of this maneuver is dependent on the time of separation, the capture scheme to be employed by the orbiter and the intended altitude for the MOI maneuver. This analysis assumes that the time of separation will minimize the MDM maneuver. The altitude for the MOI maneuver is assumed to be 250 km for both propulsive capture and aerobraking. However, for aerocapture, the MDM is not required as the orbiter will also enter the atmosphere.

Using the single launch vehicle assumptions stated previously, the post-capture orbiter mass for the propulsive capture scheme was calculated. The results are summarized in Table 7.3. The TCMs are accomplished using the attitude control system

(ACS) of the orbiter. As expected, the post-capture orbiter mass decreases for later launch dates. This is once again attributed to the increasing $V_{\infty_A}$ and subsequent increase of the $\Delta V$ required at MOI.

Table 7.2: Scenario 7 - Injected Mass Capabilities for Various Launch Vehicles

| Launch | C3 | Delta III (2 stage) (10% margin) +/- 25 kg | Delta III (3 stage) (10 % margin) +/- 25 kg | Atlas IIAS (10% margin) +/- 10 kg | Atlas IIAS (Star 48B) (10% margin) +/- 10 kg | Atlas IIAR (13% margin) +/- 10 kg |
|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 1985.6 | 2061.4 | 1981.3 | 2007.1 | 2142.0 |
| 11/27/06 | 9.657 | 1990.0 | 2064.9 | 1985.3 | 2010.4 | 2146.1 |
| 11/28/06 | 9.572 | 1993.9 | 2067.9 | 1988.8 | 2013.3 | 2149.7 |
| 11/29/06 | 9.497 | 1997.3 | 2070.6 | 1991.9 | 2015.8 | 2152.9 |
| 11/30/06 | 9.434 | 2000.2 | 2072.9 | 1994.5 | 2018.0 | 2155.6 |
| 12/1/06 | 9.379 | 2002.7 | 2074.9 | 1996.7 | 2019.8 | 2157.9 |
| 12/2/06 | 9.328 | 2005.0 | 2076.7 | 1998.8 | 2021.6 | 2160.1 |
| 12/3/06 | 9.285 | 2007.0 | 2078.3 | 2000.6 | 2023.0 | 2161.9 |
| 12/4/06 | 9.247 | 2008.7 | 2079.7 | 2002.2 | 2024.3 | 2163.5 |
| 12/5/06 | 9.209 | 2010.4 | 2081.0 | 2003.7 | 2025.6 | 2165.1 |
| 12/6/06 | 9.173 | 2012.1 | 2082.3 | 2005.2 | 2026.8 | 2166.6 |
| 12/7/06 | 9.136 | 2013.7 | 2083.7 | 2006.8 | 2028.1 | 2168.2 |
| 12/8/06 | 9.103 | 2015.2 | 2084.9 | 2008.1 | 2029.2 | 2169.6 |
| 12/9/06 | 9.070 | 2016.7 | 2086.1 | 2009.5 | 2030.3 | 2171.0 |
| 12/10/06 | 9.037 | 2018.2 | 2087.3 | 2010.8 | 2031.5 | 2172.4 |
| 12/11/06 | 9.004 | 2019.7 | 2088.5 | 2012.2 | 2032.6 | 2173.8 |
| 12/12/06 | 8.962 | 2021.7 | 2090.0 | 2013.9 | 2034.0 | 2175.6 |
| 12/13/06 | 8.922 | 2023.5 | 2091.5 | 2015.6 | 2035.4 | 2177.4 |
| 12/14/06 | 8.892 | 2024.8 | 2092.6 | 2016.8 | 2036.4 | 2178.6 |
| 12/15/06 | 8.878 | 2025.5 | 2093.1 | 2017.4 | 2036.9 | 2179.2 |
| **Launch** | **C3** | **Atlas IIAR (Star 48B) (13% margin) +/- 10 kg** | **Atlas IIARS (13% margin) +/- 10 kg** | **Atlas IIARS (Star 48B) (13% margin) +/- 10 kg** | **Ariane 4 HM7B (10% margin)** | **Ariane 5 L9 (10% margin)** |
| 11/26/06 | 9.755 | 2135.4 | 2292.1 | 2187.4 | 6176.5 | 3277.3 |
| 11/27/06 | 9.657 | 2138.8 | 2296.4 | 2190.5 | 6189.3 | 3286.1 |
| 11/28/06 | 9.572 | 2141.7 | 2300.1 | 2193.1 | 6200.4 | 3293.8 |
| 11/29/06 | 9.497 | 2144.3 | 2303.4 | 2195.5 | 6210.2 | 3300.6 |
| 11/30/06 | 9.434 | 2146.5 | 2306.2 | 2197.5 | 6218.4 | 3306.3 |
| 12/1/06 | 9.379 | 2148.4 | 2308.6 | 2199.2 | 6225.7 | 3311.3 |
| 12/2/06 | 9.328 | 2150.2 | 2310.8 | 2200.8 | 6232.4 | 3315.9 |
| 12/3/06 | 9.285 | 2151.7 | 2312.7 | 2202.2 | 6238.0 | 3319.9 |
| 12/4/06 | 9.247 | 2153.0 | 2314.4 | 2203.4 | 6243.0 | 3323.3 |
| 12/5/06 | 9.209 | 2154.4 | 2316.1 | 2204.6 | 6248.0 | 3326.8 |
| 12/6/06 | 9.173 | 2155.6 | 2317.7 | 2205.7 | 6252.8 | 3330.1 |
| 12/7/06 | 9.136 | 2156.9 | 2319.3 | 2206.9 | 6257.7 | 3333.5 |
| 12/8/06 | 9.103 | 2158.1 | 2320.8 | 2208.0 | 6262.1 | 3336.5 |
| 12/9/06 | 9.070 | 2159.2 | 2322.2 | 2209.0 | 6266.4 | 3339.5 |
| 12/10/06 | 9.037 | 2160.4 | 2323.7 | 2210.0 | 6270.8 | 3342.6 |
| 12/11/06 | 9.004 | 2161.5 | 2325.2 | 2211.1 | 6275.2 | 3345.6 |
| 12/12/06 | 8.962 | 2163.0 | 2327.0 | 2212.4 | 6280.8 | 3349.5 |
| 12/13/06 | 8.922 | 2164.4 | 2328.8 | 2213.7 | 6286.1 | 3353.1 |
| 12/14/06 | 8.892 | 2165.5 | 2330.1 | 2214.6 | 6290.1 | 3355.9 |
| 12/15/06 | 8.878 | 2165.9 | 2330.8 | 2215.1 | 6291.9 | 3357.2 |

The packaging of two aeroshells inside a single launch vehicle fairing would be difficult given current orbiter and lander designs. However, for purposes of completeness, aerocapture is still included here as an option. The calculation of the orbiter post-capture mass using aerocapture as shown in Figure 7.4 follows the

Table 7.3: Scenario 7 – (Propulsive) Orbiter Post-Capture Mass (Delta III 2-stage)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1985.6 | 75.0 | 9.1 | 9.0 | 2.286 | 484.9 | 452.3 |
| 11/27/06 | 9.656 | 3.090 | 1990.0 | 75.0 | 9.1 | 9.0 | 2.288 | 486.3 | 453.1 |
| 11/28/06 | 9.572 | 3.097 | 1993.9 | 75.0 | 9.1 | 9.0 | 2.292 | 487.8 | 453.5 |
| 11/29/06 | 9.497 | 3.104 | 1997.3 | 75.0 | 9.1 | 9.1 | 2.296 | 489.2 | 453.7 |
| 11/30/06 | 9.433 | 3.116 | 2000.2 | 75.0 | 9.2 | 9.1 | 2.302 | 490.9 | 453.5 |
| 12/1/06 | 9.378 | 3.130 | 2002.7 | 75.0 | 9.2 | 9.1 | 2.310 | 492.6 | 452.9 |
| 12/2/06 | 9.327 | 3.143 | 2005.0 | 75.0 | 9.2 | 9.1 | 2.317 | 494.2 | 452.5 |
| 12/3/06 | 9.284 | 3.163 | 2007.0 | 75.0 | 9.2 | 9.1 | 2.328 | 496.3 | 451.4 |
| 12/4/06 | 9.245 | 3.189 | 2008.7 | 75.0 | 9.2 | 9.1 | 2.342 | 498.8 | 449.7 |
| 12/5/06 | 9.207 | 3.215 | 2010.4 | 75.0 | 9.2 | 9.1 | 2.356 | 501.3 | 448.1 |
| 12/6/06 | 9.171 | 3.246 | 2012.1 | 75.0 | 9.2 | 9.1 | 2.373 | 504.2 | 446.0 |
| 12/7/06 | 9.135 | 3.274 | 2013.7 | 75.0 | 9.2 | 9.1 | 2.389 | 506.8 | 444.2 |
| 12/8/06 | 9.102 | 3.315 | 2015.2 | 75.0 | 9.2 | 9.1 | 2.412 | 510.5 | 441.3 |
| 12/9/06 | 9.068 | 3.354 | 2016.7 | 75.0 | 9.2 | 9.1 | 2.434 | 513.9 | 438.5 |
| 12/10/06 | 9.035 | 3.399 | 2018.2 | 75.0 | 9.2 | 9.2 | 2.460 | 517.9 | 435.3 |
| 12/11/06 | 8.997 | 3.437 | 2019.7 | 75.0 | 9.3 | 9.2 | 2.481 | 521.3 | 432.6 |
| 12/12/06 | 8.960 | 3.486 | 2021.7 | 75.0 | 9.3 | 9.2 | 2.510 | 525.8 | 429.1 |
| 12/13/06 | 8.920 | 3.522 | 2023.5 | 75.0 | 9.3 | 9.2 | 2.531 | 529.1 | 426.7 |
| 12/14/06 | 8.889 | 3.570 | 2024.8 | 75.0 | 9.3 | 9.2 | 2.559 | 533.3 | 423.1 |
| 12/15/06 | 8.865 | 3.621 | 2025.5 | 75.0 | 9.3 | 9.2 | 2.589 | 537.6 | 419.2 |

same procedure used in the previous examples. Again, for the single launch vehicle case, the TCMs are accomplished using the orbiter ACS. Only the TPS mass and the post-aerocapture maneuver propellant need be considered. The results indicate that the post-capture mass is almost constant during the launch period. This is similar to the results obtained in scenarios 2 and 5 when aerocapture was also utilized.

The last case in this scenario employs aerobraking as the orbiter capture scheme. As before, three different capture orbits were included for this analysis. The calculation for the post-capture orbiter mass includes the MDM maneuver after orbiter-lander separation, the MOI maneuver for initial capture, and a nominal 50 kg of propellant for various orbital trim maneuvers (OTMs) during the aerobraking phase. Table 7.5 summarizes the results of the post-capture mass calculations. As was noted with

Table 7.4: Scenario 7 – (Aerocapture) Orbiter Post-Capture Mass (Delta III 2-stage)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | TPS Mass (kg) | Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1985.6 | 75 | 9.1 | 203.3 | 30.1 | 712.8 |
| 11/27/06 | 9.657 | 3.090 | 1990.0 | 75 | 9.1 | 203.3 | 30.2 | 714.9 |
| 11/28/06 | 9.572 | 3.097 | 1993.9 | 75 | 9.1 | 203.4 | 30.3 | 716.6 |
| 11/29/06 | 9.497 | 3.104 | 1997.3 | 75 | 9.1 | 203.6 | 30.4 | 718.1 |
| 11/30/06 | 9.434 | 3.116 | 2000.2 | 75 | 9.2 | 203.8 | 30.4 | 719.2 |
| 12/1/06 | 9.378 | 3.131 | 2002.7 | 75 | 9.2 | 204.0 | 30.5 | 720.2 |
| 12/2/06 | 9.327 | 3.143 | 2005.0 | 75 | 9.2 | 204.2 | 30.5 | 721.1 |
| 12/3/06 | 9.284 | 3.163 | 2007.0 | 75 | 9.2 | 204.6 | 30.5 | 721.7 |
| 12/4/06 | 9.245 | 3.189 | 2008.7 | 75 | 9.2 | 205.1 | 30.5 | 722.0 |
| 12/5/06 | 9.207 | 3.215 | 2010.4 | 75 | 9.2 | 205.5 | 30.6 | 722.4 |
| 12/6/06 | 9.171 | 3.246 | 2012.1 | 75 | 9.2 | 206.1 | 30.6 | 722.6 |
| 12/7/06 | 9.135 | 3.275 | 2013.7 | 75 | 9.2 | 206.7 | 30.6 | 722.9 |
| 12/8/06 | 9.101 | 3.312 | 2015.2 | 75 | 9.2 | 207.4 | 30.6 | 722.9 |
| 12/9/06 | 9.068 | 3.354 | 2016.7 | 75 | 9.2 | 208.2 | 30.6 | 722.8 |
| 12/10/06 | 9.035 | 3.399 | 2018.2 | 75 | 9.2 | 209.2 | 30.6 | 722.6 |
| 12/11/06 | 9.002 | 3.438 | 2019.7 | 75 | 9.3 | 210.0 | 30.6 | 722.6 |
| 12/12/06 | 8.960 | 3.487 | 2021.7 | 75 | 9.3 | 211.1 | 30.6 | 722.4 |
| 12/13/06 | 8.920 | 3.522 | 2023.5 | 75 | 9.3 | 211.8 | 30.6 | 722.6 |
| 12/14/06 | 8.889 | 3.570 | 2024.8 | 75 | 9.3 | 212.9 | 30.5 | 722.2 |
| 12/15/06 | 8.876 | 3.647 | 2025.5 | 75 | 9.3 | 214.7 | 30.5 | 720.7 |

scenarios 3 and 6 in which aerobraking was used, the post-capture mass begins to decrease about halfway through the launch period. This is attributed to the effect of the increasing $V_{\infty_A}$ during the launch period. As the slope of both $V_{\infty_A}$ and C3 decrease for later launch dates, the rate of change for the $V_{\infty_A}$ is still faster than that of the C3. Therefore, the magnitude of associated $\Delta V$ requirement at MOI also increases at a faster rate. This explains the sudden decrease in the post-capture orbiter mass for the aerobraking cases.

As the time to aerobrake is not as critical given the single launch vehicle assumptions, the initial capture orbit will most likely be selected based on the orbiter mission requirements. If the orbiter is required to provide the communications relay function for the lander however, aerobraking will not be a viable option. In either case, the time to aerobrake is provided in Table 7.5. As expected, the longer the period of the

Table 7.5: Scenario 7 - (Aerobraking) Orbiter Post-Capture Mass (Delta III 2-stage)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1985.6 | 75.0 | 9.1 | 9.1 | 1.207 | 1.105 |
| 11/27/06 | 9.656 | 3.090 | 1990.0 | 75.0 | 9.1 | 9.1 | 1.209 | 1.106 |
| 11/28/06 | 9.572 | 3.097 | 1993.9 | 75.0 | 9.1 | 9.1 | 1.213 | 1.110 |
| 11/29/06 | 9.497 | 3.104 | 1997.3 | 75.0 | 9.1 | 9.1 | 1.217 | 1.114 |
| 11/30/06 | 9.433 | 3.116 | 2000.2 | 75.0 | 9.2 | 9.2 | 1.223 | 1.120 |
| 12/1/06 | 9.378 | 3.130 | 2002.7 | 75.0 | 9.2 | 9.2 | 1.231 | 1.128 |
| 12/2/06 | 9.327 | 3.143 | 2005.0 | 75.0 | 9.2 | 9.2 | 1.238 | 1.135 |
| 12/3/06 | 9.284 | 3.163 | 2007.0 | 75.0 | 9.2 | 9.2 | 1.249 | 1.146 |
| 12/4/06 | 9.245 | 3.189 | 2008.7 | 75.0 | 9.2 | 9.2 | 1.263 | 1.160 |
| 12/5/06 | 9.207 | 3.215 | 2010.4 | 75.0 | 9.2 | 9.2 | 1.277 | 1.174 |
| 12/6/06 | 9.171 | 3.246 | 2012.1 | 75.0 | 9.2 | 9.2 | 1.294 | 1.191 |
| 12/7/06 | 9.135 | 3.274 | 2013.7 | 75.0 | 9.2 | 9.2 | 1.310 | 1.207 |
| 12/8/06 | 9.102 | 3.315 | 2015.2 | 75.0 | 9.2 | 9.2 | 1.333 | 1.230 |
| 12/9/06 | 9.068 | 3.354 | 2016.7 | 75.0 | 9.2 | 9.2 | 1.355 | 1.252 |
| 12/10/06 | 9.035 | 3.399 | 2018.2 | 75.0 | 9.2 | 9.2 | 1.381 | 1.278 |
| 12/11/06 | 8.997 | 3.437 | 2019.7 | 75.0 | 9.3 | 9.3 | 1.402 | 1.300 |
| 12/12/06 | 8.960 | 3.486 | 2021.7 | 75.0 | 9.3 | 9.3 | 1.431 | 1.328 |
| 12/13/06 | 8.920 | 3.522 | 2023.5 | 75.0 | 9.3 | 9.3 | 1.452 | 1.349 |
| 12/14/06 | 8.889 | 3.570 | 2024.8 | 75.0 | 9.3 | 9.3 | 1.480 | 1.377 |
| 12/15/06 | 8.865 | 3.621 | 2025.5 | 75.0 | 9.3 | 9.3 | 1.511 | 1.408 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 1.059 | 302.2 | 280.8 | 271.0 | 50 | 584.9 | 606.3 | 616.1 |
| 11/27/06 | 1.060 | 303.3 | 281.8 | 272.0 | 50 | 586.0 | 607.5 | 617.3 |
| 11/28/06 | 1.064 | 304.6 | 283.2 | 273.3 | 50 | 586.5 | 608.0 | 617.9 |
| 11/29/06 | 1.068 | 306.0 | 284.5 | 274.6 | 50 | 586.9 | 608.4 | 618.2 |
| 11/30/06 | 1.074 | 307.8 | 286.3 | 276.4 | 50 | 586.5 | 608.0 | 617.8 |
| 12/1/06 | 1.082 | 309.7 | 288.2 | 278.4 | 50 | 585.8 | 607.3 | 617.1 |
| 12/2/06 | 1.089 | 311.5 | 290.1 | 280.3 | 50 | 585.1 | 606.6 | 616.3 |
| 12/3/06 | 1.100 | 314.1 | 292.7 | 282.9 | 50 | 583.5 | 604.9 | 614.7 |
| 12/4/06 | 1.114 | 317.3 | 296.0 | 286.2 | 50 | 581.2 | 602.5 | 612.2 |
| 12/5/06 | 1.128 | 320.5 | 299.2 | 289.5 | 50 | 578.8 | 600.1 | 609.8 |
| 12/6/06 | 1.146 | 324.2 | 303.1 | 293.4 | 50 | 575.9 | 597.0 | 606.7 |
| 12/7/06 | 1.161 | 327.7 | 306.6 | 297.0 | 50 | 573.3 | 594.3 | 603.9 |
| 12/8/06 | 1.184 | 332.5 | 311.6 | 302.1 | 50 | 569.1 | 590.0 | 599.6 |
| 12/9/06 | 1.206 | 337.2 | 316.4 | 306.9 | 50 | 565.2 | 586.0 | 595.5 |
| 12/10/06 | 1.232 | 342.5 | 321.9 | 312.5 | 50 | 560.6 | 581.2 | 590.6 |
| 12/11/06 | 1.254 | 347.1 | 326.6 | 317.2 | 50 | 556.7 | 577.3 | 586.6 |
| 12/12/06 | 1.282 | 353.0 | 332.7 | 323.4 | 50 | 551.8 | 572.1 | 581.4 |
| 12/13/06 | 1.303 | 357.4 | 337.2 | 328.0 | 50 | 548.3 | 568.5 | 577.7 |
| 12/14/06 | 1.332 | 363.1 | 343.1 | 333.9 | 50 | 543.2 | 563.3 | 572.4 |
| 12/15/06 | 1.362 | 369.1 | 349.2 | 340.1 | 50 | 537.6 | 557.5 | 566.6 |

initial capture orbit, the longer it takes to aerobrake. Additionally, the decreasing post-capture mass during the launch period reduces the amount of time required to aerobrake.

A comparison of the post-capture orbiter masses corresponding to the three capture schemes is presented in Figure 7.2. Aerocapture appears to be the most cost-

Table 7.6: Scenario 7 - Aerobraking Time Estimate

| AB Time<br>Launch | 15 hr orbit<br>(days) | 27 hr orbit<br>(days) | 39 hr orbit<br>(days) |
|---|---|---|---|
| 11/26/06 | 46.8 | 74.5 | 94.9 |
| 11/27/06 | 46.9 | 74.7 | 95.1 |
| 11/28/06 | 47.0 | 74.8 | 95.2 |
| 11/29/06 | 47.0 | 74.8 | 95.3 |
| 11/30/06 | 47.0 | 74.8 | 95.2 |
| 12/1/06 | 46.9 | 74.7 | 95.1 |
| 12/2/06 | 46.8 | 74.6 | 95.0 |
| 12/3/06 | 46.7 | 74.4 | 94.7 |
| 12/4/06 | 46.5 | 74.1 | 94.3 |
| 12/5/06 | 46.3 | 73.8 | 94.0 |
| 12/6/06 | 46.1 | 73.4 | 93.5 |
| 12/7/06 | 45.9 | 73.1 | 93.1 |
| 12/8/06 | 45.6 | 72.6 | 92.4 |
| 12/9/06 | 45.3 | 72.1 | 91.8 |
| 12/10/06 | 44.9 | 71.5 | 91.0 |
| 12/11/06 | 44.6 | 71.0 | 90.4 |
| 12/12/06 | 44.2 | 70.4 | 89.6 |
| 12/13/06 | 43.9 | 69.9 | 89.1 |
| 12/14/06 | 43.5 | 69.3 | 88.3 |
| 12/15/06 | 43.1 | 68.6 | 87.4 |

effective capture scheme for the single launch vehicle with orbiter and lander travelling on a type 4- trajectory. Note the significant difference between the aerocapture post-capture mass and the mass corresponding to the other two capture schemes. Recall for the two launch vehicle scenarios that the orbiter included a cruise stage when using aerocapture. The orbiter, when using propulsive capture or aerobraking however, did not include a cruise stage. In this scenario for a single launch vehicle, the extra mass of the cruise stage when subtracted from the injected mass decreases the post-capture mass for the propulsive and aerobraking capture schemes thus explaining the larger disparity.

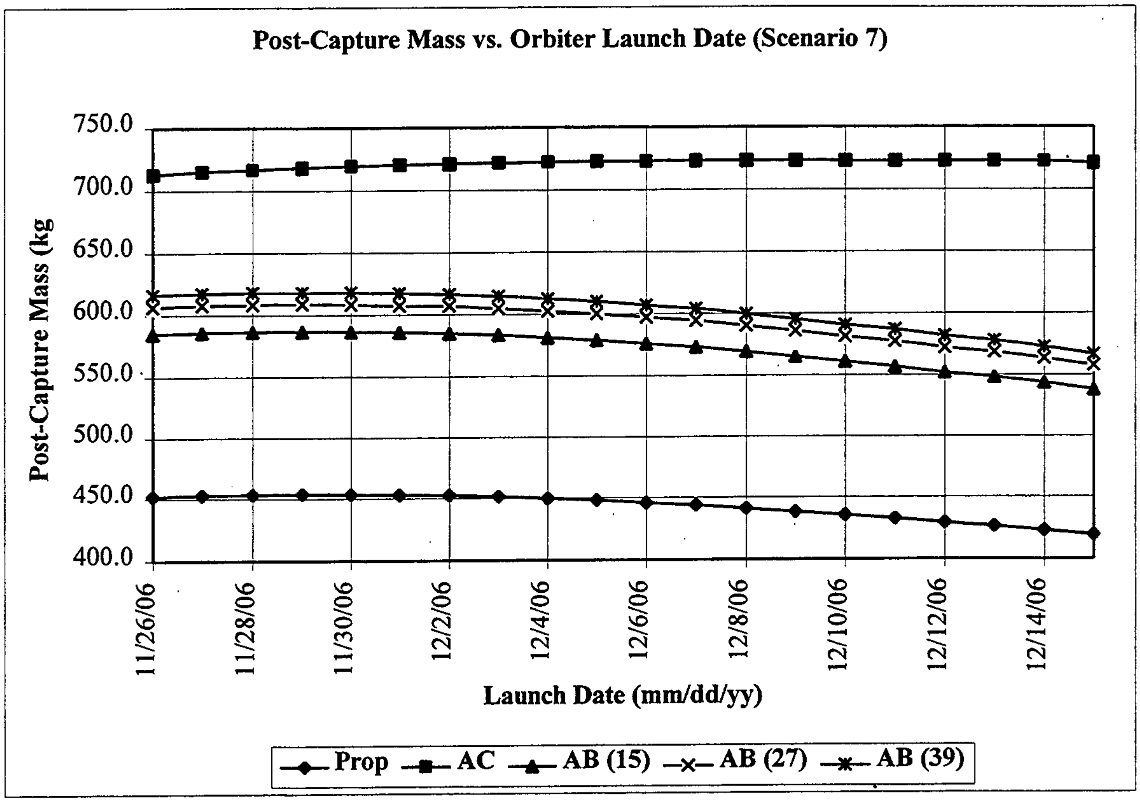


Figure 7.1: Scenario 7 - Post-Capture Orbiter Mass

## C. SCENARIO 8: ANALYSIS

The analysis for scenario 8 follows closely to the analysis in scenario 7. The post-capture orbiter mass calculations for each capture scheme are presented in Tables 7.7-7.9. The time for the orbiter to aerobrake is also included in Table 7.10. For the post-capture mass calculations, a Delta III (2-stage) launch vehicle was also used in this scenario for purposes of comparison. The post-capture orbiter masses for the corresponding capture schemes are shown together in Figure 7.2. Note a similar outcome to that observed in scenario 7. The aerocapture again appears to be the most cost-effective capture scheme for this scenario.

Table 7.7: Scenario 8 – (Propulsive) Orbiter Post-Capture Mass (Delta III 2-stage)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Captur (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 1828.7 | 75.0 | 8.3 | 8.3 | 2.025 | 409.1 | 451.2 |
| 9/14/07 | 13.130 | 2.578 | 1833.6 | 75.0 | 8.4 | 8.3 | 2.030 | 411.0 | 451.7 |
| 9/15/07 | 13.040 | 2.594 | 1837.6 | 75.0 | 8.4 | 8.3 | 2.038 | 413.0 | 451.6 |
| 9/16/07 | 12.960 | 2.637 | 1841.1 | 75.0 | 8.4 | 8.3 | 2.058 | 416.7 | 449.6 |
| 9/17/07 | 12.910 | 2.634 | 1843.8 | 75.0 | 8.4 | 8.3 | 2.057 | 417.2 | 450.5 |
| 9/18/07 | 12.860 | 2.662 | 1845.6 | 75.0 | 8.4 | 8.3 | 2.070 | 419.5 | 449.0 |
| 9/19/07 | 12.820 | 2.696 | 1847.4 | 75.0 | 8.4 | 8.4 | 2.087 | 422.3 | 447.1 |
| 9/20/07 | 12.800 | 2.709 | 1848.3 | 75.0 | 8.4 | 8.4 | 2.093 | 423.4 | 446.5 |
| 9/21/07 | 12.790 | 2.757 | 1848.7 | 75.0 | 8.4 | 8.4 | 2.116 | 426.8 | 443.2 |
| 9/22/07 | 12.800 | 2.774 | 1848.7 | 75.0 | 8.4 | 8.4 | 2.125 | 428.0 | 442.0 |
| 9/23/07 | 12.810 | 2.796 | 1847.8 | 75.0 | 8.4 | 8.4 | 2.136 | 429.3 | 440.3 |
| 9/24/07 | 12.830 | 2.858 | 1847.4 | 75.0 | 8.4 | 8.4 | 2.167 | 433.6 | 435.8 |
| 9/25/07 | 12.850 | 2.876 | 1846.1 | 75.0 | 8.4 | 8.3 | 2.176 | 434.5 | 434.2 |
| 9/26/07 | 12.890 | 2.901 | 1844.7 | 75.0 | 8.4 | 8.3 | 2.189 | 436.0 | 432.1 |
| 9/27/07 | 12.920 | 2.956 | 1842.9 | 75.0 | 8.4 | 8.3 | 2.217 | 439.4 | 427.8 |
| 9/28/07 | 12.950 | 3.008 | 1841.6 | 75.0 | 8.4 | 8.3 | 2.244 | 442.8 | 423.8 |
| 9/29/07 | 12.990 | 3.053 | 1839.8 | 75.0 | 8.4 | 8.3 | 2.268 | 445.5 | 420.2 |
| 9/30/07 | 13.040 | 3.142 | 1837.6 | 75.0 | 8.4 | 8.3 | 2.316 | 451.3 | 413.3 |
| 10/1/07 | 13.110 | 3.150 | 1834.5 | 75.0 | 8.4 | 8.3 | 2.320 | 451.1 | 412.0 |
| 10/2/07 | 12.990 | 3.235 | 1839.8 | 75.0 | 8.4 | 8.3 | 2.367 | 458.5 | 407.1 |

Table 7.8: Scenario 8 - (Aerocapture) Orbiter Post-Capture Mass (Delta III 2-stage)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | TPS Mass (kg) | Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 1828.7 | 75 | 8.3 | 196.7 | 27.3 | 644.5 |
| 9/14/07 | 13.130 | 2.578 | 1833.6 | 75 | 8.4 | 196.8 | 27.4 | 646.8 |
| 9/15/07 | 13.040 | 2.594 | 1837.6 | 75 | 8.4 | 196.9 | 27.4 | 648.5 |
| 9/16/07 | 12.960 | 2.637 | 1841.1 | 75 | 8.4 | 197.3 | 27.5 | 649.9 |
| 9/17/07 | 12.910 | 2.634 | 1843.8 | 75 | 8.4 | 197.3 | 27.5 | 651.2 |
| 9/18/07 | 12.860 | 2.662 | 1845.6 | 75 | 8.4 | 197.6 | 27.6 | 651.7 |
| 9/19/07 | 12.820 | 2.696 | 1847.4 | 75 | 8.4 | 197.9 | 27.6 | 652.3 |
| 9/20/07 | 12.800 | 2.709 | 1848.3 | 75 | 8.4 | 198.1 | 27.6 | 652.5 |
| 9/21/07 | 12.790 | 2.757 | 1848.7 | 75 | 8.4 | 198.6 | 27.6 | 652.2 |
| 9/22/07 | 12.800 | 2.774 | 1848.7 | 75 | 8.4 | 198.8 | 27.6 | 652.1 |
| 9/23/07 | 12.810 | 2.796 | 1847.8 | 75 | 8.4 | 199.1 | 27.6 | 651.4 |
| 9/24/07 | 12.830 | 2.858 | 1847.4 | 75 | 8.4 | 199.8 | 27.5 | 650.4 |
| 9/25/07 | 12.850 | 2.876 | 1846.1 | 75 | 8.4 | 200.1 | 27.5 | 649.5 |
| 9/26/07 | 12.890 | 2.901 | 1844.7 | 75 | 8.4 | 200.4 | 27.4 | 648.6 |
| 9/27/07 | 12.920 | 2.956 | 1842.9 | 75 | 8.4 | 201.2 | 27.4 | 647.0 |
| 9/28/07 | 12.950 | 3.008 | 1841.6 | 75 | 8.4 | 202.0 | 27.3 | 645.6 |
| 9/29/07 | 12.990 | 3.053 | 1839.8 | 75 | 8.4 | 202.7 | 27.2 | 644.1 |
| 9/30/07 | 13.040 | 3.142 | 1837.6 | 75 | 8.4 | 204.2 | 27.1 | 641.6 |
| 10/1/07 | 13.110 | 3.150 | 1834.5 | 75 | 8.4 | 204.4 | 27.1 | 639.9 |
| 10/2/07 | 12.990 | 3.235 | 1839.8 | 75 | 8.4 | 205.9 | 27.1 | 641.0 |

Table 7.9: Scenario 8 - (Aerobraking) Orbiter Post-Capture Mass (Delta III 2-stage)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 1828.7 | 75.0 | 8.3 | 8.3 | 0.946 | 0.843 |
| 9/14/07 | 13.130 | 2.578 | 1833.6 | 75.0 | 8.4 | 8.4 | 0.951 | 0.849 |
| 9/15/07 | 13.040 | 2.594 | 1837.6 | 75.0 | 8.4 | 8.4 | 0.959 | 0.856 |
| 9/16/07 | 12.960 | 2.637 | 1841.1 | 75.0 | 8.4 | 8.4 | 0.979 | 0.876 |
| 9/17/07 | 12.910 | 2.634 | 1843.8 | 75.0 | 8.4 | 8.4 | 0.978 | 0.875 |
| 9/18/07 | 12.860 | 2.662 | 1845.6 | 75.0 | 8.4 | 8.4 | 0.991 | 0.888 |
| 9/19/07 | 12.820 | 2.696 | 1847.4 | 75.0 | 8.4 | 8.4 | 1.008 | 0.905 |
| 9/20/07 | 12.800 | 2.709 | 1848.3 | 75.0 | 8.4 | 8.4 | 1.014 | 0.911 |
| 9/21/07 | 12.790 | 2.757 | 1848.7 | 75.0 | 8.4 | 8.4 | 1.037 | 0.935 |
| 9/22/07 | 12.800 | 2.774 | 1848.7 | 75.0 | 8.4 | 8.4 | 1.046 | 0.943 |
| 9/23/07 | 12.810 | 2.796 | 1847.8 | 75.0 | 8.4 | 8.4 | 1.057 | 0.954 |
| 9/24/07 | 12.830 | 2.858 | 1847.4 | 75.0 | 8.4 | 8.4 | 1.088 | 0.985 |
| 9/25/07 | 12.850 | 2.876 | 1846.1 | 75.0 | 8.4 | 8.4 | 1.097 | 0.994 |
| 9/26/07 | 12.890 | 2.901 | 1844.7 | 75.0 | 8.4 | 8.4 | 1.110 | 1.007 |
| 9/27/07 | 12.920 | 2.956 | 1842.9 | 75.0 | 8.4 | 8.4 | 1.138 | 1.036 |
| 9/28/07 | 12.950 | 3.008 | 1841.6 | 75.0 | 8.4 | 8.4 | 1.165 | 1.063 |
| 9/29/07 | 12.990 | 3.053 | 1839.8 | 75.0 | 8.4 | 8.4 | 1.189 | 1.087 |
| 9/30/07 | 13.040 | 3.142 | 1837.6 | 75.0 | 8.4 | 8.4 | 1.237 | 1.134 |
| 10/1/07 | 13.110 | 3.150 | 1834.5 | 75.0 | 8.4 | 8.4 | 1.241 | 1.139 |
| 10/2/07 | 12.990 | 3.235 | 1839.8 | 75.0 | 8.4 | 8.4 | 1.288 | 1.185 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 0.798 | 226.1 | 204.7 | 194.9 | 50 | 584.1 | 605.4 | 615.2 |
| 9/14/07 | 0.803 | 227.8 | 206.4 | 196.6 | 50 | 584.8 | 606.2 | 616.0 |
| 9/15/07 | 0.810 | 229.8 | 208.4 | 198.7 | 50 | 584.7 | 606.1 | 615.9 |
| 9/16/07 | 0.831 | 234.5 | 213.2 | 203.4 | 50 | 581.8 | 603.1 | 612.8 |
| 9/17/07 | 0.829 | 234.5 | 213.2 | 203.4 | 50 | 583.0 | 604.4 | 614.1 |
| 9/18/07 | 0.843 | 237.5 | 216.2 | 206.5 | 50 | 580.9 | 602.2 | 612.0 |
| 9/19/07 | 0.859 | 241.1 | 219.9 | 210.2 | 50 | 578.2 | 599.4 | 609.1 |
| 9/20/07 | 0.865 | 242.5 | 221.3 | 211.7 | 50 | 577.3 | 598.4 | 608.1 |
| 9/21/07 | 0.889 | 247.3 | 226.3 | 216.7 | 50 | 572.7 | 593.7 | 603.3 |
| 9/22/07 | 0.897 | 249.0 | 228.0 | 218.5 | 50 | 571.0 | 592.0 | 601.5 |
| 9/23/07 | 0.908 | 251.1 | 230.2 | 220.6 | 50 | 568.5 | 589.4 | 598.9 |
| 9/24/07 | 0.939 | 257.2 | 236.5 | 227.1 | 50 | 562.1 | 582.8 | 592.2 |
| 9/25/07 | 0.949 | 258.8 | 238.2 | 228.8 | 50 | 559.9 | 580.5 | 589.9 |
| 9/26/07 | 0.961 | 261.1 | 240.6 | 231.3 | 50 | 556.9 | 577.4 | 586.7 |
| 9/27/07 | 0.990 | 266.4 | 246.1 | 236.8 | 50 | 550.8 | 571.0 | 580.3 |
| 9/28/07 | 1.017 | 271.4 | 251.3 | 242.2 | 50 | 545.1 | 565.1 | 574.3 |
| 9/29/07 | 1.041 | 275.7 | 255.8 | 246.7 | 50 | 539.9 | 559.8 | 568.9 |
| 9/30/07 | 1.089 | 284.4 | 264.8 | 255.9 | 50 | 530.1 | 549.7 | 558.7 |
| 10/1/07 | 1.093 | 284.7 | 265.2 | 256.3 | 50 | 528.3 | 547.8 | 556.7 |
| 10/2/07 | 1.140 | 294.3 | 275.0 | 266.2 | 50 | 521.3 | 540.6 | 549.5 |

Table 7.10: Scenario 8 - Aerobraking Time Estimate

| AB Time Launch | 15 hr orbit (days) | 27 hr orbit (days) | 39 hr orbit (days) |
|---|---|---|---|
| 9/13/07 | 46.8 | 74.4 | 94.8 |
| 9/14/07 | 46.8 | 74.5 | 94.9 |
| 9/15/07 | 46.8 | 74.5 | 94.9 |
| 9/16/07 | 46.6 | 74.2 | 94.4 |
| 9/17/07 | 46.7 | 74.3 | 94.6 |
| 9/18/07 | 46.5 | 74.0 | 94.3 |
| 9/19/07 | 46.3 | 73.7 | 93.9 |
| 9/20/07 | 46.2 | 73.6 | 93.7 |
| 9/21/07 | 45.9 | 73.0 | 93.0 |
| 9/22/07 | 45.7 | 72.8 | 92.7 |
| 9/23/07 | 45.5 | 72.5 | 92.3 |
| 9/24/07 | 45.0 | 71.7 | 91.3 |
| 9/25/07 | 44.8 | 71.4 | 90.9 |
| 9/26/07 | 44.6 | 71.0 | 90.5 |
| 9/27/07 | 44.1 | 70.2 | 89.5 |
| 9/28/07 | 43.7 | 69.5 | 88.6 |
| 9/29/07 | 43.3 | 68.9 | 87.7 |
| 9/30/07 | 42.5 | 67.6 | 86.2 |
| 10/1/07 | 42.3 | 67.4 | 85.9 |
| 10/2/07 | 41.8 | 66.5 | 84.8 |

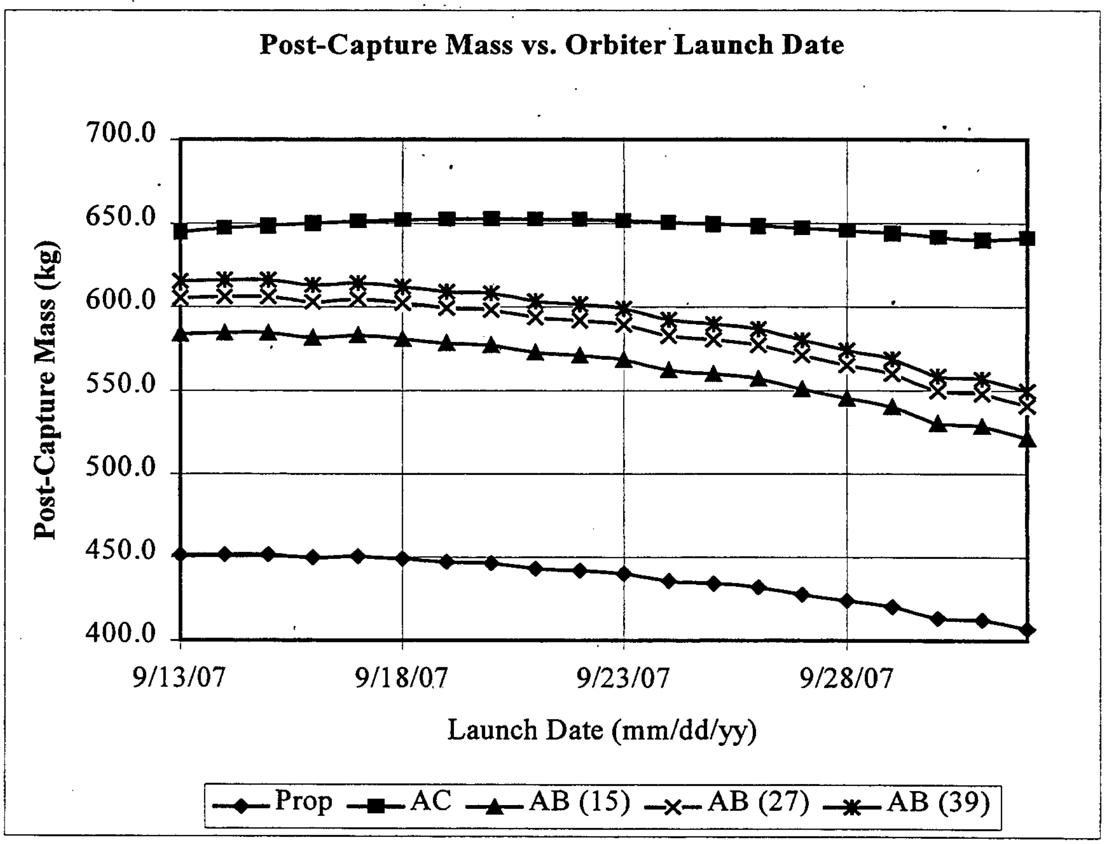


Figure 7.2: Scenario 8 - Post Capture Orbiter Mass

## D. COMPARISON OF SCENARIOS 7 AND 8

Now that the analysis for the single launch vehicle is complete, the type 4- trajectory cases from scenario 7 can be compared with the type 2 trajectory cases from scenario 8 to determine the best trajectory and capture scheme for the single launch vehicle configuration. This determination is made using the post-capture orbiter mass as the criterion for comparison. Figures 7.1 and 7.2 demonstrated that aerocapture was clearly the best capture scheme for both trajectories, but it was unclear without comparison, which trajectory was better. Therefore, the results from each trajectory were categorized by capture scheme and compared on the same plot. For example, Figure 7.3 depicts the type 4- propulsive capture case as compared against the type 2 propulsive capture case. This plot shows graphically that the type 4- propulsive capture case

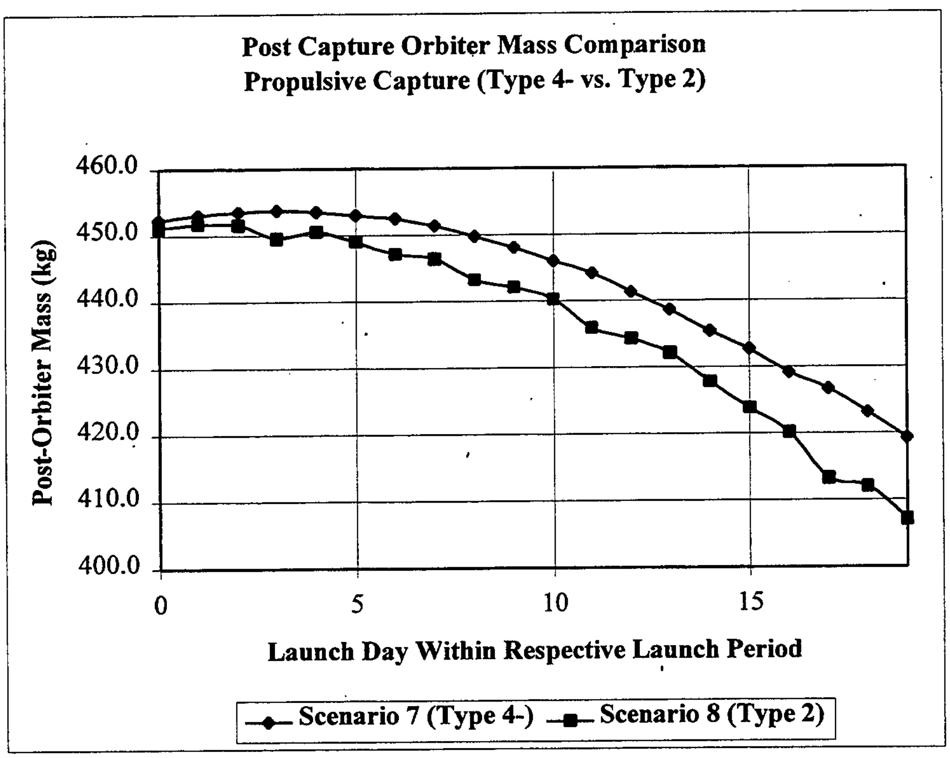


Figure 7.3: Single Launch Vehicle Type 4- vs. Type 2 Comparison (Propulsive)

is better than the type 2 propulsive capture case. Similar comparisons for the other capture schemes can also be made to show which trajectory is better should the mission designer be constrained to a particular capture scheme. Figure 7.4 shows an overwhelming advantage to launching on a type 4- trajectory when aerocapture is the

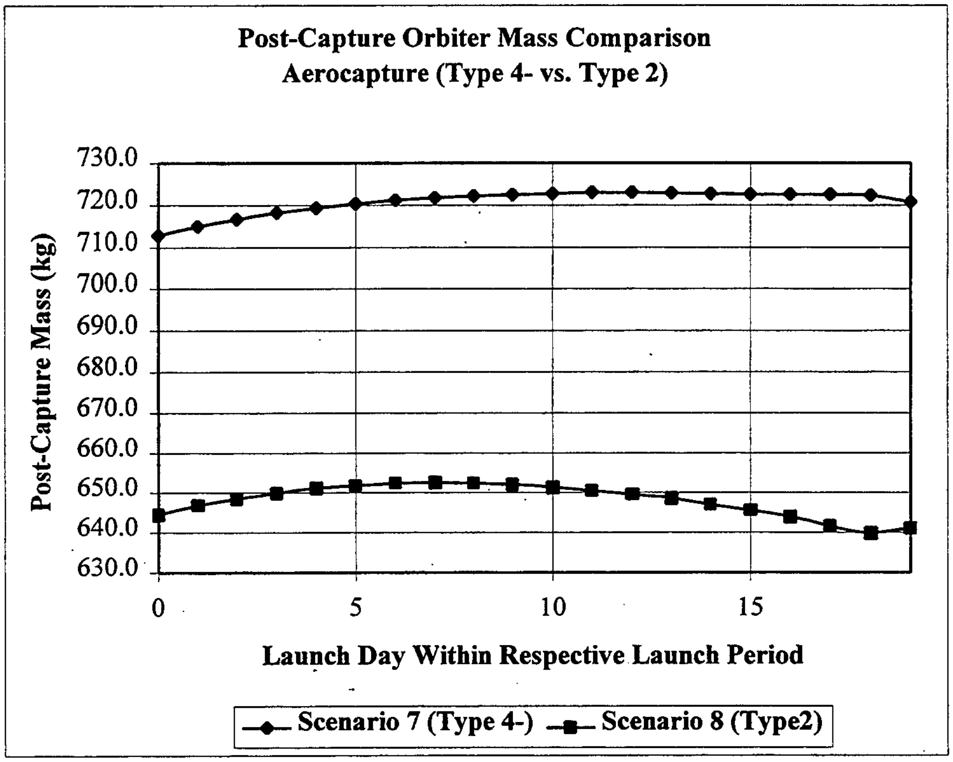


Figure 7.4: Single Launch Vehicle Type 4- vs. Type 2 Comparison (Aerocapture)

capture scheme selected. Lastly, Figure 7.5 depicts the comparison for aerobraking using the 27 hour initial capture orbit. The 15 hour and 39 hour initial orbits produced similar results. The aerobraking supports the results from the propulsive and aerocapture cases in that the type 4- trajectory is more beneficial to post-capture orbiter mass than the type 2 trajectory when using aerobraking. A conclusion can be drawn from these observations. The type 4- trajectory for the single launch vehicle configuration is more cost-effective in terms of post-capture orbiter mass than the type 2 trajectory.

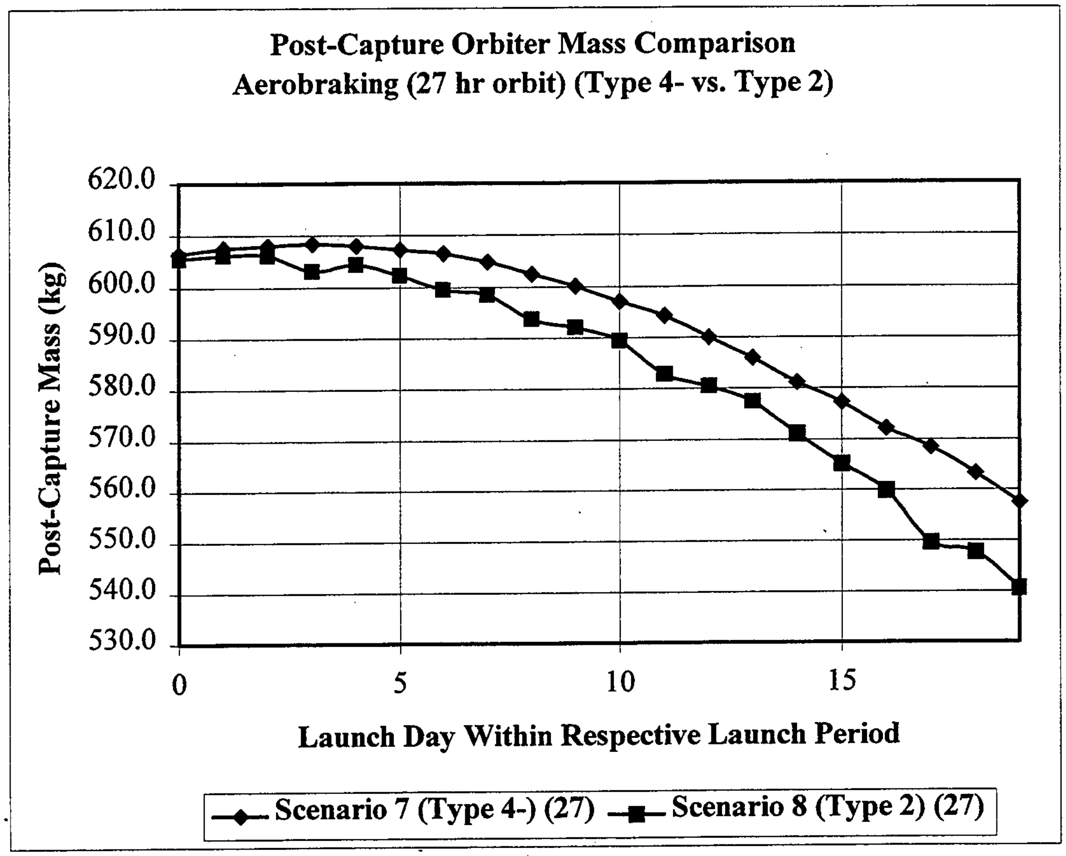


Figure 7.5: Single Launch Vehicle Type 4- vs. Type 2 Comparison (Aerobraking)

# VIII. SAMPLE RETURN CONTINGENCY

## A. BACKGROUND

The baseline mission objective for 2007 does not include a sample return. However, with recent changes in the Mars program coupled with uncertainties in the 2005 Sample Return mission, it is prudent to plan for this possibility in 2007. Additionally, the 2009 Mars mission is scheduled to be a sample return. The possibility always exists that the 2007 mission could be used to fulfill that objective. A typical sample return mission would consist of an orbiter, a lander/rover/Mars ascent vehicle (MAV). The samples to be returned will either already be cached by a previous mission or will be cached by the lander/rover upon arrival. If previously cached, the rover will require additional time to locate, collect, and return them to the MAV. It is desirable to minimize the time on the surface as the likelihood of problem occurrence will increase as a function of time. Once the samples have been successfully gathered and loaded, the MAV will launch from the surface into some nominal orbit and await rendezvous with the orbiter. The burden of rendezvous is placed on the orbiter. The MAV will not be capable of conducting plane changes although an off-azimuth launch is a possibility to meet a particular inclination requirement. The penalty for doing so must be weighed against the launch capability of the MAV. For rendezvous, the orbiter will conduct a series of maneuvers as necessary to enter the same orbital plane as the MAV. When the phasing is right, the orbiter will maneuver to intercept the MAV. Once rendezvous has been accomplished, the orbiter will retain the sample canister and jettison the MAV before conducting additional phasing maneuvers in preparation for the trans-Earth injection. Upon arrival at Earth, the sample canister will be jettisoned at some pre-determined time and will enter into a ballistic re-entry trajectory at Earth. The Utah Test and Training Range (UTTR) is the preferred landing site for the return sample although the possibility of this depends on the arrival trajectory parameters.

## B. RETURN TRAJECTORY SELECTION, TRAJECTORY ANALYSIS

This analysis is preliminary in nature with the focus on the following:

1) Defining a suitable return trajectory given the analysis of the previous scenarios;
2) Determination of the possible duration of surface operations given the timeline constraints of the trajectories;
3) Calculation of the $\Delta V$ requirements for injection back to Earth;
4) Geometry calculations;
5) Determination of the ballistic landing region upon arrival back at Earth;

Identifying the possible return trajectories began using the JPL study that had already determined the feasible ballistic trajectories for the Mars-Earth transfer. Table 8.1 summarizes those possible Mars-Earth trajectories that could be used in conjunction with

Table 8.1: Feasible Ballistic Trajectories for Mars-Earth Transfer

| Year | Trajectory Type | C3 | DLA | VHP | DAP | Launch Date | Arrival Date |
|---|---|---|---|---|---|---|---|
| 2009 | 1 | 9.4 | 9.8 | 3.2 | -21.4 | 8/22/09 | 5/10/10 |
| 2009 | 2 | 7.8 | 11.1 | 2.9 | -21.9 | 7/28/09 | 5/16/10 |
| 2010 | 3- | 13.1 | 6 | 4.8 | -5.5 | 6/9/10 | 9/10/12 |
| 2010 | 3+ | 10.6 | 12.7 | 3.7 | 21.2 | 8/26/10 | 9/17/12 |
| 2010 | 4- | 9.2 | -29.6 | 4.5 | 50.9 | 4/6/10 | 9/29/12 |
| 2011 | 1 | 6.8 | 3.3 | 3.5 | 5.1 | 8/12/11 | 7/10/12 |
| 2011 | 2 | 6.8 | 3.3 | 3.5 | 5.1 | 8/12/11 | 7/10/12 |
| 2011 | 4- | 15 | -21.8 | 4.3 | 46.4 | 12/29/11 | 9/14/14 |

the Earth-Mars trajectories previously analyzed in scenarios 1-8 (Matousek and Sergeyevsky, 1998, p.4-5). The trajectories also include the nominal launch and arrival dates. As the return trajectory only requires a single injection date, a launch period is not considered in the same sense as would be the case for a Earth-Mars transfer. Therefore, in the initial selection process for the return trajectory, only the nominal injection date was considered.

A review of the trajectory parameters in Table 8.1 indicates two possible return trajectories in the 2009 timeframe corresponding to a 2010 arrival at Earth. The type 1 and type 2 trajectories present reasonable opportunities with the type 2 trajectory requiring less $\Delta V$ for injection back to Earth. The 2010 trajectories all require considerably more $\Delta V$ to inject back to Earth. In addition, all are either type 3 or type 4 trajectories requiring much longer flight times. Utilizing one of these options for return to Earth would result in a 2012 arrival and almost 6 years of mission operations. The other possibility to consider would be the 2011 trajectories. While the type 1 and type 2 in this timeframe provide excellent opportunities with respect to the trajectory parameters, they require a considerably longer stay time for the lander/return vehicle. The 2011 type 4 trajectory also requires a relatively large amount of $\Delta V$ for injection. Therefore, the 2011 type 4 trajectory was not considered further.

Before selecting the optimum return trajectory, the arrival dates for the lander should be considered to ensure that there are a sufficient number of days available for surface operations, MAV launch, rendezvous, and orbiter phasing prior to return. Given the lander arrival dates from scenarios 1-8, Table 8.2 summarizes the calculations for the minimum and maximum operations time corresponding to each Mars-Earth return trajectory. Note that the only two trajectories not requiring the orbiter to remain at Mars for more than a year are the 2009 type 1 and type 2 trajectories. The operations time using either of these trajectories are sufficient in most cases with the exception of scenario 3 in which the minimum operations time is not adequate. In this scenario, recall that the orbiter and lander are both launched on separate launch vehicles using a type 4-

Table 8.2: Calculations for Minimum/Maximum Operations Time

| | | | Scenarios 1,2 | Type 4- | Scenario 3 | Type 4- | Scenarios 4,5, | Type 4- |
|---|---|---|---|---|---|---|---|---|
| | | | Arrival 1 | Arrival 2 | Arrival 1 | Arrival 2 | Arrival 1 | Arrival 2 |
| | | | 3/30/09 | 5/1/09 | 4/28/09 | 7/17/09 | 1/23/09 | 3/3/09 |
| Year | Type | Launch Date | Max Ops | Min Ops | Max Ops | Min Ops | Max Ops | Min Ops |
| 2009 | 1 | 8/22/09 | 145 | 113 | 116 | 36 | 211 | 172 |
| 2009 | 2 | 7/28/09 | 120 | 88 | 91 | 11 | 186 | 147 |
| 2010 | 3- | 6/9/10 | 436 | 404 | 407 | 327 | 502 | 463 |
| 2010 | 3+ | 8/26/10 | 514 | 482 | 485 | 405 | 580 | 541 |
| 2010 | 4- | 4/6/10 | 372 | 340 | 343 | 263 | 438 | 399 |
| 2011 | 1 | 8/12/11 | 865 | 833 | 836 | 756 | 931 | 892 |
| 2011 | 2 | 8/12/11 | 865 | 833 | 836 | 756 | 931 | 892 |
| 2011 | 4- | 12/29/11 | 1004 | 972 | 975 | 895 | 1070 | 1031 |

| | | | Scenario 6 | Type 4- | Scenario 8 | Type 2 |
|---|---|---|---|---|---|---|
| | | | Arrival 1 | Arrival 2 | Arrival 1 | Arrival 2 |
| | | | 1/23/09 | 4/8/09 | 8/31/08 | 10/23/08 |
| Year | Type | Launch Date | Max Ops | Min Ops | Max Ops | Min Ops |
| 2009 | 1 | 8/22/09 | 211 | 136 | 356 | 303 |
| 2009 | 2 | 7/28/09 | 186 | 111 | 331 | 278 |
| 2010 | 3- | 6/9/10 | 502 | 427 | 647 | 594 |
| 2010 | 3+ | 8/26/10 | 580 | 505 | 725 | 672 |
| 2010 | 4- | 4/6/10 | 438 | 363 | 583 | 530 |
| 2011 | 1 | 8/12/11 | 931 | 856 | 1076 | 1023 |
| 2011 | 2 | 8/12/11 | 931 | 856 | 1076 | 1023 |
| 2011 | 4- | 12/29/11 | 1070 | 995 | 1215 | 1162 |

trajectory. The lander arrival time is delayed due to the time required for the orbiter to aerobrake. In addition to the extended wait time to return, the higher $\Delta V$ requirements and longer flight times for the 2010 return trajectories preclude further consideration of these. Therefore, the 2009 type 1 and type 2 trajectories are the likely candidates.

A contour plot of the 2009 type 1 and 2 trajectories is shown in Figure 8.1. Note the relatively large region on the type 2 side in which the C3 is below 10. Also note the location of the minimum energy contour for the type 1 trajectory. The proximity to the "ridge" may require a broken plane maneuver to accomplish the trajectory making it cost

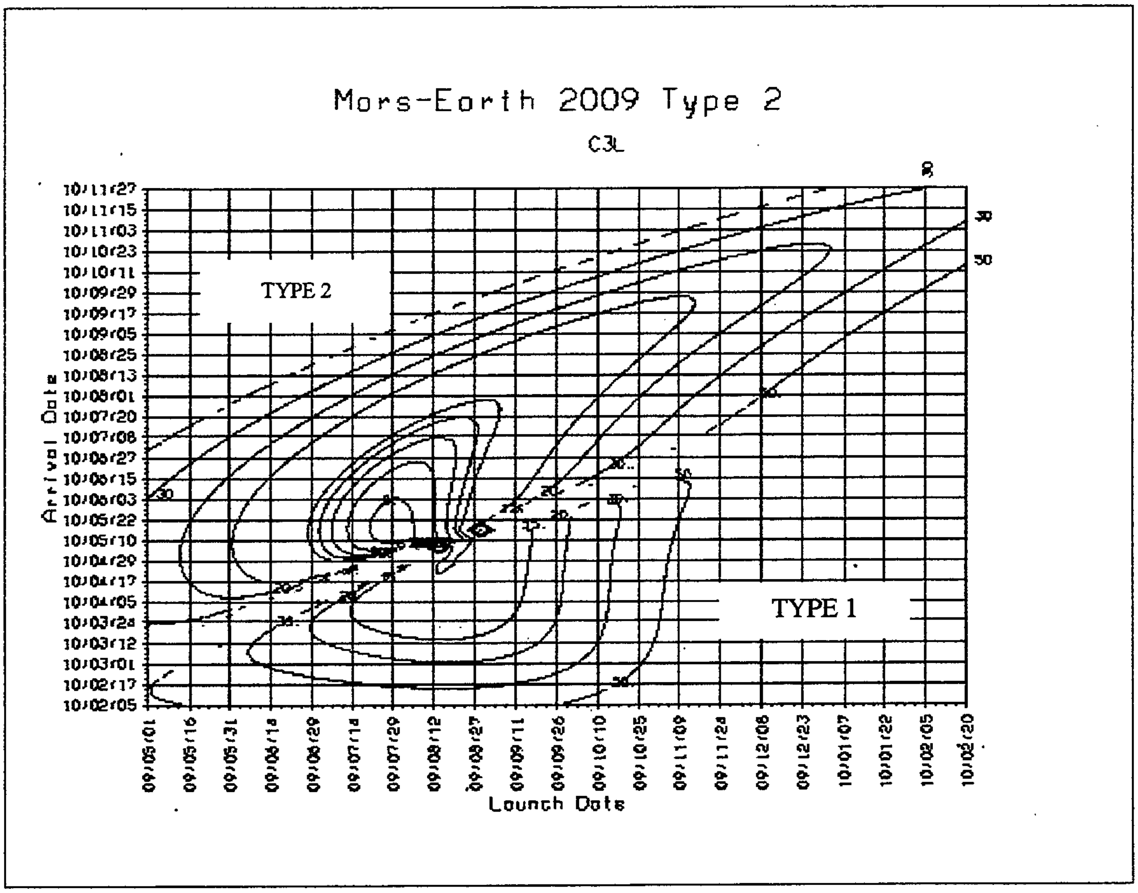


Figure 8.1: Mars-Earth 2009 Type 1 / 2 Trajectory C3 Contour (MSR)

prohibitive from a propellant standpoint. As the type 2 trajectory appears to provide more promising parameters $\left(C3, V_{\infty_A}\right)$ while incurring only a small penalty in flight time, it was chosen as the primary return trajectory for the Mars-Earth transfer.

The analysis for the type 2 trajectory does not include CATO. As sample return is not the baseline mission for 2007, MIDAS was deemed sufficient for purposes of this preliminary analysis. As mentioned previously, the injection date is rather arbitrary provided the return vehicle is properly positioned. Table 8.3 depicts the trajectory analysis parameters from the minimum C3 trajectories associated with each injection date. The trajectories were designed to depart Mars from an altitude of 250 km and arrive at an altitude of 300 km at Earth. The results in Table 8.3 were obtained using MIDAS.

Table 8.3: Mars-Earth 2009 Trajectory Analysis Parameters (MIDAS)

| Injection | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 06/30/09 | 05/15/10 | 9.637 | 16.5 | 46.4 | 2.931 | -28.0 | 339.7 |
| 07/01/09 | 05/15/10 | 9.515 | 16.5 | 46.2 | 2.933 | -27.8 | 340.2 |
| 07/02/09 | 05/16/10 | 9.398 | 16.2 | 46.0 | 2.926 | -27.5 | 340.1 |
| 07/03/09 | 05/16/10 | 9.285 | 16.0 | 45.9 | 2.922 | -27.3 | 340.1 |
| 07/04/09 | 05/16/10 | 9.175 | 15.9 | 45.7 | 2.923 | -27.1 | 340.6 |
| 07/05/09 | 05/16/10 | 9.070 | 15.7 | 45.5 | 2.917 | -26.9 | 340.5 |
| 07/06/09 | 05/16/10 | 8.968 | 15.7 | 45.2 | 2.921 | -26.7 | 341.1 |
| 07/07/09 | 05/16/10 | 8.871 | 15.4 | 45.0 | 2.917 | -26.5 | 341.1 |
| 07/08/09 | 05/16/10 | 8.778 | 15.2 | 44.8 | 2.912 | -26.3 | 341.1 |
| 07/09/09 | 05/17/10 | 8.689 | 15.0 | 44.6 | 2.910 | -26.1 | 341.3 |
| 07/10/09 | 05/17/10 | 8.604 | 14.8 | 44.3 | 2.910 | -25.8 | 341.6 |
| 07/11/09 | 05/17/10 | 8.523 | 14.7 | 44.0 | 2.912 | -25.7 | 341.9 |
| 07/12/09 | 05/17/10 | 8.447 | 14.4 | 43.8 | 2.905 | -25.4 | 341.8 |
| 07/13/09 | 05/17/10 | 8.375 | 14.2 | 43.6 | 2.902 | -25.2 | 341.8 |
| 07/14/09 | 05/17/10 | 8.307 | 13.9 | 43.3 | 2.900 | -25.0 | 342.0 |
| 07/15/09 | 05/17/10 | 8.243 | 13.8 | 42.9 | 2.903 | -24.8 | 342.4 |
| 07/16/09 | 05/17/10 | 8.184 | 13.5 | 42.7 | 2.898 | -24.5 | 342.3 |
| 07/17/09 | 05/17/10 | 8.129 | 13.3 | 42.3 | 2.898 | -24.3 | 342.5 |
| 07/18/09 | 05/18/10 | 8.079 | 13.0 | 42.0 | 2.893 | -24.0 | 342.4 |
| 07/19/09 | 05/18/10 | 8.033 | 12.8 | 41.7 | 2.893 | -23.8 | 342.6 |
| 07/20/09 | 05/17/10 | 7.991 | 12.7 | 41.3 | 2.897 | -23.6 | 343.0 |
| 07/21/09 | 05/17/10 | 7.954 | 12.4 | 40.9 | 2.897 | -23.4 | 343.2 |
| 07/22/09 | 05/18/10 | 7.922 | 12.1 | 40.6 | 2.892 | -23.1 | 343.0 |

Table 8.3(continued): Mars-Earth 2009 Trajectory Analysis Parameters (MIDAS)

| Injection | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 07/23/09 | 05/17/10 | 7.894 | 11.9 | 40.2 | 2.895 | -22.9 | 343.4 |
| 07/24/09 | 05/17/10 | 7.871 | 11.7 | 39.9 | 2.894 | -22.7 | 343.5 |
| 07/25/09 | 05/17/10 | 7.852 | 11.6 | 39.4 | 2.904 | -22.5 | 344.1 |
| 07/26/09 | 05/17/10 | 7.839 | 11.2 | 39.1 | 2.896 | -22.2 | 343.8 |
| 07/27/09 | 05/17/10 | 7.830 | 11.0 | 38.7 | 2.901 | -22.0 | 344.2 |
| 07/28/09 | 05/17/10 | 7.826 | 10.8 | 38.3 | 2.903 | -21.8 | 344.4 |
| 07/29/09 | 05/17/10 | 7.827 | 10.6 | 37.8 | 2.907 | -21.6 | 344.6 |
| 07/30/09 | 05/16/10 | 7.832 | 10.8 | 37.2 | 2.930 | -21.6 | 345.7 |
| 07/31/09 | 05/16/10 | 7.843 | 10.5 | 36.9 | 2.927 | -21.3 | 345.6 |
| 08/01/09 | 05/16/10 | 7.859 | 10.3 | 36.5 | 2.932 | -21.0 | 345.9 |
| 08/02/09 | 05/15/10 | 7.879 | 10.4 | 35.9 | 2.954 | -21.0 | 346.7 |
| 08/03/09 | 05/14/10 | 7.904 | 10.8 | 35.3 | 2.985 | -21.2 | 347.8 |
| 08/04/09 | 05/14/10 | 7.935 | 10.3 | 34.9 | 2.978 | -20.8 | 347.6 |
| 08/05/09 | 05/13/10 | 7.970 | 10.5 | 34.4 | 2.999 | -20.9 | 348.2 |
| 08/06/09 | 05/12/10 | 8.009 | 11.1 | 33.7 | 3.042 | -21.4 | 349.5 |
| 08/07/09 | 05/12/10 | 8.054 | 10.9 | 33.2 | 3.046 | -21.2 | 349.6 |
| 08/08/09 | 05/12/10 | 8.104 | 10.9 | 32.7 | 3.060 | -21.2 | 349.9 |
| 08/09/09 | 05/11/10 | 8.158 | 11.2 | 32.1 | 3.085 | -21.5 | 350.5 |
| 08/10/09 | 05/11/10 | 8.217 | 11.5 | 31.6 | 3.107 | -21.8 | 351.1 |
| 08/11/09 | 05/11/10 | 8.282 | 11.3 | 31.1 | 3.113 | -21.8 | 351.2 |
| 08/12/09 | 05/11/10 | 8.351 | 11.8 | 30.5 | 3.140 | -22.3 | 351.8 |
| 08/13/09 | 05/10/10 | 8.426 | 11.7 | 30.0 | 3.148 | -22.3 | 352.0 |
| 08/14/09 | 05/10/10 | 8.507 | 11.6 | 29.5 | 3.156 | -22.4 | 352.1 |
| 08/15/09 | 05/10/10 | 8.592 | 12.0 | 28.9 | 3.176 | -22.9 | 352.6 |
| 08/16/09 | 05/10/10 | 8.684 | 11.6 | 28.6 | 3.175 | -22.6 | 352.5 |
| 08/17/09 | 05/10/10 | 8.781 | 11.8 | 28.0 | 3.189 | -23.0 | 352.8 |
| 08/18/09 | 06/24/10 | 8.855 | 2.8 | 40.9 | 3.679 | -15.4 | 322.4 |
| 08/19/09 | 06/27/10 | 8.920 | 2.9 | 41.6 | 3.796 | -15.1 | 322.4 |
| 08/20/09 | 06/28/10 | 8.984 | 3.0 | 42.0 | 3.881 | -14.9 | 322.4 |
| 08/21/09 | 06/30/10 | 9.048 | 3.1 | 42.5 | 3.979 | -14.6 | 322.6 |
| 08/22/09 | 07/02/10 | 9.112 | 3.3 | 43.2 | 4.090 | -14.4 | 322.8 |
| 08/23/09 | 07/05/10 | 9.176 | 3.5 | 43.9 | 4.209 | -14.1 | 323.1 |
| 08/24/09 | 07/06/10 | 9.239 | 3.7 | 44.4 | 4.298 | -13.9 | 323.4 |
| 08/25/09 | 07/08/10 | 9.302 | 3.9 | 45.0 | 4.394 | -13.6 | 323.7 |
| 08/26/09 | 07/10/10 | 9.365 | 4.0 | 45.5 | 4.491 | -13.3 | 324.1 |
| 08/27/09 | 07/11/10 | 9.427 | 4.2 | 45.9 | 4.566 | -13.1 | 324.5 |
| 08/28/09 | 07/13/10 | 9.490 | 4.4 | 46.6 | 4.672 | -12.8 | 325.0 |
| 08/29/09 | 07/14/10 | 9.552 | 4.5 | 47.0 | 4.750 | -12.6 | 325.4 |
| 08/30/09 | 07/15/10 | 9.614 | 4.7 | 47.5 | 4.834 | -12.3 | 325.8 |
| 08/31/09 | 07/17/10 | 9.676 | 4.9 | 48.0 | 4.919 | -12.0 | 326.3 |

The rendezvous of the orbiter with the ascent vehicle is a difficult task. Ideally, the latitude of the lander will correspond with the inclination of the orbiter. However, this is most likely not going to be the case. The minimum inclination of the orbiter without a plane change is determined by the $\delta_{\infty_A}$. Using Equation 2.11, targeting a B-plane angle of $0°$ will result in an orbiter inclination equal to the $\delta_{\infty_A}$. The latitude of the lander is also a consideration when planning the rendezvous. The worst case latitude for the lander corresponds to the equator which would result in the maximum plane change requirement between the orbiter and MAV. The plane change must be accounted for by either the orbiter or the MAV. Current MAV designs do not provide for a plane change capability. However, a plane change could be effected by an off-azimuthal launch using Equation 2.1. In this case, the performance penalty on the MAV for a launch other than due East will have to be weighed against the propellant required for the orbiter to conduct the same plane change. Once the orbiter and lander are in the same plane at different altitudes, the orbiter can perform various phasing maneuvers until the geometry is appropriate for an intercept. The declination of the return trajectory factors into this problem as well. Chapter 2 addressed the significance of the declination with respect to the inclination of the parking orbit. If the declination is less than the inclination of the parking orbit, then the return vehicle would have two opportunities for injection. The declination of the departure asymptote for the type 2 return trajectory is shown in Figure 8.2. The discontinuity will be explained later. This plot shows that as long as the orbiter inclination is greater than about 17°, a plane change would not be required.

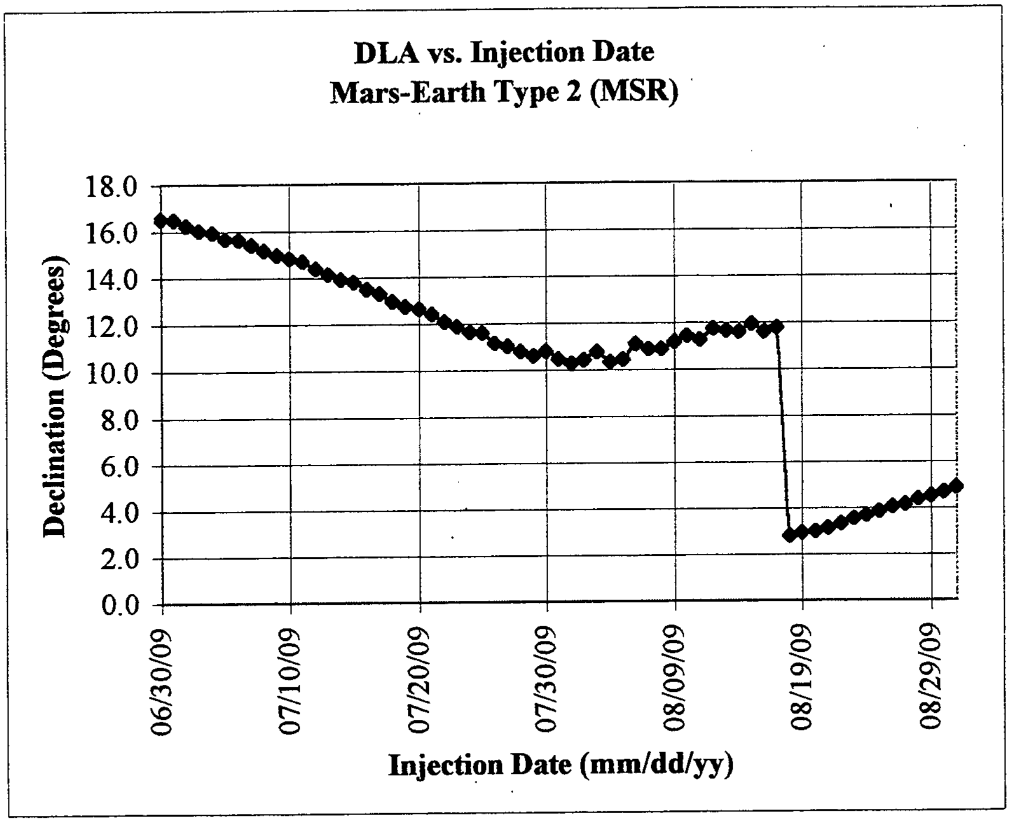


Figure 8.2: Mars-Earth Type 2 DLA vs. Injection Date (MSR)

The corresponding C3 values from the trajectories in Table 8.3 can be used to obtain the $\Delta V$ required for injection back to Earth thus defining the propellant requirements for the orbiter. Figure 8.3 shows a plot of the $\Delta V$ requirements vs. injection date for the type 2 return trajectory. As expected, the minimum $\Delta V$ requirement corresponds to the nominal injection date. A large number of injection dates were chosen in the analysis to provide additional flexibility in the selection of the injection date. The number of days available for mission operations at Mars depends directly on the date selected for injection. Recall that Table 8.2 presented the minimum

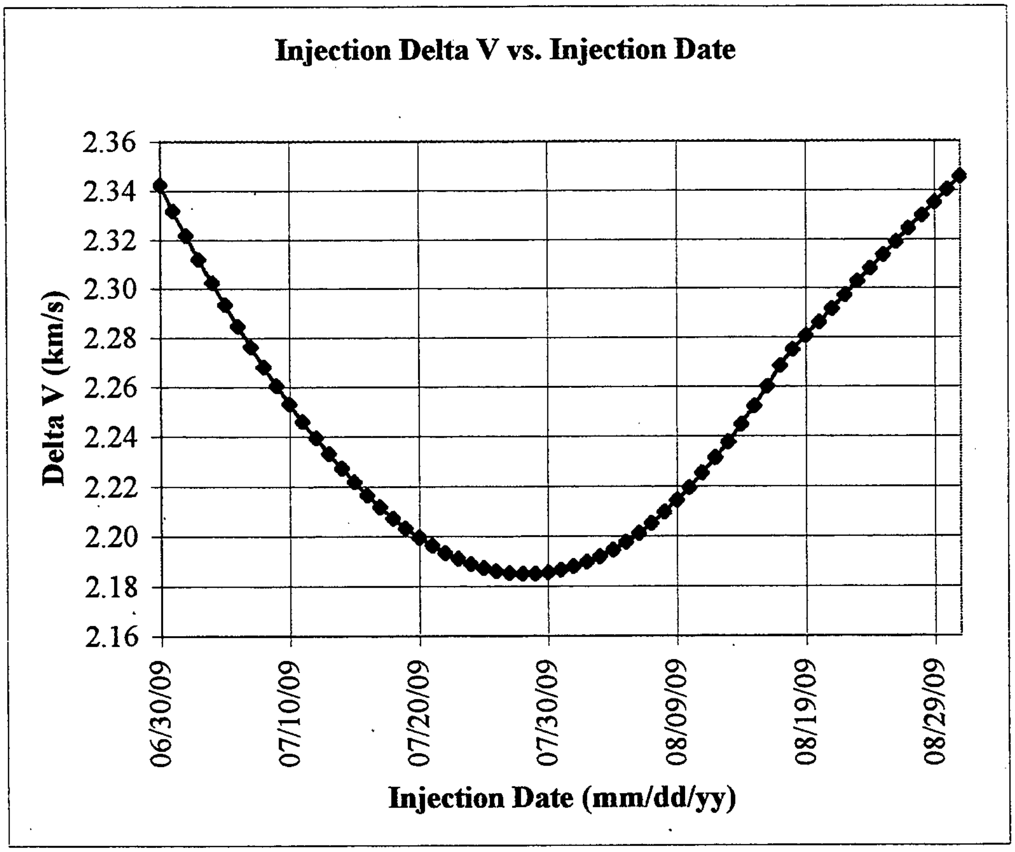


Figure 8.3: Mars-Earth 2009 Type 2 Trajectory - Injection $\Delta V$

and maximum times for mission operations. These calculations were based solely on the nominal injection dates for each return trajectory. As the 2009 type 2 is now the primary return trajectory, the minimum and maximum times for mission operations may be re-calculated using the extended range of injection dates. Table 8.4 summarizes the results. The type 2 return trajectory is compatible with all trajectories and capture schemes analyzed in scenarios 1-8. Adjustment of the injection date away from the nominal case may be required to maximize the time for mission operations. This adjustment will however, result in a performance penalty on the orbiter due to the additional $\Delta V$ required. Even so, the maximum $\Delta V$ difference for the injection dates provided is about

175 m/s which roughly equates to 30-40 kg of propellant depending on the orbiter mass at that time.

Table 8.4: Minimum/Maximum Times for Mission Operations

**Mars- Earth 2009**
**Type 2 Trajectory**

Earliest Injection Date 6/30/09

Latest Injection Date 8/31/09

| Scenarios 1,2 (Type 4-) | | Scenario 3 (Type 4-) | | Scenarios 4,5,7 (Type 4-) | |
|---|---|---|---|---|---|
| Earliest Arrival 3/30/09 | Latest Arrival 5/1/09 | Earliest Arrival 4/28/09 | Latest Arrival 7/17/09 | Earliest Arrival 1/23/09 | Latest Arrival 3/3/09 |
| Min Ops* | Max Ops** | Min Ops*** | Max Ops | Min Ops | Max Ops |
| 60 | 154 | -17 | 125 | 119 | 220 |

| Scenario 6 (Type 4-) | | Scenario 8 (Type 2) | |
|---|---|---|---|
| Earliest Arrival 1/23/09 | Latest Arrival 4/8/09 | Earliest Arrival 8/31/08 | Latest Arrival 10/23/08 |
| Min Ops | Max Ops | Min Ops | Max Ops |
| 83 | 220 | 250 | 365 |

* Minimum Operations Time = Earliest Injection Date - Latest Lander Arrival
** Maximum Operations Time = Latest Injection Date - Earliest Lander Arrival
*** Minus sign indicates that the lander arrives after injection date

The return vehicle injection period is shown in Figure 8.4 with the corresponding arrival dates at Earth. The transfer times vary from 293-320 days depending on the injection date. There are certainly possible injection dates beyond this range, but the additional $\Delta V$ requirement would most likely be prohibitive.

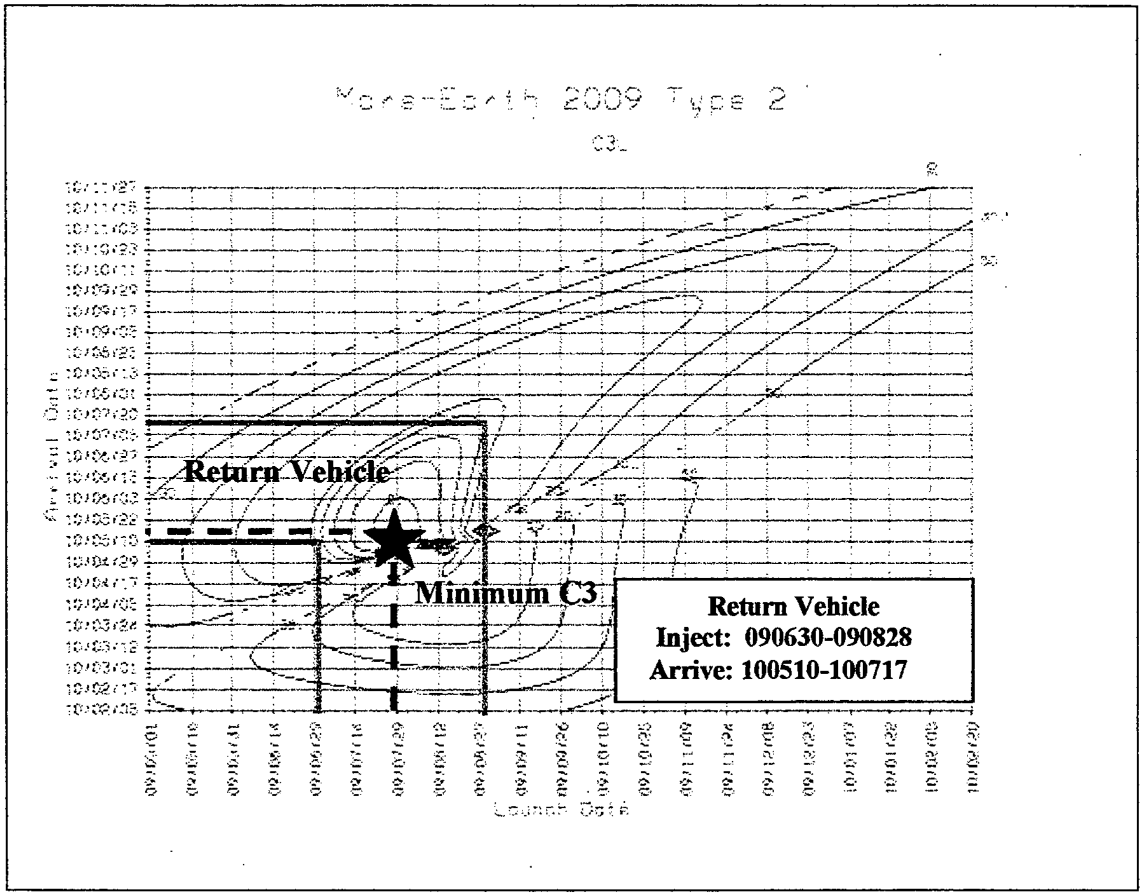


Figure 8.4: Return Vehicle Injection and Arrival Dates (MSR)

## C. GEOMETRY

The return transfer is a standard type 2 trajectory from Mars to Earth. Using the nominal injection date, a Kplot was generated in Figure 8.5. It can be seen that the true anomaly of the transfer trajectory is very close to the 180° point. As such, the arrival date is almost fixed for the first half of the injection period. However, as the injection date nears the "ridge" the corresponding arrival date begins to increase slowly at first. Finally, the arrival just jumps to a region on the contour that is now the minimum energy trajectory for that injection date.

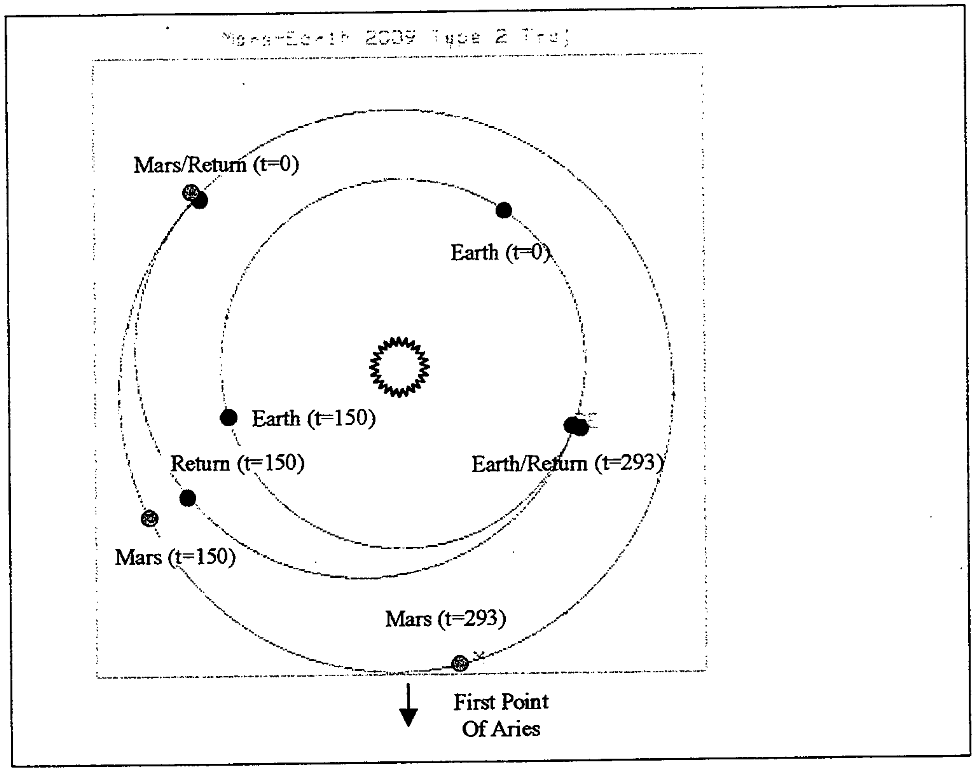


Figure 8.5: Kplot for Mars-Earth Type 2 Trajectory (Return Vehicle 090728 Injection)

The geometric relationship between the bodies of interest in this trajectory are of special concern for the sample return case as the knowledge of the sample location will be critical. Figures 8.6 shows the Sun-spacecraft, spacecraft-Earth, and Earth-Mars distances while Figure 8.7 shows the Sun-spacecraft-Earth angle for solar array and antenna pointing. The SPE angle ranges from approximately 10-80° during the transfer.

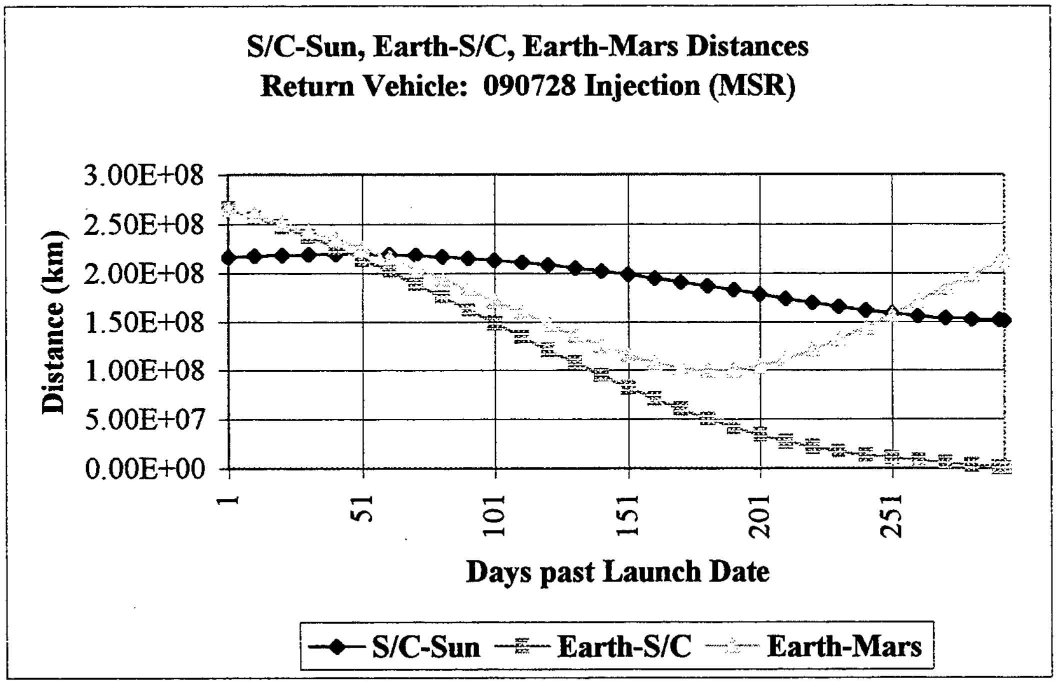


Figure 8.6: Return Vehicle Distances (090728 Injection)

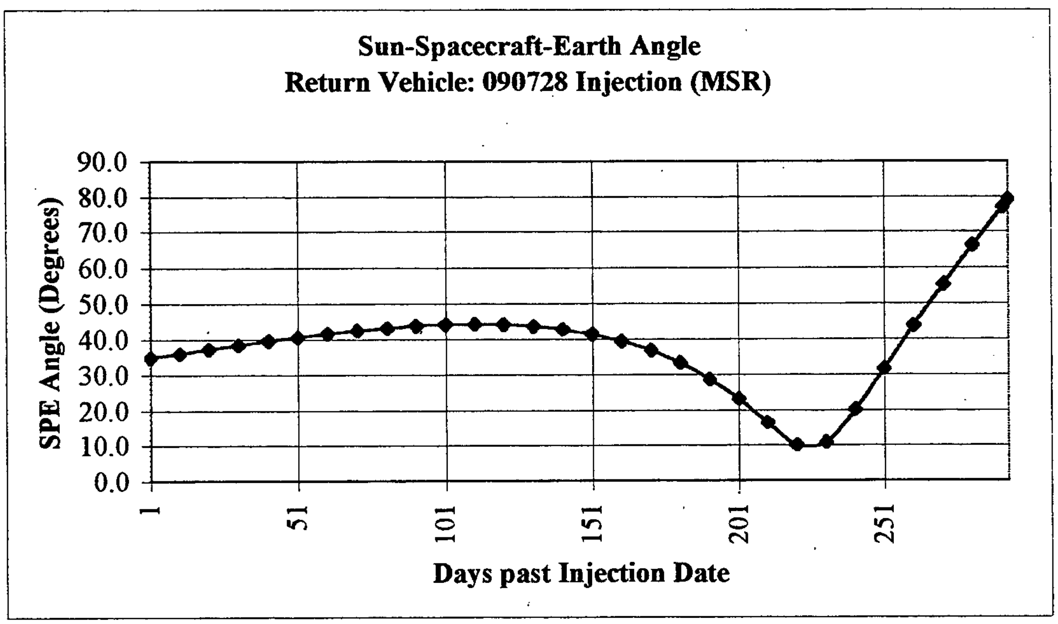


Figure 8.7: SPE Angle for Return Vehicle (090728 Injection)

The determination of the ballistic landing region for the returned sample will depend on the parameters of the incoming trajectory ($V_{\infty_A}$ and $\delta_{\infty_A}$) as noted earlier. The low values for $V_{\infty_A}$ in this case tend to negate any differences generally seen when different flight path angles are used for the entry profile. Figure 8.8 shows the minimum and maximum latitudes possible for a ballistic entry given an initial altitude of 300 km. The flight path angle was varied for this analysis, but as very little difference was noted, only the maximum and minimum achievable latitudes in which the flight path angle was -12° are presented in the figure. Additionally the latitude for the UTTR is provided for comparison. Given the type 2 return trajectory, only those injection dates corresponding to the latest injection dates in the range provided enable a successful return of the sample to the UTTR. Note the discontinuity in the data. This depicts the jump in the arrival date alluded to earlier due to the minimum energy injection and arrival dates being in close proximity to the ridge.

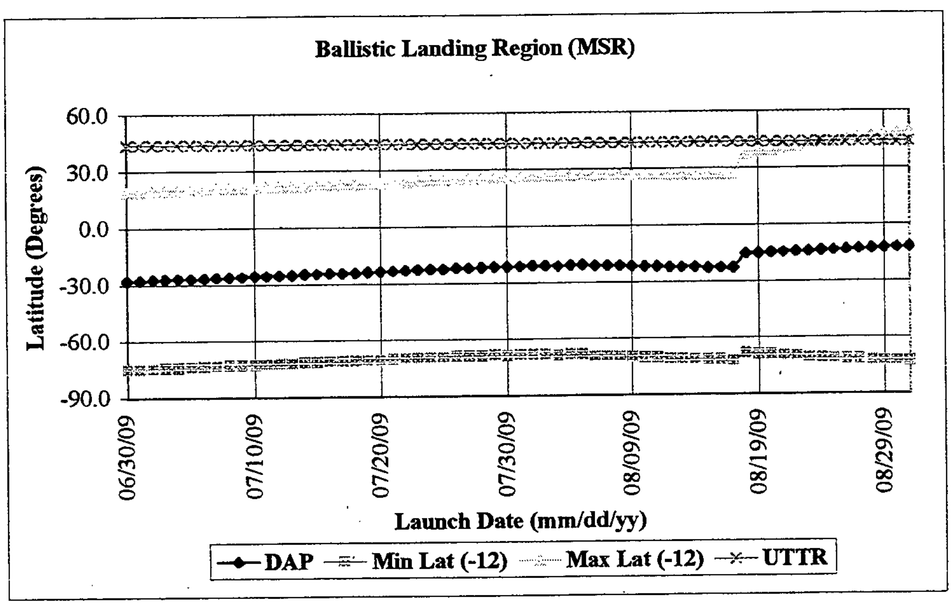


Figure 8.8: Ballistic Landing Region for Mars-Earth Type 2 Trajectory

# IX. CONCLUSIONS

This analysis effectively summarizes many of the critical aspects in the mission design process. The lack of clearly defined science objectives however, necessitated a broader examination of the various possible scenarios for the Mars 2007 mission. The emphasis in the analysis is on the process for conducting a trajectory analysis for any mission. The goal was to identify the trajectory type and capture scheme that provides the largest post-capture orbiter mass. Through the use of JPL software, the NASA Ames re-entry simulation, and a fellow student's Aerocapture simulation, a procedure was developed which enables a means to calculate the post-capture orbiter mass for a given trajectory that corresponds to each capture scheme. Estimation of the orbiter TPS mass required for aerocapture was critical in this evaluation. The development of a basic relationship between TPS mass and $V_{\infty_A}$ facilitates a more rapid analysis of the trajectory data using the standard formatted output from programs like MIDAS and CATO.

The data presented provides clear insight into the advantages and disadvantages of the trajectories analyzed in each scenario. The comprehensive nature of this analysis provides a solid foundation for the mission designers once planning is initiated for the Mars 2007 mission.

# APPENDIX A: CATO GENERATED TRAJECTORY PARAMETERS

Table A.1: Scenarios 1,2,3,7 – CATO Output for Orbiter

| Departure | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 61126 | 90123 | 9.755 | 21.3 | 193.9 | 3.087 | -28.8 | 289.3 |
| 61127 | 90125 | 9.656 | 21.9 | 194.0 | 3.090 | -29.3 | 289.1 |
| 61128 | 90126 | 9.572 | 22.3 | 194.2 | 3.097 | -29.7 | 288.6 |
| 61129 | 90128 | 9.497 | 22.8 | 194.3 | 3.104 | -30.1 | 288.2 |
| 61130 | 90130 | 9.433 | 23.2 | 194.5 | 3.116 | -30.5 | 287.6 |
| 61201 | 90131 | 9.378 | 23.6 | 194.7 | 3.130 | -30.9 | 287.1 |
| 61202 | 90202 | 9.327 | 24.0 | 194.9 | 3.143 | -31.3 | 286.8 |
| 61203 | 90204 | 9.284 | 24.3 | 195.2 | 3.163 | -31.7 | 286.2 |
| 61204 | 90206 | 9.245 | 24.4 | 195.5 | 3.189 | -32.0 | 285.6 |
| 61205 | 90208 | 9.207 | 24.6 | 195.8 | 3.215 | -32.3 | 285.1 |
| 61206 | 90210 | 9.171 | 24.7 | 196.2 | 3.246 | -32.5 | 284.6 |
| 61207 | 90212 | 9.135 | 24.9 | 196.5 | 3.274 | -32.8 | 284.2 |
| 61208 | 90215 | 9.102 | 24.9 | 197.1 | 3.315 | -33.0 | 283.7 |
| 61209 | 90217 | 9.068 | 24.9 | 197.5 | 3.354 | -33.2 | 283.3 |
| 61210 | 90219 | 9.035 | 24.8 | 198.1 | 3.399 | -33.3 | 283.0 |
| 61211 | 90221 | 8.997 | 24.9 | 198.5 | 3.437 | -33.5 | 282.8 |
| 61212 | 90224 | 8.960 | 24.8 | 199.0 | 3.486 | -33.6 | 282.6 |
| 61213 | 90225 | 8.920 | 25.0 | 199.2 | 3.522 | -33.8 | 282.6 |
| 61214 | 90228 | 8.889 | 24.8 | 199.7 | 3.570 | -33.9 | 282.5 |
| 61215 | 90302 | 8.865 | 24.6 | 200.2 | 3.621 | -34.0 | 282.5 |

Table A.2: Scenarios 1,2 - CATO Output for Lander

| Departure | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 61229 | 90330 | 8.838 | 20.1 | 208.5 | 4.345 | -34.4 | 287.1 |
| 61230 | 90401 | 8.851 | 19.8 | 209.0 | 4.388 | -34.4 | 287.6 |
| 61231 | 90403 | 8.866 | 19.3 | 209.8 | 4.442 | -34.3 | 288.2 |
| 70101 | 90404 | 8.876 | 19.0 | 210.4 | 4.486 | -34.3 | 288.7 |
| 70102 | 90406 | 8.886 | 18.4 | 211.2 | 4.540 | -34.3 | 289.4 |
| 70103 | 90408 | 8.890 | 18.0 | 211.4 | 4.583 | -34.2 | 290.0 |
| 70104 | 90410 | 8.892 | 17.6 | 212.7 | 4.631 | -34.2 | 290.6 |
| 70105 | 90412 | 8.894 | 16.9 | 213.6 | 4.685 | -34.1 | 291.3 |
| 70106 | 90413 | 8.886 | 16.7 | 214.1 | 4.722 | -34.1 | 291.9 |
| 70107 | 90415 | 8.877 | 16.3 | 214.9 | 4.767 | -34.0 | 292.5 |
| 70108 | 90417 | 8.867 | 15.8 | 215.7 | 4.814 | -33.9 | 293.2 |
| 70109 | 90418 | 8.852 | 15.4 | 216.5 | 4.859 | -33.8 | 293.9 |
| 70110 | 90420 | 8.832 | 15.1 | 217.3 | 4.903 | -33.7 | 294.6 |
| 70111 | 90422 | 8.810 | 14.6 | 218.0 | 4.947 | -33.6 | 295.3 |
| 70112 | 90423 | 8.787 | 14.3 | 218.5 | 4.985 | -33.6 | 296.0 |
| 70113 | 90425 | 8.769 | 13.8 | 219.2 | 5.026 | -33.5 | 296.6 |
| 70114 | 90426 | 8.753 | 13.4 | 219.8 | 5.064 | -33.4 | 297.3 |
| 70115 | 90428 | 8.738 | 13.1 | 220.4 | 5.101 | -33.3 | 298.0 |
| 70116 | 90429 | 8.724 | 12.7 | 221.1 | 5.138 | -33.2 | 298.7 |
| 70117 | 90501 | 8.711 | 12.3 | 221.7 | 5.175 | -33.1 | 299.3 |

Table A.3: Scenario 3 - CATO Output for Lander

| Departure | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 70115 | 90428 | 8.738 | 13.1 | 220.4 | 5.101 | -33.3 | 298.0 |
| 70116 | 90429 | 8.724 | 12.7 | 221.1 | 5.138 | -33.2 | 298.7 |
| 70117 | 90501 | 8.711 | 12.3 | 221.7 | 5.175 | -33.1 | 299.3 |
| 70118 | 90503 | 8.704 | 11.7 | 222.6 | 5.219 | -32.9 | 300.2 |
| 70119 | 90504 | 8.695 | 11.3 | 223.2 | 5.253 | -32.8 | 300.8 |
| 70120 | 90506 | 8.690 | 11.0 | 223.8 | 5.286 | -32.7 | 301.5 |
| 70121 | 90507 | 8.689 | 10.7 | 224.3 | 5.32 | -32.6 | 302.1 |
| 70122 | 90509 | 8.694 | 10.1 | 225.2 | 5.359 | -32.4 | 302.9 |
| 70123 | 90510 | 8.700 | 9.7 | 225.8 | 5.392 | -32.3 | 303.6 |
| 70124 | 90511 | 8.709 | 9.3 | 226.4 | 5.424 | -32.2 | 304.3 |
| 70125 | 90513 | 8.725 | 8.6 | 227.4 | 5.464 | -32.0 | 305.1 |
| 70126 | 90515 | 8.737 | 8.2 | 227.9 | 5.493 | -31.8 | 305.8 |
| 70127 | 90516 | 8.750 | 7.8 | 228.6 | 5.523 | -31.7 | 306.5 |
| 70128 | 90517 | 8.765 | 7.3 | 229.4 | 5.556 | -31.5 | 307.2 |
| 70129 | 90519 | 8.777 | 6.8 | 230.1 | 5.585 | -31.4 | 307.9 |
| 70130 | 90520 | 8.787 | 6.4 | 230.8 | 5.615 | -31.2 | 308.6 |
| 70131 | 90522 | 8.796 | 5.9 | 231.7 | 5.645 | -31.0 | 309.3 |
| 70201 | 90523 | 8.800 | 5.6 | 232.2 | 5.669 | -31.7 | 310.0 |
| 70202 | 90524 | 8.803 | 5.0 | 233.1 | 5.699 | -30.7 | 310.7 |
| 70203 | 90526 | 8.805 | 4.4 | 234.1 | 5.73 | -30.4 | 311.5 |
| 70204 | 90527 | 8.800 | 4.1 | 234.7 | 5.753 | -30.3 | 312.1 |
| 70205 | 90529 | 8.793 | 3.8 | 235.4 | 5.777 | -30.1 | 312.8 |
| 70206 | 90530 | 8.785 | 3.4 | 236.2 | 5.802 | -29.9 | 313.5 |
| 70207 | 90531 | 8.774 | 3.0 | 237.0 | 5.827 | -29.7 | 314.2 |
| 70208 | 90602 | 8.757 | 2.6 | 237.8 | 5.852 | -29.5 | 314.9 |
| 70209 | 90603 | 8.731 | 2.4 | 238.3 | 5.872 | -29.4 | 315.5 |
| 70210 | 90604 | 8.707 | 1.9 | 239.1 | 5.897 | -29.1 | 316.3 |
| 70211 | 90606 | 8.687 | 1.4 | 239.7 | 5.918 | -28.9 | 316.9 |
| 70212 | 90607 | 8.671 | 1.0 | 240.4 | 5.939 | -28.7 | 317.6 |
| 70213 | 90609 | 8.660 | 0.4 | 241.3 | 5.962 | -28.4 | 318.3 |
| 70214 | 90610 | 8.644 | 0.2 | 241.8 | 5.979 | -28.3 | 318.9 |
| 70215 | 90611 | 8.637 | -0.5 | 242.9 | 6.004 | -27.9 | 319.7 |
| 70216 | 90613 | 8.627 | -0.9 | 243.6 | 6.023 | -27.7 | 320.4 |
| 70217 | 90614 | 8.617 | -1.1 | 244.0 | 6.036 | -27.6 | 320.9 |
| 70218 | 90615 | 8.617 | -1.6 | 244.9 | 6.057 | -27.3 | 321.7 |
| 70219 | 90616 | 8.618 | -2.0 | 245.5 | 6.074 | -27.1 | 322.3 |
| 70220 | 90618 | 8.626 | -2.5 | 246.4 | 6.092 | -26.8 | 323.0 |
| 70221 | 90619 | 8.635 | -2.8 | 246.8 | 6.105 | -26.6 | 323.6 |
| 70222 | 90620 | 8.648 | -3.2 | 247.4 | 6.12 | -26.4 | 324.2 |
| 70223 | 90622 | 8.666 | -3.7 | 248.3 | 6.136 | -26.1 | 324.9 |
| 70224 | 90623 | 8.687 | -4.2 | 249.3 | 6.153 | -25.8 | 325.6 |
| 70225 | 90624 | 8.706 | -4.5 | 249.9 | 6.166 | -25.6 | 326.2 |
| 70226 | 90625 | 8.724 | -4.9 | 250.5 | 6.178 | -25.4 | 326.8 |
| 70227 | 90627 | 8.745 | -5.5 | 251.5 | 6.193 | -25.0 | 327.5 |
| 70228 | 90628 | 8.764 | -6.0 | 252.4 | 6.206 | -24.7 | 328.2 |
| 70301 | 90629 | 8.778 | -6.3 | 253.0 | 6.216 | -24.5 | 328.8 |
| 70302 | 90701 | 8.791 | -6.7 | 253.7 | 6.227 | -24.3 | 329.4 |
| 70303 | 90702 | 8.806 | -7.3 | 254.9 | 6.24 | -23.9 | 330.1 |
| 70304 | 90703 | 8.814 | -7.6 | 255.7 | 6.249 | -23.6 | 330.7 |
| 70305 | 90704 | 8.818 | -7.9 | 256.2 | 6.257 | -23.4 | 331.2 |
| 70306 | 90706 | 8.823 | -8.3 | 257.0 | 6.266 | -23.1 | 331.9 |
| 70307 | 90707 | 8.829 | -8.8 | 258.1 | 6.276 | -22.8 | 332.6 |
| 70308 | 90708 | 8.830 | -9.2 | 259.0 | 6.284 | -22.5 | 333.2 |
| 70309 | 90710 | 8.826 | -9.4 | 259.7 | 6.29 | -22.2 | 333.8 |
| 70310 | 90711 | 8.814 | -9.6 | 260.4 | 6.297 | -21.9 | 334.4 |
| 70311 | 90712 | 8.784 | -9.9 | 261.0 | 6.303 | -21.6 | 335.0 |
| 70312 | 90713 | 8.766 | -10.4 | 261.4 | 6.307 | -21.4 | 335.5 |
| 70313 | 90714 | 8.763 | -10.9 | 262.2 | 6.313 | -21.1 | 336.1 |
| 70314 | 90716 | 8.766 | -11.5 | 263.5 | 6.319 | -20.6 | 336.9 |
| 70315 | 90717 | 8.763 | -11.8 | 264.0 | 6.322 | -20.4 | 337.4 |

Table A.4: Scenario 4,5,6,8 - CATO Output for Orbiter

| Departure | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 70913 | 80831 | 13.240 | 11.9 | 90.9 | 2.568 | 19.8 | 190.8 |
| 70914 | 80902 | 13.130 | 12.1 | 90.4 | 2.579 | 19.6 | 190.9 |
| 70915 | 80904 | 13.040 | 12.4 | 90.1 | 2.594 | 19.3 | 190.8 |
| 70916 | 80908 | 12.970 | 13.8 | 90.5 | 2.640 | 18.2 | 190.3 |
| 70917 | 80908 | 12.910 | 13.4 | 89.6 | 2.635 | 18.5 | 190.6 |
| 70918 | 80911 | 12.860 | 14.1 | 89.5 | 2.662 | 17.9 | 190.4 |
| 70919 | 80914 | 12.820 | 15.0 | 89.6 | 2.697 | 17.1 | 190.2 |
| 70920 | 80915 | 12.800 | 15.2 | 89.2 | 2.710 | 16.9 | 190.3 |
| 70921 | 80920 | 12.790 | 16.4 | 89.6 | 2.758 | 15.9 | 190.1 |
| 70922 | 80921 | 12.800 | 16.6 | 89.2 | 2.774 | 15.6 | 190.2 |
| 70923 | 80923 | 12.810 | 17.1 | 89.0 | 2.797 | 15.2 | 190.2 |
| 70924 | 80928 | 12.830 | 18.4 | 89.7 | 2.858 | 13.9 | 190.1 |
| 70925 | 80929 | 12.850 | 18.7 | 89.4 | 2.876 | 13.6 | 190.2 |
| 70926 | 81001 | 12.890 | 19.1 | 89.2 | 2.902 | 13.2 | 190.3 |
| 70927 | 81005 | 12.920 | 20.1 | 89.8 | 2.957 | 12.2 | 190.5 |
| 70928 | 81009 | 12.950 | 21.0 | 90.3 | 3.008 | 11.3 | 190.7 |
| 70929 | 81012 | 12.990 | 21.7 | 90.7 | 3.053 | 10.5 | 190.9 |
| 70930 | 81018 | 13.040 | 22.9 | 92.9 | 3.143 | 9.1 | 191.4 |
| 71001 | 81018 | 13.110 | 22.8 | 91.8 | 3.151 | 8.9 | 191.5 |
| 71002 | 81023 | 12.990 | 23.7 | 91.9 | 3.235 | 7.6 | 192.2 |

Figure A.5: Scenario 4,5,7 - CATO Output for Lander

| Departure | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 61126 | 90123 | 9.755 | 21.3 | 193.9 | 3.088 | -28.9 | 289.4 |
| 61127 | 90125 | 9.657 | 21.9 | 194.0 | 3.091 | -29.3 | 289.1 |
| 61128 | 90126 | 9.572 | 22.4 | 194.2 | 3.097 | -29.8 | 288.6 |
| 61129 | 90128 | 9.497 | 22.9 | 194.3 | 3.104 | -30.2 | 288.3 |
| 61130 | 90130 | 9.434 | 23.3 | 194.5 | 3.116 | -30.6 | 287.7 |
| 61201 | 90131 | 9.379 | 23.6 | 194.7 | 3.131 | -31.0 | 287.1 |
| 61202 | 90202 | 9.328 | 24.1 | 194.9 | 3.143 | -31.4 | 286.8 |
| 61203 | 90204 | 9.285 | 24.3 | 195.2 | 3.163 | -31.7 | 286.3 |
| 61204 | 90206 | 9.247 | 24.5 | 195.5 | 3.190 | -32.0 | 285.6 |
| 61205 | 90208 | 9.209 | 24.7 | 195.8 | 3.216 | -32.3 | 285.1 |
| 61206 | 90210 | 9.173 | 24.8 | 196.2 | 3.247 | -32.6 | 284.6 |
| 61207 | 90212 | 9.136 | 25.0 | 196.5 | 3.275 | -32.9 | 284.2 |
| 61208 | 90215 | 9.103 | 25.0 | 197.0 | 3.313 | -33.1 | 283.8 |
| 61209 | 90217 | 9.070 | 24.9 | 197.5 | 3.355 | -33.3 | 283.4 |
| 61210 | 90219 | 9.037 | 24.8 | 198.1 | 3.400 | -33.4 | 283.1 |
| 61211 | 90221 | 9.004 | 24.8 | 198.5 | 3.439 | -33.5 | 282.8 |
| 61212 | 90224 | 8.962 | 24.8 | 199.0 | 3.488 | -33.7 | 282.7 |
| 61213 | 90225 | 8.922 | 25.0 | 199.2 | 3.523 | -33.9 | 282.6 |
| 61214 | 90228 | 8.892 | 24.8 | 199.7 | 3.571 | -34.0 | 282.6 |
| 61215 | 90303 | 8.878 | 25.0 | 200.6 | 3.649 | -33.9 | 282.5 |

Table A.6: Scenario 6 - CATO Output for Lander

| Departure | Arrival | C3 | DLA | RLA | V inf at Arr | DAP | RAP |
|---|---|---|---|---|---|---|---|
| 61126 | 90123 | 9.755 | 21.3 | 193.9 | 3.088 | -28.9 | 289.4 |
| 61127 | 90125 | 9.657 | 21.9 | 194.0 | 3.091 | -29.3 | 289.1 |
| 61128 | 90126 | 9.572 | 22.4 | 194.2 | 3.097 | -29.8 | 288.6 |
| 61129 | 90128 | 9.497 | 22.9 | 194.3 | 3.104 | -30.2 | 288.3 |
| 61130 | 90130 | 9.434 | 23.3 | 194.5 | 3.116 | -30.6 | 287.7 |
| 61201 | 90131 | 9.379 | 23.6 | 194.7 | 3.131 | -31.0 | 287.1 |
| 61202 | 90202 | 9.328 | 24.1 | 194.9 | 3.143 | -31.4 | 286.8 |
| 61203 | 90204 | 9.285 | 24.3 | 195.2 | 3.163 | -31.7 | 286.3 |
| 61204 | 90206 | 9.247 | 24.5 | 195.5 | 3.190 | -32.0 | 285.6 |
| 61205 | 90208 | 9.209 | 24.7 | 195.8 | 3.216 | -32.3 | 285.1 |
| 61206 | 90210 | 9.173 | 24.8 | 196.2 | 3.247 | -32.6 | 284.6 |
| 61207 | 90212 | 9.136 | 25.0 | 196.5 | 3.275 | -32.9 | 284.2 |
| 61208 | 90215 | 9.103 | 25.0 | 197.0 | 3.313 | -33.1 | 283.8 |
| 61209 | 90217 | 9.070 | 24.9 | 197.5 | 3.355 | -33.3 | 283.4 |
| 61210 | 90219 | 9.037 | 24.8 | 198.1 | 3.400 | -33.4 | 283.1 |
| 61211 | 90221 | 9.004 | 24.8 | 198.5 | 3.439 | -33.5 | 282.8 |
| 61212 | 90224 | 8.962 | 24.8 | 199.0 | 3.488 | -33.7 | 282.7 |
| 61213 | 90225 | 8.922 | 25.0 | 199.2 | 3.523 | -33.9 | 282.6 |
| 61214 | 90228 | 8.892 | 24.8 | 199.7 | 3.571 | -34.0 | 282.6 |
| 61215 | 90303 | 8.878 | 25.0 | 200.6 | 3.649 | -33.9 | 282.5 |
| 61216 | 90304 | 8.846 | 24.4 | 200.7 | 3.672 | -34.2 | 282.6 |
| 61217 | 90307 | 8.830 | 23.9 | 201.4 | 3.734 | -34.2 | 282.7 |
| 61218 | 90309 | 8.812 | 23.8 | 201.9 | 3.780 | -34.3 | 282.9 |
| 61219 | 90310 | 8.796 | 23.7 | 202.3 | 3.824 | -34.3 | 283.1 |
| 61220 | 90312 | 8.785 | 23.4 | 202.9 | 3.876 | -34.4 | 283.3 |
| 61221 | 90315 | 8.781 | 22.9 | 203.7 | 3.940 | -34.3 | 283.6 |
| 61222 | 90317 | 8.777 | 22.5 | 204.3 | 3.994 | -34.3 | 284.0 |
| 61223 | 90319 | 8.774 | 22.4 | 204.6 | 4.036 | -34.4 | 284.3 |
| 61224 | 90321 | 8.777 | 22.1 | 205.2 | 4.086 | -34.5 | 284.7 |
| 61225 | 90322 | 8.784 | 21.8 | 205.8 | 4.136 | -34.5 | 285.1 |
| 61226 | 90324 | 8.795 | 21.4 | 206.5 | 4.189 | -34.5 | 285.6 |
| 61227 | 90327 | 8.809 | 20.9 | 207.1 | 4.246 | -34.4 | 286.1 |
| 61228 | 90328 | 8.822 | 20.6 | 207.7 | 4.289 | -34.5 | 286.5 |
| 61229 | 90330 | 8.838 | 20.1 | 208.5 | 4.345 | -34.4 | 287.1 |
| 61230 | 90401 | 8.851 | 19.8 | 209.0 | 4.388 | -34.4 | 287.6 |
| 61231 | 90403 | 8.866 | 19.3 | 209.8 | 4.442 | -34.3 | 288.2 |
| 70101 | 90404 | 8.876 | 19.0 | 210.4 | 4.486 | -34.3 | 288.7 |
| 70102 | 90406 | 8.866 | 18.4 | 211.2 | 4.540 | -34.3 | 289.4 |
| 70103 | 90408 | 8.890 | 18.0 | 211.4 | 4.583 | -34.2 | 290.0 |

## APPENDIX B: $\delta_{\infty}, \alpha_{\infty}, \delta_{\infty_A}, \alpha_{\infty_A}$ vs. Launch Date Plots

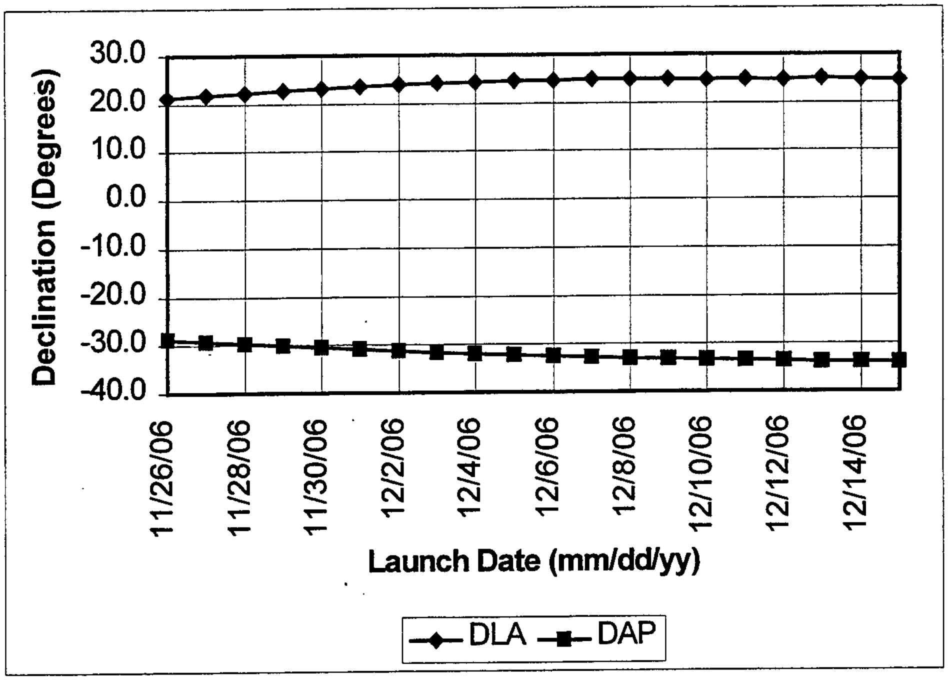


Figure B.1: Scenarios 1,2,3,7 - Orbiter Declination vs. Launch Date

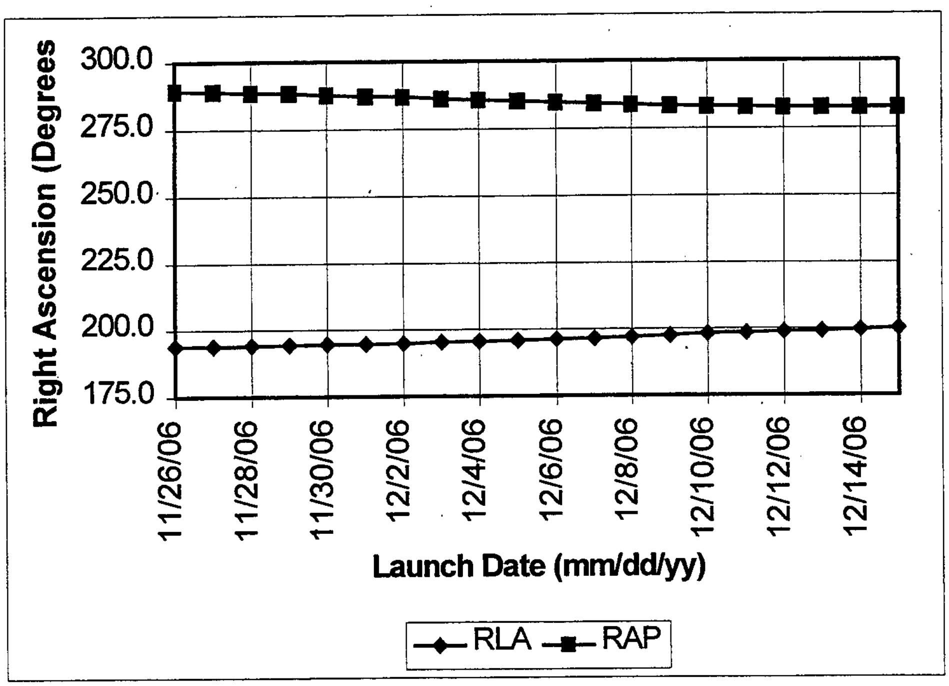


Figure B.2: Scenarios 1,2,3,7 - Orbiter Right Ascension vs. Launch Date

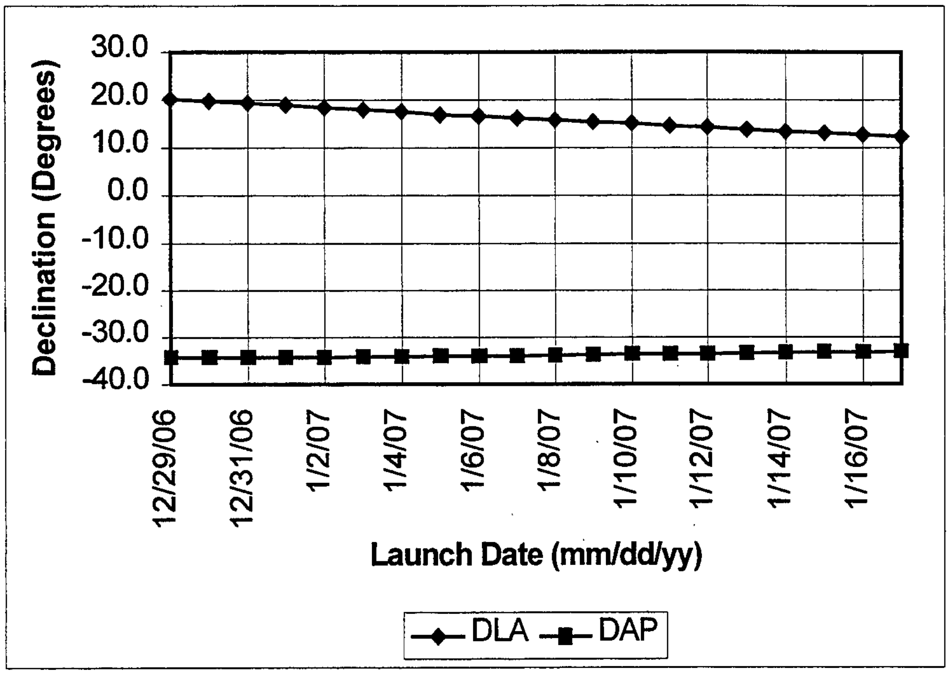


Figure B.3: Scenarios 1,2 - Lander Declination vs. Launch Date

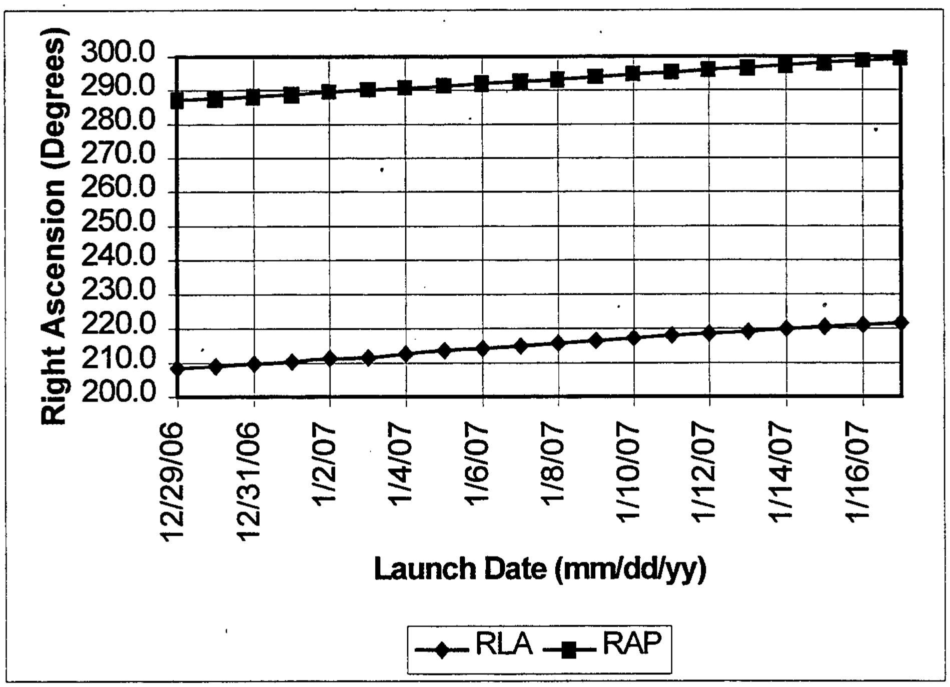


Figure B.4: Scenarios 1,2 - Lander Right Ascension vs. Launch Date

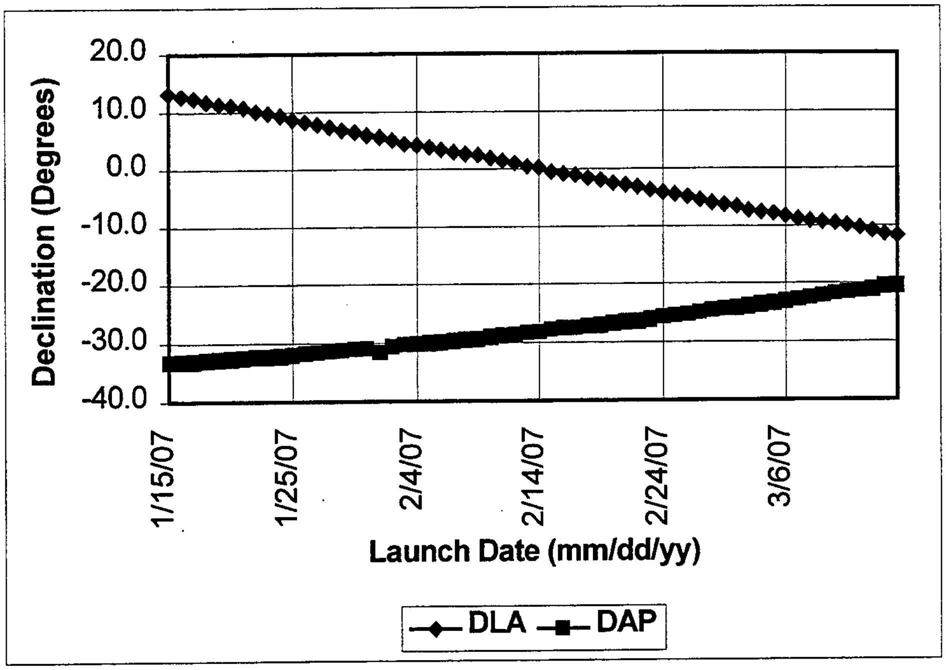


Figure B5: Scenario 3 - Lander Declination vs. Launch Date

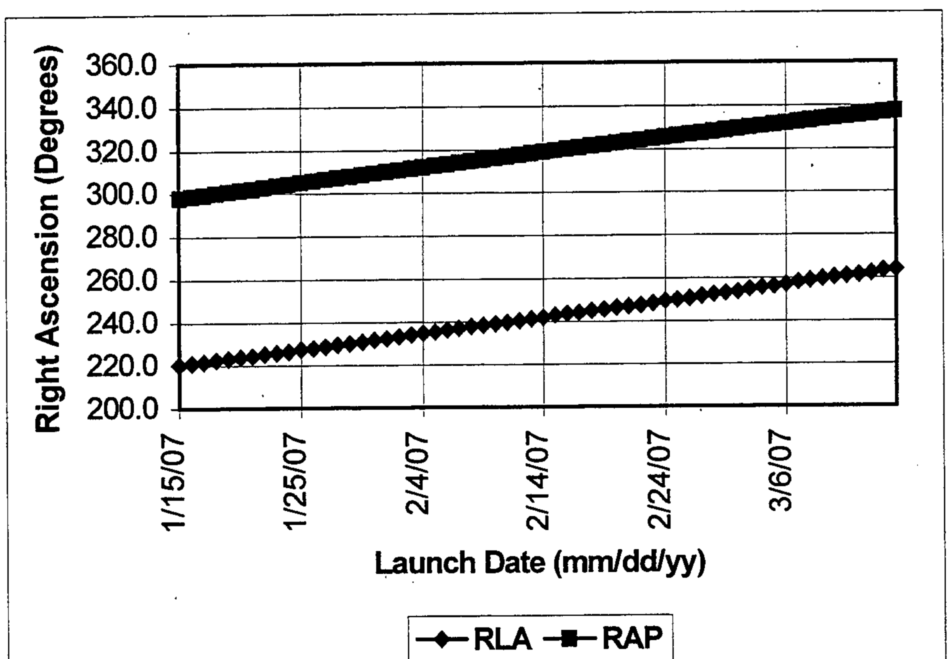


Figure B6: Scenario 3 - Lander Right Ascension vs. Launch Date

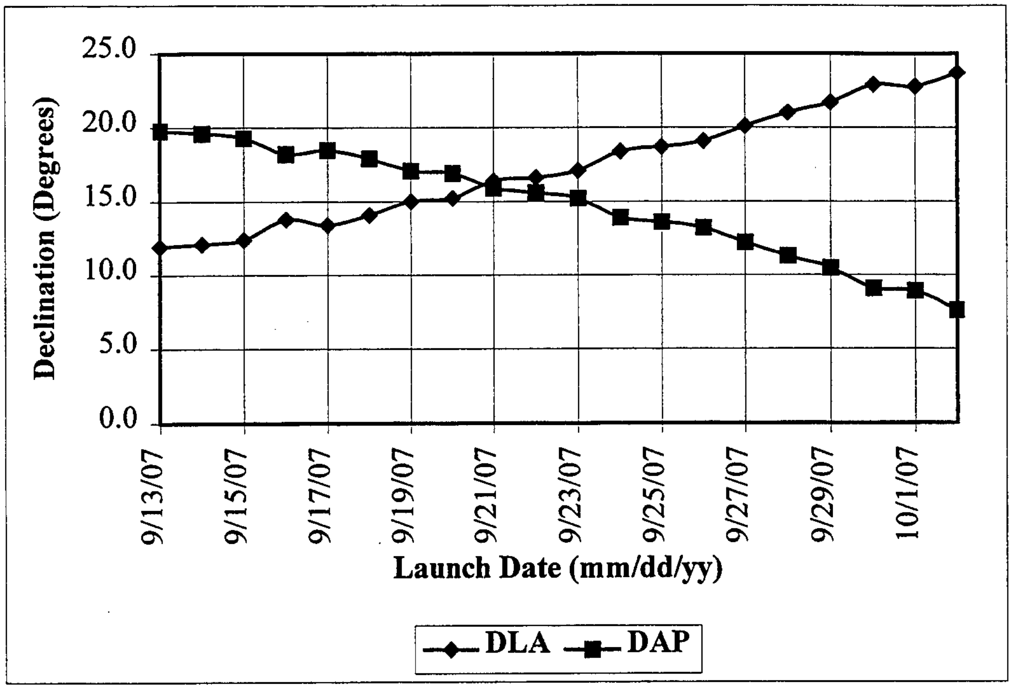


Figure B7: Scenarios 4, 5, 6, 8 – Orbiter Declination vs. Launch Date

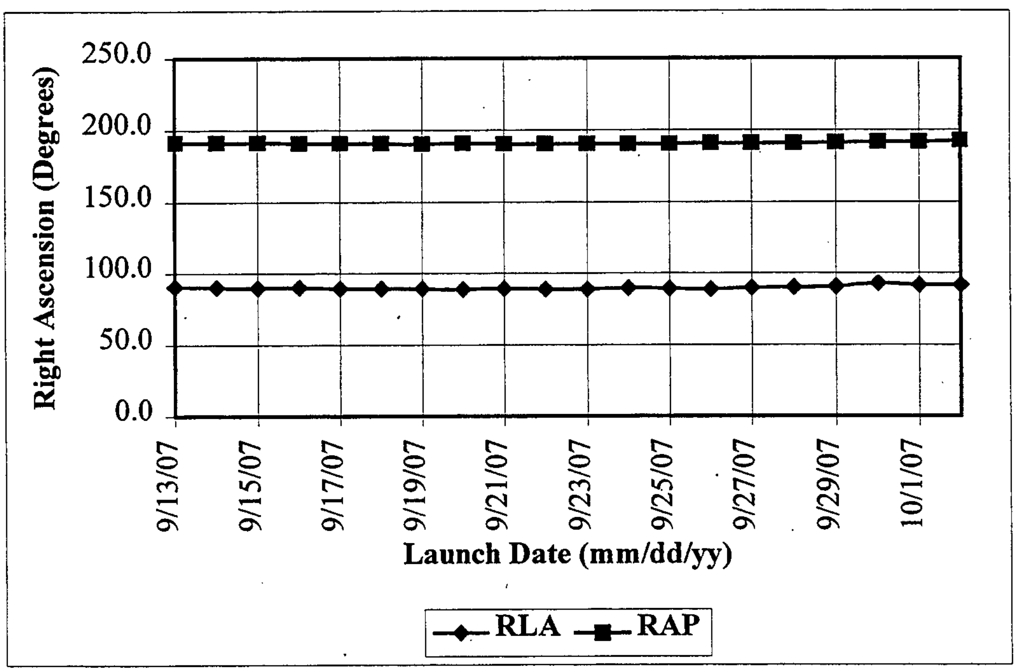


Figure B8: Scenarios 4, 5, 6, 8 – Orbiter Right Ascension vs. Launch Date

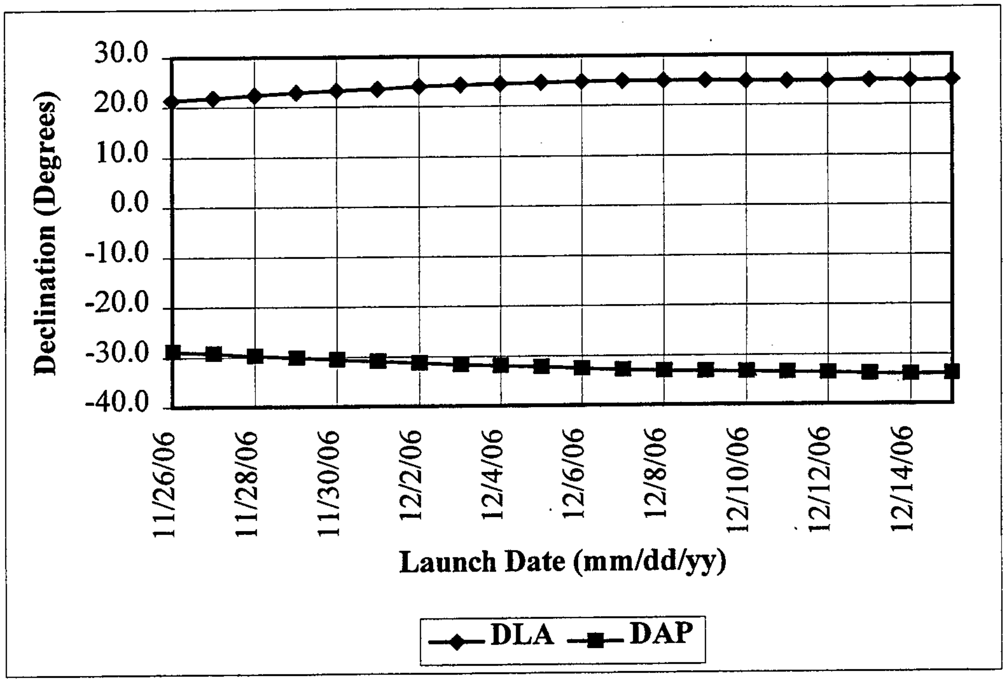


Figure B9: Scenarios 4, 5– Lander Declination vs. Launch Date

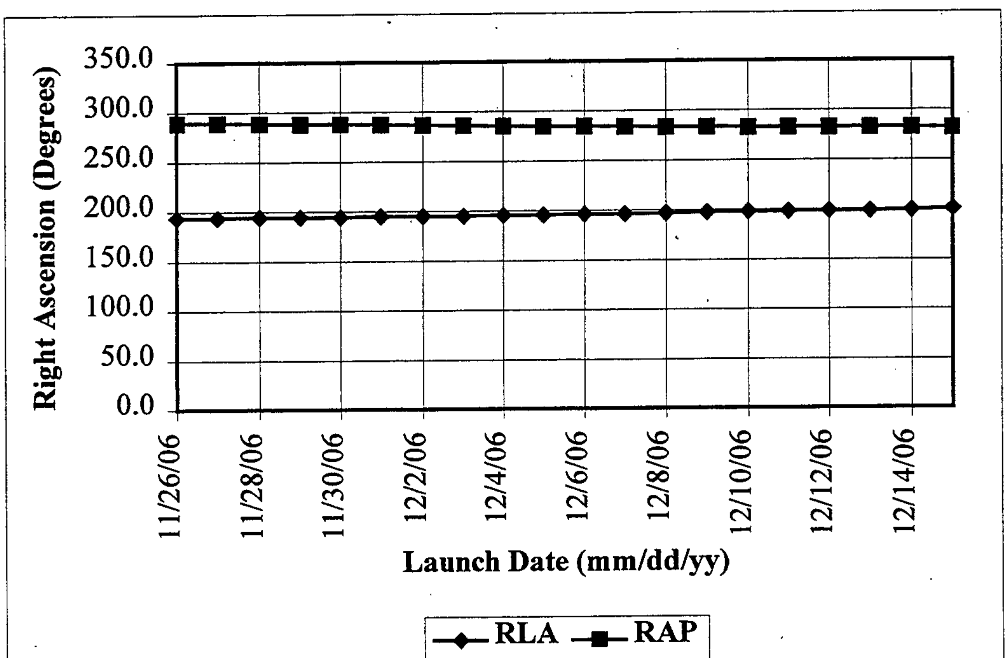


Figure B10: Scenarios 4, 5– Lander Right Ascension vs. Launch Date

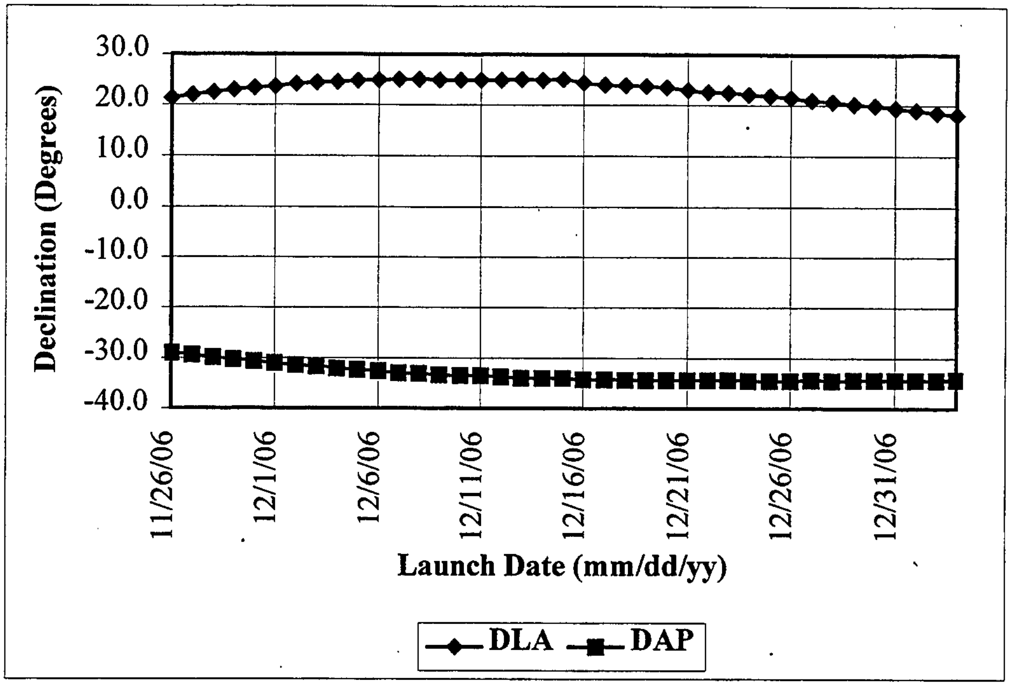

Figure B11: Scenario 6 – Lander Declination vs. Launch Date

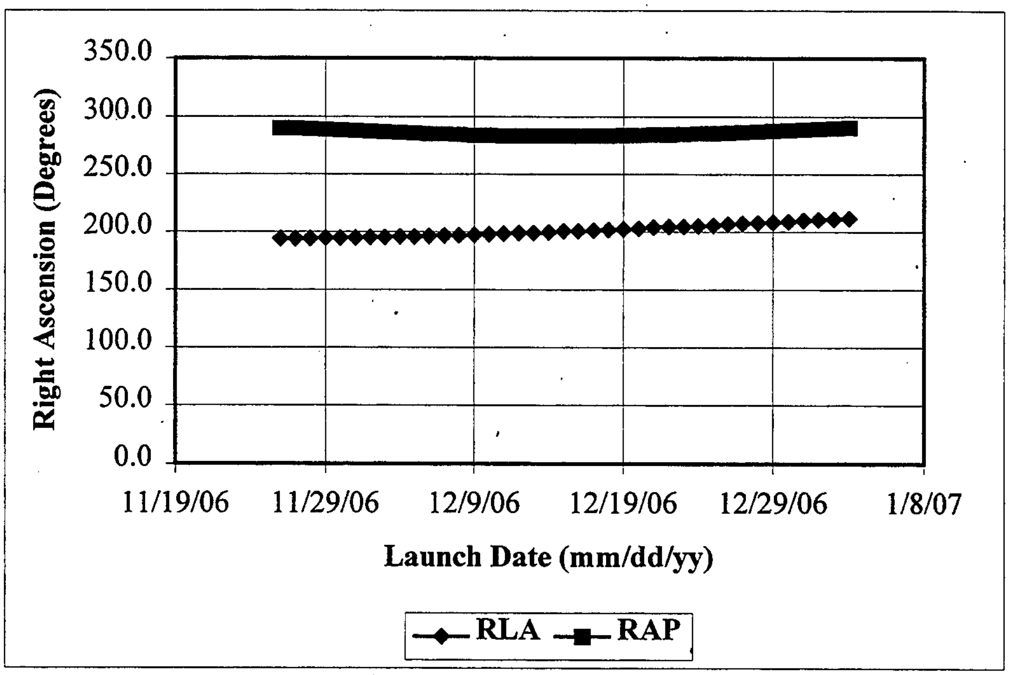

Figure B12: Scenario 6 – Lander Right Ascension vs. Launch Date

# APPENDIX C. LAUNCH VEHICLE PERFORMANCE TABLES

Scenario 1,2,3 - C3 vs. Injected Mass (Orbiter)

| Launch | C3 | Delta II 7925 (7% margin) | Delta II 7925 H (10% margin) | Delta II 7325 (Star 48B) (7% margin) | Delta II 7425 (Star 48B) (7% margin) |
|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 984.3 | 1100.9 | 557.6 | 649.2 |
| 11/27/06 | 9.657 | 986.3 | 1103.2 | 558.7 | 650.6 |
| 11/28/06 | 9.572 | 988.1 | 1105.2 | 559.7 | 651.8 |
| 11/29/06 | 9.497 | 989.6 | 1106.9 | 560.6 | 652.8 |
| 11/30/06 | 9.434 | 990.9 | 1108.3 | 561.4 | 653.7 |
| 12/1/06 | 9.379 | 992.1 | 1109.6 | 562.0 | 654.5 |
| 12/2/06 | 9.328 | 993.1 | 1110.8 | 562.6 | 655.2 |
| 12/3/06 | 9.285 | 994.0 | 1111.8 | 563.1 | 655.9 |
| 12/4/06 | 9.247 | 994.8 | 1112.7 | 563.6 | 656.4 |
| 12/5/06 | 9.209 | 995.6 | 1113.6 | 564.0 | 656.9 |
| 12/6/06 | 9.173 | 996.3 | 1114.4 | 564.4 | 657.5 |
| 12/7/06 | 9.136 | 997.1 | 1115.3 | 564.9 | 658.0 |
| 12/8/06 | 9.103 | 997.8 | 1116.0 | 565.3 | 658.5 |
| 12/9/06 | 9.070 | 998.5 | 1116.8 | 565.6 | 658.9 |
| 12/10/06 | 9.037 | 999.2 | 1117.6 | 566.0 | 659.4 |
| 12/11/06 | 9.004 | 999.8 | 1118.4 | 566.4 | 659.9 |
| 12/12/06 | 8.962 | 1000.7 | 1119.3 | 566.9 | 660.5 |
| 12/13/06 | 8.922 | 1001.6 | 1120.3 | 567.4 | 661.1 |
| 12/14/06 | 8.892 | 1002.2 | 1121.0 | 567.8 | 661.5 |
| 12/15/06 | 8.878 | 1002.5 | 1121.3 | 567.9 | 661.7 |

Scenario 1,2 - C3 vs. Injected Mass (Lander)

| Launch | C3 | Delta II 7925<br>(7% margin) | Delta II 7925 H<br>(10% margin) | Delta II 7325<br>(Star 48B)<br>(7% margin) | Delta II 7425<br>(Star 48B)<br>(7% margin) |
|---|---|---|---|---|---|
| 12/29/06 | 8.838 | 1003.3 | 1122.2 | 568.4 | 662.3 |
| 12/30/06 | 8.851 | 1003.0 | 1121.9 | 568.2 | 662.1 |
| 12/31/06 | 8.866 | 1002.7 | 1121.6 | 568.1 | 661.9 |
| 1/1/07 | 8.876 | 1002.5 | 1121.4 | 567.9 | 661.7 |
| 1/2/07 | 8.886 | 1002.3 | 1121.1 | 567.8 | 661.6 |
| 1/3/07 | 8.890 | 1002.2 | 1121.0 | 567.8 | 661.5 |
| 1/4/07 | 8.892 | 1002.2 | 1121.0 | 567.8 | 661.5 |
| 1/5/07 | 8.894 | 1002.1 | 1120.9 | 567.7 | 661.5 |
| 1/6/07 | 8.886 | 1002.3 | 1121.1 | 567.8 | 661.6 |
| 1/7/07 | 8.877 | 1002.5 | 1121.3 | 567.9 | 661.7 |
| 1/8/07 | 8.867 | 1002.7 | 1121.6 | 568.1 | 661.9 |
| 1/9/07 | 8.852 | 1003.0 | 1121.9 | 568.2 | 662.1 |
| 1/10/07 | 8.832 | 1003.4 | 1122.4 | 568.5 | 662.4 |
| 1/11/07 | 8.810 | 1003.9 | 1122.9 | 568.7 | 662.7 |
| 1/12/07 | 8.787 | 1004.4 | 1123.4 | 569.0 | 663.0 |
| 1/13/07 | 8.769 | 1004.8 | 1123.9 | 569.2 | 663.3 |
| 1/14/07 | 8.753 | 1005.1 | 1124.2 | 569.4 | 663.5 |
| 1/15/07 | 8.738 | 1005.4 | 1124.6 | 569.6 | 663.7 |
| 1/16/07 | 8.724 | 1005.7 | 1124.9 | 569.7 | 663.9 |
| 1/17/07 | 8.711 | 1006.0 | 1125.2 | 569.9 | 664.1 |

Scenario 3 - C3 vs. Injected Mass (Lander)

| Launch | C3 | Delta II 7925 (7% margin) | Delta II 7925 H (10% margin) | Delta II 7325 (Star 48B) (7% margin) | Delta II 7425 (Star 48B) (7% margin) |
|---|---|---|---|---|---|
| 1/15/07 | 8.738 | 1005.4 | 1124.6 | 569.6 | 663.7 |
| 1/16/07 | 8.724 | 1005.7 | 1124.9 | 569.7 | 663.9 |
| 1/17/07 | 8.711 | 1006.0 | 1125.2 | 569.9 | 664.1 |
| 1/18/07 | 8.704 | 1006.1 | 1125.4 | 570.0 | 664.2 |
| 1/19/07 | 8.695 | 1006.3 | 1125.6 | 570.1 | 664.3 |
| 1/20/07 | 8.690 | 1006.4 | 1125.7 | 570.2 | 664.4 |
| 1/21/07 | 8.689 | 1006.4 | 1125.7 | 570.2 | 664.4 |
| 1/22/07 | 8.694 | 1006.3 | 1125.6 | 570.1 | 664.4 |
| 1/23/07 | 8.700 | 1006.2 | 1125.5 | 570.0 | 664.3 |
| 1/24/07 | 8.709 | 1006.0 | 1125.3 | 569.9 | 664.1 |
| 1/25/07 | 8.725 | 1005.7 | 1124.9 | 569.7 | 663.9 |
| 1/26/07 | 8.737 | 1005.4 | 1124.6 | 569.6 | 663.7 |
| 1/27/07 | 8.750 | 1005.2 | 1124.3 | 569.4 | 663.5 |
| 1/28/07 | 8.765 | 1004.8 | 1124.0 | 569.3 | 663.3 |
| 1/29/07 | 8.777 | 1004.6 | 1123.7 | 569.1 | 663.2 |
| 1/30/07 | 8.787 | 1004.4 | 1123.4 | 569.0 | 663.0 |
| 1/31/07 | 8.796 | 1004.2 | 1123.2 | 568.9 | 662.9 |
| 2/1/07 | 8.800 | 1004.1 | 1123.1 | 568.8 | 662.8 |
| 2/2/07 | 8.803 | 1004.0 | 1123.1 | 568.8 | 662.8 |
| 2/3/07 | 8.805 | 1004.0 | 1123.0 | 568.8 | 662.7 |
| 2/4/07 | 8.800 | 1004.1 | 1123.1 | 568.8 | 662.8 |
| 2/5/07 | 8.793 | 1004.3 | 1123.3 | 568.9 | 662.9 |
| 2/6/07 | 8.785 | 1004.4 | 1123.5 | 569.0 | 663.0 |
| 2/7/07 | 8.774 | 1004.7 | 1123.7 | 569.2 | 663.2 |
| 2/8/07 | 8.757 | 1005.0 | 1124.1 | 569.4 | 663.4 |
| 2/9/07 | 8.731 | 1005.6 | 1124.8 | 569.7 | 663.8 |
| 2/10/07 | 8.707 | 1006.1 | 1125.3 | 570.0 | 664.2 |
| 2/11/07 | 8.687 | 1006.5 | 1125.8 | 570.2 | 664.5 |
| 2/12/07 | 8.671 | 1006.8 | 1126.2 | 570.4 | 664.7 |
| 2/13/07 | 8.660 | 1007.0 | 1126.4 | 570.5 | 664.8 |
| 2/14/07 | 8.644 | 1007.4 | 1126.8 | 570.7 | 665.1 |
| 2/15/07 | 8.637 | 1007.5 | 1127.0 | 570.8 | 665.2 |
| 2/16/07 | 8.627 | 1007.7 | 1127.2 | 570.9 | 665.3 |
| 2/17/07 | 8.617 | 1007.9 | 1127.4 | 571.0 | 665.5 |
| 2/18/07 | 8.617 | 1007.9 | 1127.4 | 571.0 | 665.5 |
| 2/19/07 | 8.618 | 1007.9 | 1127.4 | 571.0 | 665.5 |
| 2/20/07 | 8.626 | 1007.8 | 1127.2 | 570.9 | 665.3 |
| 2/21/07 | 8.635 | 1007.6 | 1127.0 | 570.8 | 665.2 |
| 2/22/07 | 8.648 | 1007.3 | 1126.7 | 570.7 | 665.0 |
| 2/23/07 | 8.666 | 1006.9 | 1126.3 | 570.4 | 664.8 |
| 2/24/07 | 8.687 | 1006.5 | 1125.8 | 570.2 | 664.5 |
| 2/25/07 | 8.706 | 1006.1 | 1125.3 | 570.0 | 664.2 |
| 2/26/07 | 8.724 | 1005.7 | 1124.9 | 569.7 | 663.9 |
| 2/27/07 | 8.745 | 1005.3 | 1124.4 | 569.5 | 663.6 |
| 2/28/07 | 8.764 | 1004.9 | 1124.0 | 569.3 | 663.3 |
| 3/1/07 | 8.778 | 1004.6 | 1123.7 | 569.1 | 663.1 |
| 3/2/07 | 8.791 | 1004.3 | 1123.3 | 569.0 | 662.9 |
| 3/3/07 | 8.806 | 1004.0 | 1123.0 | 568.8 | 662.7 |
| 3/4/07 | 8.814 | 1003.8 | 1122.8 | 568.7 | 662.6 |
| 3/5/07 | 8.818 | 1003.7 | 1122.7 | 568.6 | 662.6 |
| 3/6/07 | 8.823 | 1003.6 | 1122.6 | 568.6 | 662.5 |
| 3/7/07 | 8.829 | 1003.5 | 1122.5 | 568.5 | 662.4 |
| 3/8/07 | 8.830 | 1003.5 | 1122.4 | 568.5 | 662.4 |
| 3/9/07 | 8.826 | 1003.6 | 1122.5 | 568.5 | 662.4 |
| 3/10/07 | 8.814 | 1003.8 | 1122.8 | 568.7 | 662.6 |
| 3/11/07 | 8.784 | 1004.4 | 1123.5 | 569.0 | 663.0 |
| 3/12/07 | 8.766 | 1004.8 | 1123.9 | 569.3 | 663.3 |
| 3/13/07 | 8.763 | 1004.9 | 1124.0 | 569.3 | 663.4 |
| 3/14/07 | 8.766 | 1004.8 | 1123.9 | 569.3 | 663.3 |
| 3/15/07 | 8.763 | 1004.9 | 1124.0 | 569.3 | 663.4 |

Scenarios 4,5,6 - C3 vs. Injected Mass (Orbiter)

| Launch | C3 | Delta II 7925 (7% margin) | Delta II 7925 H (10% margin) | Delta II 7325 (Star 48B) (7% margin) | Delta II 7425 (Star 48B) (7% margin) |
|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 915.5 | 1034.0 | 518.1 | 602.2 |
| 9/14/07 | 13.130 | 917.6 | 1035.9 | 519.3 | 603.6 |
| 9/15/07 | 13.040 | 919.3 | 1037.5 | 520.3 | 604.8 |
| 9/16/07 | 12.970 | 920.7 | 1038.7 | 521.1 | 605.7 |
| 9/17/07 | 12.910 | 921.8 | 1039.8 | 521.7 | 606.5 |
| 9/18/07 | 12.860 | 922.8 | 1040.7 | 522.3 | 607.1 |
| 9/19/07 | 12.820 | 923.5 | 1041.4 | 522.7 | 607.7 |
| 9/20/07 | 12.800 | 923.9 | 1041.7 | 523.0 | 607.9 |
| 9/21/07 | 12.790 | 924.1 | 1041.9 | 523.1 | 608.0 |
| 9/22/07 | 12.800 | 923.9 | 1041.7 | 523.0 | 607.9 |
| 9/23/07 | 12.810 | 923.7 | 1041.5 | 522.8 | 607.8 |
| 9/24/07 | 12.830 | 923.3 | 1041.2 | 522.6 | 607.5 |
| 9/25/07 | 12.850 | 923.0 | 1040.8 | 522.4 | 607.3 |
| 9/26/07 | 12.890 | 922.2 | 1040.1 | 522.0 | 606.7 |
| 9/27/07 | 12.920 | 921.6 | 1039.6 | 521.6 | 606.4 |
| 9/28/07 | 12.950 | 921.0 | 1039.1 | 521.3 | 606.0 |
| 9/29/07 | 12.990 | 920.3 | 1038.4 | 520.9 | 605.4 |
| 9/30/07 | 13.040 | 919.3 | 1037.5 | 520.3 | 604.8 |
| 10/1/07 | 13.110 | 918.0 | 1036.3 | 519.5 | 603.9 |
| 10/2/07 | 12.990 | 920.3 | 1038.4 | 520.9 | 605.4 |

Scenarios 4,5 - C3 vs. Injected Mass (Lander)

| Launch | C3 | Delta II 7925 (7% margin) | Delta II 7925 H (10% margin) | Delta II 7325 (Star 48B) (7% margin) | Delta II 7425 (Star 48B) (7% margin) |
|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 984.3 | 1065.4 | 557.6 | 649.2 |
| 11/27/06 | 9.657 | 986.3 | 1067.6 | 558.7 | 650.6 |
| 11/28/06 | 9.572 | 988.1 | 1069.5 | 559.7 | 651.8 |
| 11/29/06 | 9.497 | 989.6 | 1071.2 | 560.6 | 652.8 |
| 11/30/06 | 9.434 | 990.9 | 1072.6 | 561.4 | 653.7 |
| 12/1/06 | 9.379 | 992.1 | 1073.8 | 562.0 | 654.5 |
| 12/2/06 | 9.328 | 993.1 | 1075.0 | 562.6 | 655.2 |
| 12/3/06 | 9.285 | 994.0 | 1075.9 | 563.1 | 655.9 |
| 12/4/06 | 9.247 | 994.8 | 1076.8 | 563.6 | 656.4 |
| 12/5/06 | 9.209 | 995.6 | 1077.6 | 564.0 | 656.9 |
| 12/6/06 | 9.173 | 996.3 | 1078.5 | 564.4 | 657.5 |
| 12/7/06 | 9.136 | 997.1 | 1079.3 | 564.9 | 658.0 |
| 12/8/06 | 9.103 | 997.8 | 1080.0 | 565.3 | 658.5 |
| 12/9/06 | 9.070 | 998.5 | 1080.8 | 565.6 | 658.9 |
| 12/10/06 | 9.037 | 999.2 | 1081.5 | 566.0 | 659.4 |
| 12/11/06 | 9.004 | 999.8 | 1082.3 | 566.4 | 659.9 |
| 12/12/06 | 8.962 | 1000.7 | 1083.2 | 566.9 | 660.5 |
| 12/13/06 | 8.922 | 1001.6 | 1084.1 | 567.4 | 661.1 |
| 12/14/06 | 8.892 | 1002.2 | 1084.8 | 567.8 | 661.5 |
| 12/15/06 | 8.878 | 1002.5 | 1085.1 | 567.9 | 661.7 |

Scenario 6 - C3 vs. Injected Mass (Lander)

| Launch | C3 | Delta II 7925 (7% margin) | Delta II 7925 H (10% margin) | Delta II 7325 (Star 48B) (7% margin) | Delta II 7425 (Star 48B) (7% margin) |
|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 984.3 | 1065.4 | 557.6 | 649.2 |
| 11/27/06 | 9.657 | 986.3 | 1067.6 | 558.7 | 650.6 |
| 11/28/06 | 9.572 | 988.1 | 1069.5 | 559.7 | 651.8 |
| 11/29/06 | 9.497 | 989.6 | 1071.2 | 560.6 | 652.8 |
| 11/30/06 | 9.434 | 990.9 | 1072.6 | 561.4 | 653.7 |
| 12/1/06 | 9.379 | 992.1 | 1073.8 | 562.0 | 654.5 |
| 12/2/06 | 9.328 | 993.1 | 1075.0 | 562.6 | 655.2 |
| 12/3/06 | 9.285 | 994.0 | 1075.9 | 563.1 | 655.9 |
| 12/4/06 | 9.247 | 994.8 | 1076.8 | 563.6 | 656.4 |
| 12/5/06 | 9.209 | 995.6 | 1077.6 | 564.0 | 656.9 |
| 12/6/06 | 9.173 | 996.3 | 1078.5 | 564.4 | 657.5 |
| 12/7/06 | 9.136 | 997.1 | 1079.3 | 564.9 | 658.0 |
| 12/8/06 | 9.103 | 997.8 | 1080.0 | 565.3 | 658.5 |
| 12/9/06 | 9.070 | 998.5 | 1080.8 | 565.6 | 658.9 |
| 12/10/06 | 9.037 | 999.2 | 1081.5 | 566.0 | 659.4 |
| 12/11/06 | 9.004 | 999.8 | 1082.3 | 566.4 | 659.9 |
| 12/12/06 | 8.962 | 1000.7 | 1083.2 | 566.9 | 660.5 |
| 12/13/06 | 8.922 | 1001.6 | 1084.1 | 567.4 | 661.1 |
| 12/14/06 | 8.892 | 1002.2 | 1084.8 | 567.8 | 661.5 |
| 12/15/06 | 8.878 | 1002.5 | 1085.1 | 567.9 | 661.7 |
| 12/16/06 | 8.846 | 1003.1 | 1085.9 | 568.3 | 662.2 |
| 12/17/06 | 8.830 | 1003.5 | 1086.2 | 568.5 | 662.4 |
| 12/18/06 | 8.812 | 1003.9 | 1086.6 | 568.7 | 662.6 |
| 12/19/06 | 8.796 | 1004.2 | 1087.0 | 568.9 | 662.9 |
| 12/20/06 | 8.785 | 1004.4 | 1087.2 | 569.0 | 663.0 |
| 12/21/06 | 8.781 | 1004.5 | 1087.3 | 569.1 | 663.1 |
| 12/22/06 | 8.777 | 1004.6 | 1087.4 | 569.1 | 663.2 |
| 12/23/06 | 8.774 | 1004.7 | 1087.5 | 569.2 | 663.2 |
| 12/24/06 | 8.777 | 1004.6 | 1087.4 | 569.1 | 663.2 |
| 12/25/06 | 8.784 | 1004.4 | 1087.3 | 569.0 | 663.0 |
| 12/26/06 | 8.795 | 1004.2 | 1087.0 | 568.9 | 662.9 |
| 12/27/06 | 8.809 | 1003.9 | 1086.7 | 568.7 | 662.7 |
| 12/28/06 | 8.822 | 1003.6 | 1086.4 | 568.6 | 662.5 |
| 12/29/06 | 8.838 | 1003.3 | 1086.0 | 568.4 | 662.3 |
| 12/30/06 | 8.851 | 1003.0 | 1085.7 | 568.2 | 662.1 |
| 12/31/06 | 8.866 | 1002.7 | 1085.4 | 568.1 | 661.9 |
| 1/1/07 | 8.876 | 1002.5 | 1085.2 | 567.9 | 661.7 |
| 1/2/07 | 8.866 | 1002.7 | 1085.4 | 568.1 | 661.9 |
| 1/3/07 | 8.890 | 1002.2 | 1084.9 | 567.8 | 661.5 |

# Scenario 8 - C3 vs. Injected Mass (Orbiter/Lander)

| Launch | C3 | Delta III (2 stage) (10% margin) +/- 25 kg | Delta III (3 stage) (10 % margin) +/- 25 kg | Atlas IIAS (10% margin) +/- 10 kg | Atlas IIAS (Star 48B) (10% margin) +/- 10 kg | Atlas IIAR (13% margin) +/- 10 kg |
|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 1828.7 | 1942.1 | 1841.9 | 1893.2 | 2000.0 |
| 9/14/07 | 13.130 | 1833.6 | 1945.7 | 1846.2 | 1896.7 | 2004.3 |
| 9/15/07 | 13.040 | 1837.6 | 1948.7 | 1849.7 | 1899.5 | 2007.9 |
| 9/16/07 | 12.960 | 1841.1 | 1951.3 | 1852.8 | 1902.1 | 2011.0 |
| 9/17/07 | 12.900 | 1843.8 | 1953.3 | 1855.2 | 1904.0 | 2013.4 |
| 9/18/07 | 12.860 | 1845.6 | 1954.6 | 1856.7 | 1905.3 | 2015.0 |
| 9/19/07 | 12.820 | 1847.4 | 1955.9 | 1858.3 | 1906.5 | 2016.6 |
| 9/20/07 | 12.800 | 1848.3 | 1956.6 | 1859.1 | 1907.2 | 2017.4 |
| 9/21/07 | 12.790 | 1848.7 | 1956.9 | 1859.5 | 1907.5 | 2017.8 |
| 9/22/07 | 12.790 | 1848.7 | 1956.9 | 1859.5 | 1907.5 | 2017.8 |
| 9/23/07 | 12.810 | 1847.8 | 1956.3 | 1858.7 | 1906.9 | 2017.0 |
| 9/24/07 | 12.820 | 1847.4 | 1955.9 | 1858.3 | 1906.5 | 2016.6 |
| 9/25/07 | 12.850 | 1846.1 | 1954.9 | 1857.1 | 1905.6 | 2015.4 |
| 9/26/07 | 12.880 | 1844.7 | 1954.0 | 1856.0 | 1904.6 | 2014.2 |
| 9/27/07 | 12.920 | 1842.9 | 1952.6 | 1854.4 | 1903.4 | 2012.6 |
| 9/28/07 | 12.950 | 1841.6 | 1951.6 | 1853.2 | 1902.4 | 2011.4 |
| 9/29/07 | 12.990 | 1839.8 | 1950.3 | 1851.6 | 1901.1 | 2009.9 |
| 9/30/07 | 13.040 | 1837.6 | 1948.7 | 1849.7 | 1899.5 | 2007.9 |
| 10/1/07 | 13.110 | 1834.5 | 1946.4 | 1847.0 | 1897.3 | 2005.1 |
| 10/2/07 | 12.990 | 1839.8 | 1950.3 | 1851.6 | 1901.1 | 2009.9 |
| Launch | C3 | Atlas IIAR (Star 48B) (13% margin) +/- 10 kg | Atlas IIARS (13% margin) +/- 10 kg | tlas IIARS (Star 48 (13% margin) +/- 10 kg | Ariane 4 (10% margin) | Ariane 5 (10% margin) |
| 9/13/07 | 13.240 | 2018.1 | 2143.2 | 2079.8 | 6433.2 | 2979.7 |
| 9/14/07 | 13.130 | 2021.7 | 2147.8 | 2083.2 | 6447.7 | 2988.6 |
| 9/15/07 | 13.040 | 2024.6 | 2151.6 | 2085.9 | 6459.5 | 2996.0 |
| 9/16/07 | 12.960 | 2027.3 | 2154.9 | 2088.3 | 6470.0 | 3002.5 |
| 9/17/07 | 12.900 | 2029.2 | 2157.4 | 2090.1 | 6478.0 | 3007.4 |
| 9/18/07 | 12.860 | 2030.6 | 2159.1 | 2091.4 | 6483.3 | 3010.7 |
| 9/19/07 | 12.820 | 2031.9 | 2160.7 | 2092.6 | 6488.6 | 3014.0 |
| 9/20/07 | 12.800 | 2032.5 | 2161.6 | 2093.2 | 6491.2 | 3015.6 |
| 9/21/07 | 12.790 | 2032.9 | 2162.0 | 2093.5 | 6492.6 | 3016.4 |
| 9/22/07 | 12.790 | 2032.9 | 2162.0 | 2093.5 | 6492.6 | 3016.4 |
| 9/23/07 | 12.810 | 2032.2 | 2161.2 | 2092.9 | 6489.9 | 3014.8 |
| 9/24/07 | 12.820 | 2031.9 | 2160.7 | 2092.6 | 6488.6 | 3014.0 |
| 9/25/07 | 12.850 | 2030.9 | 2159.5 | 2091.7 | 6484.6 | 3011.5 |
| 9/26/07 | 12.880 | 2029.9 | 2158.2 | 2090.8 | 6480.6 | 3009.1 |
| 9/27/07 | 12.920 | 2028.6 | 2156.6 | 2089.5 | 6475.3 | 3005.8 |
| 9/28/07 | 12.950 | 2027.6 | 2155.3 | 2088.6 | 6471.4 | 3003.3 |
| 9/29/07 | 12.990 | 2026.3 | 2153.6 | 2087.4 | 6466.1 | 3000.1 |
| 9/30/07 | 13.040 | 2024.6 | 2151.6 | 2085.9 | 6459.5 | 2996.0 |
| 10/1/07 | 13.110 | 2022.3 | 2148.6 | 2083.8 | 6450.3 | 2990.3 |
| 10/2/07 | 12.990 | 2026.3 | 2153.6 | 2087.4 | 6466.1 | 3000.1 |

# APPENDIX D. POST-CAPTURE ORBITER MASS CALCULATIONS

Scenario 1 - Post-Capture Orbiter Mass (Delta II 7925H)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | MOI ΔV (km/s) | MOI Prop (kg) | Post-Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1100.9 | 10.5 | 2.286 | 564.2 | 526.2 |
| 11/27/06 | 9.656 | 3.090 | 1103.2 | 10.5 | 2.288 | 565.7 | 527.0 |
| 11/28/06 | 9.572 | 3.097 | 1105.2 | 10.5 | 2.292 | 567.3 | 527.3 |
| 11/29/06 | 9.497 | 3.104 | 1106.9 | 10.5 | 2.296 | 568.8 | 527.5 |
| 11/30/06 | 9.433 | 3.116 | 1108.3 | 10.5 | 2.302 | 570.7 | 527.1 |
| 12/1/06 | 9.378 | 3.130 | 1109.6 | 10.6 | 2.310 | 572.6 | 526.5 |
| 12/2/06 | 9.327 | 3.143 | 1110.8 | 10.6 | 2.317 | 574.4 | 525.8 |
| 12/3/06 | 9.284 | 3.163 | 1111.8 | 10.6 | 2.328 | 576.7 | 524.5 |
| 12/4/06 | 9.245 | 3.189 | 1112.7 | 10.6 | 2.342 | 579.6 | 522.5 |
| 12/5/06 | 9.207 | 3.215 | 1113.6 | 10.6 | 2.356 | 582.4 | 520.6 |
| 12/6/06 | 9.171 | 3.246 | 1114.4 | 10.6 | 2.373 | 585.7 | 518.1 |
| 12/7/06 | 9.135 | 3.274 | 1115.3 | 10.6 | 2.389 | 588.7 | 516.0 |
| 12/8/06 | 9.102 | 3.315 | 1116.0 | 10.6 | 2.412 | 592.9 | 512.5 |
| 12/9/06 | 9.068 | 3.354 | 1116.8 | 10.6 | 2.434 | 596.9 | 509.3 |
| 12/10/06 | 9.035 | 3.399 | 1117.6 | 10.6 | 2.460 | 601.4 | 505.5 |
| 12/11/06 | 8.997 | 3.437 | 1118.4 | 10.6 | 2.481 | 605.4 | 502.3 |
| 12/12/06 | 8.960 | 3.486 | 1119.3 | 10.7 | 2.510 | 610.4 | 498.2 |
| 12/13/06 | 8.920 | 3.522 | 1120.3 | 10.7 | 2.531 | 614.3 | 495.3 |
| 12/14/06 | 8.889 | 3.570 | 1121.0 | 10.7 | 2.559 | 619.1 | 491.2 |
| 12/15/06 | 8.865 | 3.621 | 1121.3 | 10.7 | 2.589 | 624.0 | 486.6 |

Scenario 1- Post-Capture Orbiter Mass (Delta II 7325 Star 48B)

| Launch | C3 | V inf (km/s) | Inj Mass (kg) | TCMs (30 m/s) Prop Mass | MOI ΔV (km/s) | Prop Mass (kg) | Post MOI Mass (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 984.6 | 9.4 | 2.286 | 504.6 | 470.6 |
| 11/27/06 | 9.656 | 3.090 | 986.4 | 9.4 | 2.288 | 505.8 | 471.2 |
| 11/28/06 | 9.572 | 3.097 | 988.1 | 9.4 | 2.292 | 507.2 | 471.5 |
| 11/29/06 | 9.497 | 3.104 | 989.6 | 9.4 | 2.296 | 508.6 | 471.6 |
| 11/30/06 | 9.433 | 3.116 | 990.9 | 9.4 | 2.302 | 510.2 | 471.3 |
| 12/1/06 | 9.378 | 3.130 | 992.1 | 9.4 | 2.310 | 512.0 | 470.7 |
| 12/2/06 | 9.327 | 3.143 | 993.1 | 9.4 | 2.317 | 513.5 | 470.1 |
| 12/3/06 | 9.284 | 3.163 | 994.0 | 9.5 | 2.328 | 515.6 | 468.9 |
| 12/4/06 | 9.245 | 3.189 | 994.8 | 9.5 | 2.342 | 518.2 | 467.2 |
| 12/5/06 | 9.207 | 3.215 | 995.6 | 9.5 | 2.356 | 520.7 | 465.4 |
| 12/6/06 | 9.171 | 3.246 | 996.4 | 9.5 | 2.373 | 523.7 | 463.3 |
| 12/7/06 | 9.135 | 3.274 | 997.1 | 9.5 | 2.389 | 526.3 | 461.3 |
| 12/8/06 | 9.102 | 3.315 | 997.8 | 9.5 | 2.412 | 530.1 | 458.2 |
| 12/9/06 | 9.068 | 3.354 | 998.5 | 9.5 | 2.434 | 533.7 | 455.3 |
| 12/10/06 | 9.035 | 3.399 | 999.2 | 9.5 | 2.460 | 537.7 | 452.0 |
| 12/11/06 | 8.997 | 3.437 | 1000.0 | 9.5 | 2.481 | 541.3 | 449.2 |
| 12/12/06 | 8.960 | 3.486 | 1000.8 | 9.5 | 2.510 | 545.8 | 445.5 |
| 12/13/06 | 8.920 | 3.522 | 1001.6 | 9.5 | 2.531 | 549.2 | 442.9 |
| 12/14/06 | 8.889 | 3.570 | 1002.2 | 9.5 | 2.559 | 553.5 | 439.1 |
| 12/15/06 | 8.865 | 3.621 | 1002.7 | 9.5 | 2.589 | 558.0 | 435.1 |

Scenario 1- Post-Capture Orbiter Mass (Delta II 7425 Star 48B)

| Launch | C3 | V inf (km/s) | Inj Mass (kg) | TCMs (30 m/s) Prop Mass | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 649.2 | 6.2 | 2.286 | 332.7 | 310.3 |
| 11/27/06 | 9.656 | 3.090 | 650.6 | 6.2 | 2.288 | 333.6 | 310.8 |
| 11/28/06 | 9.572 | 3.097 | 651.8 | 6.2 | 2.292 | 334.6 | 311.0 |
| 11/29/06 | 9.497 | 3.104 | 652.8 | 6.2 | 2.296 | 335.5 | 311.1 |
| 11/30/06 | 9.433 | 3.116 | 653.7 | 6.2 | 2.302 | 336.6 | 310.9 |
| 12/1/06 | 9.378 | 3.130 | 654.5 | 6.2 | 2.310 | 337.7 | 310.5 |
| 12/2/06 | 9.327 | 3.143 | 655.2 | 6.2 | 2.317 | 338.8 | 310.2 |
| 12/3/06 | 9.284 | 3.163 | 655.9 | 6.2 | 2.328 | 340.2 | 309.4 |
| 12/4/06 | 9.245 | 3.189 | 656.4 | 6.2 | 2.342 | 341.9 | 308.3 |
| 12/5/06 | 9.207 | 3.215 | 656.9 | 6.3 | 2.356 | 343.6 | 307.1 |
| 12/6/06 | 9.171 | 3.246 | 657.5 | 6.3 | 2.373 | 345.5 | 305.7 |
| 12/7/06 | 9.135 | 3.274 | 658.0 | 6.3 | 2.389 | 347.3 | 304.4 |
| 12/8/06 | 9.102 | 3.315 | 658.5 | 6.3 | 2.412 | 349.8 | 302.4 |
| 12/9/06 | 9.068 | 3.354 | 658.9 | 6.3 | 2.434 | 352.2 | 300.5 |
| 12/10/06 | 9.035 | 3.399 | 659.4 | 6.3 | 2.460 | 354.9 | 298.3 |
| 12/11/06 | 8.997 | 3.437 | 659.9 | 6.3 | 2.481 | 357.2 | 296.4 |
| 12/12/06 | 8.960 | 3.486 | 660.5 | 6.3 | 2.510 | 360.2 | 294.0 |
| 12/13/06 | 8.920 | 3.522 | 661.1 | 6.3 | 2.531 | 362.5 | 292.3 |
| 12/14/06 | 8.889 | 3.570 | 661.5 | 6.3 | 2.559 | 365.3 | 289.9 |
| 12/15/06 | 8.865 | 3.621 | 661.7 | 6.3 | 2.589 | 368.2 | 287.2 |

Scenario 2- Post-Capture Orbiter Mass (Delta II 7925H)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (30 m/s) (kg) | Cruise Stage (kg) | TPS Mass (kg) | Post AC (kg) | Post AC Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1100.9 | 10.5 | 75 | 203.3 | 822.7 | 33.4 | 789.3 |
| 11/27/06 | 9.657 | 3.090 | 1103.2 | 10.5 | 75 | 203.3 | 824.9 | 33.5 | 791.4 |
| 11/28/06 | 9.572 | 3.097 | 1105.2 | 10.5 | 75 | 203.4 | 826.7 | 33.6 | 793.2 |
| 11/29/06 | 9.497 | 3.104 | 1106.9 | 10.5 | 75 | 203.6 | 828.3 | 33.6 | 794.7 |
| 11/30/06 | 9.434 | 3.116 | 1108.3 | 10.5 | 75 | 203.8 | 829.6 | 33.7 | 795.9 |
| 12/1/06 | 9.378 | 3.131 | 1109.6 | 10.6 | 75 | 204.0 | 830.6 | 33.7 | 796.9 |
| 12/2/06 | 9.327 | 3.143 | 1110.8 | 10.6 | 75 | 204.2 | 831.6 | 33.7 | 797.8 |
| 12/3/06 | 9.284 | 3.163 | 1111.8 | 10.6 | 75 | 204.6 | 832.2 | 33.8 | 798.4 |
| 12/4/06 | 9.245 | 3.189 | 1112.7 | 10.6 | 75 | 205.1 | 832.6 | 33.8 | 798.8 |
| 12/5/06 | 9.207 | 3.215 | 1113.6 | 10.6 | 75 | 205.5 | 833.0 | 33.8 | 799.2 |
| 12/6/06 | 9.171 | 3.246 | 1114.4 | 10.6 | 75 | 206.1 | 833.3 | 33.8 | 799.5 |
| 12/7/06 | 9.135 | 3.275 | 1115.3 | 10.6 | 75 | 206.7 | 833.6 | 33.8 | 799.8 |
| 12/8/06 | 9.101 | 3.312 | 1116.0 | 10.6 | 75 | 207.4 | 833.6 | 33.8 | 799.8 |
| 12/9/06 | 9.068 | 3.354 | 1116.8 | 10.6 | 75 | 208.2 | 833.6 | 33.8 | 799.7 |
| 12/10/06 | 9.035 | 3.399 | 1117.6 | 10.6 | 75 | 209.2 | 833.4 | 33.8 | 799.6 |
| 12/11/06 | 9.002 | 3.438 | 1118.4 | 10.6 | 75 | 210.0 | 833.4 | 33.8 | 799.5 |
| 12/12/06 | 8.960 | 3.487 | 1119.3 | 10.7 | 75 | 211.1 | 833.3 | 33.8 | 799.5 |
| 12/13/06 | 8.920 | 3.522 | 1120.3 | 10.7 | 75 | 211.8 | 833.4 | 33.8 | 799.6 |
| 12/14/06 | 8.889 | 3.570 | 1121.0 | 10.7 | 75 | 212.9 | 833.0 | 33.8 | 799.2 |
| 12/15/06 | 8.876 | 3.647 | 1121.3 | 10.7 | 75 | 214.7 | 831.6 | 33.7 | 797.8 |

## Scenario 2- Post-Capture Orbiter Mass (Delta II 7325 Star 48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (30 m/s) (kg) | Cruise Stage (kg) | TPS Mass (kg) | Post AC (kg) | Post AC Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 557.6 | 5.3 | 75 | 203.3 | 279.3 | 11.3 | 268.0 |
| 11/27/06 | 9.657 | 3.090 | 558.7 | 5.3 | 75 | 203.3 | 280.4 | 11.4 | 269.0 |
| 11/28/06 | 9.572 | 3.097 | 559.7 | 5.3 | 75 | 203.4 | 281.3 | 11.4 | 269.9 |
| 11/29/06 | 9.497 | 3.104 | 560.6 | 5.3 | 75 | 203.6 | 282.1 | 11.4 | 270.6 |
| 11/30/06 | 9.434 | 3.116 | 561.4 | 5.3 | 75 | 203.8 | 282.6 | 11.5 | 271.1 |
| 12/1/06 | 9.378 | 3.131 | 562.0 | 5.3 | 75 | 204.0 | 283.0 | 11.5 | 271.5 |
| 12/2/06 | 9.327 | 3.143 | 562.6 | 5.4 | 75 | 204.2 | 283.4 | 11.5 | 271.9 |
| 12/3/06 | 9.284 | 3.163 | 563.1 | 5.4 | 75 | 204.6 | 283.5 | 11.5 | 272.0 |
| 12/4/06 | 9.245 | 3.189 | 563.6 | 5.4 | 75 | 205.1 | 283.5 | 11.5 | 272.0 |
| 12/5/06 | 9.207 | 3.215 | 564.0 | 5.4 | 75 | 205.5 | 283.5 | 11.5 | 272.0 |
| 12/6/06 | 9.171 | 3.246 | 564.4 | 5.4 | 75 | 206.1 | 283.3 | 11.5 | 271.8 |
| 12/7/06 | 9.135 | 3.275 | 564.9 | 5.4 | 75 | 206.7 | 283.2 | 11.5 | 271.7 |
| 12/8/06 | 9.101 | 3.312 | 565.3 | 5.4 | 75 | 207.4 | 282.9 | 11.5 | 271.4 |
| 12/9/06 | 9.068 | 3.354 | 565.6 | 5.4 | 75 | 208.2 | 282.4 | 11.5 | 270.9 |
| 12/10/06 | 9.035 | 3.399 | 566.0 | 5.4 | 75 | 209.2 | 281.9 | 11.4 | 270.4 |
| 12/11/06 | 9.002 | 3.438 | 566.4 | 5.4 | 75 | 210.0 | 281.4 | 11.4 | 270.0 |
| 12/12/06 | 8.960 | 3.487 | 566.9 | 5.4 | 75 | 211.1 | 280.9 | 11.4 | 269.5 |
| 12/13/06 | 8.920 | 3.522 | 567.4 | 5.4 | 75 | 211.8 | 280.6 | 11.4 | 269.2 |
| 12/14/06 | 8.889 | 3.570 | 567.8 | 5.4 | 75 | 212.9 | 279.8 | 11.4 | 268.5 |
| 12/15/06 | 8.876 | 3.647 | 567.9 | 5.4 | 75 | 214.7 | 278.2 | 11.3 | 266.9 |

## Scenario 2- Post-Capture Orbiter Mass (Delta II 7425 Star 48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (30 m/s) (kg) | Cruise Stage (kg) | TPS Mass (kg) | Post AC (kg) | Post AC Mvr (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 649.2 | 6.2 | 75 | 203.3 | 370.9 | 15.1 |
| 11/27/06 | 9.657 | 3.090 | 650.6 | 6.2 | 75 | 203.3 | 372.3 | 15.1 |
| 11/28/06 | 9.572 | 3.097 | 651.8 | 6.2 | 75 | 203.4 | 373.3 | 15.2 |
| 11/29/06 | 9.497 | 3.104 | 652.8 | 6.2 | 75 | 203.6 | 374.3 | 15.2 |
| 11/30/06 | 9.434 | 3.116 | 653.7 | 6.2 | 75 | 203.8 | 375.0 | 15.2 |
| 12/1/06 | 9.378 | 3.131 | 654.5 | 6.2 | 75 | 204.0 | 375.5 | 15.2 |
| 12/2/06 | 9.327 | 3.143 | 655.2 | 6.2 | 75 | 204.2 | 376.0 | 15.3 |
| 12/3/06 | 9.284 | 3.163 | 655.9 | 6.2 | 75 | 204.6 | 376.3 | 15.3 |
| 12/4/06 | 9.245 | 3.189 | 656.4 | 6.2 | 75 | 205.1 | 376.3 | 15.3 |
| 12/5/06 | 9.207 | 3.215 | 656.9 | 6.3 | 75 | 205.5 | 376.4 | 15.3 |
| 12/6/06 | 9.171 | 3.246 | 657.5 | 6.3 | 75 | 206.1 | 376.3 | 15.3 |
| 12/7/06 | 9.135 | 3.275 | 658.0 | 6.3 | 75 | 206.7 | 376.3 | 15.3 |
| 12/8/06 | 9.101 | 3.312 | 658.5 | 6.3 | 75 | 207.4 | 376.1 | 15.3 |
| 12/9/06 | 9.068 | 3.354 | 658.9 | 6.3 | 75 | 208.2 | 375.7 | 15.2 |
| 12/10/06 | 9.035 | 3.399 | 659.4 | 6.3 | 75 | 209.2 | 375.2 | 15.2 |
| 12/11/06 | 9.002 | 3.438 | 659.9 | 6.3 | 75 | 210.0 | 374.9 | 15.2 |
| 12/12/06 | 8.960 | 3.487 | 660.5 | 6.3 | 75 | 211.1 | 374.4 | 15.2 |
| 12/13/06 | 8.920 | 3.522 | 661.1 | 6.3 | 75 | 211.8 | 374.2 | 15.2 |
| 12/14/06 | 8.889 | 3.570 | 661.5 | 6.3 | 75 | 212.9 | 373.6 | 15.2 |
| 12/15/06 | 8.876 | 3.647 | 661.7 | 6.3 | 75 | 214.7 | 372.0 | 15.1 |

## Scenario 3 - Post-Capture Orbiter Mass (Delta II 7925H)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1034.0 | 9.8 | 1.207 | 1.105 | 1.059 |
| 11/27/06 | 9.656 | 3.090 | 1035.9 | 9.9 | 1.209 | 1.106 | 1.060 |
| 11/28/06 | 9.572 | 3.097 | 1037.5 | 9.9 | 1.213 | 1.110 | 1.064 |
| 11/29/06 | 9.497 | 3.104 | 1038.7 | 9.9 | 1.217 | 1.114 | 1.068 |
| 11/30/06 | 9.433 | 3.116 | 1039.8 | 9.9 | 1.223 | 1.120 | 1.074 |
| 12/1/06 | 9.378 | 3.130 | 1040.7 | 9.9 | 1.231 | 1.128 | 1.082 |
| 12/2/06 | 9.327 | 3.143 | 1041.4 | 9.9 | 1.238 | 1.135 | 1.089 |
| 12/3/06 | 9.284 | 3.163 | 1041.7 | 9.9 | 1.249 | 1.146 | 1.100 |
| 12/4/06 | 9.245 | 3.189 | 1041.9 | 9.9 | 1.263 | 1.160 | 1.114 |
| 12/5/06 | 9.207 | 3.215 | 1041.7 | 9.9 | 1.277 | 1.174 | 1.128 |
| 12/6/06 | 9.171 | 3.246 | 1041.5 | 9.9 | 1.294 | 1.191 | 1.146 |
| 12/7/06 | 9.135 | 3.274 | 1041.2 | 9.9 | 1.310 | 1.207 | 1.161 |
| 12/8/06 | 9.102 | 3.315 | 1040.8 | 9.9 | 1.333 | 1.230 | 1.184 |
| 12/9/06 | 9.068 | 3.354 | 1040.1 | 9.9 | 1.355 | 1.252 | 1.206 |
| 12/10/06 | 9.035 | 3.399 | 1039.6 | 9.9 | 1.381 | 1.278 | 1.232 |
| 12/11/06 | 8.997 | 3.437 | 1039.1 | 9.9 | 1.402 | 1.300 | 1.254 |
| 12/12/06 | 8.960 | 3.486 | 1038.4 | 9.9 | 1.431 | 1.328 | 1.282 |
| 12/13/06 | 8.920 | 3.522 | 1037.5 | 9.9 | 1.452 | 1.349 | 1.303 |
| 12/14/06 | 8.889 | 3.570 | 1036.3 | 9.9 | 1.480 | 1.377 | 1.332 |
| 12/15/06 | 8.865 | 3.621 | 1038.4 | 9.9 | 1.511 | 1.408 | 1.362 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 327.1 | 303.9 | 293.3 | 50 | 697.0 | 720.2 | 730.8 |
| 11/27/06 | 328.1 | 304.9 | 294.2 | 50 | 698.0 | 721.2 | 731.8 |
| 11/28/06 | 329.4 | 306.2 | 295.6 | 50 | 698.2 | 721.4 | 732.1 |
| 11/29/06 | 330.7 | 307.4 | 296.8 | 50 | 698.2 | 721.4 | 732.0 |
| 11/30/06 | 332.4 | 309.2 | 298.6 | 50 | 697.5 | 720.7 | 731.3 |
| 12/1/06 | 334.4 | 311.2 | 300.6 | 50 | 696.4 | 719.6 | 730.1 |
| 12/2/06 | 336.2 | 313.0 | 302.5 | 50 | 695.3 | 718.4 | 729.0 |
| 12/3/06 | 338.7 | 315.6 | 305.1 | 50 | 693.1 | 716.2 | 726.7 |
| 12/4/06 | 341.9 | 318.9 | 308.4 | 50 | 690.1 | 713.1 | 723.6 |
| 12/5/06 | 345.0 | 322.1 | 311.7 | 50 | 686.8 | 709.7 | 720.1 |
| 12/6/06 | 348.7 | 325.9 | 315.5 | 50 | 683.0 | 705.7 | 716.1 |
| 12/7/06 | 351.9 | 329.3 | 319.0 | 50 | 679.4 | 702.0 | 712.3 |
| 12/8/06 | 356.8 | 334.3 | 324.1 | 50 | 674.2 | 696.6 | 706.9 |
| 12/9/06 | 361.3 | 339.0 | 328.8 | 50 | 669.0 | 691.3 | 701.4 |
| 12/10/06 | 366.5 | 344.4 | 334.4 | 50 | 663.2 | 685.3 | 695.4 |
| 12/11/06 | 370.9 | 349.0 | 339.0 | 50 | 658.3 | 680.2 | 690.2 |
| 12/12/06 | 376.6 | 354.9 | 345.0 | 50 | 651.9 | 673.6 | 683.5 |
| 12/13/06 | 380.7 | 359.1 | 349.3 | 50 | 647.0 | 668.5 | 678.4 |
| 12/14/06 | 386.0 | 364.7 | 354.9 | 50 | 640.4 | 661.7 | 671.5 |
| 12/15/06 | 393.0 | 371.8 | 362.1 | 50 | 635.5 | 656.7 | 666.4 |

## Scenario 3 - Post-Capture Orbiter Mass (Delta II 7325 Star 48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 518.1 | 4.9 | 1.207 | 1.105 | 1.059 |
| 11/27/06 | 9.656 | 3.090 | 519.3 | 4.9 | 1.209 | 1.106 | 1.060 |
| 11/28/06 | 9.572 | 3.097 | 520.3 | 5.0 | 1.213 | 1.110 | 1.064 |
| 11/29/06 | 9.497 | 3.104 | 521.1 | 5.0 | 1.217 | 1.114 | 1.068 |
| 11/30/06 | 9.433 | 3.116 | 521.7 | 5.0 | 1.223 | 1.120 | 1.074 |
| 12/1/06 | 9.378 | 3.130 | 522.3 | 5.0 | 1.231 | 1.128 | 1.082 |
| 12/2/06 | 9.327 | 3.143 | 522.7 | 5.0 | 1.238 | 1.135 | 1.089 |
| 12/3/06 | 9.284 | 3.163 | 523.0 | 5.0 | 1.249 | 1.146 | 1.100 |
| 12/4/06 | 9.245 | 3.189 | 523.1 | 5.0 | 1.263 | 1.160 | 1.114 |
| 12/5/06 | 9.207 | 3.215 | 523.0 | 5.0 | 1.277 | 1.174 | 1.128 |
| 12/6/06 | 9.171 | 3.246 | 522.8 | 5.0 | 1.294 | 1.191 | 1.146 |
| 12/7/06 | 9.135 | 3.274 | 522.6 | 5.0 | 1.310 | 1.207 | 1.161 |
| 12/8/06 | 9.102 | 3.315 | 522.4 | 5.0 | 1.333 | 1.230 | 1.184 |
| 12/9/06 | 9.068 | 3.354 | 522.0 | 5.0 | 1.355 | 1.252 | 1.206 |
| 12/10/06 | 9.035 | 3.399 | 521.6 | 5.0 | 1.381 | 1.278 | 1.232 |
| 12/11/06 | 8.997 | 3.437 | 521.3 | 5.0 | 1.402 | 1.300 | 1.254 |
| 12/12/06 | 8.960 | 3.486 | 520.9 | 5.0 | 1.431 | 1.328 | 1.282 |
| 12/13/06 | 8.920 | 3.522 | 520.3 | 5.0 | 1.452 | 1.349 | 1.303 |
| 12/14/06 | 8.889 | 3.570 | 519.5 | 4.9 | 1.480 | 1.377 | 1.332 |
| 12/15/06 | 8.865 | 3.621 | 520.9 | 5.0 | 1.511 | 1.408 | 1.362 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 163.9 | 152.3 | 147.0 | 50 | 349.3 | 360.9 | 366.2 |
| 11/27/06 | 164.5 | 152.8 | 147.5 | 50 | 349.9 | 361.5 | 366.9 |
| 11/28/06 | 165.2 | 153.5 | 148.2 | 50 | 350.1 | 361.8 | 367.1 |
| 11/29/06 | 165.9 | 154.2 | 148.9 | 50 | 350.2 | 361.9 | 367.2 |
| 11/30/06 | 166.8 | 155.2 | 149.8 | 50 | 350.0 | 361.6 | 366.9 |
| 12/1/06 | 167.8 | 156.2 | 150.9 | 50 | 349.5 | 361.1 | 366.4 |
| 12/2/06 | 168.8 | 157.1 | 151.8 | 50 | 349.0 | 360.6 | 365.9 |
| 12/3/06 | 170.0 | 158.5 | 153.2 | 50 | 347.9 | 359.5 | 364.8 |
| 12/4/06 | 171.6 | 160.1 | 154.8 | 50 | 346.4 | 358.0 | 363.2 |
| 12/5/06 | 173.2 | 161.7 | 156.5 | 50 | 344.8 | 356.3 | 361.5 |
| 12/6/06 | 175.0 | 163.6 | 158.4 | 50 | 342.8 | 354.3 | 359.5 |
| 12/7/06 | 176.7 | 165.3 | 160.1 | 50 | 341.0 | 352.3 | 357.5 |
| 12/8/06 | 179.1 | 167.8 | 162.7 | 50 | 338.4 | 349.6 | 354.8 |
| 12/9/06 | 181.3 | 170.1 | 165.0 | 50 | 335.7 | 346.9 | 352.0 |
| 12/10/06 | 183.9 | 172.8 | 167.8 | 50 | 332.8 | 343.8 | 348.9 |
| 12/11/06 | 186.1 | 175.1 | 170.1 | 50 | 330.2 | 341.2 | 346.3 |
| 12/12/06 | 188.9 | 178.0 | 173.1 | 50 | 327.0 | 337.9 | 342.8 |
| 12/13/06 | 190.9 | 180.1 | 175.2 | 50 | 324.5 | 335.3 | 340.2 |
| 12/14/06 | 193.5 | 182.8 | 177.9 | 50 | 321.1 | 331.8 | 336.6 |
| 12/15/06 | 197.1 | 186.5 | 181.6 | 50 | 318.8 | 329.4 | 334.2 |

## Scenario 3 - Post-Capture Orbiter Mass (Delta II 7425 Star 48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 602.2 | 5.7 | 1.207 | 1.105 | 1.059 |
| 11/27/06 | 9.656 | 3.090 | 603.6 | 5.7 | 1.209 | 1.106 | 1.060 |
| 11/28/06 | 9.572 | 3.097 | 604.8 | 5.8 | 1.213 | 1.110 | 1.064 |
| 11/29/06 | 9.497 | 3.104 | 605.7 | 5.8 | 1.217 | 1.114 | 1.068 |
| 11/30/06 | 9.433 | 3.116 | 606.5 | 5.8 | 1.223 | 1.120 | 1.074 |
| 12/1/06 | 9.378 | 3.130 | 607.1 | 5.8 | 1.231 | 1.128 | 1.082 |
| 12/2/06 | 9.327 | 3.143 | 607.7 | 5.8 | 1.238 | 1.135 | 1.089 |
| 12/3/06 | 9.284 | 3.163 | 607.9 | 5.8 | 1.249 | 1.146 | 1.100 |
| 12/4/06 | 9.245 | 3.189 | 608.0 | 5.8 | 1.263 | 1.160 | 1.114 |
| 12/5/06 | 9.207 | 3.215 | 607.9 | 5.8 | 1.277 | 1.174 | 1.128 |
| 12/6/06 | 9.171 | 3.246 | 607.8 | 5.8 | 1.294 | 1.191 | 1.146 |
| 12/7/06 | 9.135 | 3.274 | 607.5 | 5.8 | 1.310 | 1.207 | 1.161 |
| 12/8/06 | 9.102 | 3.315 | 607.3 | 5.8 | 1.333 | 1.230 | 1.184 |
| 12/9/06 | 9.068 | 3.354 | 606.7 | 5.8 | 1.355 | 1.252 | 1.206 |
| 12/10/06 | 9.035 | 3.399 | 606.4 | 5.8 | 1.381 | 1.278 | 1.232 |
| 12/11/06 | 8.997 | 3.437 | 606.0 | 5.8 | 1.402 | 1.300 | 1.254 |
| 12/12/06 | 8.960 | 3.486 | 605.4 | 5.8 | 1.431 | 1.328 | 1.282 |
| 12/13/06 | 8.920 | 3.522 | 604.8 | 5.8 | 1.452 | 1.349 | 1.303 |
| 12/14/06 | 8.889 | 3.570 | 603.9 | 5.7 | 1.480 | 1.377 | 1.332 |
| 12/15/06 | 8.865 | 3.621 | 605.4 | 5.8 | 1.511 | 1.408 | 1.362 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 11/26/06 | 190.5 | 177.0 | 170.8 | 50 | 405.9 | 419.5 | 425.6 |
| 11/27/06 | 191.2 | 177.6 | 171.5 | 50 | 406.7 | 420.2 | 426.4 |
| 11/28/06 | 192.0 | 178.5 | 172.3 | 50 | 407.0 | 420.5 | 426.7 |
| 11/29/06 | 192.8 | 179.3 | 173.1 | 50 | 407.1 | 420.7 | 426.9 |
| 11/30/06 | 193.9 | 180.4 | 174.2 | 50 | 406.8 | 420.4 | 426.5 |
| 12/1/06 | 195.1 | 181.6 | 175.4 | 50 | 406.3 | 419.8 | 426.0 |
| 12/2/06 | 196.2 | 182.7 | 176.5 | 50 | 405.7 | 419.2 | 425.4 |
| 12/3/06 | 197.7 | 184.2 | 178.0 | 50 | 404.5 | 417.9 | 424.1 |
| 12/4/06 | 199.5 | 186.1 | 180.0 | 50 | 402.7 | 416.1 | 422.3 |
| 12/5/06 | 201.3 | 188.0 | 181.9 | 50 | 400.8 | 414.2 | 420.3 |
| 12/6/06 | 203.5 | 190.2 | 184.1 | 50 | 398.5 | 411.8 | 417.9 |
| 12/7/06 | 205.3 | 192.2 | 186.1 | 50 | 396.4 | 409.6 | 415.6 |
| 12/8/06 | 208.2 | 195.1 | 189.1 | 50 | 393.3 | 406.4 | 412.4 |
| 12/9/06 | 210.7 | 197.7 | 191.8 | 50 | 390.2 | 403.2 | 409.2 |
| 12/10/06 | 213.8 | 200.9 | 195.0 | 50 | 386.8 | 399.7 | 405.6 |
| 12/11/06 | 216.3 | 203.5 | 197.7 | 50 | 383.9 | 396.7 | 402.5 |
| 12/12/06 | 219.6 | 206.9 | 201.2 | 50 | 380.1 | 392.7 | 398.5 |
| 12/13/06 | 221.9 | 209.3 | 203.6 | 50 | 377.1 | 389.7 | 395.4 |
| 12/14/06 | 224.9 | 212.5 | 206.8 | 50 | 373.2 | 385.6 | 391.3 |
| 12/15/06 | 229.1 | 216.8 | 211.1 | 50 | 370.6 | 382.9 | 388.5 |

Scenario 4 - Post-Capture Orbiter Mass (Delta II 7925H)

| Launch | C3 | V inf at Arr | Inj Mass | TCM Mass | MOI ΔV | Δ Mass | Post Capture |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 1034.0 | 9.8 | 2.026 | 487.1 | 537.0 |
| 9/14/07 | 13.130 | 2.579 | 1035.9 | 9.9 | 2.031 | 488.9 | 537.2 |
| 9/15/07 | 13.040 | 2.594 | 1037.5 | 9.9 | 2.038 | 490.9 | 536.8 |
| 9/16/07 | 12.970 | 2.640 | 1038.7 | 9.9 | 2.060 | 495.2 | 533.7 |
| 9/17/07 | 12.910 | 2.635 | 1039.8 | 9.9 | 2.057 | 495.3 | 534.6 |
| 9/18/07 | 12.860 | 2.662 | 1040.7 | 9.9 | 2.070 | 497.9 | 532.9 |
| 9/19/07 | 12.820 | 2.697 | 1041.4 | 9.9 | 2.087 | 501.1 | 530.4 |
| 9/20/07 | 12.800 | 2.710 | 1041.7 | 9.9 | 2.093 | 502.3 | 529.5 |
| 9/21/07 | 12.790 | 2.758 | 1041.9 | 9.9 | 2.117 | 506.4 | 525.6 |
| 9/22/07 | 12.800 | 2.774 | 1041.7 | 9.9 | 2.125 | 507.6 | 524.2 |
| 9/23/07 | 12.810 | 2.797 | 1041.5 | 9.9 | 2.136 | 509.4 | 522.2 |
| 9/24/07 | 12.830 | 2.858 | 1041.2 | 9.9 | 2.167 | 514.3 | 517.0 |
| 9/25/07 | 12.850 | 2.876 | 1040.8 | 9.9 | 2.176 | 515.6 | 515.3 |
| 9/26/07 | 12.890 | 2.902 | 1040.1 | 9.9 | 2.189 | 517.5 | 512.8 |
| 9/27/07 | 12.920 | 2.957 | 1039.6 | 9.9 | 2.218 | 521.8 | 507.9 |
| 9/28/07 | 12.950 | 3.008 | 1039.1 | 9.9 | 2.244 | 525.9 | 503.3 |
| 9/29/07 | 12.990 | 3.053 | 1038.4 | 9.9 | 2.268 | 529.3 | 499.2 |
| 9/30/07 | 13.040 | 3.143 | 1037.5 | 9.9 | 2.317 | 536.5 | 491.1 |
| 10/1/07 | 13.110 | 3.151 | 1036.3 | 9.9 | 2.321 | 536.5 | 489.9 |
| 10/2/07 | 12.990 | 3.235 | 1038.4 | 9.9 | 2.367 | 544.8 | 483.7 |

Scenario 4 - Post-Capture Orbiter Mass (Delta II 7325 Star 48B)

| Launch | C3 | V inf at Arr | Inj Mass | TCM Mass | MOI ΔV | Δ Mass | Post Capture |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 518.1 | 4.9 | 2.026 | 244.1 | 269.1 |
| 9/14/07 | 13.130 | 2.579 | 519.3 | 4.9 | 2.031 | 245.1 | 269.3 |
| 9/15/07 | 13.040 | 2.594 | 520.3 | 5.0 | 2.038 | 246.2 | 269.2 |
| 9/16/07 | 12.970 | 2.640 | 521.1 | 5.0 | 2.060 | 248.4 | 267.7 |
| 9/17/07 | 12.910 | 2.635 | 521.7 | 5.0 | 2.057 | 248.5 | 268.3 |
| 9/18/07 | 12.860 | 2.662 | 522.3 | 5.0 | 2.070 | 249.9 | 267.4 |
| 9/19/07 | 12.820 | 2.697 | 522.7 | 5.0 | 2.087 | 251.5 | 266.2 |
| 9/20/07 | 12.800 | 2.710 | 523.0 | 5.0 | 2.093 | 252.2 | 265.8 |
| 9/21/07 | 12.790 | 2.758 | 523.1 | 5.0 | 2.117 | 254.2 | 263.9 |
| 9/22/07 | 12.800 | 2.774 | 523.0 | 5.0 | 2.125 | 254.8 | 263.2 |
| 9/23/07 | 12.810 | 2.797 | 522.8 | 5.0 | 2.136 | 255.7 | 262.2 |
| 9/24/07 | 12.830 | 2.858 | 522.6 | 5.0 | 2.167 | 258.2 | 259.5 |
| 9/25/07 | 12.850 | 2.876 | 522.4 | 5.0 | 2.176 | 258.8 | 258.6 |
| 9/26/07 | 12.890 | 2.902 | 522.0 | 5.0 | 2.189 | 259.7 | 257.3 |
| 9/27/07 | 12.920 | 2.957 | 521.6 | 5.0 | 2.218 | 261.8 | 254.8 |
| 9/28/07 | 12.950 | 3.008 | 521.3 | 5.0 | 2.244 | 263.8 | 252.5 |
| 9/29/07 | 12.990 | 3.053 | 520.9 | 5.0 | 2.268 | 265.5 | 250.4 |
| 9/30/07 | 13.040 | 3.143 | 520.3 | 5.0 | 2.317 | 269.0 | 246.3 |
| 10/1/07 | 13.110 | 3.151 | 519.5 | 4.9 | 2.321 | 269.0 | 245.6 |
| 10/2/07 | 12.990 | 3.235 | 520.9 | 5.0 | 2.367 | 273.3 | 242.6 |

## Scenario 4 - Post-Capture Orbiter Mass (Delta II 7425 Star 48B)

| Launch | C3 | V inf at Arr | Inj Mass | TCM Mass | MOI ΔV | Δ Mass | Post Capture |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 602.2 | 5.7 | 2.026 | 283.7 | 312.8 |
| 9/14/07 | 13.130 | 2.579 | 603.6 | 5.7 | 2.031 | 284.9 | 313.0 |
| 9/15/07 | 13.040 | 2.594 | 604.8 | 5.8 | 2.038 | 286.1 | 312.9 |
| 9/16/07 | 12.970 | 2.640 | 605.7 | 5.8 | 2.060 | 288.7 | 311.2 |
| 9/17/07 | 12.910 | 2.635 | 606.5 | 5.8 | 2.057 | 288.9 | 311.8 |
| 9/18/07 | 12.860 | 2.662 | 607.1 | 5.8 | 2.070 | 290.5 | 310.9 |
| 9/19/07 | 12.820 | 2.697 | 607.7 | 5.8 | 2.087 | 292.4 | 309.5 |
| 9/20/07 | 12.800 | 2.710 | 607.9 | 5.8 | 2.093 | 293.1 | 309.0 |
| 9/21/07 | 12.790 | 2.758 | 608.0 | 5.8 | 2.117 | 295.5 | 306.8 |
| 9/22/07 | 12.800 | 2.774 | 607.9 | 5.8 | 2.125 | 296.2 | 305.9 |
| 9/23/07 | 12.810 | 2.797 | 607.8 | 5.8 | 2.136 | 297.3 | 304.7 |
| 9/24/07 | 12.830 | 2.858 | 607.5 | 5.8 | 2.167 | 300.1 | 301.6 |
| 9/25/07 | 12.850 | 2.876 | 607.3 | 5.8 | 2.176 | 300.8 | 300.6 |
| 9/26/07 | 12.890 | 2.902 | 606.7 | 5.8 | 2.189 | 301.9 | 299.1 |
| 9/27/07 | 12.920 | 2.957 | 606.4 | 5.8 | 2.218 | 304.4 | 296.2 |
| 9/28/07 | 12.950 | 3.008 | 606.0 | 5.8 | 2.244 | 306.7 | 293.5 |
| 9/29/07 | 12.990 | 3.053 | 605.4 | 5.8 | 2.268 | 308.6 | 291.1 |
| 9/30/07 | 13.040 | 3.143 | 604.8 | 5.8 | 2.317 | 312.7 | 286.3 |
| 10/1/07 | 13.110 | 3.151 | 603.9 | 5.7 | 2.321 | 312.7 | 285.5 |
| 10/2/07 | 12.990 | 3.235 | 605.4 | 5.8 | 2.367 | 317.6 | 282.0 |

## Scenario 5 - Post-Capture Orbiter Mass (Delta II 7925H)

| | | V inf at Arr | Inj Mass | TCMs (30 m/s) | Cruise Stage | TPS Mass | Post AC Mass | Post AC Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|---|
| Launch | C3 | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 9/13/07 | 13.240 | 2.567 | 1034.0 | 9.8 | 75 | 196.7 | 762.3 | 30.9 | 731.4 |
| 9/14/07 | 13.130 | 2.578 | 1035.9 | 9.9 | 75 | 196.8 | 764.1 | 31.0 | 733.1 |
| 9/15/07 | 13.040 | 2.594 | 1037.5 | 9.9 | 75 | 196.9 | 765.6 | 31.1 | 734.5 |
| 9/16/07 | 12.960 | 2.637 | 1038.7 | 9.9 | 75 | 197.3 | 766.4 | 31.1 | 735.3 |
| 9/17/07 | 12.910 | 2.634 | 1039.8 | 9.9 | 75 | 197.3 | 767.5 | 31.1 | 736.3 |
| 9/18/07 | 12.860 | 2.662 | 1040.7 | 9.9 | 75 | 197.6 | 768.1 | 31.2 | 736.9 |
| 9/19/07 | 12.820 | 2.696 | 1041.4 | 9.9 | 75 | 197.9 | 768.5 | 31.2 | 737.3 |
| 9/20/07 | 12.800 | 2.709 | 1041.7 | 9.9 | 75 | 198.1 | 768.7 | 31.2 | 737.5 |
| 9/21/07 | 12.790 | 2.757 | 1041.9 | 9.9 | 75 | 198.6 | 768.3 | 31.2 | 737.1 |
| 9/22/07 | 12.800 | 2.774 | 1041.7 | 9.9 | 75 | 198.8 | 767.9 | 31.2 | 736.8 |
| 9/23/07 | 12.810 | 2.796 | 1041.5 | 9.9 | 75 | 199.1 | 767.5 | 31.1 | 736.3 |
| 9/24/07 | 12.830 | 2.858 | 1041.2 | 9.9 | 75 | 199.8 | 766.4 | 31.1 | 735.3 |
| 9/25/07 | 12.850 | 2.876 | 1040.8 | 9.9 | 75 | 200.1 | 765.8 | 31.1 | 734.7 |
| 9/26/07 | 12.890 | 2.901 | 1040.1 | 9.9 | 75 | 200.4 | 764.7 | 31.0 | 733.7 |
| 9/27/07 | 12.920 | 2.956 | 1039.6 | 9.9 | 75 | 201.2 | 763.4 | 31.0 | 732.4 |
| 9/28/07 | 12.950 | 3.008 | 1039.1 | 9.9 | 75 | 202.0 | 762.1 | 30.9 | 731.2 |
| 9/29/07 | 12.990 | 3.053 | 1038.4 | 9.9 | 75 | 202.7 | 760.7 | 30.9 | 729.8 |
| 9/30/07 | 13.040 | 3.142 | 1037.5 | 9.9 | 75 | 204.2 | 758.3 | 30.8 | 727.5 |
| 10/1/07 | 13.110 | 3.150 | 1036.3 | 9.9 | 75 | 204.4 | 756.9 | 30.7 | 726.2 |
| 10/2/07 | 12.990 | 3.235 | 1038.4 | 9.9 | 75 | 205.9 | 757.5 | 30.7 | 726.7 |

## Scenario 5 - Post-Capture Orbiter Mass (Delta II 7325 Star 48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCMs (30 m/s) (kg) | Cruise Stage (kg) | TPS Mass (kg) | Post AC Mass (kg) | Post AC Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 518.1 | 4.9 | 75 | 196.7 | 246.4 | 10.0 | 236.4 |
| 9/14/07 | 13.130 | 2.578 | 519.3 | 4.9 | 75 | 196.8 | 247.5 | 10.0 | 237.5 |
| 9/15/07 | 13.040 | 2.594 | 520.3 | 5.0 | 75 | 196.9 | 248.4 | 10.1 | 238.3 |
| 9/16/07 | 12.960 | 2.637 | 521.1 | 5.0 | 75 | 197.3 | 248.7 | 10.1 | 238.7 |
| 9/17/07 | 12.910 | 2.634 | 521.7 | 5.0 | 75 | 197.3 | 249.4 | 10.1 | 239.3 |
| 9/18/07 | 12.860 | 2.662 | 522.3 | 5.0 | 75 | 197.6 | 249.7 | 10.1 | 239.6 |
| 9/19/07 | 12.820 | 2.696 | 522.7 | 5.0 | 75 | 197.9 | 249.8 | 10.1 | 239.7 |
| 9/20/07 | 12.800 | 2.709 | 523.0 | 5.0 | 75 | 198.1 | 249.9 | 10.1 | 239.8 |
| 9/21/07 | 12.790 | 2.757 | 523.1 | 5.0 | 75 | 198.6 | 249.5 | 10.1 | 239.3 |
| 9/22/07 | 12.800 | 2.774 | 523.0 | 5.0 | 75 | 198.8 | 249.2 | 10.1 | 239.1 |
| 9/23/07 | 12.810 | 2.796 | 522.8 | 5.0 | 75 | 199.1 | 248.8 | 10.1 | 238.7 |
| 9/24/07 | 12.830 | 2.858 | 522.6 | 5.0 | 75 | 199.8 | 247.8 | 10.1 | 237.7 |
| 9/25/07 | 12.850 | 2.876 | 522.4 | 5.0 | 75 | 200.1 | 247.3 | 10.0 | 237.3 |
| 9/26/07 | 12.890 | 2.901 | 522.0 | 5.0 | 75 | 200.4 | 246.5 | 10.0 | 236.5 |
| 9/27/07 | 12.920 | 2.956 | 521.6 | 5.0 | 75 | 201.2 | 245.4 | 10.0 | 235.5 |
| 9/28/07 | 12.950 | 3.008 | 521.3 | 5.0 | 75 | 202.0 | 244.3 | 9.9 | 234.4 |
| 9/29/07 | 12.990 | 3.053 | 520.9 | 5.0 | 75 | 202.7 | 243.1 | 9.9 | 233.3 |
| 9/30/07 | 13.040 | 3.142 | 520.3 | 5.0 | 75 | 204.2 | 241.1 | 9.8 | 231.3 |
| 10/1/07 | 13.110 | 3.150 | 519.5 | 4.9 | 75 | 204.4 | 240.2 | 9.7 | 230.4 |
| 10/2/07 | 12.990 | 3.235 | 520.9 | 5.0 | 75 | 205.9 | 239.9 | 9.7 | 230.2 |

## Scenario 5 - Post-Capture Orbiter Mass (Delta II 7425 Star 48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCMs (30 m/s) (kg) | Cruise Stage (kg) | TPS Mass (kg) | Post AC Mass (kg) | Post AC Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 602.2 | 5.7 | 75 | 196.7 | 330.5 | 13.4 | 317.1 |
| 9/14/07 | 13.130 | 2.578 | 603.6 | 5.7 | 75 | 196.8 | 331.8 | 13.5 | 318.4 |
| 9/15/07 | 13.040 | 2.594 | 604.8 | 5.8 | 75 | 196.9 | 332.9 | 13.5 | 319.4 |
| 9/16/07 | 12.960 | 2.637 | 605.7 | 5.8 | 75 | 197.3 | 333.4 | 13.5 | 319.8 |
| 9/17/07 | 12.910 | 2.634 | 606.5 | 5.8 | 75 | 197.3 | 334.2 | 13.6 | 320.6 |
| 9/18/07 | 12.860 | 2.662 | 607.1 | 5.8 | 75 | 197.6 | 334.6 | 13.6 | 321.0 |
| 9/19/07 | 12.820 | 2.696 | 607.7 | 5.8 | 75 | 197.9 | 334.7 | 13.6 | 321.2 |
| 9/20/07 | 12.800 | 2.709 | 607.9 | 5.8 | 75 | 198.1 | 334.9 | 13.6 | 321.3 |
| 9/21/07 | 12.790 | 2.757 | 608.0 | 5.8 | 75 | 198.6 | 334.5 | 13.6 | 320.9 |
| 9/22/07 | 12.800 | 2.774 | 607.9 | 5.8 | 75 | 198.8 | 334.1 | 13.6 | 320.6 |
| 9/23/07 | 12.810 | 2.796 | 607.8 | 5.8 | 75 | 199.1 | 333.7 | 13.5 | 320.2 |
| 9/24/07 | 12.830 | 2.858 | 607.5 | 5.8 | 75 | 199.8 | 332.7 | 13.5 | 319.2 |
| 9/25/07 | 12.850 | 2.876 | 607.3 | 5.8 | 75 | 200.1 | 332.2 | 13.5 | 318.7 |
| 9/26/07 | 12.890 | 2.901 | 606.7 | 5.8 | 75 | 200.4 | 331.3 | 13.4 | 317.9 |
| 9/27/07 | 12.920 | 2.956 | 606.4 | 5.8 | 75 | 201.2 | 330.1 | 13.4 | 316.7 |
| 9/28/07 | 12.950 | 3.008 | 606.0 | 5.8 | 75 | 202.0 | 329.0 | 13.4 | 315.6 |
| 9/29/07 | 12.990 | 3.053 | 605.4 | 5.8 | 75 | 202.7 | 327.7 | 13.3 | 314.4 |
| 9/30/07 | 13.040 | 3.142 | 604.8 | 5.8 | 75 | 204.2 | 325.6 | 13.2 | 312.4 |
| 10/1/07 | 13.110 | 3.150 | 603.9 | 5.7 | 75 | 204.4 | 324.5 | 13.2 | 311.4 |
| 10/2/07 | 12.990 | 3.235 | 605.4 | 5.8 | 75 | 205.9 | 324.5 | 13.2 | 311.4 |

# Scenario 6 - Post-Capture Orbiter Mass (Delta II 7925H)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 1034.0 | 9.8 | 0.947 | 0.844 | 0.798 |
| 9/14/07 | 13.130 | 2.579 | 1035.9 | 9.9 | 0.952 | 0.849 | 0.803 |
| 9/15/07 | 13.040 | 2.594 | 1037.5 | 9.9 | 0.959 | 0.856 | 0.810 |
| 9/16/07 | 12.970 | 2.640 | 1038.7 | 9.9 | 0.981 | 0.878 | 0.832 |
| 9/17/07 | 12.910 | 2.635 | 1039.8 | 9.9 | 0.978 | 0.876 | 0.830 |
| 9/18/07 | 12.860 | 2.662 | 1040.7 | 9.9 | 0.991 | 0.888 | 0.843 |
| 9/19/07 | 12.820 | 2.697 | 1041.4 | 9.9 | 1.008 | 0.905 | 0.860 |
| 9/20/07 | 12.800 | 2.710 | 1041.7 | 9.9 | 1.014 | 0.912 | 0.866 |
| 9/21/07 | 12.790 | 2.758 | 1041.9 | 9.9 | 1.038 | 0.935 | 0.889 |
| 9/22/07 | 12.800 | 2.774 | 1041.7 | 9.9 | 1.046 | 0.943 | 0.897 |
| 9/23/07 | 12.810 | 2.797 | 1041.5 | 9.9 | 1.057 | 0.955 | 0.909 |
| 9/24/07 | 12.830 | 2.858 | 1041.2 | 9.9 | 1.088 | 0.985 | 0.939 |
| 9/25/07 | 12.850 | 2.876 | 1040.8 | 9.9 | 1.097 | 0.994 | 0.949 |
| 9/26/07 | 12.890 | 2.902 | 1040.1 | 9.9 | 1.110 | 1.008 | 0.962 |
| 9/27/07 | 12.920 | 2.957 | 1039.6 | 9.9 | 1.139 | 1.036 | 0.990 |
| 9/28/07 | 12.950 | 3.008 | 1039.1 | 9.9 | 1.165 | 1.063 | 1.017 |
| 9/29/07 | 12.990 | 3.053 | 1038.4 | 9.9 | 1.189 | 1.087 | 1.041 |
| 9/30/07 | 13.040 | 3.143 | 1037.5 | 9.9 | 1.238 | 1.135 | 1.089 |
| 10/1/07 | 13.110 | 3.151 | 1036.3 | 9.9 | 1.242 | 1.139 | 1.093 |
| 10/2/07 | 12.990 | 3.235 | 1038.4 | 9.9 | 1.288 | 1.185 | 1.140 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 266.7 | 241.5 | 230.0 | 50 | 757.4 | 782.6 | 794.2 |
| 9/14/07 | 268.5 | 243.2 | 231.7 | 50 | 757.6 | 782.8 | 794.3 |
| 9/15/07 | 270.6 | 245.4 | 233.9 | 50 | 757.0 | 782.3 | 793.8 |
| 9/16/07 | 276.1 | 251.1 | 239.6 | 50 | 752.7 | 777.8 | 789.2 |
| 9/17/07 | 275.9 | 250.8 | 239.3 | 50 | 754.0 | 779.1 | 790.6 |
| 9/18/07 | 279.2 | 254.2 | 242.7 | 50 | 751.6 | 776.6 | 788.0 |
| 9/19/07 | 283.4 | 258.5 | 247.1 | 50 | 748.0 | 772.9 | 784.3 |
| 9/20/07 | 285.0 | 260.2 | 248.8 | 50 | 746.8 | 771.7 | 783.0 |
| 9/21/07 | 290.6 | 266.0 | 254.7 | 50 | 741.3 | 766.0 | 777.3 |
| 9/22/07 | 292.5 | 267.9 | 256.6 | 50 | 739.3 | 764.0 | 775.2 |
| 9/23/07 | 295.1 | 270.6 | 259.4 | 50 | 736.5 | 761.1 | 772.3 |
| 9/24/07 | 302.2 | 277.9 | 266.8 | 50 | 729.1 | 753.4 | 764.5 |
| 9/25/07 | 304.2 | 280.0 | 268.9 | 50 | 726.8 | 750.9 | 762.0 |
| 9/26/07 | 307.0 | 283.0 | 272.0 | 50 | 723.2 | 747.3 | 758.3 |
| 9/27/07 | 313.4 | 289.6 | 278.7 | 50 | 716.3 | 740.2 | 751.1 |
| 9/28/07 | 319.3 | 295.7 | 284.9 | 50 | 709.9 | 733.5 | 744.3 |
| 9/29/07 | 324.5 | 301.0 | 290.3 | 50 | 704.0 | 727.5 | 738.2 |
| 9/30/07 | 334.9 | 311.9 | 301.3 | 50 | 692.7 | 715.8 | 726.3 |
| 10/1/07 | 335.5 | 312.5 | 302.0 | 50 | 690.9 | 713.9 | 724.4 |
| 10/2/07 | 346.3 | 323.6 | 313.2 | 50 | 682.2 | 704.9 | 715.3 |

## Scenario 6 - Post-Capture Orbiter Mass (Delta II 7325 Star48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 518.1 | 4.9 | 0.947 | 0.844 | 0.798 |
| 9/14/07 | 13.130 | 2.579 | 519.3 | 4.9 | 0.952 | 0.849 | 0.803 |
| 9/15/07 | 13.040 | 2.594 | 520.3 | 5.0 | 0.959 | 0.856 | 0.810 |
| 9/16/07 | 12.970 | 2.640 | 521.1 | 5.0 | 0.981 | 0.878 | 0.832 |
| 9/17/07 | 12.910 | 2.635 | 521.7 | 5.0 | 0.978 | 0.876 | 0.830 |
| 9/18/07 | 12.860 | 2.662 | 522.3 | 5.0 | 0.991 | 0.888 | 0.843 |
| 9/19/07 | 12.820 | 2.697 | 522.7 | 5.0 | 1.008 | 0.905 | 0.860 |
| 9/20/07 | 12.800 | 2.710 | 523.0 | 5.0 | 1.014 | 0.912 | 0.866 |
| 9/21/07 | 12.790 | 2.758 | 523.1 | 5.0 | 1.038 | 0.935 | 0.889 |
| 9/22/07 | 12.800 | 2.774 | 523.0 | 5.0 | 1.046 | 0.943 | 0.897 |
| 9/23/07 | 12.810 | 2.797 | 522.8 | 5.0 | 1.057 | 0.955 | 0.909 |
| 9/24/07 | 12.830 | 2.858 | 522.6 | 5.0 | 1.088 | 0.985 | 0.939 |
| 9/25/07 | 12.850 | 2.876 | 522.4 | 5.0 | 1.097 | 0.994 | 0.949 |
| 9/26/07 | 12.890 | 2.902 | 522.0 | 5.0 | 1.110 | 1.008 | 0.962 |
| 9/27/07 | 12.920 | 2.957 | 521.6 | 5.0 | 1.139 | 1.036 | 0.990 |
| 9/28/07 | 12.950 | 3.008 | 521.3 | 5.0 | 1.165 | 1.063 | 1.017 |
| 9/29/07 | 12.990 | 3.053 | 520.9 | 5.0 | 1.189 | 1.087 | 1.041 |
| 9/30/07 | 13.040 | 3.143 | 520.3 | 5.0 | 1.238 | 1.135 | 1.089 |
| 10/1/07 | 13.110 | 3.151 | 519.5 | 4.9 | 1.242 | 1.139 | 1.093 |
| 10/2/07 | 12.990 | 3.235 | 520.9 | 5.0 | 1.288 | 1.185 | 1.140 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 133.6 | 121.0 | 115.2 | 50 | 379.5 | 392.2 | 397.9 |
| 9/14/07 | 134.6 | 121.9 | 116.2 | 50 | 379.8 | 392.4 | 398.2 |
| 9/15/07 | 135.7 | 123.1 | 117.3 | 50 | 379.7 | 392.3 | 398.1 |
| 9/16/07 | 138.5 | 126.0 | 120.2 | 50 | 377.6 | 390.2 | 395.9 |
| 9/17/07 | 138.4 | 125.8 | 120.1 | 50 | 378.4 | 391.0 | 396.7 |
| 9/18/07 | 140.1 | 127.6 | 121.8 | 50 | 377.2 | 389.8 | 395.5 |
| 9/19/07 | 142.3 | 129.8 | 124.1 | 50 | 375.5 | 388.0 | 393.7 |
| 9/20/07 | 143.1 | 130.6 | 124.9 | 50 | 374.9 | 387.4 | 393.1 |
| 9/21/07 | 145.9 | 133.5 | 127.9 | 50 | 372.2 | 384.6 | 390.2 |
| 9/22/07 | 146.8 | 134.5 | 128.8 | 50 | 371.2 | 383.5 | 389.2 |
| 9/23/07 | 148.1 | 135.8 | 130.2 | 50 | 369.7 | 382.0 | 387.7 |
| 9/24/07 | 151.7 | 139.5 | 133.9 | 50 | 366.0 | 378.2 | 383.7 |
| 9/25/07 | 152.7 | 140.5 | 135.0 | 50 | 364.8 | 376.9 | 382.4 |
| 9/26/07 | 154.1 | 142.0 | 136.5 | 50 | 362.9 | 375.0 | 380.5 |
| 9/27/07 | 157.3 | 145.3 | 139.8 | 50 | 359.4 | 371.4 | 376.8 |
| 9/28/07 | 160.2 | 148.3 | 142.9 | 50 | 356.1 | 368.0 | 373.4 |
| 9/29/07 | 162.7 | 151.0 | 145.6 | 50 | 353.2 | 364.9 | 370.3 |
| 9/30/07 | 168.0 | 156.4 | 151.1 | 50 | 347.4 | 358.9 | 364.2 |
| 10/1/07 | 168.2 | 156.7 | 151.4 | 50 | 346.4 | 357.9 | 363.2 |
| 10/2/07 | 173.7 | 162.3 | 157.1 | 50 | 342.2 | 353.6 | 358.8 |

## Scenario 6 - Post-Capture Orbiter Mass (Delta II 7425 Star48B)

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | TCM Mass (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) | 39 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.568 | 602.2 | 5.7 | 0.947 | 0.844 | 0.798 |
| 9/14/07 | 13.130 | 2.579 | 603.6 | 5.7 | 0.952 | 0.849 | 0.803 |
| 9/15/07 | 13.040 | 2.594 | 604.8 | 5.8 | 0.959 | 0.856 | 0.810 |
| 9/16/07 | 12.970 | 2.640 | 605.7 | 5.8 | 0.981 | 0.878 | 0.832 |
| 9/17/07 | 12.910 | 2.635 | 606.5 | 5.8 | 0.978 | 0.876 | 0.830 |
| 9/18/07 | 12.860 | 2.662 | 607.1 | 5.8 | 0.991 | 0.888 | 0.843 |
| 9/19/07 | 12.820 | 2.697 | 607.7 | 5.8 | 1.008 | 0.905 | 0.860 |
| 9/20/07 | 12.800 | 2.710 | 607.9 | 5.8 | 1.014 | 0.912 | 0.866 |
| 9/21/07 | 12.790 | 2.758 | 608.0 | 5.8 | 1.038 | 0.935 | 0.889 |
| 9/22/07 | 12.800 | 2.774 | 607.9 | 5.8 | 1.046 | 0.943 | 0.897 |
| 9/23/07 | 12.810 | 2.797 | 607.8 | 5.8 | 1.057 | 0.955 | 0.909 |
| 9/24/07 | 12.830 | 2.858 | 607.5 | 5.8 | 1.088 | 0.985 | 0.939 |
| 9/25/07 | 12.850 | 2.876 | 607.3 | 5.8 | 1.097 | 0.994 | 0.949 |
| 9/26/07 | 12.890 | 2.902 | 606.7 | 5.8 | 1.110 | 1.008 | 0.962 |
| 9/27/07 | 12.920 | 2.957 | 606.4 | 5.8 | 1.139 | 1.036 | 0.990 |
| 9/28/07 | 12.950 | 3.008 | 606.0 | 5.8 | 1.165 | 1.063 | 1.017 |
| 9/29/07 | 12.990 | 3.053 | 605.4 | 5.8 | 1.189 | 1.087 | 1.041 |
| 9/30/07 | 13.040 | 3.143 | 604.8 | 5.8 | 1.238 | 1.135 | 1.089 |
| 10/1/07 | 13.110 | 3.151 | 603.9 | 5.7 | 1.242 | 1.139 | 1.093 |
| 10/2/07 | 12.990 | 3.235 | 605.4 | 5.8 | 1.288 | 1.185 | 1.140 |

| Launch | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop Mass (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|
| 9/13/07 | 155.3 | 140.7 | 133.9 | 50 | 441.1 | 455.8 | 462.5 |
| 9/14/07 | 156.4 | 141.7 | 135.0 | 50 | 441.4 | 456.1 | 462.9 |
| 9/15/07 | 157.7 | 143.0 | 136.3 | 50 | 441.3 | 456.0 | 462.7 |
| 9/16/07 | 161.0 | 146.4 | 139.7 | 50 | 438.9 | 453.5 | 460.2 |
| 9/17/07 | 160.9 | 146.3 | 139.6 | 50 | 439.8 | 454.5 | 461.1 |
| 9/18/07 | 162.9 | 148.3 | 141.6 | 50 | 438.5 | 453.1 | 459.7 |
| 9/19/07 | 165.4 | 150.8 | 144.2 | 50 | 436.5 | 451.0 | 457.7 |
| 9/20/07 | 166.3 | 151.8 | 145.2 | 50 | 435.8 | 450.3 | 456.9 |
| 9/21/07 | 169.6 | 155.2 | 148.6 | 50 | 432.6 | 447.0 | 453.6 |
| 9/22/07 | 170.7 | 156.3 | 149.8 | 50 | 431.5 | 445.8 | 452.4 |
| 9/23/07 | 172.2 | 157.9 | 151.4 | 50 | 429.8 | 444.1 | 450.6 |
| 9/24/07 | 176.3 | 162.1 | 155.7 | 50 | 425.4 | 439.6 | 446.1 |
| 9/25/07 | 177.5 | 163.4 | 156.9 | 50 | 424.0 | 438.1 | 444.6 |
| 9/26/07 | 179.1 | 165.1 | 158.7 | 50 | 421.9 | 435.9 | 442.3 |
| 9/27/07 | 182.8 | 168.9 | 162.5 | 50 | 417.8 | 431.7 | 438.1 |
| 9/28/07 | 186.2 | 172.4 | 166.1 | 50 | 414.0 | 427.8 | 434.1 |
| 9/29/07 | 189.2 | 175.5 | 169.3 | 50 | 410.5 | 424.2 | 430.4 |
| 9/30/07 | 195.2 | 181.8 | 175.7 | 50 | 403.8 | 417.2 | 423.4 |
| 10/1/07 | 195.5 | 182.1 | 176.0 | 50 | 402.6 | 416.0 | 422.1 |
| 10/2/07 | 201.9 | 188.7 | 182.6 | 50 | 397.8 | 411.0 | 417.1 |

## Scenario 7 - Post-Capture Orbiter Mass (Atlas IIAS) -Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1981.3 | 75.0 | 9.1 | 9.0 | 2.286 | 483.9 | 451.2 |
| 11/27/06 | 9.656 | 3.090 | 1985.3 | 75.0 | 9.1 | 9.0 | 2.288 | 485.1 | 452.0 |
| 11/28/06 | 9.572 | 3.097 | 1988.8 | 75.0 | 9.1 | 9.0 | 2.292 | 486.5 | 452.3 |
| 11/29/06 | 9.497 | 3.104 | 1991.9 | 75.0 | 9.1 | 9.0 | 2.296 | 487.9 | 452.4 |
| 11/30/06 | 9.433 | 3.116 | 1994.5 | 75.0 | 9.1 | 9.0 | 2.302 | 489.5 | 452.1 |
| 12/1/06 | 9.378 | 3.130 | 1996.7 | 75.0 | 9.1 | 9.1 | 2.310 | 491.1 | 451.6 |
| 12/2/06 | 9.327 | 3.143 | 1998.8 | 75.0 | 9.2 | 9.1 | 2.317 | 492.7 | 451.0 |
| 12/3/06 | 9.284 | 3.163 | 2000.6 | 75.0 | 9.2 | 9.1 | 2.328 | 494.7 | 449.9 |
| 12/4/06 | 9.245 | 3.189 | 2002.2 | 75.0 | 9.2 | 9.1 | 2.342 | 497.1 | 448.2 |
| 12/5/06 | 9.207 | 3.215 | 2003.7 | 75.0 | 9.2 | 9.1 | 2.356 | 499.6 | 446.5 |
| 12/6/06 | 9.171 | 3.246 | 2005.2 | 75.0 | 9.2 | 9.1 | 2.373 | 502.4 | 444.4 |
| 12/7/06 | 9.135 | 3.274 | 2006.8 | 75.0 | 9.2 | 9.1 | 2.389 | 505.0 | 442.6 |
| 12/8/06 | 9.102 | 3.315 | 2008.1 | 75.0 | 9.2 | 9.1 | 2.412 | 508.6 | 439.7 |
| 12/9/06 | 9.068 | 3.354 | 2009.5 | 75.0 | 9.2 | 9.1 | 2.434 | 512.0 | 436.9 |
| 12/10/06 | 9.035 | 3.399 | 2010.8 | 75.0 | 9.2 | 9.1 | 2.460 | 515.9 | 433.6 |
| 12/11/06 | 8.997 | 3.437 | 2012.2 | 75.0 | 9.2 | 9.1 | 2.481 | 519.3 | 430.9 |
| 12/12/06 | 8.960 | 3.486 | 2013.9 | 75.0 | 9.2 | 9.1 | 2.510 | 523.7 | 427.4 |
| 12/13/06 | 8.920 | 3.522 | 2015.6 | 75.0 | 9.2 | 9.1 | 2.531 | 527.0 | 424.9 |
| 12/14/06 | 8.889 | 3.570 | 2016.8 | 75.0 | 9.2 | 9.2 | 2.559 | 531.1 | 421.4 |
| 12/15/06 | 8.865 | 3.621 | 2017.4 | 75.0 | 9.2 | 9.2 | 2.589 | 535.4 | 417.5 |

## Scenario 7 - Post-Capture Orbiter Mass (Atlas IIAR) - Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 2142.0 | 75.0 | 9.8 | 9.7 | 2.286 | 524.6 | 489.3 |
| 11/27/06 | 9.656 | 3.090 | 2146.1 | 75.0 | 9.9 | 9.8 | 2.288 | 525.9 | 490.0 |
| 11/28/06 | 9.572 | 3.097 | 2149.7 | 75.0 | 9.9 | 9.8 | 2.292 | 527.4 | 490.3 |
| 11/29/06 | 9.497 | 3.104 | 2152.9 | 75.0 | 9.9 | 9.8 | 2.296 | 528.8 | 490.4 |
| 11/30/06 | 9.433 | 3.116 | 2155.6 | 75.0 | 9.9 | 9.8 | 2.302 | 530.5 | 490.1 |
| 12/1/06 | 9.378 | 3.130 | 2157.9 | 75.0 | 9.9 | 9.8 | 2.310 | 532.3 | 489.4 |
| 12/2/06 | 9.327 | 3.143 | 2160.1 | 75.0 | 9.9 | 9.8 | 2.317 | 534.0 | 488.8 |
| 12/3/06 | 9.284 | 3.163 | 2161.9 | 75.0 | 9.9 | 9.8 | 2.328 | 536.1 | 487.6 |
| 12/4/06 | 9.245 | 3.189 | 2163.5 | 75.0 | 9.9 | 9.8 | 2.342 | 538.7 | 485.7 |
| 12/5/06 | 9.207 | 3.215 | 2165.1 | 75.0 | 9.9 | 9.8 | 2.356 | 541.4 | 483.9 |
| 12/6/06 | 9.171 | 3.246 | 2166.6 | 75.0 | 10.0 | 9.9 | 2.373 | 544.4 | 481.6 |
| 12/7/06 | 9.135 | 3.274 | 2168.2 | 75.0 | 10.0 | 9.9 | 2.389 | 547.2 | 479.6 |
| 12/8/06 | 9.102 | 3.315 | 2169.6 | 75.0 | 10.0 | 9.9 | 2.412 | 551.1 | 476.4 |
| 12/9/06 | 9.068 | 3.354 | 2171.0 | 75.0 | 10.0 | 9.9 | 2.434 | 554.8 | 473.4 |
| 12/10/06 | 9.035 | 3.399 | 2172.4 | 75.0 | 10.0 | 9.9 | 2.460 | 559.0 | 469.8 |
| 12/11/06 | 8.997 | 3.437 | 2173.8 | 75.0 | 10.0 | 9.9 | 2.481 | 562.7 | 466.9 |
| 12/12/06 | 8.960 | 3.486 | 2175.6 | 75.0 | 10.0 | 9.9 | 2.510 | 567.4 | 463.1 |
| 12/13/06 | 8.920 | 3.522 | 2177.4 | 75.0 | 10.0 | 9.9 | 2.531 | 570.9 | 460.4 |
| 12/14/06 | 8.889 | 3.570 | 2178.6 | 75.0 | 10.0 | 9.9 | 2.559 | 575.4 | 456.5 |
| 12/15/06 | 8.865 | 3.621 | 2179.2 | 75.0 | 10.0 | 9.9 | 2.589 | 580.0 | 452.2 |

## Scenario 7 - Post-Capture Orbiter Mass (Atlas IIARS) - Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 2292.1 | 75.0 | 10.5 | 10.4 | 2.286 | 562.7 | 524.8 |
| 11/27/06 | 9.656 | 3.090 | 2296.4 | 75.0 | 10.6 | 10.5 | 2.288 | 564.1 | 525.6 |
| 11/28/06 | 9.572 | 3.097 | 2300.1 | 75.0 | 10.6 | 10.5 | 2.292 | 565.7 | 525.8 |
| 11/29/06 | 9.497 | 3.104 | 2303.4 | 75.0 | 10.6 | 10.5 | 2.296 | 567.1 | 526.0 |
| 11/30/06 | 9.433 | 3.116 | 2306.2 | 75.0 | 10.6 | 10.5 | 2.302 | 568.9 | 525.5 |
| 12/1/06 | 9.378 | 3.130 | 2308.6 | 75.0 | 10.6 | 10.5 | 2.310 | 570.8 | 524.8 |
| 12/2/06 | 9.327 | 3.143 | 2310.8 | 75.0 | 10.6 | 10.5 | 2.317 | 572.6 | 524.2 |
| 12/3/06 | 9.284 | 3.163 | 2312.7 | 75.0 | 10.6 | 10.5 | 2.328 | 574.9 | 522.8 |
| 12/4/06 | 9.245 | 3.189 | 2314.4 | 75.0 | 10.7 | 10.6 | 2.342 | 577.7 | 520.8 |
| 12/5/06 | 9.207 | 3.215 | 2316.1 | 75.0 | 10.7 | 10.6 | 2.356 | 580.5 | 518.9 |
| 12/6/06 | 9.171 | 3.246 | 2317.7 | 75.0 | 10.7 | 10.6 | 2.373 | 583.7 | 516.4 |
| 12/7/06 | 9.135 | 3.274 | 2319.3 | 75.0 | 10.7 | 10.6 | 2.389 | 586.7 | 514.2 |
| 12/8/06 | 9.102 | 3.315 | 2320.8 | 75.0 | 10.7 | 10.6 | 2.412 | 590.8 | 510.8 |
| 12/9/06 | 9.068 | 3.354 | 2322.2 | 75.0 | 10.7 | 10.6 | 2.434 | 594.8 | 507.5 |
| 12/10/06 | 9.035 | 3.399 | 2323.7 | 75.0 | 10.7 | 10.6 | 2.460 | 599.3 | 503.7 |
| 12/11/06 | 8.997 | 3.437 | 2325.2 | 75.0 | 10.7 | 10.6 | 2.481 | 603.2 | 500.6 |
| 12/12/06 | 8.960 | 3.486 | 2327.0 | 75.0 | 10.7 | 10.6 | 2.510 | 608.2 | 496.5 |
| 12/13/06 | 8.920 | 3.522 | 2328.8 | 75.0 | 10.7 | 10.6 | 2.531 | 612.0 | 493.5 |
| 12/14/06 | 8.889 | 3.570 | 2330.1 | 75.0 | 10.7 | 10.6 | 2.559 | 616.8 | 489.4 |
| 12/15/06 | 8.865 | 3.621 | 2330.8 | 75.0 | 10.7 | 10.6 | 2.589 | 621.7 | 484.8 |

## Scenario 7 - Post-Capture Orbiter Mass (Delta III 3-stage) - Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 2061.4 | 75.0 | 9.5 | 9.4 | 2.286 | 504.2 | 470.2 |
| 11/27/06 | 9.656 | 3.090 | 2064.9 | 75.0 | 9.5 | 9.4 | 2.288 | 505.3 | 470.8 |
| 11/28/06 | 9.572 | 3.097 | 2067.9 | 75.0 | 9.5 | 9.4 | 2.292 | 506.6 | 470.9 |
| 11/29/06 | 9.497 | 3.104 | 2070.6 | 75.0 | 9.5 | 9.4 | 2.296 | 507.9 | 471.0 |
| 11/30/06 | 9.433 | 3.116 | 2072.9 | 75.0 | 9.5 | 9.4 | 2.302 | 509.4 | 470.6 |
| 12/1/06 | 9.378 | 3.130 | 2074.9 | 75.0 | 9.5 | 9.4 | 2.310 | 511.1 | 469.9 |
| 12/2/06 | 9.327 | 3.143 | 2076.7 | 75.0 | 9.5 | 9.4 | 2.317 | 512.6 | 469.3 |
| 12/3/06 | 9.284 | 3.163 | 2078.3 | 75.0 | 9.5 | 9.4 | 2.328 | 514.6 | 468.0 |
| 12/4/06 | 9.245 | 3.189 | 2079.7 | 75.0 | 9.5 | 9.4 | 2.342 | 517.1 | 466.2 |
| 12/5/06 | 9.207 | 3.215 | 2081.0 | 75.0 | 9.5 | 9.5 | 2.356 | 519.6 | 464.4 |
| 12/6/06 | 9.171 | 3.246 | 2082.3 | 75.0 | 9.5 | 9.5 | 2.373 | 522.5 | 462.2 |
| 12/7/06 | 9.135 | 3.274 | 2083.7 | 75.0 | 9.6 | 9.5 | 2.389 | 525.1 | 460.2 |
| 12/8/06 | 9.102 | 3.315 | 2084.9 | 75.0 | 9.6 | 9.5 | 2.412 | 528.8 | 457.1 |
| 12/9/06 | 9.068 | 3.354 | 2086.1 | 75.0 | 9.6 | 9.5 | 2.434 | 532.3 | 454.2 |
| 12/10/06 | 9.035 | 3.399 | 2087.3 | 75.0 | 9.6 | 9.5 | 2.460 | 536.3 | 450.8 |
| 12/11/06 | 8.997 | 3.437 | 2088.5 | 75.0 | 9.6 | 9.5 | 2.481 | 539.8 | 447.9 |
| 12/12/06 | 8.960 | 3.486 | 2090.0 | 75.0 | 9.6 | 9.5 | 2.510 | 544.2 | 444.2 |
| 12/13/06 | 8.920 | 3.522 | 2091.5 | 75.0 | 9.6 | 9.5 | 2.531 | 547.6 | 441.5 |
| 12/14/06 | 8.889 | 3.570 | 2092.6 | 75.0 | 9.6 | 9.5 | 2.559 | 551.8 | 437.8 |
| 12/15/06 | 8.865 | 3.621 | 2093.1 | 75.0 | 9.6 | 9.5 | 2.589 | 556.2 | 433.7 |

Scenario 7 - Post-Capture Orbiter Mass (Atlas IIAS) – Aerocapture

| Launch | C3 | V inf at Arr | Inj Mass | Cruise Stage | TCMs | TPS Mass | Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|
| | | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 11/26/06 | 9.755 | 3.087 | 1981.3 | 75 | 9.1 | 203.3 | 30.1 | 710.8 |
| 11/27/06 | 9.657 | 3.090 | 1985.3 | 75 | 9.1 | 203.3 | 30.1 | 712.6 |
| 11/28/06 | 9.572 | 3.097 | 1988.8 | 75 | 9.1 | 203.4 | 30.2 | 714.2 |
| 11/29/06 | 9.497 | 3.104 | 1991.9 | 75 | 9.1 | 203.6 | 30.3 | 715.5 |
| 11/30/06 | 9.434 | 3.116 | 1994.5 | 75 | 9.1 | 203.8 | 30.3 | 716.5 |
| 12/1/06 | 9.378 | 3.131 | 1996.7 | 75 | 9.1 | 204.0 | 30.3 | 717.4 |
| 12/2/06 | 9.327 | 3.143 | 1998.8 | 75 | 9.2 | 204.2 | 30.4 | 718.2 |
| 12/3/06 | 9.284 | 3.163 | 2000.6 | 75 | 9.2 | 204.6 | 30.4 | 718.7 |
| 12/4/06 | 9.245 | 3.189 | 2002.2 | 75 | 9.2 | 205.1 | 30.4 | 719.0 |
| 12/5/06 | 9.207 | 3.215 | 2003.7 | 75 | 9.2 | 205.5 | 30.4 | 719.2 |
| 12/6/06 | 9.171 | 3.246 | 2005.2 | 75 | 9.2 | 206.1 | 30.4 | 719.4 |
| 12/7/06 | 9.135 | 3.275 | 2006.8 | 75 | 9.2 | 206.7 | 30.4 | 719.6 |
| 12/8/06 | 9.101 | 3.312 | 2008.1 | 75 | 9.2 | 207.4 | 30.4 | 719.5 |
| 12/9/06 | 9.068 | 3.354 | 2009.5 | 75 | 9.2 | 208.2 | 30.4 | 719.4 |
| 12/10/06 | 9.035 | 3.399 | 2010.8 | 75 | 9.2 | 209.2 | 30.4 | 719.1 |
| 12/11/06 | 9.002 | 3.438 | 2012.2 | 75 | 9.2 | 210.0 | 30.4 | 719.0 |
| 12/12/06 | 8.960 | 3.487 | 2013.9 | 75 | 9.2 | 211.1 | 30.4 | 718.8 |
| 12/13/06 | 8.920 | 3.522 | 2015.6 | 75 | 9.2 | 211.8 | 30.4 | 718.8 |
| 12/14/06 | 8.889 | 3.570 | 2016.8 | 75 | 9.2 | 212.9 | 30.4 | 718.4 |
| 12/15/06 | 8.876 | 3.647 | 2017.4 | 75 | 9.2 | 214.7 | 30.3 | 716.9 |

Scenario 7 - Post-Capture Orbiter Mass (Atlas IIAR) – Aerocapture

| Launch | C3 | V inf at Arr | Inj Mass | Cruise Stage | TCMs | TPS Mass | Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|
| | | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 11/26/06 | 9.755 | 3.087 | 2142.0 | 75 | 9.8 | 203.3 | 33.3 | 787.1 |
| 11/27/06 | 9.657 | 3.090 | 2146.1 | 75 | 9.9 | 203.3 | 33.4 | 789.0 |
| 11/28/06 | 9.572 | 3.097 | 2149.7 | 75 | 9.9 | 203.4 | 33.4 | 790.6 |
| 11/29/06 | 9.497 | 3.104 | 2152.9 | 75 | 9.9 | 203.6 | 33.5 | 792.0 |
| 11/30/06 | 9.434 | 3.116 | 2155.6 | 75 | 9.9 | 203.8 | 33.5 | 793.1 |
| 12/01/06 | 9.378 | 3.131 | 2157.9 | 75 | 9.9 | 204.0 | 33.6 | 793.9 |
| 12/02/06 | 9.327 | 3.143 | 2160.1 | 75 | 9.9 | 204.2 | 33.6 | 794.8 |
| 12/03/06 | 9.284 | 3.163 | 2161.9 | 75 | 9.9 | 204.6 | 33.6 | 795.3 |
| 12/04/06 | 9.245 | 3.189 | 2163.5 | 75 | 9.9 | 205.1 | 33.7 | 795.6 |
| 12/05/06 | 9.207 | 3.215 | 2165.1 | 75 | 9.9 | 205.5 | 33.7 | 795.9 |
| 12/06/06 | 9.171 | 3.246 | 2166.6 | 75 | 10.0 | 206.1 | 33.7 | 796.1 |
| 12/07/06 | 9.135 | 3.275 | 2168.2 | 75 | 10.0 | 206.7 | 33.7 | 796.3 |
| 12/08/06 | 9.101 | 3.312 | 2169.6 | 75 | 10.0 | 207.4 | 33.7 | 796.3 |
| 12/09/06 | 9.068 | 3.354 | 2171.0 | 75 | 10.0 | 208.2 | 33.7 | 796.1 |
| 12/10/06 | 9.035 | 3.399 | 2172.4 | 75 | 10.0 | 209.2 | 33.7 | 795.9 |
| 12/11/06 | 9.002 | 3.438 | 2173.8 | 75 | 10.0 | 210.0 | 33.7 | 795.8 |
| 12/12/06 | 8.960 | 3.487 | 2175.6 | 75 | 10.0 | 211.1 | 33.7 | 795.6 |
| 12/13/06 | 8.920 | 3.522 | 2177.4 | 75 | 10.0 | 211.8 | 33.7 | 795.7 |
| 12/14/06 | 8.889 | 3.570 | 2178.6 | 75 | 10.0 | 212.9 | 33.6 | 795.2 |
| 12/15/06 | 8.876 | 3.647 | 2179.2 | 75 | 10.0 | 214.7 | 33.6 | 793.8 |

Scenario 7 - Post-Capture Orbiter Mass (Atlas IIARS) – Aerocapture

| Launch | C3 | V inf at Arr | Inj Mass | Cruise Stage | TCMs | TPS Mass | Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|
| | | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 11/26/06 | 9.755 | 3.087 | 2292.1 | 75 | 10.5 | 203.3 | 36.3 | 858.4 |
| 11/27/06 | 9.657 | 3.090 | 2296.4 | 75 | 10.6 | 203.3 | 36.4 | 860.4 |
| 11/28/06 | 9.572 | 3.097 | 2300.1 | 75 | 10.6 | 203.4 | 36.5 | 862.1 |
| 11/29/06 | 9.497 | 3.104 | 2303.4 | 75 | 10.6 | 203.6 | 36.5 | 863.5 |
| 11/30/06 | 9.434 | 3.116 | 2306.2 | 75 | 10.6 | 203.8 | 36.6 | 864.6 |
| 12/01/06 | 9.378 | 3.131 | 2308.6 | 75 | 10.6 | 204.0 | 36.6 | 865.5 |
| 12/02/06 | 9.327 | 3.143 | 2310.8 | 75 | 10.6 | 204.2 | 36.6 | 866.4 |
| 12/03/06 | 9.284 | 3.163 | 2312.7 | 75 | 10.6 | 204.6 | 36.7 | 867.0 |
| 12/04/06 | 9.245 | 3.189 | 2314.4 | 75 | 10.7 | 205.1 | 36.7 | 867.3 |
| 12/05/06 | 9.207 | 3.215 | 2316.1 | 75 | 10.7 | 205.5 | 36.7 | 867.7 |
| 12/06/06 | 9.171 | 3.246 | 2317.7 | 75 | 10.7 | 206.1 | 36.7 | 867.9 |
| 12/07/06 | 9.135 | 3.275 | 2319.3 | 75 | 10.7 | 206.7 | 36.7 | 868.1 |
| 12/08/06 | 9.101 | 3.312 | 2320.8 | 75 | 10.7 | 207.4 | 36.7 | 868.1 |
| 12/09/06 | 9.068 | 3.354 | 2322.2 | 75 | 10.7 | 208.2 | 36.7 | 868.0 |
| 12/10/06 | 9.035 | 3.399 | 2323.7 | 75 | 10.7 | 209.2 | 36.7 | 867.8 |
| 12/11/06 | 9.002 | 3.438 | 2325.2 | 75 | 10.7 | 210.0 | 36.7 | 867.7 |
| 12/12/06 | 8.960 | 3.487 | 2327.0 | 75 | 10.7 | 211.1 | 36.7 | 867.5 |
| 12/13/06 | 8.920 | 3.522 | 2328.8 | 75 | 10.7 | 211.8 | 36.7 | 867.6 |
| 12/14/06 | 8.889 | 3.570 | 2330.1 | 75 | 10.7 | 212.9 | 36.7 | 867.2 |
| 12/15/06 | 8.876 | 3.647 | 2330.8 | 75 | 10.7 | 214.7 | 36.6 | 865.8 |

Scenario 7 - Post-Capture Orbiter Mass (Delta III 3-stage) – Aerocapture

| Launch | C3 | V inf at Arr | Inj Mass | Cruise Stage | TCM | TPS Mass | Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|
| | | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 11/26/06 | 9.755 | 3.087 | 2061.4 | 75 | 9.5 | 203.3 | 31.7 | 748.8 |
| 11/27/06 | 9.657 | 3.090 | 2064.9 | 75 | 9.5 | 203.3 | 31.7 | 750.4 |
| 11/28/06 | 9.572 | 3.097 | 2067.9 | 75 | 9.5 | 203.4 | 31.8 | 751.7 |
| 11/29/06 | 9.497 | 3.104 | 2070.6 | 75 | 9.5 | 203.6 | 31.8 | 752.9 |
| 11/30/06 | 9.434 | 3.116 | 2072.9 | 75 | 9.5 | 203.8 | 31.9 | 753.8 |
| 12/1/06 | 9.378 | 3.131 | 2074.9 | 75 | 9.5 | 204.0 | 31.9 | 754.5 |
| 12/2/06 | 9.327 | 3.143 | 2076.7 | 75 | 9.5 | 204.2 | 31.9 | 755.2 |
| 12/3/06 | 9.284 | 3.163 | 2078.3 | 75 | 9.5 | 204.6 | 32.0 | 755.6 |
| 12/4/06 | 9.245 | 3.189 | 2079.7 | 75 | 9.5 | 205.1 | 32.0 | 755.8 |
| 12/5/06 | 9.207 | 3.215 | 2081.0 | 75 | 9.5 | 205.5 | 32.0 | 756.0 |
| 12/6/06 | 9.171 | 3.246 | 2082.3 | 75 | 9.5 | 206.1 | 32.0 | 756.0 |
| 12/7/06 | 9.135 | 3.275 | 2083.7 | 75 | 9.6 | 206.7 | 32.0 | 756.1 |
| 12/8/06 | 9.101 | 3.312 | 2084.9 | 75 | 9.6 | 207.4 | 32.0 | 756.0 |
| 12/9/06 | 9.068 | 3.354 | 2086.1 | 75 | 9.6 | 208.2 | 32.0 | 755.8 |
| 12/10/06 | 9.035 | 3.399 | 2087.3 | 75 | 9.6 | 209.2 | 32.0 | 755.4 |
| 12/11/06 | 9.002 | 3.438 | 2088.5 | 75 | 9.6 | 210.0 | 31.9 | 755.2 |
| 12/12/06 | 8.960 | 3.487 | 2090.0 | 75 | 9.6 | 211.1 | 31.9 | 754.9 |
| 12/13/06 | 8.920 | 3.522 | 2091.5 | 75 | 9.6 | 211.8 | 31.9 | 754.9 |
| 12/14/06 | 8.889 | 3.570 | 2092.6 | 75 | 9.6 | 212.9 | 31.9 | 754.3 |
| 12/15/06 | 8.876 | 3.647 | 2093.1 | 75 | 9.6 | 214.7 | 31.8 | 752.9 |

## Scenario 7 - Post-Capture Orbiter Mass (Atlas IIAS) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 1981.3 | 75.0 | 9.1 | 9.1 | 1.207 | 1.105 |
| 11/27/06 | 9.656 | 3.090 | 1985.3 | 75.0 | 9.1 | 9.1 | 1.209 | 1.106 |
| 11/28/06 | 9.572 | 3.097 | 1988.8 | 75.0 | 9.1 | 9.1 | 1.213 | 1.110 |
| 11/29/06 | 9.497 | 3.104 | 1991.9 | 75.0 | 9.1 | 9.1 | 1.217 | 1.114 |
| 11/30/06 | 9.433 | 3.116 | 1994.5 | 75.0 | 9.1 | 9.1 | 1.223 | 1.120 |
| 12/1/06 | 9.378 | 3.130 | 1996.7 | 75.0 | 9.1 | 9.1 | 1.231 | 1.128 |
| 12/2/06 | 9.327 | 3.143 | 1998.8 | 75.0 | 9.2 | 9.2 | 1.238 | 1.135 |
| 12/3/06 | 9.284 | 3.163 | 2000.6 | 75.0 | 9.2 | 9.2 | 1.249 | 1.146 |
| 12/4/06 | 9.245 | 3.189 | 2002.2 | 75.0 | 9.2 | 9.2 | 1.263 | 1.160 |
| 12/5/06 | 9.207 | 3.215 | 2003.7 | 75.0 | 9.2 | 9.2 | 1.277 | 1.174 |
| 12/6/06 | 9.171 | 3.246 | 2005.2 | 75.0 | 9.2 | 9.2 | 1.294 | 1.191 |
| 12/7/06 | 9.135 | 3.274 | 2006.8 | 75.0 | 9.2 | 9.2 | 1.310 | 1.207 |
| 12/8/06 | 9.102 | 3.315 | 2008.1 | 75.0 | 9.2 | 9.2 | 1.333 | 1.230 |
| 12/9/06 | 9.068 | 3.354 | 2009.5 | 75.0 | 9.2 | 9.2 | 1.355 | 1.252 |
| 12/10/06 | 9.035 | 3.399 | 2010.8 | 75.0 | 9.2 | 9.2 | 1.381 | 1.278 |
| 12/11/06 | 8.997 | 3.437 | 2012.2 | 75.0 | 9.2 | 9.2 | 1.402 | 1.300 |
| 12/12/06 | 8.960 | 3.486 | 2013.9 | 75.0 | 9.2 | 9.2 | 1.431 | 1.328 |
| 12/13/06 | 8.920 | 3.522 | 2015.6 | 75.0 | 9.2 | 9.2 | 1.452 | 1.349 |
| 12/14/06 | 8.889 | 3.570 | 2016.8 | 75.0 | 9.2 | 9.2 | 1.480 | 1.377 |
| 12/15/06 | 8.865 | 3.621 | 2017.4 | 75.0 | 9.2 | 9.2 | 1.511 | 1.408 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 1.059 | 301.6 | 280.2 | 270.4 | 50 | 583.5 | 604.9 | 614.6 |
| 11/27/06 | 1.060 | 302.5 | 281.1 | 271.3 | 50 | 584.5 | 605.9 | 615.7 |
| 11/28/06 | 1.064 | 303.8 | 282.4 | 272.6 | 50 | 584.9 | 606.3 | 616.1 |
| 11/29/06 | 1.068 | 305.1 | 283.7 | 273.9 | 50 | 585.1 | 606.6 | 616.3 |
| 11/30/06 | 1.074 | 306.8 | 285.4 | 275.6 | 50 | 584.6 | 606.1 | 615.9 |
| 12/1/06 | 1.082 | 308.8 | 287.4 | 277.6 | 50 | 583.8 | 605.2 | 615.0 |
| 12/2/06 | 1.089 | 310.5 | 289.2 | 279.4 | 50 | 583.1 | 604.5 | 614.2 |
| 12/3/06 | 1.100 | 313.0 | 291.7 | 282.0 | 50 | 581.4 | 602.8 | 612.5 |
| 12/4/06 | 1.114 | 316.2 | 295.0 | 285.3 | 50 | 579.0 | 600.3 | 610.0 |
| 12/5/06 | 1.128 | 319.4 | 298.2 | 288.5 | 50 | 576.7 | 597.8 | 607.5 |
| 12/6/06 | 1.146 | 323.1 | 302.0 | 292.4 | 50 | 573.7 | 594.7 | 604.4 |
| 12/7/06 | 1.161 | 326.5 | 305.5 | 295.9 | 50 | 571.0 | 592.0 | 601.6 |
| 12/8/06 | 1.184 | 331.3 | 310.5 | 301.0 | 50 | 566.9 | 587.7 | 597.2 |
| 12/9/06 | 1.206 | 335.9 | 315.2 | 305.8 | 50 | 562.9 | 583.6 | 593.1 |
| 12/10/06 | 1.232 | 341.2 | 320.7 | 311.3 | 50 | 558.3 | 578.8 | 588.2 |
| 12/11/06 | 1.254 | 345.8 | 325.3 | 316.0 | 50 | 554.4 | 574.8 | 584.2 |
| 12/12/06 | 1.282 | 351.6 | 331.4 | 322.1 | 50 | 549.4 | 569.7 | 578.9 |
| 12/13/06 | 1.303 | 356.0 | 335.9 | 326.7 | 50 | 545.8 | 566.0 | 575.2 |
| 12/14/06 | 1.332 | 361.7 | 341.7 | 332.6 | 50 | 540.8 | 560.8 | 569.9 |
| 12/15/06 | 1.362 | 367.5 | 347.7 | 338.7 | 50 | 535.2 | 555.0 | 564.0 |

## Scenario 8 - Post-Capture Orbiter Mass (Delta III 3-stage) – Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 1981.3 | 75.0 | 9.1 | 9.0 | 2.025 | 444.7 | 490.4 |
| 9/14/07 | 13.130 | 2.578 | 1985.3 | 75.0 | 9.1 | 9.0 | 2.030 | 446.4 | 490.7 |
| 9/15/07 | 13.040 | 2.594 | 1988.8 | 75.0 | 9.1 | 9.0 | 2.038 | 448.4 | 490.4 |
| 9/16/07 | 12.960 | 2.637 | 1991.9 | 75.0 | 9.1 | 9.0 | 2.058 | 452.3 | 488.0 |
| 9/17/07 | 12.910 | 2.634 | 1994.5 | 75.0 | 9.1 | 9.0 | 2.057 | 452.7 | 488.9 |
| 9/18/07 | 12.860 | 2.662 | 1996.7 | 75.0 | 9.1 | 9.1 | 2.070 | 455.3 | 487.4 |
| 9/19/07 | 12.820 | 2.696 | 1998.8 | 75.0 | 9.2 | 9.1 | 2.087 | 458.4 | 485.3 |
| 9/20/07 | 12.800 | 2.709 | 2000.6 | 75.0 | 9.2 | 9.1 | 2.093 | 459.8 | 484.8 |
| 9/21/07 | 12.790 | 2.757 | 2002.2 | 75.0 | 9.2 | 9.1 | 2.116 | 463.8 | 481.6 |
| 9/22/07 | 12.800 | 2.774 | 2003.7 | 75.0 | 9.2 | 9.1 | 2.125 | 465.4 | 480.7 |
| 9/23/07 | 12.810 | 2.796 | 2005.2 | 75.0 | 9.2 | 9.1 | 2.136 | 467.5 | 479.4 |
| 9/24/07 | 12.830 | 2.858 | 2006.8 | 75.0 | 9.2 | 9.1 | 2.167 | 472.6 | 475.0 |
| 9/25/07 | 12.850 | 2.876 | 2008.1 | 75.0 | 9.2 | 9.1 | 2.176 | 474.3 | 474.0 |
| 9/26/07 | 12.890 | 2.901 | 2009.5 | 75.0 | 9.2 | 9.1 | 2.189 | 476.5 | 472.4 |
| 9/27/07 | 12.920 | 2.956 | 2010.8 | 75.0 | 9.2 | 9.1 | 2.217 | 481.1 | 468.4 |
| 9/28/07 | 12.950 | 3.008 | 2012.2 | 75.0 | 9.2 | 9.1 | 2.244 | 485.5 | 464.7 |
| 9/29/07 | 12.990 | 3.053 | 2013.9 | 75.0 | 9.2 | 9.1 | 2.268 | 489.5 | 461.6 |
| 9/30/07 | 13.040 | 3.142 | 2015.6 | 75.0 | 9.2 | 9.1 | 2.316 | 496.9 | 455.0 |
| 10/1/07 | 13.110 | 3.150 | 2016.8 | 75.0 | 9.2 | 9.2 | 2.320 | 497.8 | 454.7 |
| 10/2/07 | 12.990 | 3.235 | 2017.4 | 75.0 | 9.2 | 9.2 | 2.367 | 504.7 | 448.1 |

# Scenario 7 - Post-Capture Orbiter Mass (Atlas IIAR) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 2142.0 | 75.0 | 9.8 | 9.8 | 1.207 | 1.105 |
| 11/27/06 | 9.656 | 3.090 | 2146.1 | 75.0 | 9.9 | 9.9 | 1.209 | 1.106 |
| 11/28/06 | 9.572 | 3.097 | 2149.7 | 75.0 | 9.9 | 9.9 | 1.213 | 1.110 |
| 11/29/06 | 9.497 | 3.104 | 2152.9 | 75.0 | 9.9 | 9.9 | 1.217 | 1.114 |
| 11/30/06 | 9.433 | 3.116 | 2155.6 | 75.0 | 9.9 | 9.9 | 1.223 | 1.120 |
| 12/1/06 | 9.378 | 3.130 | 2157.9 | 75.0 | 9.9 | 9.9 | 1.231 | 1.128 |
| 12/2/06 | 9.327 | 3.143 | 2160.1 | 75.0 | 9.9 | 9.9 | 1.238 | 1.135 |
| 12/3/06 | 9.284 | 3.163 | 2161.9 | 75.0 | 9.9 | 9.9 | 1.249 | 1.146 |
| 12/4/06 | 9.245 | 3.189 | 2163.5 | 75.0 | 9.9 | 9.9 | 1.263 | 1.160 |
| 12/5/06 | 9.207 | 3.215 | 2165.1 | 75.0 | 9.9 | 9.9 | 1.277 | 1.174 |
| 12/6/06 | 9.171 | 3.246 | 2166.6 | 75.0 | 10.0 | 10.0 | 1.294 | 1.191 |
| 12/7/06 | 9.135 | 3.274 | 2168.2 | 75.0 | 10.0 | 10.0 | 1.310 | 1.207 |
| 12/8/06 | 9.102 | 3.315 | 2169.6 | 75.0 | 10.0 | 10.0 | 1.333 | 1.230 |
| 12/9/06 | 9.068 | 3.354 | 2171.0 | 75.0 | 10.0 | 10.0 | 1.355 | 1.252 |
| 12/10/06 | 9.035 | 3.399 | 2172.4 | 75.0 | 10.0 | 10.0 | 1.381 | 1.278 |
| 12/11/06 | 8.997 | 3.437 | 2173.8 | 75.0 | 10.0 | 10.0 | 1.402 | 1.300 |
| 12/12/06 | 8.960 | 3.486 | 2175.6 | 75.0 | 10.0 | 10.0 | 1.431 | 1.328 |
| 12/13/06 | 8.920 | 3.522 | 2177.4 | 75.0 | 10.0 | 10.0 | 1.452 | 1.349 |
| 12/14/06 | 8.889 | 3.570 | 2178.6 | 75.0 | 10.0 | 10.0 | 1.480 | 1.377 |
| 12/15/06 | 8.865 | 3.621 | 2179.2 | 75.0 | 10.0 | 10.0 | 1.511 | 1.408 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 1.059 | 327.0 | 303.8 | 293.2 | 50 | 636.9 | 660.1 | 670.6 |
| 11/27/06 | 1.060 | 328.0 | 304.8 | 294.1 | 50 | 637.9 | 661.1 | 671.7 |
| 11/28/06 | 1.064 | 329.4 | 306.1 | 295.5 | 50 | 638.2 | 661.5 | 672.1 |
| 11/29/06 | 1.068 | 330.7 | 307.5 | 296.9 | 50 | 638.4 | 661.7 | 672.3 |
| 11/30/06 | 1.074 | 332.6 | 309.4 | 298.8 | 50 | 637.9 | 661.1 | 671.7 |
| 12/1/06 | 1.082 | 334.6 | 311.4 | 300.8 | 50 | 637.0 | 660.2 | 670.8 |
| 12/2/06 | 1.089 | 336.6 | 313.4 | 302.8 | 50 | 636.1 | 659.3 | 669.9 |
| 12/3/06 | 1.100 | 339.3 | 316.2 | 305.6 | 50 | 634.3 | 657.4 | 668.0 |
| 12/4/06 | 1.114 | 342.7 | 319.6 | 309.1 | 50 | 631.7 | 654.7 | 665.2 |
| 12/5/06 | 1.128 | 346.1 | 323.1 | 312.7 | 50 | 629.1 | 652.0 | 662.5 |
| 12/6/06 | 1.146 | 350.1 | 327.3 | 316.8 | 50 | 625.8 | 648.7 | 659.1 |
| 12/7/06 | 1.161 | 353.8 | 331.0 | 320.6 | 50 | 622.9 | 645.7 | 656.0 |
| 12/8/06 | 1.184 | 359.0 | 336.4 | 326.1 | 50 | 618.4 | 641.0 | 651.3 |
| 12/9/06 | 1.206 | 364.0 | 341.5 | 331.3 | 50 | 614.1 | 636.5 | 646.8 |
| 12/10/06 | 1.232 | 369.7 | 347.5 | 337.3 | 50 | 609.0 | 631.3 | 641.5 |
| 12/11/06 | 1.254 | 374.6 | 352.5 | 342.4 | 50 | 604.8 | 627.0 | 637.1 |
| 12/12/06 | 1.282 | 380.9 | 359.0 | 349.0 | 50 | 599.4 | 621.3 | 631.4 |
| 12/13/06 | 1.303 | 385.7 | 363.8 | 353.9 | 50 | 595.5 | 617.3 | 627.3 |
| 12/14/06 | 1.332 | 391.8 | 370.2 | 360.3 | 50 | 590.0 | 611.7 | 621.5 |
| 12/15/06 | 1.362 | 398.2 | 376.7 | 366.9 | 50 | 583.9 | 605.4 | 615.2 |

## Scenario 7 - Post-Capture Orbiter Mass (Atlas IIARS) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 2292.1 | 75.0 | 10.5 | 10.5 | 1.207 | 1.105 |
| 11/27/06 | 9.656 | 3.090 | 2296.4 | 75.0 | 10.6 | 10.6 | 1.209 | 1.106 |
| 11/28/06 | 9.572 | 3.097 | 2300.1 | 75.0 | 10.6 | 10.6 | 1.213 | 1.110 |
| 11/29/06 | 9.497 | 3.104 | 2303.4 | 75.0 | 10.6 | 10.6 | 1.217 | 1.114 |
| 11/30/06 | 9.433 | 3.116 | 2306.2 | 75.0 | 10.6 | 10.6 | 1.223 | 1.120 |
| 12/1/06 | 9.378 | 3.130 | 2308.6 | 75.0 | 10.6 | 10.6 | 1.231 | 1.128 |
| 12/2/06 | 9.327 | 3.143 | 2310.8 | 75.0 | 10.6 | 10.6 | 1.238 | 1.135 |
| 12/3/06 | 9.284 | 3.163 | 2312.7 | 75.0 | 10.6 | 10.6 | 1.249 | 1.146 |
| 12/4/06 | 9.245 | 3.189 | 2314.4 | 75.0 | 10.7 | 10.7 | 1.263 | 1.160 |
| 12/5/06 | 9.207 | 3.215 | 2316.1 | 75.0 | 10.7 | 10.7 | 1.277 | 1.174 |
| 12/6/06 | 9.171 | 3.246 | 2317.7 | 75.0 | 10.7 | 10.7 | 1.294 | 1.191 |
| 12/7/06 | 9.135 | 3.274 | 2319.3 | 75.0 | 10.7 | 10.7 | 1.310 | 1.207 |
| 12/8/06 | 9.102 | 3.315 | 2320.8 | 75.0 | 10.7 | 10.7 | 1.333 | 1.230 |
| 12/9/06 | 9.068 | 3.354 | 2322.2 | 75.0 | 10.7 | 10.7 | 1.355 | 1.252 |
| 12/10/06 | 9.035 | 3.399 | 2323.7 | 75.0 | 10.7 | 10.7 | 1.381 | 1.278 |
| 12/11/06 | 8.997 | 3.437 | 2325.2 | 75.0 | 10.7 | 10.7 | 1.402 | 1.300 |
| 12/12/06 | 8.960 | 3.486 | 2327.0 | 75.0 | 10.7 | 10.7 | 1.431 | 1.328 |
| 12/13/06 | 8.920 | 3.522 | 2328.8 | 75.0 | 10.7 | 10.7 | 1.452 | 1.349 |
| 12/14/06 | 8.889 | 3.570 | 2330.1 | 75.0 | 10.7 | 10.7 | 1.480 | 1.377 |
| 12/15/06 | 8.865 | 3.621 | 2330.8 | 75.0 | 10.7 | 10.7 | 1.511 | 1.408 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 1.059 | 350.7 | 325.8 | 314.5 | 50 | 686.7 | 711.6 | 723.0 |
| 11/27/06 | 1.060 | 351.8 | 326.9 | 315.5 | 50 | 687.8 | 712.7 | 724.1 |
| 11/28/06 | 1.064 | 353.3 | 328.3 | 317.0 | 50 | 688.1 | 713.0 | 724.4 |
| 11/29/06 | 1.068 | 354.7 | 329.8 | 318.4 | 50 | 688.3 | 713.2 | 724.6 |
| 11/30/06 | 1.074 | 356.7 | 331.8 | 320.4 | 50 | 687.7 | 712.6 | 724.0 |
| 12/1/06 | 1.082 | 358.9 | 334.0 | 322.6 | 50 | 686.7 | 711.6 | 722.9 |
| 12/2/06 | 1.089 | 360.9 | 336.1 | 324.7 | 50 | 685.8 | 710.6 | 721.9 |
| 12/3/06 | 1.100 | 363.8 | 339.0 | 327.7 | 50 | 683.8 | 708.6 | 719.9 |
| 12/4/06 | 1.114 | 367.4 | 342.7 | 331.5 | 50 | 681.0 | 705.7 | 716.9 |
| 12/5/06 | 1.128 | 371.1 | 346.5 | 335.3 | 50 | 678.1 | 702.7 | 714.0 |
| 12/6/06 | 1.146 | 375.4 | 350.9 | 339.7 | 50 | 674.6 | 699.1 | 710.3 |
| 12/7/06 | 1.161 | 379.3 | 354.9 | 343.8 | 50 | 671.5 | 695.9 | 707.0 |
| 12/8/06 | 1.184 | 384.9 | 360.7 | 349.6 | 50 | 666.6 | 690.8 | 701.9 |
| 12/9/06 | 1.206 | 390.2 | 366.2 | 355.2 | 50 | 662.0 | 686.1 | 697.0 |
| 12/10/06 | 1.232 | 396.4 | 372.5 | 361.6 | 50 | 656.6 | 680.4 | 691.3 |
| 12/11/06 | 1.254 | 401.6 | 377.9 | 367.1 | 50 | 652.0 | 675.8 | 686.6 |
| 12/12/06 | 1.282 | 408.4 | 384.9 | 374.1 | 50 | 646.2 | 669.7 | 680.5 |
| 12/13/06 | 1.303 | 413.5 | 390.1 | 379.4 | 50 | 642.0 | 665.4 | 676.1 |
| 12/14/06 | 1.332 | 420.0 | 396.8 | 386.2 | 50 | 636.1 | 659.3 | 669.9 |
| 12/15/06 | 1.362 | 426.8 | 403.8 | 393.4 | 50 | 629.6 | 652.6 | 663.1 |

# Scenario 7 - Post-Capture Orbiter Mass (Delta III 3-stage) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 9.755 | 3.087 | 2061.4 | 75.0 | 9.5 | 9.5 | 1.207 | 1.105 |
| 11/27/06 | 9.656 | 3.090 | 2064.9 | 75.0 | 9.5 | 9.5 | 1.209 | 1.106 |
| 11/28/06 | 9.572 | 3.097 | 2067.9 | 75.0 | 9.5 | 9.5 | 1.213 | 1.110 |
| 11/29/06 | 9.497 | 3.104 | 2070.6 | 75.0 | 9.5 | 9.5 | 1.217 | 1.114 |
| 11/30/06 | 9.433 | 3.116 | 2072.9 | 75.0 | 9.5 | 9.5 | 1.223 | 1.120 |
| 12/1/06 | 9.378 | 3.130 | 2074.9 | 75.0 | 9.5 | 9.5 | 1.231 | 1.128 |
| 12/2/06 | 9.327 | 3.143 | 2076.7 | 75.0 | 9.5 | 9.5 | 1.238 | 1.135 |
| 12/3/06 | 9.284 | 3.163 | 2078.3 | 75.0 | 9.5 | 9.5 | 1.249 | 1.146 |
| 12/4/06 | 9.245 | 3.189 | 2079.7 | 75.0 | 9.5 | 9.5 | 1.263 | 1.160 |
| 12/5/06 | 9.207 | 3.215 | 2081.0 | 75.0 | 9.5 | 9.5 | 1.277 | 1.174 |
| 12/6/06 | 9.171 | 3.246 | 2082.3 | 75.0 | 9.5 | 9.5 | 1.294 | 1.191 |
| 12/7/06 | 9.135 | 3.274 | 2083.7 | 75.0 | 9.6 | 9.6 | 1.310 | 1.207 |
| 12/8/06 | 9.102 | 3.315 | 2084.9 | 75.0 | 9.6 | 9.6 | 1.333 | 1.230 |
| 12/9/06 | 9.068 | 3.354 | 2086.1 | 75.0 | 9.6 | 9.6 | 1.355 | 1.252 |
| 12/10/06 | 9.035 | 3.399 | 2087.3 | 75.0 | 9.6 | 9.6 | 1.381 | 1.278 |
| 12/11/06 | 8.997 | 3.437 | 2088.5 | 75.0 | 9.6 | 9.6 | 1.402 | 1.300 |
| 12/12/06 | 8.960 | 3.486 | 2090.0 | 75.0 | 9.6 | 9.6 | 1.431 | 1.328 |
| 12/13/06 | 8.920 | 3.522 | 2091.5 | 75.0 | 9.6 | 9.6 | 1.452 | 1.349 |
| 12/14/06 | 8.889 | 3.570 | 2092.6 | 75.0 | 9.6 | 9.6 | 1.480 | 1.377 |
| 12/15/06 | 8.865 | 3.621 | 2093.1 | 75.0 | 9.6 | 9.6 | 1.511 | 1.408 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 11/26/06 | 1.059 | 314.2 | 291.9 | 281.7 | 50 | 610.1 | 632.4 | 642.5 |
| 11/27/06 | 1.060 | 315.1 | 292.8 | 282.6 | 50 | 610.9 | 633.2 | 643.4 |
| 11/28/06 | 1.064 | 316.4 | 294.1 | 283.9 | 50 | 611.1 | 633.4 | 643.6 |
| 11/29/06 | 1.068 | 317.6 | 295.3 | 285.1 | 50 | 611.2 | 633.5 | 643.7 |
| 11/30/06 | 1.074 | 319.4 | 297.1 | 286.9 | 50 | 610.6 | 632.9 | 643.1 |
| 12/1/06 | 1.082 | 321.3 | 299.0 | 288.9 | 50 | 609.6 | 631.9 | 642.1 |
| 12/2/06 | 1.089 | 323.1 | 300.9 | 290.7 | 50 | 608.7 | 631.0 | 641.1 |
| 12/3/06 | 1.100 | 325.7 | 303.5 | 293.4 | 50 | 606.9 | 629.1 | 639.2 |
| 12/4/06 | 1.114 | 328.9 | 306.8 | 296.7 | 50 | 604.3 | 626.4 | 636.5 |
| 12/5/06 | 1.128 | 332.2 | 310.1 | 300.1 | 50 | 601.8 | 623.8 | 633.8 |
| 12/6/06 | 1.146 | 336.0 | 314.1 | 304.1 | 50 | 598.6 | 620.5 | 630.5 |
| 12/7/06 | 1.161 | 339.5 | 317.7 | 307.7 | 50 | 595.7 | 617.6 | 627.5 |
| 12/8/06 | 1.184 | 344.5 | 322.8 | 312.9 | 50 | 591.3 | 613.0 | 622.9 |
| 12/9/06 | 1.206 | 349.2 | 327.7 | 317.9 | 50 | 587.2 | 608.7 | 618.5 |
| 12/10/06 | 1.232 | 354.7 | 333.3 | 323.6 | 50 | 582.3 | 603.6 | 613.4 |
| 12/11/06 | 1.254 | 359.4 | 338.2 | 328.5 | 50 | 578.2 | 599.4 | 609.1 |
| 12/12/06 | 1.282 | 365.4 | 344.4 | 334.7 | 50 | 572.9 | 594.0 | 603.6 |
| 12/13/06 | 1.303 | 369.9 | 349.0 | 339.4 | 50 | 569.1 | 590.1 | 599.6 |
| 12/14/06 | 1.332 | 375.8 | 355.0 | 345.5 | 50 | 563.8 | 584.6 | 594.1 |
| 12/15/06 | 1.362 | 381.9 | 361.3 | 351.9 | 50 | 558.0 | 578.5 | 587.9 |

## Scenario 8 - Post-Capture Orbiter Mass (Atlas IIAS) – Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 1981.3 | 75.0 | 9.1 | 9.0 | 2.025 | 444.7 | 490.4 |
| 9/14/07 | 13.130 | 2.578 | 1985.3 | 75.0 | 9.1 | 9.0 | 2.030 | 446.4 | 490.7 |
| 9/15/07 | 13.040 | 2.594 | 1988.8 | 75.0 | 9.1 | 9.0 | 2.038 | 448.4 | 490.4 |
| 9/16/07 | 12.960 | 2.637 | 1991.9 | 75.0 | 9.1 | 9.0 | 2.058 | 452.3 | 488.0 |
| 9/17/07 | 12.910 | 2.634 | 1994.5 | 75.0 | 9.1 | 9.0 | 2.057 | 452.7 | 488.9 |
| 9/18/07 | 12.860 | 2.662 | 1996.7 | 75.0 | 9.1 | 9.1 | 2.070 | 455.3 | 487.4 |
| 9/19/07 | 12.820 | 2.696 | 1998.8 | 75.0 | 9.2 | 9.1 | 2.087 | 458.4 | 485.3 |
| 9/20/07 | 12.800 | 2.709 | 2000.6 | 75.0 | 9.2 | 9.1 | 2.093 | 459.8 | 484.8 |
| 9/21/07 | 12.790 | 2.757 | 2002.2 | 75.0 | 9.2 | 9.1 | 2.116 | 463.8 | 481.6 |
| 9/22/07 | 12.800 | 2.774 | 2003.7 | 75.0 | 9.2 | 9.1 | 2.125 | 465.4 | 480.7 |
| 9/23/07 | 12.810 | 2.796 | 2005.2 | 75.0 | 9.2 | 9.1 | 2.136 | 467.5 | 479.4 |
| 9/24/07 | 12.830 | 2.858 | 2006.8 | 75.0 | 9.2 | 9.1 | 2.167 | 472.6 | 475.0 |
| 9/25/07 | 12.850 | 2.876 | 2008.1 | 75.0 | 9.2 | 9.1 | 2.176 | 474.3 | 474.0 |
| 9/26/07 | 12.890 | 2.901 | 2009.5 | 75.0 | 9.2 | 9.1 | 2.189 | 476.5 | 472.4 |
| 9/27/07 | 12.920 | 2.956 | 2010.8 | 75.0 | 9.2 | 9.1 | 2.217 | 481.1 | 468.4 |
| 9/28/07 | 12.950 | 3.008 | 2012.2 | 75.0 | 9.2 | 9.1 | 2.244 | 485.5 | 464.7 |
| 9/29/07 | 12.990 | 3.053 | 2013.9 | 75.0 | 9.2 | 9.1 | 2.268 | 489.5 | 461.6 |
| 9/30/07 | 13.040 | 3.142 | 2015.6 | 75.0 | 9.2 | 9.1 | 2.316 | 496.9 | 455.0 |
| 10/1/07 | 13.110 | 3.150 | 2016.8 | 75.0 | 9.2 | 9.2 | 2.320 | 497.8 | 454.7 |
| 10/2/07 | 12.990 | 3.235 | 2017.4 | 75.0 | 9.2 | 9.2 | 2.367 | 504.7 | 448.1 |

## Scenario 8 - Post-Capture Orbiter Mass (Atlas IIAR) – Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2142.0 | 75.0 | 9.8 | 9.7 | 2.025 | 482.2 | 531.8 |
| 9/14/07 | 13.130 | 2.578 | 2146.1 | 75.0 | 9.9 | 9.8 | 2.030 | 484.0 | 531.9 |
| 9/15/07 | 13.040 | 2.594 | 2149.7 | 75.0 | 9.9 | 9.8 | 2.038 | 486.1 | 531.6 |
| 9/16/07 | 12.960 | 2.637 | 2152.9 | 75.0 | 9.9 | 9.8 | 2.058 | 490.3 | 529.0 |
| 9/17/07 | 12.910 | 2.634 | 2155.6 | 75.0 | 9.9 | 9.8 | 2.057 | 490.7 | 529.9 |
| 9/18/07 | 12.860 | 2.662 | 2157.9 | 75.0 | 9.9 | 9.8 | 2.070 | 493.5 | 528.2 |
| 9/19/07 | 12.820 | 2.696 | 2160.1 | 75.0 | 9.9 | 9.8 | 2.087 | 496.8 | 526.0 |
| 9/20/07 | 12.800 | 2.709 | 2161.9 | 75.0 | 9.9 | 9.8 | 2.093 | 498.3 | 525.4 |
| 9/21/07 | 12.790 | 2.757 | 2163.5 | 75.0 | 9.9 | 9.8 | 2.116 | 502.6 | 521.9 |
| 9/22/07 | 12.800 | 2.774 | 2165.1 | 75.0 | 9.9 | 9.8 | 2.125 | 504.4 | 520.9 |
| 9/23/07 | 12.810 | 2.796 | 2166.6 | 75.0 | 10.0 | 9.9 | 2.136 | 506.5 | 519.5 |
| 9/24/07 | 12.830 | 2.858 | 2168.2 | 75.0 | 10.0 | 9.9 | 2.167 | 512.1 | 514.7 |
| 9/25/07 | 12.850 | 2.876 | 2169.6 | 75.0 | 10.0 | 9.9 | 2.176 | 513.9 | 513.6 |
| 9/26/07 | 12.890 | 2.901 | 2171.0 | 75.0 | 10.0 | 9.9 | 2.189 | 516.3 | 511.8 |
| 9/27/07 | 12.920 | 2.956 | 2172.4 | 75.0 | 10.0 | 9.9 | 2.217 | 521.3 | 507.6 |
| 9/28/07 | 12.950 | 3.008 | 2173.8 | 75.0 | 10.0 | 9.9 | 2.244 | 526.0 | 503.5 |
| 9/29/07 | 12.990 | 3.053 | 2175.6 | 75.0 | 10.0 | 9.9 | 2.268 | 530.3 | 500.1 |
| 9/30/07 | 13.040 | 3.142 | 2177.4 | 75.0 | 10.0 | 9.9 | 2.316 | 538.3 | 493.0 |
| 10/1/07 | 13.110 | 3.150 | 2178.6 | 75.0 | 10.0 | 9.9 | 2.320 | 539.3 | 492.6 |
| 10/2/07 | 12.990 | 3.235 | 2179.2 | 75.0 | 10.0 | 9.9 | 2.367 | 546.7 | 485.5 |

## Scenario 8 - Post-Capture Orbiter Mass (Atlas IIARS) – Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2292.1 | 75.0 | 10.5 | 10.4 | 2.025 | 517.2 | 570.4 |
| 9/14/07 | 13.130 | 2.578 | 2296.4 | 75.0 | 10.6 | 10.5 | 2.030 | 519.1 | 570.5 |
| 9/15/07 | 13.040 | 2.594 | 2300.1 | 75.0 | 10.6 | 10.5 | 2.038 | 521.4 | 570.1 |
| 9/16/07 | 12.960 | 2.637 | 2303.4 | 75.0 | 10.6 | 10.5 | 2.058 | 525.8 | 567.3 |
| 9/17/07 | 12.910 | 2.634 | 2306.2 | 75.0 | 10.6 | 10.5 | 2.057 | 526.2 | 568.2 |
| 9/18/07 | 12.860 | 2.662 | 2308.6 | 75.0 | 10.6 | 10.5 | 2.070 | 529.2 | 566.4 |
| 9/19/07 | 12.820 | 2.696 | 2310.8 | 75.0 | 10.6 | 10.5 | 2.087 | 532.7 | 564.1 |
| 9/20/07 | 12.800 | 2.709 | 2312.7 | 75.0 | 10.6 | 10.5 | 2.093 | 534.3 | 563.4 |
| 9/21/07 | 12.790 | 2.757 | 2314.4 | 75.0 | 10.7 | 10.6 | 2.116 | 538.9 | 559.6 |
| 9/22/07 | 12.800 | 2.774 | 2316.1 | 75.0 | 10.7 | 10.6 | 2.125 | 540.8 | 558.5 |
| 9/23/07 | 12.810 | 2.796 | 2317.7 | 75.0 | 10.7 | 10.6 | 2.136 | 543.1 | 557.0 |
| 9/24/07 | 12.830 | 2.858 | 2319.3 | 75.0 | 10.7 | 10.6 | 2.167 | 549.0 | 551.9 |
| 9/25/07 | 12.850 | 2.876 | 2320.8 | 75.0 | 10.7 | 10.6 | 2.176 | 551.0 | 550.6 |
| 9/26/07 | 12.890 | 2.901 | 2322.2 | 75.0 | 10.7 | 10.6 | 2.189 | 553.6 | 548.7 |
| 9/27/07 | 12.920 | 2.956 | 2323.7 | 75.0 | 10.7 | 10.6 | 2.217 | 558.9 | 544.2 |
| 9/28/07 | 12.950 | 3.008 | 2325.2 | 75.0 | 10.7 | 10.6 | 2.244 | 564.0 | 539.8 |
| 9/29/07 | 12.990 | 3.053 | 2327.0 | 75.0 | 10.7 | 10.6 | 2.268 | 568.5 | 536.2 |
| 9/30/07 | 13.040 | 3.142 | 2328.8 | 75.0 | 10.7 | 10.6 | 2.316 | 577.1 | 528.5 |
| 10/1/07 | 13.110 | 3.150 | 2330.1 | 75.0 | 10.7 | 10.6 | 2.320 | 578.1 | 528.1 |
| 10/2/07 | 12.990 | 3.235 | 2330.8 | 75.0 | 10.7 | 10.6 | 2.367 | 586.1 | 520.4 |

## Scenario 8 - Post-Capture Orbiter Mass (Delta III 3-stage) – Propulsive

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | MOI ΔV (km/s) | Prop Mass (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2061.4 | 75.0 | 9.5 | 9.4 | 2.025 | 463.4 | 511.0 |
| 9/14/07 | 13.130 | 2.578 | 2064.9 | 75.0 | 9.5 | 9.4 | 2.030 | 465.0 | 511.1 |
| 9/15/07 | 13.040 | 2.594 | 2067.9 | 75.0 | 9.5 | 9.4 | 2.038 | 467.0 | 510.6 |
| 9/16/07 | 12.960 | 2.637 | 2070.6 | 75.0 | 9.5 | 9.4 | 2.058 | 470.9 | 508.0 |
| 9/17/07 | 12.910 | 2.634 | 2072.9 | 75.0 | 9.5 | 9.4 | 2.057 | 471.2 | 508.8 |
| 9/18/07 | 12.860 | 2.662 | 2074.9 | 75.0 | 9.5 | 9.4 | 2.070 | 473.8 | 507.2 |
| 9/19/07 | 12.820 | 2.696 | 2076.7 | 75.0 | 9.5 | 9.4 | 2.087 | 476.9 | 505.0 |
| 9/20/07 | 12.800 | 2.709 | 2078.3 | 75.0 | 9.5 | 9.4 | 2.093 | 478.3 | 504.4 |
| 9/21/07 | 12.790 | 2.757 | 2079.7 | 75.0 | 9.5 | 9.4 | 2.116 | 482.4 | 500.9 |
| 9/22/07 | 12.800 | 2.774 | 2081.0 | 75.0 | 9.5 | 9.5 | 2.125 | 484.1 | 499.9 |
| 9/23/07 | 12.810 | 2.796 | 2082.3 | 75.0 | 9.5 | 9.5 | 2.136 | 486.1 | 498.5 |
| 9/24/07 | 12.830 | 2.858 | 2083.7 | 75.0 | 9.6 | 9.5 | 2.167 | 491.4 | 493.9 |
| 9/25/07 | 12.850 | 2.876 | 2084.9 | 75.0 | 9.6 | 9.5 | 2.176 | 493.1 | 492.8 |
| 9/26/07 | 12.890 | 2.901 | 2086.1 | 75.0 | 9.6 | 9.5 | 2.189 | 495.4 | 491.1 |
| 9/27/07 | 12.920 | 2.956 | 2087.3 | 75.0 | 9.6 | 9.5 | 2.217 | 500.1 | 486.9 |
| 9/28/07 | 12.950 | 3.008 | 2088.5 | 75.0 | 9.6 | 9.5 | 2.244 | 504.6 | 483.0 |
| 9/29/07 | 12.990 | 3.053 | 2090.0 | 75.0 | 9.6 | 9.5 | 2.268 | 508.7 | 479.7 |
| 9/30/07 | 13.040 | 3.142 | 2091.5 | 75.0 | 9.6 | 9.5 | 2.316 | 516.3 | 472.8 |
| 10/1/07 | 13.110 | 3.150 | 2092.6 | 75.0 | 9.6 | 9.5 | 2.320 | 517.2 | 472.4 |
| 10/2/07 | 12.990 | 3.235 | 2093.1 | 75.0 | 9.6 | 9.5 | 2.367 | 524.3 | 465.6 |

## Scenario 8 - Post-Capture Orbiter Mass (Atlas IIAS) – Aerocapture

| Launch | C3 | V inf at Arr | Inj Mass | Cruise Stage | TCMs (30 m/s) | TPS Mass | Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|
| | | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 9/13/07 | 13.240 | 2.567 | 1981.3 | 75 | 9.1 | 196.7 | 30.3 | 717.1 |
| 9/14/07 | 13.130 | 2.578 | 1985.3 | 75 | 9.1 | 196.8 | 30.4 | 718.9 |
| 9/15/07 | 13.040 | 2.594 | 1988.8 | 75 | 9.1 | 196.9 | 30.5 | 720.4 |
| 9/16/07 | 12.960 | 2.637 | 1991.9 | 75 | 9.1 | 197.3 | 30.5 | 721.5 |
| 9/17/07 | 12.910 | 2.634 | 1994.5 | 75 | 9.1 | 197.3 | 30.6 | 722.7 |
| 9/18/07 | 12.860 | 2.662 | 1996.7 | 75 | 9.1 | 197.6 | 30.6 | 723.6 |
| 9/19/07 | 12.820 | 2.696 | 1998.8 | 75 | 9.2 | 197.9 | 30.6 | 724.2 |
| 9/20/07 | 12.800 | 2.709 | 2000.6 | 75 | 9.2 | 198.1 | 30.7 | 724.9 |
| 9/21/07 | 12.790 | 2.757 | 2002.2 | 75 | 9.2 | 198.6 | 30.7 | 725.2 |
| 9/22/07 | 12.800 | 2.774 | 2003.7 | 75 | 9.2 | 198.8 | 30.7 | 725.7 |
| 9/23/07 | 12.810 | 2.796 | 2005.2 | 75 | 9.2 | 199.1 | 30.7 | 726.2 |
| 9/24/07 | 12.830 | 2.858 | 2006.8 | 75 | 9.2 | 199.8 | 30.7 | 726.1 |
| 9/25/07 | 12.850 | 2.876 | 2008.1 | 75 | 9.2 | 200.1 | 30.7 | 726.5 |
| 9/26/07 | 12.890 | 2.901 | 2009.5 | 75 | 9.2 | 200.4 | 30.7 | 726.9 |
| 9/27/07 | 12.920 | 2.956 | 2010.8 | 75 | 9.2 | 201.2 | 30.7 | 726.8 |
| 9/28/07 | 12.950 | 3.008 | 2012.2 | 75 | 9.2 | 202.0 | 30.7 | 726.7 |
| 9/29/07 | 12.990 | 3.053 | 2013.9 | 75 | 9.2 | 202.7 | 30.7 | 726.8 |
| 9/30/07 | 13.040 | 3.142 | 2015.6 | 75 | 9.2 | 204.2 | 30.7 | 726.1 |
| 10/1/07 | 13.110 | 3.150 | 2016.8 | 75 | 9.2 | 204.4 | 30.7 | 726.6 |
| 10/2/07 | 12.990 | 3.235 | 2017.4 | 75 | 9.2 | 205.9 | 30.7 | 725.4 |

## Scenario 8 - Post-Capture Orbiter Mass (Atlas IIAR) – Aerocapture

| Launch | C3 | V inf at Arr | Inj Mass | Cruise Stage | TCMs (30 m/s) | TPS Mass | Mvr | Post Capture |
|---|---|---|---|---|---|---|---|---|
| | | (km/s) | (kg) | (kg) | (kg) | (kg) | (kg) | (kg) |
| 9/13/07 | 13.240 | 2.567 | 2142.0 | 75 | 9.8 | 196.7 | 33.6 | 793.4 |
| 9/14/07 | 13.130 | 2.578 | 2146.1 | 75 | 9.9 | 196.8 | 33.6 | 795.3 |
| 9/15/07 | 13.040 | 2.594 | 2149.7 | 75 | 9.9 | 196.9 | 33.7 | 796.9 |
| 9/16/07 | 12.960 | 2.637 | 2152.9 | 75 | 9.9 | 197.3 | 33.8 | 798.0 |
| 9/17/07 | 12.910 | 2.634 | 2155.6 | 75 | 9.9 | 197.3 | 33.8 | 799.3 |
| 9/18/07 | 12.860 | 2.662 | 2157.9 | 75 | 9.9 | 197.6 | 33.8 | 800.1 |
| 9/19/07 | 12.820 | 2.696 | 2160.1 | 75 | 9.9 | 197.9 | 33.9 | 800.8 |
| 9/20/07 | 12.800 | 2.709 | 2161.9 | 75 | 9.9 | 198.1 | 33.9 | 801.5 |
| 9/21/07 | 12.790 | 2.757 | 2163.5 | 75 | 9.9 | 198.6 | 33.9 | 801.8 |
| 9/22/07 | 12.800 | 2.774 | 2165.1 | 75 | 9.9 | 198.8 | 33.9 | 802.4 |
| 9/23/07 | 12.810 | 2.796 | 2166.6 | 75 | 10.0 | 199.1 | 34.0 | 802.9 |
| 9/24/07 | 12.830 | 2.858 | 2168.2 | 75 | 10.0 | 199.8 | 34.0 | 802.9 |
| 9/25/07 | 12.850 | 2.876 | 2169.6 | 75 | 10.0 | 200.1 | 34.0 | 803.3 |
| 9/26/07 | 12.890 | 2.901 | 2171.0 | 75 | 10.0 | 200.4 | 34.0 | 803.6 |
| 9/27/07 | 12.920 | 2.956 | 2172.4 | 75 | 10.0 | 201.2 | 34.0 | 803.5 |
| 9/28/07 | 12.950 | 3.008 | 2173.8 | 75 | 10.0 | 202.0 | 34.0 | 803.5 |
| 9/29/07 | 12.990 | 3.053 | 2175.6 | 75 | 10.0 | 202.7 | 34.0 | 803.6 |
| 9/30/07 | 13.040 | 3.142 | 2177.4 | 75 | 10.0 | 204.2 | 34.0 | 803.0 |
| 10/1/07 | 13.110 | 3.150 | 2178.6 | 75 | 10.0 | 204.4 | 34.0 | 803.5 |
| 10/2/07 | 12.990 | 3.235 | 2179.2 | 75 | 10.0 | 205.9 | 33.9 | 802.3 |

### Scenario 8 - Post-Capture Orbiter Mass (Atlas IIARS) – Aerocapture

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (30 m/s) (kg) | TPS Mass (kg) | Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2292.1 | 75 | 10.5 | 196.7 | 36.6 | 864.7 |
| 9/14/07 | 13.130 | 2.578 | 2296.4 | 75 | 10.6 | 196.8 | 36.7 | 866.7 |
| 9/15/07 | 13.040 | 2.594 | 2300.1 | 75 | 10.6 | 196.9 | 36.7 | 868.3 |
| 9/16/07 | 12.960 | 2.637 | 2303.4 | 75 | 10.6 | 197.3 | 36.8 | 869.5 |
| 9/17/07 | 12.910 | 2.634 | 2306.2 | 75 | 10.6 | 197.3 | 36.8 | 870.8 |
| 9/18/07 | 12.860 | 2.662 | 2308.6 | 75 | 10.6 | 197.6 | 36.9 | 871.7 |
| 9/19/07 | 12.820 | 2.696 | 2310.8 | 75 | 10.6 | 197.9 | 36.9 | 872.5 |
| 9/20/07 | 12.800 | 2.709 | 2312.7 | 75 | 10.6 | 198.1 | 36.9 | 873.2 |
| 9/21/07 | 12.790 | 2.757 | 2314.4 | 75 | 10.7 | 198.6 | 36.9 | 873.5 |
| 9/22/07 | 12.800 | 2.774 | 2316.1 | 75 | 10.7 | 198.8 | 37.0 | 874.1 |
| 9/23/07 | 12.810 | 2.796 | 2317.7 | 75 | 10.7 | 199.1 | 37.0 | 874.6 |
| 9/24/07 | 12.830 | 2.858 | 2319.3 | 75 | 10.7 | 199.8 | 37.0 | 874.6 |
| 9/25/07 | 12.850 | 2.876 | 2320.8 | 75 | 10.7 | 200.1 | 37.0 | 875.1 |
| 9/26/07 | 12.890 | 2.901 | 2322.2 | 75 | 10.7 | 200.4 | 37.0 | 875.5 |
| 9/27/07 | 12.920 | 2.956 | 2323.7 | 75 | 10.7 | 201.2 | 37.0 | 875.4 |
| 9/28/07 | 12.950 | 3.008 | 2325.2 | 75 | 10.7 | 202.0 | 37.0 | 875.4 |
| 9/29/07 | 12.990 | 3.053 | 2327.0 | 75 | 10.7 | 202.7 | 37.0 | 875.6 |
| 9/30/07 | 13.040 | 3.142 | 2328.8 | 75 | 10.7 | 204.2 | 37.0 | 875.0 |
| 10/1/07 | 13.110 | 3.150 | 2330.1 | 75 | 10.7 | 204.4 | 37.0 | 875.5 |
| 10/2/07 | 12.990 | 3.235 | 2330.8 | 75 | 10.7 | 205.9 | 37.0 | 874.3 |

### Scenario 8 - Post-Capture Orbiter Mass (Atlas IIARS) – Aerocapture

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (30 m/s) (kg) | TPS Mass (kg) | Mvr (kg) | Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2061.4 | 75 | 9.5 | 196.7 | 31.9 | 755.1 |
| 9/14/07 | 13.130 | 2.578 | 2064.9 | 75 | 9.5 | 196.8 | 32.0 | 756.7 |
| 9/15/07 | 13.040 | 2.594 | 2067.9 | 75 | 9.5 | 196.9 | 32.1 | 758.0 |
| 9/16/07 | 12.960 | 2.637 | 2070.6 | 75 | 9.5 | 197.3 | 32.1 | 758.9 |
| 9/17/07 | 12.910 | 2.634 | 2072.9 | 75 | 9.5 | 197.3 | 32.1 | 760.0 |
| 9/18/07 | 12.860 | 2.662 | 2074.9 | 75 | 9.5 | 197.6 | 32.2 | 760.7 |
| 9/19/07 | 12.820 | 2.696 | 2076.7 | 75 | 9.5 | 197.9 | 32.2 | 761.2 |
| 9/20/07 | 12.800 | 2.709 | 2078.3 | 75 | 9.5 | 198.1 | 32.2 | 761.8 |
| 9/21/07 | 12.790 | 2.757 | 2079.7 | 75 | 9.5 | 198.6 | 32.2 | 762.0 |
| 9/22/07 | 12.800 | 2.774 | 2081.0 | 75 | 9.5 | 198.8 | 32.2 | 762.4 |
| 9/23/07 | 12.810 | 2.796 | 2082.3 | 75 | 9.5 | 199.1 | 32.3 | 762.8 |
| 9/24/07 | 12.830 | 2.858 | 2083.7 | 75 | 9.6 | 199.8 | 32.3 | 762.7 |
| 9/25/07 | 12.850 | 2.876 | 2084.9 | 75 | 9.6 | 200.1 | 32.3 | 763.0 |
| 9/26/07 | 12.890 | 2.901 | 2086.1 | 75 | 9.6 | 200.4 | 32.3 | 763.3 |
| 9/27/07 | 12.920 | 2.956 | 2087.3 | 75 | 9.6 | 201.2 | 32.3 | 763.1 |
| 9/28/07 | 12.950 | 3.008 | 2088.5 | 75 | 9.6 | 202.0 | 32.3 | 762.9 |
| 9/29/07 | 12.990 | 3.053 | 2090.0 | 75 | 9.6 | 202.7 | 32.3 | 762.9 |
| 9/30/07 | 13.040 | 3.142 | 2091.5 | 75 | 9.6 | 204.2 | 32.2 | 762.2 |
| 10/1/07 | 13.110 | 3.150 | 2092.6 | 75 | 9.6 | 204.4 | 32.3 | 762.6 |
| 10/2/07 | 12.990 | 3.235 | 2093.1 | 75 | 9.6 | 205.9 | 32.2 | 761.3 |

## Scenario 8 - Post-Capture Orbiter Mass (Atlas IIAS) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 1981.3 | 75.0 | 9.1 | 9.1 | 0.946 | 0.843 |
| 9/14/07 | 13.130 | 2.578 | 1985.3 | 75.0 | 9.1 | 9.1 | 0.951 | 0.849 |
| 9/15/07 | 13.040 | 2.594 | 1988.8 | 75.0 | 9.1 | 9.1 | 0.959 | 0.856 |
| 9/16/07 | 12.960 | 2.637 | 1991.9 | 75.0 | 9.1 | 9.1 | 0.979 | 0.876 |
| 9/17/07 | 12.910 | 2.634 | 1994.5 | 75.0 | 9.1 | 9.1 | 0.978 | 0.875 |
| 9/18/07 | 12.860 | 2.662 | 1996.7 | 75.0 | 9.1 | 9.1 | 0.991 | 0.888 |
| 9/19/07 | 12.820 | 2.696 | 1998.8 | 75.0 | 9.2 | 9.2 | 1.008 | 0.905 |
| 9/20/07 | 12.800 | 2.709 | 2000.6 | 75.0 | 9.2 | 9.2 | 1.014 | 0.911 |
| 9/21/07 | 12.790 | 2.757 | 2002.2 | 75.0 | 9.2 | 9.2 | 1.037 | 0.935 |
| 9/22/07 | 12.800 | 2.774 | 2003.7 | 75.0 | 9.2 | 9.2 | 1.046 | 0.943 |
| 9/23/07 | 12.810 | 2.796 | 2005.2 | 75.0 | 9.2 | 9.2 | 1.057 | 0.954 |
| 9/24/07 | 12.830 | 2.858 | 2006.8 | 75.0 | 9.2 | 9.2 | 1.088 | 0.985 |
| 9/25/07 | 12.850 | 2.876 | 2008.1 | 75.0 | 9.2 | 9.2 | 1.097 | 0.994 |
| 9/26/07 | 12.890 | 2.901 | 2009.5 | 75.0 | 9.2 | 9.2 | 1.110 | 1.007 |
| 9/27/07 | 12.920 | 2.956 | 2010.8 | 75.0 | 9.2 | 9.2 | 1.138 | 1.036 |
| 9/28/07 | 12.950 | 3.008 | 2012.2 | 75.0 | 9.2 | 9.2 | 1.165 | 1.063 |
| 9/29/07 | 12.990 | 3.053 | 2013.9 | 75.0 | 9.2 | 9.2 | 1.189 | 1.087 |
| 9/30/07 | 13.040 | 3.142 | 2015.6 | 75.0 | 9.2 | 9.2 | 1.237 | 1.134 |
| 10/1/07 | 13.110 | 3.150 | 2016.8 | 75.0 | 9.2 | 9.2 | 1.241 | 1.139 |
| 10/2/07 | 12.990 | 3.235 | 2017.4 | 75.0 | 9.2 | 9.2 | 1.288 | 1.185 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 0.798 | 245.8 | 222.5 | 211.9 | 50 | 639.2 | 662.5 | 673.1 |
| 9/14/07 | 0.803 | 247.4 | 224.2 | 213.6 | 50 | 639.6 | 662.8 | 673.4 |
| 9/15/07 | 0.810 | 249.6 | 226.3 | 215.7 | 50 | 639.1 | 662.4 | 673.0 |
| 9/16/07 | 0.831 | 254.5 | 231.4 | 220.8 | 50 | 635.7 | 658.9 | 669.4 |
| 9/17/07 | 0.829 | 254.5 | 231.3 | 220.8 | 50 | 637.0 | 660.1 | 670.7 |
| 9/18/07 | 0.843 | 257.8 | 234.7 | 224.1 | 50 | 634.8 | 657.9 | 668.5 |
| 9/19/07 | 0.859 | 261.7 | 238.7 | 228.2 | 50 | 631.9 | 654.9 | 665.4 |
| 9/20/07 | 0.865 | 263.3 | 240.3 | 229.8 | 50 | 631.2 | 654.1 | 664.6 |
| 9/21/07 | 0.889 | 268.7 | 245.9 | 235.4 | 50 | 626.6 | 649.4 | 659.8 |
| 9/22/07 | 0.897 | 270.7 | 248.0 | 237.6 | 50 | 625.3 | 648.1 | 658.5 |
| 9/23/07 | 0.908 | 273.3 | 250.6 | 240.2 | 50 | 623.4 | 646.1 | 656.5 |
| 9/24/07 | 0.939 | 280.3 | 257.8 | 247.5 | 50 | 617.2 | 639.7 | 650.0 |
| 9/25/07 | 0.949 | 282.5 | 260.0 | 249.7 | 50 | 615.7 | 638.2 | 648.4 |
| 9/26/07 | 0.961 | 285.4 | 263.0 | 252.8 | 50 | 613.4 | 635.8 | 646.0 |
| 9/27/07 | 0.990 | 291.7 | 269.5 | 259.3 | 50 | 607.8 | 630.0 | 640.2 |
| 9/28/07 | 1.017 | 297.6 | 275.6 | 265.6 | 50 | 602.5 | 624.6 | 634.6 |
| 9/29/07 | 1.041 | 302.9 | 281.0 | 271.0 | 50 | 598.1 | 620.0 | 630.0 |
| 9/30/07 | 1.089 | 313.1 | 291.6 | 281.7 | 50 | 588.7 | 610.3 | 620.1 |
| 10/1/07 | 1.093 | 314.2 | 292.7 | 282.8 | 50 | 588.2 | 609.8 | 619.6 |
| 10/2/07 | 1.140 | 323.9 | 302.6 | 292.9 | 50 | 578.8 | 600.1 | 609.8 |

# Scenario 8 - Post-Capture Orbiter Mass (Atlas IIAR) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2142.0 | 75.0 | 9.8 | 9.8 | 0.946 | 0.843 |
| 9/14/07 | 13.130 | 2.578 | 2146.1 | 75.0 | 9.9 | 9.9 | 0.951 | 0.849 |
| 9/15/07 | 13.040 | 2.594 | 2149.7 | 75.0 | 9.9 | 9.9 | 0.959 | 0.856 |
| 9/16/07 | 12.960 | 2.637 | 2152.9 | 75.0 | 9.9 | 9.9 | 0.979 | 0.876 |
| 9/17/07 | 12.910 | 2.634 | 2155.6 | 75.0 | 9.9 | 9.9 | 0.978 | 0.875 |
| 9/18/07 | 12.860 | 2.662 | 2157.9 | 75.0 | 9.9 | 9.9 | 0.991 | 0.888 |
| 9/19/07 | 12.820 | 2.696 | 2160.1 | 75.0 | 9.9 | 9.9 | 1.008 | 0.905 |
| 9/20/07 | 12.800 | 2.709 | 2161.9 | 75.0 | 9.9 | 9.9 | 1.014 | 0.911 |
| 9/21/07 | 12.790 | 2.757 | 2163.5 | 75.0 | 9.9 | 9.9 | 1.037 | 0.935 |
| 9/22/07 | 12.800 | 2.774 | 2165.1 | 75.0 | 9.9 | 9.9 | 1.046 | 0.943 |
| 9/23/07 | 12.810 | 2.796 | 2166.6 | 75.0 | 10.0 | 10.0 | 1.057 | 0.954 |
| 9/24/07 | 12.830 | 2.858 | 2168.2 | 75.0 | 10.0 | 10.0 | 1.088 | 0.985 |
| 9/25/07 | 12.850 | 2.876 | 2169.6 | 75.0 | 10.0 | 10.0 | 1.097 | 0.994 |
| 9/26/07 | 12.890 | 2.901 | 2171.0 | 75.0 | 10.0 | 10.0 | 1.110 | 1.007 |
| 9/27/07 | 12.920 | 2.956 | 2172.4 | 75.0 | 10.0 | 10.0 | 1.138 | 1.036 |
| 9/28/07 | 12.950 | 3.008 | 2173.8 | 75.0 | 10.0 | 10.0 | 1.165 | 1.063 |
| 9/29/07 | 12.990 | 3.053 | 2175.6 | 75.0 | 10.0 | 10.0 | 1.189 | 1.087 |
| 9/30/07 | 13.040 | 3.142 | 2177.4 | 75.0 | 10.0 | 10.0 | 1.237 | 1.134 |
| 10/1/07 | 13.110 | 3.150 | 2178.6 | 75.0 | 10.0 | 10.0 | 1.241 | 1.139 |
| 10/2/07 | 12.990 | 3.235 | 2179.2 | 75.0 | 10.0 | 10.0 | 1.288 | 1.185 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 0.798 | 266.5 | 241.3 | 229.8 | 50 | 697.3 | 722.6 | 734.1 |
| 9/14/07 | 0.803 | 268.3 | 243.0 | 231.5 | 50 | 697.6 | 722.8 | 734.3 |
| 9/15/07 | 0.810 | 270.5 | 245.3 | 233.8 | 50 | 697.1 | 722.3 | 733.8 |
| 9/16/07 | 0.831 | 275.9 | 250.8 | 239.3 | 50 | 693.3 | 718.4 | 729.8 |
| 9/17/07 | 0.829 | 275.9 | 250.8 | 239.3 | 50 | 694.6 | 719.7 | 731.2 |
| 9/18/07 | 0.843 | 279.4 | 254.4 | 242.9 | 50 | 692.2 | 717.3 | 728.7 |
| 9/19/07 | 0.859 | 283.6 | 258.7 | 247.3 | 50 | 689.1 | 714.0 | 725.4 |
| 9/20/07 | 0.865 | 285.4 | 260.5 | 249.1 | 50 | 688.2 | 713.1 | 724.5 |
| 9/21/07 | 0.889 | 291.2 | 266.4 | 255.1 | 50 | 683.2 | 707.9 | 719.2 |
| 9/22/07 | 0.897 | 293.4 | 268.7 | 257.4 | 50 | 681.8 | 706.5 | 717.7 |
| 9/23/07 | 0.908 | 296.2 | 271.6 | 260.3 | 50 | 679.7 | 704.3 | 715.6 |
| 9/24/07 | 0.939 | 303.7 | 279.3 | 268.2 | 50 | 673.0 | 697.4 | 708.5 |
| 9/25/07 | 0.949 | 306.1 | 281.7 | 270.6 | 50 | 671.3 | 695.7 | 706.8 |
| 9/26/07 | 0.961 | 309.3 | 285.0 | 273.9 | 50 | 668.8 | 693.1 | 704.2 |
| 9/27/07 | 0.990 | 316.0 | 292.0 | 281.0 | 50 | 662.7 | 686.8 | 697.8 |
| 9/28/07 | 1.017 | 322.5 | 298.6 | 287.7 | 50 | 657.0 | 680.8 | 691.7 |
| 9/29/07 | 1.041 | 328.2 | 304.5 | 293.6 | 50 | 652.2 | 675.9 | 686.7 |
| 9/30/07 | 1.089 | 339.2 | 315.9 | 305.2 | 50 | 641.9 | 665.3 | 676.0 |
| 10/1/07 | 1.093 | 340.4 | 317.1 | 306.4 | 50 | 641.4 | 664.7 | 675.4 |
| 10/2/07 | 1.140 | 350.9 | 327.9 | 317.3 | 50 | 631.2 | 654.2 | 664.8 |

# Scenario 8 - Post-Capture Orbiter Mass (Atlas IIARS) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2292.1 | 75.0 | 10.5 | 10.5 | 0.946 | 0.843 |
| 9/14/07 | 13.130 | 2.578 | 2296.4 | 75.0 | 10.6 | 10.6 | 0.951 | 0.849 |
| 9/15/07 | 13.040 | 2.594 | 2300.1 | 75.0 | 10.6 | 10.6 | 0.959 | 0.856 |
| 9/16/07 | 12.960 | 2.637 | 2303.4 | 75.0 | 10.6 | 10.6 | 0.979 | 0.876 |
| 9/17/07 | 12.910 | 2.634 | 2306.2 | 75.0 | 10.6 | 10.6 | 0.978 | 0.875 |
| 9/18/07 | 12.860 | 2.662 | 2308.6 | 75.0 | 10.6 | 10.6 | 0.991 | 0.888 |
| 9/19/07 | 12.820 | 2.696 | 2310.8 | 75.0 | 10.6 | 10.6 | 1.008 | 0.905 |
| 9/20/07 | 12.800 | 2.709 | 2312.7 | 75.0 | 10.6 | 10.6 | 1.014 | 0.911 |
| 9/21/07 | 12.790 | 2.757 | 2314.4 | 75.0 | 10.7 | 10.7 | 1.037 | 0.935 |
| 9/22/07 | 12.800 | 2.774 | 2316.1 | 75.0 | 10.7 | 10.7 | 1.046 | 0.943 |
| 9/23/07 | 12.810 | 2.796 | 2317.7 | 75.0 | 10.7 | 10.7 | 1.057 | 0.954 |
| 9/24/07 | 12.830 | 2.858 | 2319.3 | 75.0 | 10.7 | 10.7 | 1.088 | 0.985 |
| 9/25/07 | 12.850 | 2.876 | 2320.8 | 75.0 | 10.7 | 10.7 | 1.097 | 0.994 |
| 9/26/07 | 12.890 | 2.901 | 2322.2 | 75.0 | 10.7 | 10.7 | 1.110 | 1.007 |
| 9/27/07 | 12.920 | 2.956 | 2323.7 | 75.0 | 10.7 | 10.7 | 1.138 | 1.036 |
| 9/28/07 | 12.950 | 3.008 | 2325.2 | 75.0 | 10.7 | 10.7 | 1.165 | 1.063 |
| 9/29/07 | 12.990 | 3.053 | 2327.0 | 75.0 | 10.7 | 10.7 | 1.189 | 1.087 |
| 9/30/07 | 13.040 | 3.142 | 2328.8 | 75.0 | 10.7 | 10.7 | 1.237 | 1.134 |
| 10/1/07 | 13.110 | 3.150 | 2330.1 | 75.0 | 10.7 | 10.7 | 1.241 | 1.139 |
| 10/2/07 | 12.990 | 3.235 | 2330.8 | 75.0 | 10.7 | 10.7 | 1.288 | 1.185 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 0.798 | 285.8 | 258.8 | 246.4 | 50 | 751.6 | 778.6 | 791.0 |
| 9/14/07 | 0.803 | 287.7 | 260.7 | 248.3 | 50 | 751.8 | 778.9 | 791.2 |
| 9/15/07 | 0.810 | 290.2 | 263.1 | 250.8 | 50 | 751.2 | 778.3 | 790.6 |
| 9/16/07 | 0.831 | 295.8 | 269.0 | 256.7 | 50 | 747.2 | 774.0 | 786.3 |
| 9/17/07 | 0.829 | 295.8 | 268.9 | 256.6 | 50 | 748.5 | 775.5 | 787.8 |
| 9/18/07 | 0.843 | 299.6 | 272.8 | 260.5 | 50 | 745.9 | 772.8 | 785.0 |
| 9/19/07 | 0.859 | 304.1 | 277.4 | 265.2 | 50 | 742.5 | 769.3 | 781.5 |
| 9/20/07 | 0.865 | 306.0 | 279.3 | 267.1 | 50 | 741.6 | 768.3 | 780.5 |
| 9/21/07 | 0.889 | 312.2 | 285.7 | 273.6 | 50 | 736.2 | 762.7 | 774.8 |
| 9/22/07 | 0.897 | 314.6 | 288.1 | 276.0 | 50 | 734.6 | 761.1 | 773.2 |
| 9/23/07 | 0.908 | 317.6 | 291.2 | 279.1 | 50 | 732.4 | 758.8 | 770.9 |
| 9/24/07 | 0.939 | 325.7 | 299.5 | 287.6 | 50 | 725.1 | 751.3 | 763.3 |
| 9/25/07 | 0.949 | 328.2 | 302.1 | 290.1 | 50 | 723.4 | 749.5 | 761.4 |
| 9/26/07 | 0.961 | 331.6 | 305.6 | 293.7 | 50 | 720.7 | 746.7 | 758.6 |
| 9/27/07 | 0.990 | 338.8 | 313.0 | 301.2 | 50 | 714.1 | 739.9 | 751.7 |
| 9/28/07 | 1.017 | 345.7 | 320.1 | 308.5 | 50 | 707.9 | 733.5 | 745.2 |
| 9/29/07 | 1.041 | 351.8 | 326.4 | 314.8 | 50 | 702.8 | 728.2 | 739.8 |
| 9/30/07 | 1.089 | 363.7 | 338.6 | 327.2 | 50 | 691.8 | 716.8 | 728.3 |
| 10/1/07 | 1.093 | 364.9 | 339.9 | 328.5 | 50 | 691.2 | 716.2 | 727.6 |
| 10/2/07 | 1.140 | 376.1 | 351.5 | 340.2 | 50 | 680.3 | 705.0 | 716.2 |

## Scenario 8 - Post-Capture Orbiter Mass (Delta III 3-stage) – Aerobraking

| Launch | C3 | V inf at Arr (km/s) | Inj Mass (kg) | Cruise Stage (kg) | TCMs (kg) | MDM (kg) | 15 hr orbit MOI ΔV (km/s) | 27 hr orbit MOI ΔV (km/s) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 13.240 | 2.567 | 2061.4 | 75.0 | 9.5 | 9.5 | 0.946 | 0.843 |
| 9/14/07 | 13.130 | 2.578 | 2064.9 | 75.0 | 9.5 | 9.5 | 0.951 | 0.849 |
| 9/15/07 | 13.040 | 2.594 | 2067.9 | 75.0 | 9.5 | 9.5 | 0.959 | 0.856 |
| 9/16/07 | 12.960 | 2.637 | 2070.6 | 75.0 | 9.5 | 9.5 | 0.979 | 0.876 |
| 9/17/07 | 12.910 | 2.634 | 2072.9 | 75.0 | 9.5 | 9.5 | 0.978 | 0.875 |
| 9/18/07 | 12.860 | 2.662 | 2074.9 | 75.0 | 9.5 | 9.5 | 0.991 | 0.888 |
| 9/19/07 | 12.820 | 2.696 | 2076.7 | 75.0 | 9.5 | 9.5 | 1.008 | 0.905 |
| 9/20/07 | 12.800 | 2.709 | 2078.3 | 75.0 | 9.5 | 9.5 | 1.014 | 0.911 |
| 9/21/07 | 12.790 | 2.757 | 2079.7 | 75.0 | 9.5 | 9.5 | 1.037 | 0.935 |
| 9/22/07 | 12.800 | 2.774 | 2081.0 | 75.0 | 9.5 | 9.5 | 1.046 | 0.943 |
| 9/23/07 | 12.810 | 2.796 | 2082.3 | 75.0 | 9.5 | 9.5 | 1.057 | 0.954 |
| 9/24/07 | 12.830 | 2.858 | 2083.7 | 75.0 | 9.6 | 9.6 | 1.088 | 0.985 |
| 9/25/07 | 12.850 | 2.876 | 2084.9 | 75.0 | 9.6 | 9.6 | 1.097 | 0.994 |
| 9/26/07 | 12.890 | 2.901 | 2086.1 | 75.0 | 9.6 | 9.6 | 1.110 | 1.007 |
| 9/27/07 | 12.920 | 2.956 | 2087.3 | 75.0 | 9.6 | 9.6 | 1.138 | 1.036 |
| 9/28/07 | 12.950 | 3.008 | 2088.5 | 75.0 | 9.6 | 9.6 | 1.165 | 1.063 |
| 9/29/07 | 12.990 | 3.053 | 2090.0 | 75.0 | 9.6 | 9.6 | 1.189 | 1.087 |
| 9/30/07 | 13.040 | 3.142 | 2091.5 | 75.0 | 9.6 | 9.6 | 1.237 | 1.134 |
| 10/1/07 | 13.110 | 3.150 | 2092.6 | 75.0 | 9.6 | 9.6 | 1.241 | 1.139 |
| 10/2/07 | 12.990 | 3.235 | 2093.1 | 75.0 | 9.6 | 9.6 | 1.288 | 1.185 |

| Launch | 39 hr orbit MOI ΔV (km/s) | 15 hr orbit Prop Mass (kg) | 27 hr orbit Prop (kg) | 39 hr orbit Prop Mass (kg) | OTMs (kg) | 15 hr orbit Post Capture (kg) | 27 hr orbit Post Capture (kg) | 39 hr orbit Post Capture (kg) |
|---|---|---|---|---|---|---|---|---|
| 9/13/07 | 0.798 | 256.1 | 231.9 | 220.8 | 50 | 668.2 | 692.4 | 703.5 |
| 9/14/07 | 0.803 | 257.7 | 233.5 | 222.4 | 50 | 668.3 | 692.5 | 703.6 |
| 9/15/07 | 0.810 | 259.9 | 235.7 | 224.6 | 50 | 667.6 | 691.8 | 702.9 |
| 9/16/07 | 0.831 | 264.9 | 240.9 | 229.9 | 50 | 663.9 | 688.0 | 699.0 |
| 9/17/07 | 0.829 | 264.9 | 240.8 | 229.8 | 50 | 665.0 | 689.1 | 700.2 |
| 9/18/07 | 0.843 | 268.3 | 244.2 | 233.2 | 50 | 662.6 | 686.7 | 697.7 |
| 9/19/07 | 0.859 | 272.3 | 248.3 | 237.4 | 50 | 659.5 | 683.5 | 694.4 |
| 9/20/07 | 0.865 | 273.9 | 250.0 | 239.1 | 50 | 658.6 | 682.5 | 693.5 |
| 9/21/07 | 0.889 | 279.5 | 255.7 | 244.9 | 50 | 653.8 | 677.5 | 688.3 |
| 9/22/07 | 0.897 | 281.6 | 257.9 | 247.1 | 50 | 652.3 | 676.0 | 686.9 |
| 9/23/07 | 0.908 | 284.3 | 260.6 | 249.8 | 50 | 650.3 | 673.9 | 684.7 |
| 9/24/07 | 0.939 | 291.5 | 268.1 | 257.4 | 50 | 643.8 | 667.2 | 677.9 |
| 9/25/07 | 0.949 | 293.7 | 270.3 | 259.7 | 50 | 642.1 | 665.5 | 676.1 |
| 9/26/07 | 0.961 | 296.7 | 273.4 | 262.8 | 50 | 639.7 | 663.0 | 673.6 |
| 9/27/07 | 0.990 | 303.2 | 280.1 | 269.6 | 50 | 633.8 | 656.9 | 667.4 |
| 9/28/07 | 1.017 | 309.4 | 286.5 | 276.0 | 50 | 628.2 | 651.1 | 661.6 |
| 9/29/07 | 1.041 | 314.8 | 292.1 | 281.7 | 50 | 623.5 | 646.3 | 656.7 |
| 9/30/07 | 1.089 | 325.4 | 303.0 | 292.7 | 50 | 613.7 | 636.1 | 646.3 |
| 10/1/07 | 1.093 | 326.5 | 304.1 | 293,9 | 50 | 613.1 | 635.5 | 645.7 |
| 10/2/07 | 1.140 | 336.5 | 314.4 | 304.3 | 50 | 603.3 | 625.4 | 635.5 |

# APPENDIX E. EXAMPLE PROGRAMS

## QUICK Program to calculate the Sun-spacecraft, Earth-spacecraft, Earth-Mars distances and Sun-spacecraft-Earth angle

```
scgeom_061126_orbiter.q

@ program to calculate geometry for Earth, Mars, and S/C and SPE Angle

jdlaunch=date(061126)
jdarrive=date(090123)

tfl=jdarrive-jdlaunch
time=tfl*86400
cplann(81,0)
earthpos=bodpos(jdlaunch,3,0)
marspos=bodpos(jdarrive,4,0)
orbit=orbfit(earthpos,marspos,time,-1,'')
orbprt(81,0)

d=jdlaunch+seq((tfl+1)-1)

@ Calculate the Earth-Mars distance

marspost=bodpos(d,4,0)
earpost=bodpos(d,3,0)
eamardis=absv(marspost-earpost)

@ Calculate appropriate vectors

scsunvec=orbpos((d-jdlaunch)*86400)
scsun=absv(scsunvec)
easunvec=(bodpos(d,3,0))
earthsun=absv(easunvec)

earthsc=absv(easunvec-scsunvec)
snscvec=-scsunvec
eascvec=scsunvec-easunvec
spe=angv(snscvec,-eascvec)

@ Determine initial vector

scsnvec0=orbpos(0)
easnvec0=bodpos(jdlaunch,3,0)
earthsc0=absv(scsnvec0-easnvec0)

;
```

## MATLAB PROGRAM FOR TIME REQUIRED TO AEROBRAKE

```
% Program to estimate the time required to aerobrake
% Steve Zike
% NPS 1998

% Constants and parameters
```

```
time=0;                        % initialization
count=1;
mu=42828.3;                    % gravitational constant for Mars
Rm=3393.4;                     % Radius for Mars
Re=3518.4;                     % Radius for atmosphere interface
Cd=2;
m=450;
A=15;
BC=(Cd*A)/m;                   % Ballistic coefficient

% Inputs for density at periapsis, semi-major axis, and periapsis alt

rho=input('enter the density at the periapsis altitude (in kg/m^3) ');
a=input('enter the semi-major axis of the capture orbit (in km) ');
rp_alt=input('enter the periapsis altitude (in km) ');

% Loop to iteratively solve for the time
% Routine halts when semi-major axis is less than 2000 km assuming
% the perigee remains constant

while a > 5393.4

% Calculate the period

P(count)=((2*pi)*sqrt((a^3)/mu))/60;          % Period in minutes

% Calculate the eccentricity

rp=Rm+rp_alt;

e=1-(rp/a);

% Calculate the true anomaly

nu=(acos(((a*(1-e^2)/Re)-1)/e));

% Calculate the angle traversed during aerobraking

theta=2*nu;

% Calculate the velocity at the interface points
% on the elliptical orbit

V_1_2=sqrt(((2*mu)/Re)-(mu/a));

% Calculate the change in energy

Delta_E=0.5*BC*rho*(V_1_2^2)*rp*1e3*theta;

% Calculate the energy of the new orbit

Init_E=-mu/(2*a);
Final_E=Init_E-Delta_E;              % New orbit energy < old orbit

% Calculate the change in the semi-major axis due to the energy change
```

```
a_new=-mu/(2*Final_E);
Delta_a=a-a_new;
a=a_new;
time=time + P(count);
count=count + 1;

end

number_of_passes=count-1;
number_of_passes
total_ab_time=time/1440; % Aerobraking time in days
total_ab_time
```

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