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LAUNCH PERIOD ANALYSIS FOR THE JUPITER GRAVITY ASSIST OPPORTUNITIES TO PLUTO

by

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Pluto remains the last outer planet as yet unsurveyed by any passing spacecraft. The spacecraft, Pluto-Kuiper Express, is part of an approach by NASA to build, smaller, better, cheaper satellites for future space exploration. NASA's Jet Propulsion Laboratory is designing a mission that will conduct reconnaissance of the Pluto/Charon system, determining their composition, atmosphere, and geological characteristics. If successful, the spacecraft will be sent to observe objects in the Kuiper Belt, laying just beyond the boundary of the solar system. To reach Pluto in a reasonable time frame at the lowest cost, several trajectory options must be carefully considered. This thesis presents a comprehensive analysis of a trajectory consisting of a Jupiter Gravity Assist flyby to Pluto. JPL specified two nominal launch dates of November 2003 and December 2004. The daily C_3 requirements for these dates were determined by using the JPL programs MIDAS and CATO. This facilitated the creation of nominal launch periods for these two dates. By comparing the launch energy required by the trajectory on each day of the period to the performance capabilities of several medium-lift launch vehicles, launch strategies for each day were compiled. These results allow JPL to make the final decision of the most feasible arrangement for launch, and build an alternate launch plan should the primary become unavailable.

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ABSTRACT

Pluto remains the last outer planet as yet unsurveyed by any passing spacecraft. The spacecraft, Pluto-Kuiper Express, is part of an approach by NASA to build, smaller, better, cheaper satellites for future space exploration. NASA's Jet Propulsion Laboratory is designing a mission that will conduct reconnaissance of the Pluto/Charon system, determining their composition, atmosphere, and geological characteristics. If successful, the spacecraft will be sent to observe objects in the Kuiper Belt, laying just beyond the boundary of the solar system. To reach Pluto in a reasonable time frame at the lowest cost, several trajectory options must be carefully considered. This thesis presents a comprehensive analysis of a trajectory consisting of a Jupiter Gravity Assist flyby to Pluto. JPL specified two nominal launch dates of November 2003 and December 2004. The daily C_3 requirements for these dates were determined by using the JPL programs MIDAS and CATO. This facilitated the creation of nominal launch periods for these two dates. By comparing the launch energy required by the trajectory on each day of the period to the performance capabilities of several medium-lift launch vehicles, launch strategies for each day were compiled. These results allow JPL to make the final decision of the most feasible arrangement for launch, and build an alternate launch plan should the primary become unavailable.

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LIST OF SYMBOLS, ACRONYMS, AND ABBREVIATIONS

ΔV	Change in Velocity Required to Perform Desired Maneuver (km/s)
C_3	$(V_\infty)^2$ (km^2/s^2)
CATO	Computer Algorithm for Trajectory Optimization
ESA	European Space Agency
JGA	Jupiter Gravity Assist
JPL	Jet Propulsion Laboratory
GEO	Geosynchronous Orbit
GTO	Geosynchronous Transfer Orbit
LEO	Low Earth Orbit
MASL	Mission Analysis Support Library
MIDAS	Mission Integrated Design and Analysis Software
PAM	Payload Assist Module
PKE	Pluto-Kuiper Express Spacecraft
SEP	Solar-Electric Propulsion
SE-VGA	Solar-Electric Venus Gravity Assist
SE-VVJGA	Solar-Electric Venus-Venus Jupiter Gravity Assist
V_∞	Velocity at Infinity (km/s)
VVJGA	Venus-Venus Jupiter Gravity Assist
VVVJGA	Venus-Venus-Venus Jupiter Gravity Assist

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I. INTRODUCTION

A. BACKGROUND

Within the last decade, exploration of the outer planets has become of increasing interest to both the space science community and the public. The *Voyager* flybys of the 1970's and 1980's afforded a glimpse of what lay beyond Mars. The rough pictures from the two *Voyager* satellites provide the basis for a new plan to perform detailed observation and analysis of Jupiter, Saturn, Uranus, Neptune, and Pluto. NASA's Jet Propulsion Laboratory (JPL) has taken the lead in the exploration of the outer planets. This new era began as the *Galileo* satellite transmitted back its first images of Jupiter and a few of her larger moons. The possible ice fractures observed on Europa give hope to the prospect of water and life on that moon. JPL is currently planning a separate mission to determine exactly what lies on Europa. The *Cassini* spacecraft is scheduled to launch for Saturn in October 1997, and will further our understanding of the composition of the giant planets. Neptune and Uranus have yet to be named in separate space missions, but they were observed by both *Voyagers* as they flew out of the solar system. There was an opportunity for *Voyager I* to perform a Pluto flyby, but it would have precluded a close encounter with Saturn's moon, Titan. The results of this flyby eventually inspired the *Cassini* mission, so the decision to forgo Pluto was accepted and appreciated by the scientific community (Lunine, 1995, p.7).

Pluto remains the only outer planet not yet observed by any passing spacecraft. Scientists are intrigued by Pluto because it appears to be more of a cold, solid planet rather than a gas giant like Jupiter and Saturn. Pluto's high surface albedo, most likely due to surface methane or nitrogen ice, allows for some Earth-based observation of the planet, but any information about the surface geology or subsurface composition is speculation. Pluto's atmosphere is thought to be very dynamic and transient, dependent upon Pluto's relative position to the Sun; its most recent closest approach occurred in 1989. Pluto's atmosphere is probably in a great state of change, and therefore of great interest to scientists. There is also a desire to determine the exact values of the basic orbital parameters, including the semi-major axis and mean motion. Some rough calculations have been performed using ground-based observations and data from the *Hubble Space Telescope*, but they are difficult to verify since all the observational data extends back only to 1915, and Pluto has a 248-year orbital period. Only about one third of the orbital period has been observed to present (Lunine, 1995, p. 6-10).

Scientists are also curious to learn more about Pluto's moon, Charon. Very little is known about its surface appearance, composition, and origin. Charon's surface is not as reflective as Pluto's, but spectroscopic observations suggest it is largely covered by frozen water, with additional ice constituents. An initial hypothesis was that the moon was actually a captured asteroid. There has been recent speculation that Charon is not a moon at all, but part of a binary planet system resulting from a catastrophic planetary collision (McKinnon, 1989, p. L41), consisting of Pluto and Charon, revolving around the barycenter. Scientists also theorize that Pluto and Charon may provide considerable information about the early periods of planetary evolution, and perhaps even the origin of Earth's atmosphere.

A third area of interest at the outer edge of the solar system is the recently discovered band of small bodies residing just outside the planetary region. This region is widely believed to be the theorized Kuiper Belt. The objects residing in this region are most likely raw materials and remnants of the material that formed the solar system. They may contain many clues as to how the planets were formed and then placed into their stable circular orbits, rather than the highly elliptical orbits of the comets that travel very close to the Sun. The Kuiper Belt may be the source of short-period comets, in much the same way that the Oort Cloud acts as a reservoir for long-period comets.

NASA is currently developing a group of projects whose missions are to explore the hottest and coldest regions of the solar system. Dubbed the "Ice and Fire Preprojects," they include missions to explore Pluto, Jupiter's icy moon Europa, and the Sun. JPL is currently developing the "Ice" portion (as well as the "Fire"), a mission to travel to Pluto and Charon and perform various science reconnaissance experiments in order to determine their content, atmosphere, and characteristics. In addition, if the flyby of Pluto and Charon is successful, the spacecraft will continue beyond Pluto to encounter and study one or more objects in the Kuiper Belt region. The spacecraft, *Pluto-Kuiper Express* (*PKE*), is one of a series of small, low-cost, high technology spacecraft, following the *Sagan Memorial Station* (*Mars Pathfinder*) and the *New Millennium* projects. *Galileo* and *Cassini* are the last of the behemoth planetary exploration satellites. The *PKE* is part of the NASA/JPL approach that incorporates new technologies in hardware and software to reduce cost, mass, power, and volume, without jeopardizing any performance, science, or operational capabilities (Price et al., p. 1). The result is a highly integrated system that combines a spacecraft with several highly technical science instruments, called a "sciencecraft" (Lunine, 1995, p. 36). The next generation of sciencecraft will be very low mass, 100 kilograms or less, when compared to *Cassini*'s mass of 5000 kilograms. Figure 1 displays a size comparison of the *PKE* with some of the other outer-planets spacecraft. At approximately two meters in length, *PKE* is dwarfed by *Cassini* and *Galileo*.

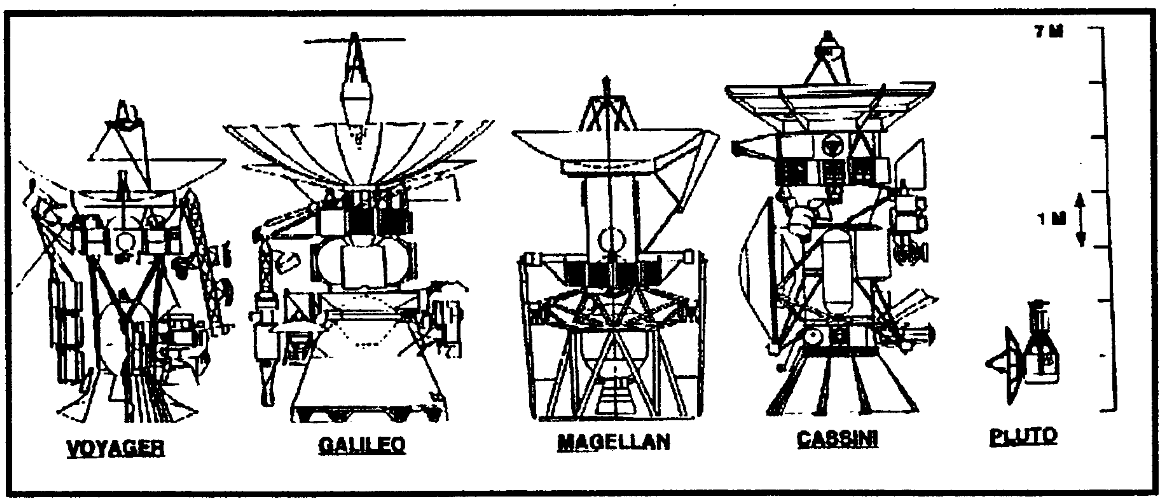

Figure 1: Size Comparison of Planetary Exploration Spacecraft

B. PLUTO-KUIPER EXPRESS SPACECRAFT DESCRIPTION

The primary objective of the *PKE* is to conduct the first science reconnaissance of the Pluto-Charon system that includes the first ever close-up observations of the system. The secondary objective of the mission to for the spacecraft itself to "serve as a pathfinder for lower cost exploration of the outer solar system, by employing pioneering technologies, operations approaches, development techniques and management styles" (Brewster, 1996, p. 2). The current mission concept is one or two satellites launched in November 2003 or December 2004 on a medium-lift expendable launch vehicle, to arrive at Pluto within ten years of launch. Time and budget constraints may limit the mission to only one spacecraft. If two spacecraft are used, they will be launched together but arrive at Pluto approximately six months apart, requiring two separate launch period analyses be performed for each of the two nominal launch dates. Table 1 summarizes the main objectives of *PKE* by dividing them into categories, based upon the importance of the data to be collected.

The sciencecraft bus portion of *PKE* will sit atop the propulsion module of the spacecraft. Attached to the sciencecraft bus will be a science package, a high gain antenna, and star sensors. The science package, called the Strawman Payload, will consist of an atmospheric entry probe and four flyby instruments: a Visible Wavelength Camera, an Infrared Mapping Spectrometer, an Ultraviolet Spectrometer, and an Uplink Radio Science Occultation. The Strawman Payload is designed to meet all of the Category 1a objectives, and perform measurements to satisfy the Category 1b and 1c objectives when and if desired. The Visible Wavelength Camera will provide images for geological mapping of the planet, as well as information pertaining to the formation, evolution, and composition of the system. The images may also reveal surface deposits of ice that will aid scientists in understanding how the atmosphere and the surface interact during the planet's long rotational cycle around the Sun. The camera will also provide the first images of Kuiper Belt objects, if the mission proceeds to that point.

Category Definition	Objective
1a: Absolutely essential to first-scientific reconnaissance mission	• Characterize global geology and morphology of Pluto and Charon • Surface composition mapping • Characterize neutral atmosphere and its escape rate
1b: Important but not mandatory	• Surface and atmosphere time variability • Stereo imaging • High resolution terminator mapping • Selected high resolution surface composition mapping • Characterize Pluto's ionosphere and solar wind interaction • Search for neutral species, hydrocarbons, and nitriles in Pluto's upper atmosphere. Obtain isotopic discrimination where possible • Search for atmosphere around Charon • Determine bolometric bond albedos • Surface temperature mapping
1c: Desirable but secondary	• Characterize energetic particle environment • Refinement of bulk parameters (radii, masses, densities) and orbit • Magnetic field strength • Additional satellite and ring searches

Table 1: Summary of Pluto Express Science Objectives (Lunine, 1995, p. 39)

The Infrared Mapping Spectrometer will determine the surface compositions of Pluto and Charon by performing spectroscopic imaging in the near-infrared part of the spectrum. The Ultraviolet Spectrometer will measure the composition and structure of the neutral atmosphere surrounding Pluto. The Uplink Radio Science Occultation will measure how radio signals are affected by the neutral atmosphere during Earth occultation. It will also aid in determining the surface temperature and pressure, and perform pressure measurements at several scale heights in order to ascertain the properties of the atmosphere (Lunine, 1995, pp. 41-43). The high gain antenna will be part of the communications system that will operate uplink at X-band (7.1 Ghz) and downlink at Ka-band (32 Ghz). Commanding and data exchange will be conducted via NASA's Deep Space Network (DSN) facilities. Finally, the star sensors will provide the primary pointing reference for the spacecraft during its flight. By comparing the centroids of the stars in its field of view with a stored onboard star map about once very second, the desired pointing accuracy of 2 milli-radians can be achieved.

To achieve a more intense study of the atmosphere, an atmospheric probe will be launched towards Pluto. Its impact on the surface will be at a high enough velocity to vaporize some of the surface material and create a flash. The resulting plume might then be analyzed by the *PKE* or instruments in Earth orbit, such as the *Hubble Space Telescope*. Currently, plans are for the Space Research Institute of Russia to supply the probe, called Drop Zond. The instruments to be carried on the Drop Zond are a Mass Spectrometer or a Retarding Potential Analyzer, an Atmospheric Imager, and an Accelerometer. The Drop Zond will perform measurements of the atmosphere and image the planet as it approaches the surface. The data will be received by the *PKE* as it flies by the planet, until the probe cannot transmit through ionized atmospheric constituents or it crashes on the surface.

C. TRAJECTORY OPTIONS

In order to achieve the "sciencecraft-approach" objectives of a smaller, cheaper, more technologically advanced spacecraft, the trajectory of the *PKE* to Pluto must be carefully considered. Because of Pluto's very long orbital period as compared to Earth's period, there are several opportunities when Pluto and Earth are positioned for certain minimum energy trajectory options. The alignment of other planets, such as those desired for gravity assist maneuvers, must also be investigated so as to permit the shortest or most fuel-efficient route for interplanetary travel. The two most important tradeoffs are time of flight and energy applied to the spacecraft by the launch vehicle. In general, the interplanetary travel time and the energy required to escape Earth orbit to reach Pluto are inversely proportional. However, the tradeoffs of time versus energy applied can best be determined by analyzing various opportunities for gravity-assist trajectories. While several viable trajectory options are mentioned in the following sections, no one option can be chosen until exhaustive analysis has been performed for each trajectory and compared to launch vehicle performance and cost figures.

1. Direct Trajectories

A preferred flight path is a direct route from Earth to Pluto. Depending on the alignment of the two planets, this not only results in short flight times, but also in the least damaging radiation environment. This is because planets with large radiation belts and magnetospheres can be avoided by flying the spacecraft directly to Pluto. Therefore, the spacecraft would not require extensive measures of radiation hardening and protection. The flight times for a direct trajectory vary from nine to eleven years, depending upon planetary alignment and energy required to achieve an Earth escape hyperbola. The flight costs are lowered because there are no extraneous maneuvers, other than flight corrections, necessary to change the flight velocity. However, a direct trajectory does entail a very high energy thrust to escape Earth orbit with enough coasting velocity to reach Pluto. A large energy requirement equates to a large launch vehicle and propulsion requirement. A large-capacity launch vehicle such as Titan IV/Centaur or Russian Proton with at least one additional upper stage would be needed (Price et al., p. 2). These large, heavy-lift rockets are extremely expensive and would violate the secondary objective of this mission to maintain very low overall costs. The rocket size tradeoff directly affects the overall cost, and outweighs the advantages gained by a short duration flight time.

2. Multiple, Non-Earth Gravity Assist Trajectories

Eliminating the possibility of a direct trajectory leads to the use of multiple planetary gravity assist flyby trajectories. These trajectories lower the mission costs by reducing the amount of energy required from the launch vehicle. Instead of flying directly toward Pluto, the spacecraft needs only enough energy to send it to an intermediate planet. Additional gains in energy are obtained by a gravity-assist from the planet. The requirement for less energy at Earth results in a smaller launch vehicle, possibly one with a medium-lift capability with or without an upper stage, depending upon flight time. However, since the spacecraft must venture closer to the planet Jupiter, it must be able to survive the harsher radiation environment. Earth itself can be utilized for a gravity assist flyby, but extra effort is needed to ensure that the probability of an Earth impact during the flyby is acceptably low.

There are several non-Earth multiple gravity-assist trajectory options under analysis at JPL for the *PKE*. An opportunity exists for a Venus-Venus-Venus Jupiter Gravity Assist (VVVJGA) trajectory with a launch in March of 2001. The spacecraft can be launched on a Delta or Russian Molniya vehicle without an upper stage, and arrive at Pluto in less than twelve years. The alternate trajectory is Venus/Venus/Jupiter Gravity Assist (VVJGA) with an opportunity for launch in July 2002. In spite of the savings due to using a smaller, cheaper launch vehicle, the extensive ground support required for these complex, multiple assist trajectories greatly increases costs, making the VVVJGA and VVJGA expensive options for traveling to Pluto.

Another option under consideration is the use of Solar-Electric Propulsion (SEP) in conjunction with multiple gravity assist flyby trajectories. An opportunity for a Solar-Electric Venus Gravity Assist trajectory (SE-VGA) occurs in 2006, using a Delta or Molniya launch vehicle. There are also at least two opportunities for Solar-Electric Venus-Venus Jupiter Gravity Assist (SE-VVJGA) trajectories in 2002 and 2004. The extra Venus and additional Jupiter gravity assist flybys allow the use of a smaller and cheaper SEP module than can be used for the SE-VGA trajectory. Despite the advantages of opportunities and size reduction, the use of SEP still appears to be more expensive because it can only provide thrust out to approximately 2 AU (Price et al., p. 2).

3. Jupiter Gravity Assist Trajectories

A straight JGA involves escaping an Earth parking orbit, flying directly to Jupiter for a gravity assist flyby, and traveling onto Pluto. This option is immediately attractive because is involves only one gravity assist flyby, simplifying the mission design and operations, and thus reducing mission support costs. This gives it a great advantage over VVVJGA, VVJGA, SE-VGA, and SE-VVJGA. The *PKE* can be launched from a medium-lift vehicle, such as a Delta or Molniya, with the addition of an upper stage. Because it is traveling first to Jupiter, the energy requirements to escape Earth orbit are not as high as those required for a direct-to-Pluto trajectory. Consequently, the upper stage required for the JGA need not be as large as that required for a direct route. This cost reduction works to offset the cost increase incurred by the additional upper stage.

There are two JGA trajectory opportunities, November 2003 and December 2004. If these are not used, then the next feasible, low cost opportunity does not appear until 2013 because Jupiter is only in the correct position in its orbit for 2-3 years out of its 12-year orbital period. The JGA provides a viable route to reach Pluto by reducing overall mission costs and mission operations. The following chapters describe the detailed analysis performed for the two JGA opportunities. Using computer programs written at JPL, the orbital parameters are calculated for a trajectory from Earth to Jupiter to Pluto. The launch date for each opportunity is then extended into a launch period of four weeks, with the accompanying orbital parameter data for each day in the period. By comparing the launch data information to the performance characteristics of various launch vehicles with and without upper stages, a launch strategy is devised that defines when the JGA is most advantageous and on what launch vehicle. These strategies can then be subjected to a cost analysis and comparison with other trajectory options in order to choose the best route for the *PKE*.

II. PRELIMINARY TRAJECTORY ANALYSIS USING JPL SOFTWARE MIDAS

A. MIDAS ALGORITHM

The JPL code, MIDAS (Mission Integrated Design and Analysis Software) was used to generate the initial JGA trajectory parameters for the launch strategy analysis. This program is currently hosted on a SUN SPARC station at JPL, and was remotely accessed for the purposes of this analysis. The actual JGA opportunities described in the previous chapter were determined using MIDAS. Given a launch year and a mission goal of traveling to Pluto, MIDAS is capable of optimizing various trajectories to determine the nominal launch date, time, Earth departure parameters, JGA flyby parameters, and Pluto arrival characteristics.

1. Capabilities

MIDAS is a powerful trajectory optimization program written by JPL engineers, for the purpose of designing patched conic interplanetary trajectories. By performing calculations of gradients and partial derivatives, MIDAS can efficiently analyze various trajectories in order to determine which is optimal given a certain set of constraints (Sauer, 1991, p.1). During the course of the optimization process, MIDAS can add a deep space maneuver, powered gravity assist flyby, or unpowered gravity assist flyby to the trajectory at calculated points. MIDAS will them recalculate the entire trajectory based on the new maneuver. The resultant trajectory is then stored for later comparison with other resultant trajectories. The final result is a list of possible trajectories to achieve the mission. They will vary in launch date and time, type of intermediate maneuver, and arrival date and time. This provides the analyst with several options to pursue further in-depth analysis.

The optimization is achieved by minimizing the total ΔV that the spacecraft will need in order to complete its mission (Sauer, 1991, p. 2). To accomplish this, MIDAS requires that certain information be available to initialize the program. This input is normally mission constraint information, such as Earth parking orbit criterion, flyby altitudes or times, or arrival orbit parameters such as circularization or flyby. In addition, planetary, asteroid, and comet information can be added in the form of state vectors to describe their motion in inertial space.

2. Gravity Assist Flyby Trajectories

Before any information is input into the program, the program defaults any inserted gravity assist flyby trajectories as powered, meaning that there is a ΔV burn associated with the flyby. The burn generally occurs at the perigee point of the flyby trajectory, augmenting the inherent increase in energy and change in direction. The hyperbolic trajectory becomes more energetic than it would have been if the flyby were unpowered. For this analysis, an unpowered JGA is desired, and the program must be reinitialized for an unpowered flyby by inputting estimated values of minimum perigee altitude and B-plane angle (approach angle).

MIDAS optimizes the desired trajectory by first reading the input parameters and calculating a tentative trajectory. MIDAS then "hypothesizes" a new set of orbital parameters, based on the first, and calculates the new trajectory. Whenever the trajectory is modified by MIDAS during the course of optimization, the data is automatically recorded in a running file. This includes the addition or deletion of a deep space maneuver, the conversion of an unpowered flyby to a powered flyby or the reverse, and reinitialization if the trajectory violates the altitude constraints required for an unpowered flyby (Sauer, 1991, p. 3). If the new parameters produce a trajectory that has a lower ΔV than the previous one, the previous is purged in favor of the new, and this data becomes the baseline for the hypothesis. This continues until the ΔV has been sufficiently minimized. At this point the program terminates, and the results can be output in various formats, dependent upon the user's preference.

B. CREATING AN INPUT FILE

In order to reinitialize MIDAS as described above for an unpowered JGA, an input file must be created that contains all of the necessary data for trajectory calculation and optimization. The input file is a simple text file containing variable names and associated values. The file must contain the suffix *.inp* in order for MIDAS to recognize it as an input file. Figure 2 displays the input file used for the 2003 JGA opportunity, while Figures 3, 4, and 5 show the input files for the 2004 opportunity. See Appendix A for an abbreviated list of input variable definitions.

In each of the input files, Earth is given as the departure body, and Pluto as the target body. The one and only intermediate body to be encountered is defined as Jupiter. For each file, the epoch time begins on January 1 of 2004 or 2005. Other variables indicate the minimum altitude of the flyby, parking orbit inclination, desired flight time, and the variables that are to be optimized. The variable *rcb* prompts the program to reinitialize for an unpowered flyby at the intermediate body.

The *jdate*, or departure date, must be chosen carefully to ensure the best optimized trajectory. It is best and most efficient to choose a date relatively close to the nominal launch date. This can be chosen by first arbitrarily choosing a launch date for the input file and executing MIDAS to obtain preliminary trajectory results. As MIDAS tries various trajectories for optimization, the nominal dates for launch, JGA, and arrival change in accordance with the solution. This new departure date is replaced into the input file, and MIDAS is executed again, calculating new dates. This iterative process is repeated until there is no longer a variance in the MIDAS's resulting departure date. The final date calculated is the nominal launch date from parking orbit.

```
HEAD='2000 V-M-V to Outer Solar System'
SHOTA='Earth',
BULSI='Pluto',
BODY='Jupiter',
JDL=2004,1,1.0,ndb=2,ALTB=71400,poi=30,nda=1,
VARYI='jdate','tpb','tm1',
nlv=28
lvcont=.1
kadp=.05
tpb=500.,
tend=9
varyn='tend',minyn=9,.5,13
vlist='+jdate','tend','c3','dvp1','rcb','mnet'
choice=0
jdate=0, $
$END
```

Figure 2: Input File (e59_03a.inp) for Year 2003 JGA Opportunity

```
HEAD='2000 V-M-V to Outer Solar System'
SHOTA='Earth',
BULSI='Pluto',
BODY='Jupiter',
JDL=2005,1,1.0,ndb=2,ALTB=357000,poi=30,nda=1,
VARYI='jdate','tpb','tm1'
tpb=500,
tend=10
choice=0
jdate=0, $
$END
```

Figure 3: Input File #1 (e59_04.inp) for Year 2004 JGA Opportunity

```
HEAD='2000 V-M-V to Outer Solar System'
SHOTA='Earth',
BULSI='Pluto',
BODY='Jupiter',
JDL=2005,1,1.0,ndb=2,ALTB=71400,poi=30,nda=1,
VARYI='jdate','tpb','tm1',
tpb=500,
nlv=28
lvcont=.1
kadp=.05
tend=8
varyn='tend',minyn=8,5,13
vlist='+jdate','tend','c3','dvpl','rcb','mnet'
choice=0
jdate=0, $
$END
```

Figure 4: Input File #2 (e59_04a.inp) for Year 2004 JGA Opportunity

```
HEAD='2000 V-M-V to Outer Solar System'
SHOTA='Earth',
BULSI='Pluto',
BODY='Jupiter'
JDL=2005,1,1.0,ndb=2,ALTB=35700,poi=30,nda=1,
VARYI='jdate','tpb','tm1'
tpb=500,
tend=10
varyn='tend',minyn=10,-.5,7
vlist='+jdate','tend','c3','dvmt','rcb'
choice=0
jdate=0, $
$END
```

Figure 5: Input File #3 (e59_04b.inp) for Year 2004 JGA Opportunity

C. OUTPUT FILES

The MIDAS optimization is executed with a single command given on the UNIX command line, that specifies the format of the output file. The command is written as a UNIX command: *midas -# <input filename>.inp*. The # is replaced by one or more letters that correspond to output file options, dictating how detailed the output is desired and how it is to be presented. Appendix B lists the command line options for various outputs (Sauer, 1991, pp. 5-6). For this JGA analysis, the most useful output option was *-S*, as it provided the most information and data pertaining to the optimized trajectory.

The detailed output file lists the parameter values in tabular form. The file is segregated into three parts: the heading and overall trajectory information, information specific to each constrained point of the trajectory given in several reference frames (see Chapter III, Section A, Subsection 2, for more information on reference frames), and a data list for each leg of the trajectory between the constrained points. The heading simply states the trajectory that was optimized in a specific reference frame. Among other information, it displays the total transit time (*tend*), minimum ΔV, its rectangular components, and minimum injected mass to which the trajectory is applicable.

The second part lists all of the calculated parameters required to model the trajectory at each constrained point. For this analysis, the points are Earth departure, Pluto arrival, and JGA. The exact date and time, in both calendar and Julian form, are given. For each constrained point, the values of position, angular measures, and velocities are listed for each of three reference frames: heliocentric ecliptic of 2000, body centered ecliptic of 2000, and body centered planet equator and equinox of date. This allows much flexibility in the use of this output data. It saves the time needed to transform one reference frame's data into another, depending on how the data is to be used next. The last part is a statement of all of the calculated data for that portion or leg of the trajectory between stated constrained points, but in a completely heliocentric reference frame. The data includes positions at beginning and end of the trajectory leg, corresponding velocities, incoming and outgoing angular positions, closest points of approach, and travel time in days.

Appendix C contains the MIDAS output files, using the *-S* flag option, that correspond to the input files displayed in Figures 2, 3, 4, and 5. At the beginning of each output is a copy of the original input file, given for the purpose of completeness. Following that, MIDAS displays all of the trajectories that are calculated to be feasible. For the purposes of this analysis, only the values of the trajectory actually used are listed in Appendix C.

III. DETAILED TRAJECTORY ANALYSIS USING JPL SOFTWARE CATO

A. CATO ALGORITHM

MIDAS produces a good model of the desired trajectory, but because it uses multi-conic type of technique, it does not provide the most accurate data results. The program will continue to execute until the difference between the calculated ΔV's is essentially zero. MIDAS only accounts for the velocity difference between the trajectories, and not the position difference. It also does not take each portion of the trajectory separately and, therefore, does not allow calculations to be performed at any specific point. Computer Algorithm for Trajectory Optimization, or CATO, is similar to MIDAS in that it is a program designed to take a desired interplanetary path and optimize it through an iterative process to find a "best fit" trajectory and its parameters. CATO calculates the trajectory values and presents an output that may be significantly different from the desired value. The program chooses a step size for each parameter for the current iteration based on the results of the previous calculations. It adds this step to each parameter value, and recalculates the trajectory. This process is repeated until CATO converges to a desired value. CATO provides an interactive method for the user to control the optimization process at each portion and ensure a solution is generated. It attempts to minimize the total ΔV required for an interplanetary mission (Bright and Byrnes, 1996, p. 1-1). Using variables contained in an input file, the spacecraft itself can be modeled for gravitational effects from the involved planets and nearby moons along the route, as well as for the effects of solar pressure. The free variables can be given upper and lower boundary limits, providing the constraints to which CATO must optimize.

1. Breakpoints and Control Points

To input a desired flight path into CATO, the intended trajectory must first be divided into a set of *breakpoints* and *control points*. This allows the program to treat each leg of the trajectory separately, applying multi-body integration methods to determine the most economical (lowest total ΔV, unpowered flyby maneuvers) trajectory parameters. Once the breakpoints and control points are determined, separate parameters and variables are used to define them.

At some point on each trajectory leg is a defined *control point*. This is generally a location in that leg that must have some constraints, in altitude, velocity, period, inclination, etc. The control point serves as the anchor for all of the integration that the program must perform during the optimization process. Each control point is given a set of *control point variables*. These variables are user-defined parameters of the initial conditions at epoch. They allow the user to utilize one of several reference coordinate systems and their associated variables. The next section provides more details about control point variables. The control point variables of all of the legs provides the program with an initial complete trajectory from which to begin the iteration process. Table 2 lists some important considerations when choosing the placement of the control points.

Lesson #1	Place control points at places on the trajectory where you want to constrain the state in some way.
Lesson #2	Select the variable set so that it includes those parameters that you need to constrain.
Lesson #3	Free those variables that your really do not care about. Allow CATO to find the values that require the least ΔV.
Lesson #4	Fix any variable that must have a certain specific value.

Table 2: Lessons for Control Point Placement (Bright and Byrnes, 1996, p. 3-5)

Between each control point is a *breakpoint*. They are arbitrarily defined positions in time that serve as termination points for the program's iterations. In cases where a maneuver is expected or desired to occur at a certain point in a trajectory, the breakpoint can be placed at that point, since all maneuvers occur at the breakpoints (Bright and Byrnes, 1996, p. 3-3). Additionally, the entire trajectory must begin and end with a breakpoint. The breakpoints and control points must alternate. For example, if the trajectory had **n** legs, the sequence would be:

$$\mathbf{BP_0, CP_1, BP_1, CP_2, ..., CP_n, BP_n,}$$

where **BP** is a breakpoint and **CP** is a control point. The subscripts denote the trajectory leg number, with "0" being the given initial breakpoint that begins the first leg. Figure 6 displays the layout of the CATO trajectory used for this JGA analysis.

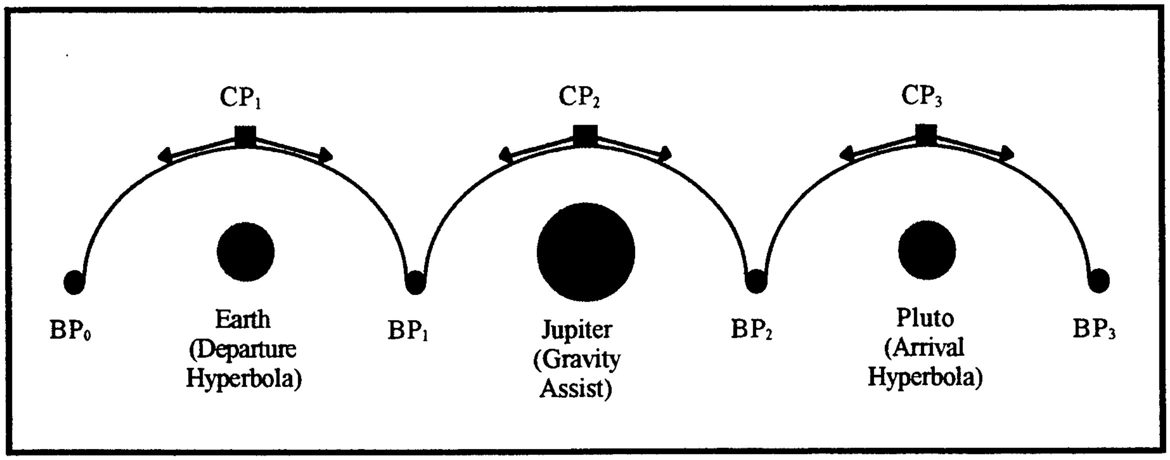

Figure 6: Schematic Representation of Trajectory (Bright and Byrnes, 1996, p. 3-2)

To generate the initial and all subsequent trajectories, CATO considers each leg separately and maps out the actual trajectory path dictated by its parameters in the input file. At each control point, the program propagates the trajectory in two directions, forwards in time and backwards in time, toward the breakpoint that lies between each pair of control points. At each breakpoint, CATO computes the incoming position and velocity from the previous control point, and the outgoing position and velocity to the proceeding control point. The difference between the incoming and outgoing position calculation produces a *position discontinuity*, and likewise a *velocity discontinuity* for the incoming and outgoing velocity. To achieve a continuous trajectory, the position discontinuity must be equal to zero. However, due to the nature of the desired trajectory, a velocity discontinuity may not always equal zero, resulting in a necessary ΔV at this breakpoint position in order to continue on this trajectory. For an unpowered interplanetary flight and JGA as required by this analysis, the velocity discontinuity must be forced to zero through interactive iteration of the trajectory.

2. Control Point Variable Selection

To define the parameters at each control point, a set of *control point variables* must be selected. These consist of seven variables: a reference epoch, and a set of six parameters associated with the chosen epoch. Any set of variables may be used at a control point, and all of the control points need not use the same set, as long as each is clearly specified. Choosing which set of variables to use is given to user's preference and format of initial trajectory information. In addition to choosing a parameterization, the user must decide whether a variable or a set is free or optimized by the program, or is fixed or to be used as boundaries for optimization.

In the input file, the epoch reference parameters are chosen using the variable *iOrbU*. This variable is assigned a string of numbers called a *flag* that define which epoch is chosen, and what variables are to be associated with that epoch. Table 3 defines the flag, which can be between two and seven digits. Each section is lettered from *A* to *G*. In section *A*, the parameterization is specified. The remaining digits in each section are chosen based on the information desired or given, for the selected parameterization.

Section B, the Reference Coordinate System, is defined as follows:

0: VECTOR standard coordinate system
1: Geocentric and equinox of 1950
2: Planet (central body) centered and equinox of epoch
3: Planet centered and prime meridian of epoch
4: Planet centered and prime meridian of date
5: Geocentric and equinox of 2000
6: Earth ecliptic and equinox of 1950
7: Planet orbit plane and equinox of epoch
8: Earth ecliptic and equinox of 2000

The flag is written with the digits ordered *GFEDCBA* and read from right to left, since the lower digits will likely be used most often in specifying the orbital elements, and leading zeroes can then be eliminated. For example, the flag *iOrbU=11* defines the variables as classical orbital elements in an Earth-centered equator reference frame and equinox of 1950. The flag *iOrbU=30083* defines the variables as hyperbolic quantities, Earth ecliptic reference frame and equinox of 2000, spherical coordinates, and B angle, B*T, and B*R defined.

A		1		2		3		4
A		Classical Elements (a, e, I, Ω, ω, M)		Position and Velocity		Hyperbolic Quantities (alt, th, t, V_∞)		Launch Planet and Date
A		1		2		3		4
B	0	Vector Standard Sys	0	Vector Standard Sys	0	Vector Standard Sys	0	Vector Standard Sys
B	1	Geocen Equinox '50	1	Geocen Equinox '50	1	Geocen Equinox '50	1	Geocen Equinox '50
B	2	Planet Equnx Epoch	2	Planet Equnx Epoch	2	Planet Equnx Epoch	2	Planet Equnx Epoch
B	3	Planet Merid Epoch	3	Planet Merid Epoch	3	Planet Merid Epoch	3	Planet Merid Epoch
B	4	Planet Merid Date	4	Planet Merid Date	4	Planet Merid Date	4	Planet Merid Date
B	5	Geocen Equinox '00	5	Geocen Equinox '00	5	Geocen Equinox '00	5	Geocen Equinox '00
B	6	Earth Ecl Equnx '50	6	Earth Ecl Equnx '50	6	Earth Ecl Equnx '50	6	Earth Ecl Equnx '50
B	7	Planet Orbit Epoch	7	Planet Orbit Epoch	7	Planet Orbit Epoch	7	Planet Orbit Epoch
B	8	Earth Ecl Equnx '00	8	Earth Ecl Equnx '00	8	Earth Ecl Equnx '00	8	Earth Ecl Equnx '00
C	0	Semi Major Axis	0	Spherical	0	Spherical	0	Spherical
C	1	Period	1	Cartesian	1	Cartesian	1	Cartesian
C	2	C3	2		2	C3	2	C3
C	3	Apoapse Altitude	3		3		3	
C	4	Apoapse Rt Asc Dec	4		4		4	
D	0	Eccentricity	0		0	B Angle	0	Date
D	1	Perigee Altitude	1		1	Inclination, B*R<0	1	JDS
D	2	Perigee Rt Asc Dec	2		2	Inclination, B*R>0	2	MODJD
D	3		3		3	Azimuth, B*R<0	3	
D	4		4		4	Azimuth, B*R>0	4	
E	0	Argument of Perigee	0		0	Altitude of Perigee	0	
E	1	Longitude of Perigee	1		1	Radius of Perigee	1	
E	2		2		2	B Length	2	
E	3		3		3	B*T, B*R	3	
E	4		4		4	Radius	4	
F	0	Mean Anomaly	0		0	Incoming V	0	
F	1	Time to Perigee	1		1	Outgoing V	1	
F	2	True Anomaly	2		2		2	
G	0		0		0	Time to Perigee	0	
G	1		1		1	Flight Path Angle	1	
G	2		2		2	True Anomaly	2	
G	3		3		3	Radius T<0	3	
G	4		4		4	Radius T>0	4	

Table 3: iOrbU Flag System Summary (Bright and Byrnes, 1996, p. E-5).

3. Interactive Mode Operation

Before executing any input files on CATO, the proper space environment must be set. This is done by using the UNIX command *setenv* on the command line, as follows:

% setenv EPHEMERIS1 /usr/lib/eph/de202.bsp

Ephemeris data is the location of planets, system barycenters, satellites, and the sun with respect to time. The ephemeris file sets these parameters in CATO, to describe the positions and movements of the planets and their moons. The file listed above is located on a server at JPL. CATO does contain a hard-wired analytic ephemeris which it uses if none of the environment variables are set manually. In these types of files, as used at JPL, the ephemeris is provided in Chebyshev polynomials. In CATO, this information, along with other planetary constants such as gravitational (GM), radius, pole vector, oblateness, etc., are used to develop the "environment" in which the spacecraft flies.

The program CATO is initiated by typing "cato" on the UNIX command line. The initial inputs can be read into the program by two methods: having the program read an input file or entering the *interactive* mode and typing in the data item by item. The input file has the obvious advantage of having all of the data for one trajectory stored in one file, which can be read in many times. When typing the data in the interactive mode, the original input information is lost when the program is terminated, and must be reentered on a subsequent execution. Once the input file is entered into the program and the optimization is initiated, CATO enters the interactive mode, so that the user can control the convergence and aid CATO with the step sizes in the iterations. In this mode, the user can modify, add, or delete breakpoints and control points, show the resultant trajectory convergence status, or save the output at its current stage by simply typing a command on the CATO command line. Appendix D summarizes the commands understood by CATO for interactive mode use (Bright and Byrnes, 1996, pp. 4-2, 4-3). The commands correspond to variable types used in the input file, or used for interactive data input. The commands themselves can be abbreviated during interactive execution, using the first three letters of the command, such as *int* for *interactive*.

The process for optimizing the trajectory in the interactive mode begins by plotting the initial trajectory defined. The parameters are entered into CATO (as described in the following section); any last minute modifications can be made to the input values either in the input file or by using the commands *:CP, :BP,* or *:MV* on the CATO command line. The *optimize* command is invoked to begin the procedure. CATO uses numerical integration, and an iterative process called "reweighting the cost function" (Bright and Byrnes, 1996, p. 3-7), which in this case are the sum of the ΔV magnitudes, when there are no finite burns in the trajectory (Bright and Byrnes, 1996, p. 3-7). CATO outputs the new trajectory parameters and velocity and position discontinuities, which are viewed using the *show* command. If the discontinuities are not zero, then the trajectory must be reset and reoptimized. Using the *modify* commands on the control line, the parameters at the control and breakpoints are redefined as needed to redirect CATO for the next optimization. The *continue* command is used to optimize the newly modified trajectory parameters. This process is repeated until the discontinuities are sufficiently close to zero, on the order of 10^{-6} or less. Chapter IV details the interactive optimization process to demonstrate how is was performed for this JGA analysis.

B. CREATING INPUT FILE USING MIDAS OUTPUT DATA

The input file is a compilation of commands (Appendix D) that feeds the program the initial trajectory information. The file is read into CATO by typing *file <filename>* on the CATO command line, in this case *file yr2003.inp* or *file yr2004.inp*. The last line in each command file, *interactive*, sets the program to interactive mode for optimization control.

Much of the information for the input files was garnered from the output files of the MIDAS runs. It is necessary to choose the information contained in the portion of the MIDAS output that matches the desired reference frame to be used in CATO. Some of the information is environmental data and parameters that JPL uses as a standard in CATO files. The MIDAS results provide a good estimate of the exact trajectory required to optimize the problem. Using this information vice arbitrary values in the input file will ensure that the initial trajectory discontinuities are within a reasonable error range. The

MIDAS results that are used in the CATO input file include: Earth departure date C_3, V_∞ declination, V_∞ right ascension; JGA date, perigee altitude, B-plane angle, V_∞ value, V_∞ declination, V_∞ right ascension; and Pluto arrival date, V_∞ value, V_∞ declination, and V_∞ right ascension. These values serve at the initial values for the free variables in the input file.

Beginning the input file are introductory comments, stating the file's creator and nominal launch information, the name of the MIDAS file upon which the input file is based. Immediately following is a list of gravitational bodies, listed with the variable *GravBods*. The string of numbers correspond to the bodies whose gravitational forces will be considered for acceleration models in this trajectory optimization. The numbering system is referred to as the Mission Analysis Support Library (MASL) Body Number System, and employed for CATO with the exception of the sun's number. Appendix E lists all of the bodies accounted for in the MASL system. After that, the file is segmented by alternating breakpoints and control points.

At each breakpoint, a range for the Julian date is given, basically stating the time range of the breakpoint. The variable *jd* is chosen to be within the date range, and is the point in time that a breakpoint maneuver or burn would occur. If there is to be no additional mid-course maneuvers, then the breakpoints can be set a fixed. The variable *cbody* uses the MASL Body Number System to specify what planetary body may affect the breakpoint, such as Earth in an Earth-departure breakpoint, otherwise the variable is set to zero to indicate it is at an intermediate point in space.

For the control points, the variable *bodyU* specifies which body will most affect the control point calculations. *iOrbU* determines the reference frame and epoch in which the parameters are defined. For each parameter, a lower bound, starting value, and upper bound are specified. The bounds are chosen somewhat arbitrarily, though there are some standard values used by JPL for certain variables. These bounds can be changed during the interactive mode optimization, to allow CATO more freedom in the iteration process. CATO will generally bounce back and forth above and below the optimal solution, while its iteration step size decreases. It CATO hit one of the boundaries during the process, this would hinder the program, believing it cannot iterate that value any further, and most likely not reduce the discontinuities to zero. Also defined for the control point are differential values, used by CATO during the integration process. Lastly, the variable *FixedU* defines which parameters are fixed, and which are free to be optimized.

Appendix F contains the input files used for this JGA trajectory analysis, one for each nominal launch date of 14 November 2003 and 15 December 2004. The comment lines at the beginning of each file indicate which MIDAS output file was used to create the CATO input file. Each breakpoint and control point is named with the *+BP* and *+CP* commands. To input the parameters at each point, FORTRAN 90 conventions are used to correlate with the way CATO will read and manipulate these numbers. The input lines must begin with an ampersand ("&") and end with a slash ("/"). The spacing and formatting of the parameters in the input file are not necessary, since the CATO code reads the symbols and not the spaces or carriage returns. They are included to make the file easier for the user to read and review, providing an organized appearance.

C. OUTPUT FILES

As CATO computes new trajectory values, it displays the breakpoint states values, the discontinuities, linearized values from the reweighting and iteration, all of the original values, the proposed step size for the next iteration, and the new proposed values based on the step size. The discontinuities are listed for each internal breakpoint, those that have control points defined both before and after them. In the input file, if *noMvr=.TRUE.* then both the position and velocity discontinuities are listed. The beginning and ending breakpoints are not included, because there are no discontinuities to calculate. If *noMvr=.FALSE.* then only the position discontinuities are listed. Also of interest are the *linearized dKs*. These are the calculate ΔV's at the internal breakpoints. For this analysis, these values are desired to be zero or on the order of 10^{-6} or less.

The parameter values after each optimization are easily displayed with the *show* command. The command tabulates all of the new parameters, separated as breakpoints and control points. At each control point, the user can compare the current values to their boundary limits, and evaluate how the value is progressing towards a limit or convergence value. To the left of the columns, CATO indicates if a parameter has hit against one of the boundaries, with "U" for upper boundary, and "L" for lower boundary. If a value comes against a boundary, this represents a problem that was not sufficiently optimized. These boundary values must be changed to lengthen the value range, and then reoptimized from the original values. These changes are performed on the command line with the *:CP* command to modify the control point values. The values of the *U* variables are treated as a vector, where any element can be designated by its numerical position, starting with zero. Once the control point values are modified, the *continue* command reinitiates the optimization using the new boundary values with the last calculated parameter values. The current set of parameter values can be saved as a CATO data file using the *checkpoint <data filename>* command. This allows the same optimization to be loaded and continued at a later time without having to reload the original input file and proceeding through the previous set of modifications.

IV. LAUNCH PERIOD DETERMINATION

A. CREATION OF NOMINAL LAUNCH PERIOD USING CATO

MIDAS and CATO provided two nominal launch dates and times for the PKE to travel to Pluto, however, it is impractical to depend solely on the nominal launch date for the actual launch. Numerous factors, including weather, budget, or natural disaster, can and will affect the launch date and time, possibly necessitating a reschedule. To prepare for this, it is necessary to have a launch period of opportunity centered around the nominal launch date. This provides for a contingency backup plan should unforeseen circumstances negate the planned launch date. Once the launch period is established, it can be compared to the performance characteristics of various launch vehicles to determine which is most capable of launching the PKE on a given day.

The majority of the data calculation for this analysis was performed by CATO, with the original data estimates coming from MIDAS. CATO offers a more accurate and complete analysis of the trajectory that was more useful in this JGA analysis. The creation of a launch period was simpler in CATO because of the interactive mode capability, where only the launch date or arrival date had to be changed, with regular modifications performed during the optimization. The following section describes how the interactive procedure was used to create the daily files.

1. Creating the Daily File

The first daily file is that created for the nominal launch date, in this case, one for 14 November 2003 and one for 15 December 2004. The input files contained in Appendix E were used to create the nominal launch date file. The procedure will be described for the 2003 opportunity, as the procedure for the 2004 opportunity is identical. As stated in the previous chapter, once CATO was initiated, the input file was read into the program using the *file yr2003.inp* command on the CATO command line.

CATO responds by echoing all of the commands issued in the input file, ending with the *interactive* command. The process is begun by invoking the *opt* command (as stated previously, each command may be abbreviated with the first three letters), which causes CATO to read the values of the variables and list them as a vector called *Previous Vars*. Since this is the first attempt at optimization, the program defaults to an iteration step-size of zero for all variables; thus the following vector, *Proposed Vars*, is the same as the previous vector values.

To perform the first iteration, the *con* command is used. After ensuring that the proper ephemeris is loaded into the computer, CATO lists *Breakpoint States* and *Breakpoint Constraint Discontinuities*. There are four numbers listed, indicating position and velocity for each internal breakpoint. This is a function of the input file, as described in Chapter III. The next line of interest is the *Linearized dKs*, which indicate the ΔV required at the corresponding breakpoint. For this JGA analysis, to obtain an unpowered flyby, the discontinuities and the dKs must be essentially zero.

The next step in the interactive optimization process is to not only check the discontinuity and dK values, but also use the *show* command to observe the parameter values. The left side will indicate if any parameters hit a boundary value which hindered the optimization. If this is so, then the current optimization must be purged from the program using the command *con ,,0*, which tells the program to perform zero steps, and resets the trajectory to the last set of saved values. Using the modify commands, the culprit boundary values can be changed to allow a greater range for CATO. Once all modifications are entered, the *con* command is used to attempt an optimization. If the *show* command reveals that no values are against a boundary, then a series of *con* commands can be issued until the discontinuities and dKs are sufficiently zero. However, the parameter values should continue to be monitored, because any time in the optimization process, a value can approach and hit a boundary.

The *con* command is very important in controlling the progress of the optimization, specifically the first and third argument. The first argument indicates the number of steps that should be taken. This allows CATO to run *con* more than once, if the step sizes are sufficiently small and the actual values are far enough from a boundary that there is no danger of it hitting one. The third argument allows the user to decrease the step size. Sometimes at the beginning of an optimization, the step sizes are relatively large, and can cause the actual values to alternate between the boundary values very often. The correct solution may in fact lie somewhere in the middle, but CATO cannot realize that because of the step size. Reducing the step size to 0.1 or 0.2 of the proposed value allows CATO to proceed at a slower pace, so that it can be steered toward the possible solution instead of passing over it. Once the proposed step sizes are have been reduced to magnitudes of 10^{-2} or less, then the user can assume a normal step size of 1.

In this analysis, the total number of *con*'s required was about 10-15 for each file. It is interesting to note that while there was no problem reducing the dKs to zero (10^{-18} to 10^{-24}), the discontinuities required considerably more manipulating to force them toward zero. The velocity discontinuities were eventually reduced to approximately 10^{-10} or less, the position discontinuities did not get as close to zero. The first internal breakpoint, *Jupiter Trajectory* or *JupTraj*, could only be reduced to the order of 10^{-3}, sometimes 10^{-4}. After consulting with JPL engineers experienced in this type of analysis, this position discontinuity was deemed acceptable, as it was on the order of meters. After many attempts, the best that could be hoped for was to reduce the discrepancy to less than five meters. The second internal breakpoint, *Pluto Trajectory* or *PluTraj*, was sufficient to the order of 10^{-6}. Figure 7 is an example of the output for 14 November 2003 input file, after several *con* commands.

```
Command > con

Breakpoint States (User-Specified Coordinates):
2452958.1040+           5.20231924406E+03 3.19502631448E+03 -2.40872643827E+03 -
7.75778833631E+00 1.29407460936E+01 4.11774814625E-01

2453249.5000-           -5.30766184351E+08 -7.10321316831E+05 1.69244830086E+07 -
1.43151940375E+01 -1.12493338901E+01 1.73648573710E-01
2453249.5000+           -5.30766184345E+08 -7.10321302505E+05 1.69244830080E+07 -
1.43151940382E+01 -1.12493338895E+01 1.73648573699E-01

2453540.1000-           -7.73702244018E+08 -2.96799201525E+08 2.08817099188E+07
2.07387582386E+00 -2.33007860642E+01 1.24674759236E+00
2453540.1000+           -7.73702244018E+08 -2.96799201525E+08 2.08817099188E+07
2.07387582386E+00 -2.33007860642E+01 1.24674759236E+00

2456610.0000-           3.27504453661E+05 -3.93637461310E+06 6.24897081066E+05
1.20089325396E+00 -1.47016010713E+01 2.30653743150E+00

Breakpoint Constraint Discontinuities:
        1.53434722557E-03
        9.06007495813E-10
        3.76443119176E-06
        3.72201668773E-13

 *** MINIMIZING ***

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

 Linearized dKs 7.19260080775E-18 2.58493941423E-26 1.14063986552E-18 1.46443224382E-24

 Linearized dCs
 SUM          0.00000000000E+00

Previous Vars:   1.06348611650E+02 2.31254766721E+01 1.46574047951E+02 1.70544217400E+08
3.57697417771E+05 8.41091213983E+00
            1.14043443967E+01 6.51776123800E-01 3.47311195413E+01 1.49192914151E+01
8.84387200296E+00 2.74775056495E+02

Proposed Step:   1.17328287288E-09 3.04551528855E-11 -3.50753626355E-09 -6.54355200657E-04
8.14033605710E-06 1.39396589384E-11
            -5.95248802282E-11 1.79103090260E-10 5.29292639265E-10 -2.11962282657E-11
6.52944513175E-13 9.02526790911E-12

Proposed Vars:   1.06348611651E+02 2.31254766721E+01 1.46574047947E+02 1.70544217400E+08
3.57697417779E+05 8.41091213985E+00
            1.14043443967E+01 6.51776123979E-01 3.47311195418E+01 1.49192914151E+01
8.84387200296E+00 2.74775056495E+02
Command > che 1114
```

Figure 7: Example CATO Output for 14 November 2003 Input File

For this trajectory, the position discontinuity at the first internal breakpoint is approximately 1.5 meters, with the remaining three discontinuities sufficiently small enough to be considered zero. The function required only one reweighting, indicating that the trajectory was optimized, and any further attempts to continue would likely cause the discrepancies to change by only slight amounts. Another indicator if this is the *Proposed Step* vector; the sizes are minutely small, resulting in very little change in the parameter values on the next optimization attempt. The final trajectory parameter values for the two nominal launch dates, as displayed by the *show* command, are listed in Appendix H. Using these two files, a series of daily files was constructed, constituting the nominal launch period for each opportunity.

2. Creating the Launch Period Files for 2003 and 2004

Creating the launch period files for each nominal launch date was simple in concept. A period of four weeks was chosen as the desired extended launch period, which would be analyzed for possible contingency launch dates and launch vehicles. The period begins exactly two weeks before the nominal date, and ends two weeks afterward, for a total of twenty nine daily files for each period. As stated in Chapter I, there is a possibility of launching two PKE spacecraft together on one launch vehicle, with the second PKE arriving at Pluto six months after the first. This necessitates two analyses for each launch period: one for a ten year flight time, and one for a ten and a half year flight time. To simplify the notation, the four launch periods will be defined as follows:

Period 1: nominal 14 November 2003, 10 year flight time

Period 1a: nominal 14 November 2003, 10.5 year flight time

Period 2: nominal 15 December 2004, 10 year flight time

Period 2a: nominal 15 December 2004, 10.5 year flight time

Basically, the entire optimization process described in the previous section had to be repeated for a total of one hundred and sixteen daily files. This was a very time consuming process due to the time required by the program and computer to break down the trajectory and perform all of the necessary calculations.

The initial step was to create Periods 1 and 2 by changing the Earth departure dates, then creating Periods 1a and 2a by changing the Pluto arrival dates of Period 1 and 2. Starting from the nominal launch file, a simple modify command was used on the CATO command line to subtract from or add to the departure date in the *Earth* breakpoint and *Departure Hyperbola* control point. In addition, the lower and upper boundary values must to be changed to reflect the correct range around the parameter value. Also, the value of *Fid JD* must to be adjusted for the control point so that all points maintain the proper order. The Fid JD defaults to the parameter value of the *Julian Date* in the input file, and updates itself automatically only for the breakpoint, not the control point. This was especially important for the daily files with launch dates that are after the nominal, because the *Earth* breakpoint Julian date and Fid JD are later than the corresponding control point values, causing the order of the first two points to switch. The modification commands for the first date after the nominal, 15 November 2003, are entered as follows:

»:bp Earth

»&bp jdMin=2452959.1039996,

jd=2452959.1039997,

jdMax=2452959.1039998/

»:cp DepHyp

»&cp Umin(0)=2452959.1039999,

U(0)=2452959.104,

Umax(0)=2452959.2,

jdFid=2452972.104/

In this case, the *Fid JD* was moved to reflect the last date of the period, 28 November 2003.

This interactive modification process eliminated the need to change, resave, and reload the original input file. This also allowed CATO to begin the next optimization from parameter values that were already within a reasonable range from the actual solution. For the 14 November 2003 date, daily files were created for the dates 31 October 2003, to 28 November 2003, and for the 15 December 2004 date, the range was 1 December 2004 to 29 December 2004. The interactive optimization process went very smoothly. By observing the calculated parameter values after each optimization, it was possible to follow the direction of each variable, and determine before a continuation if a boundary value needed to be changed before any time was spent on the actual optimization. This helped save some time during the process, where optimizing one daily file took approximately five to ten minutes.

The process to create Periods 1a and 2a is identical, except the arrival date in the *Arrival Hyperbola* control point and *Pluto Arrival and Beyond* breakpoint were increased by six months. Each daily file was re-optimized with the new arrival date. Again, there were no problems encountered, and the re-optimization proceeded must the same as the first one. Appendix H contains a summary of the parameter values for the free variables of all daily files for all four launch periods. The charts are paged by control point, allowing easy analysis of the values for each point. For this analysis, this summary was very helpful in determining the trend of the parameter values, as described in the following section. Currently, there is no output option that allows saving the final parameters in a spreadsheet type file. The numbers in Appendix H had to be manually entered into Microsoft Excel.

B. DAILY PERFORMANCE OF SPACECRAFT IN LAUNCH PERIOD

1. C_3 Requirements

The C_3 defines the amount of energy required for the spacecraft to escape Earth's gravity and inject into a hyperbolic trajectory. Mathematically, it is the square of the hyperbolic velocity at infinity. This parameter is very widely used as a measure of performance for most launch vehicles; specifically, C_3 versus injected mass capability. Generally, the higher the C_3 the less mass the launch vehicle can place into the desired orbit or trajectory. CATO, in its algorithm to optimize a cost function, also minimizes the C_3 required, which minimizes the fuel required onboard. Therefore, it is most logical that the minimum C_3 values would correspond to the nominal launch date, and slowly increase the further away one moves from the nominal.

In this JGA analysis, the minimum C_3 does occur at the nominal. For 14 November 2003, the C_3 required is 106.35 km^2/s^2, and 119.19 km^2/s^2 for 15 December 2004. Figure 8 displays a graph of the C_3 values for Periods 1 and 1a, while Figure 9 shows the C_3 values for Periods 2 and 2a. In all cases, the left ordinate indicates the values of the required C_3. The absolute minimum occurs at the nominal launch date. A comparison of the two windows yields an initial conclusion that using the 10.5 year flight time is more economical, because the overall C_3 values are less.

2. ΔV Requirements

The ΔV, or amount of velocity change required for injection into the interplanetary Earth departure trajectory, increases with the amount of C_3 required. Figures 8 and 9 display the ΔV required for trajectory insertion for all four periods; the right ordinate indicates the values. The velocity change required to achieve the desired C_3, starting from a circular parking orbit, is calculated from (Barnett and Farless, 1996, p. 28):

$$\Delta V = \sqrt{\left(C_3 + \frac{2\mu}{r_{park}}\right)} - \sqrt{\left(\frac{\mu}{r_{park}}\right)},\ \text{km/s}$$

$$\mu = 3.986(10)^5\ km^3/s^2,\ r_{park} = 6563.1\ km$$

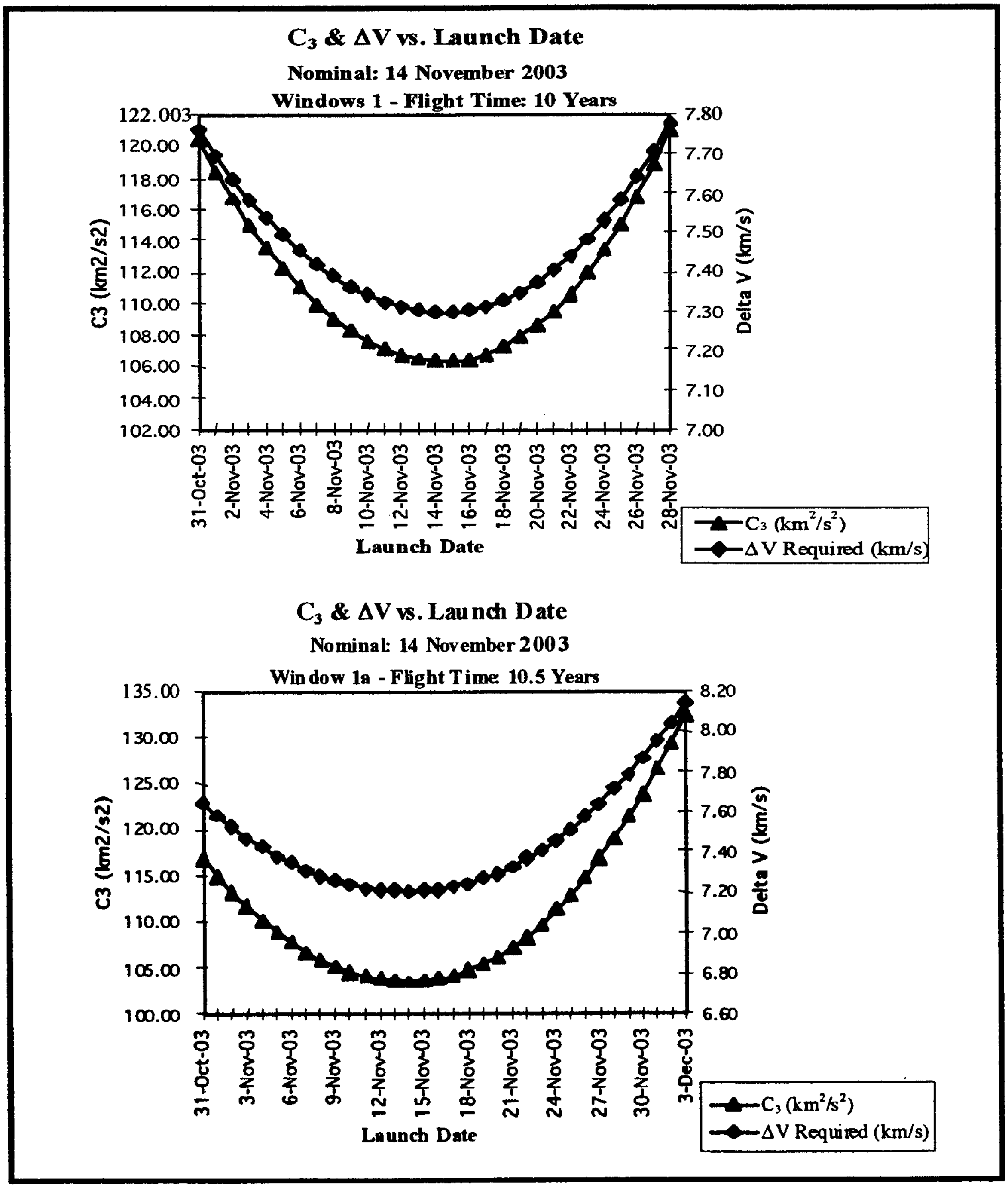

Figure 8: C_3 and ΔV for Periods 1 and 1a

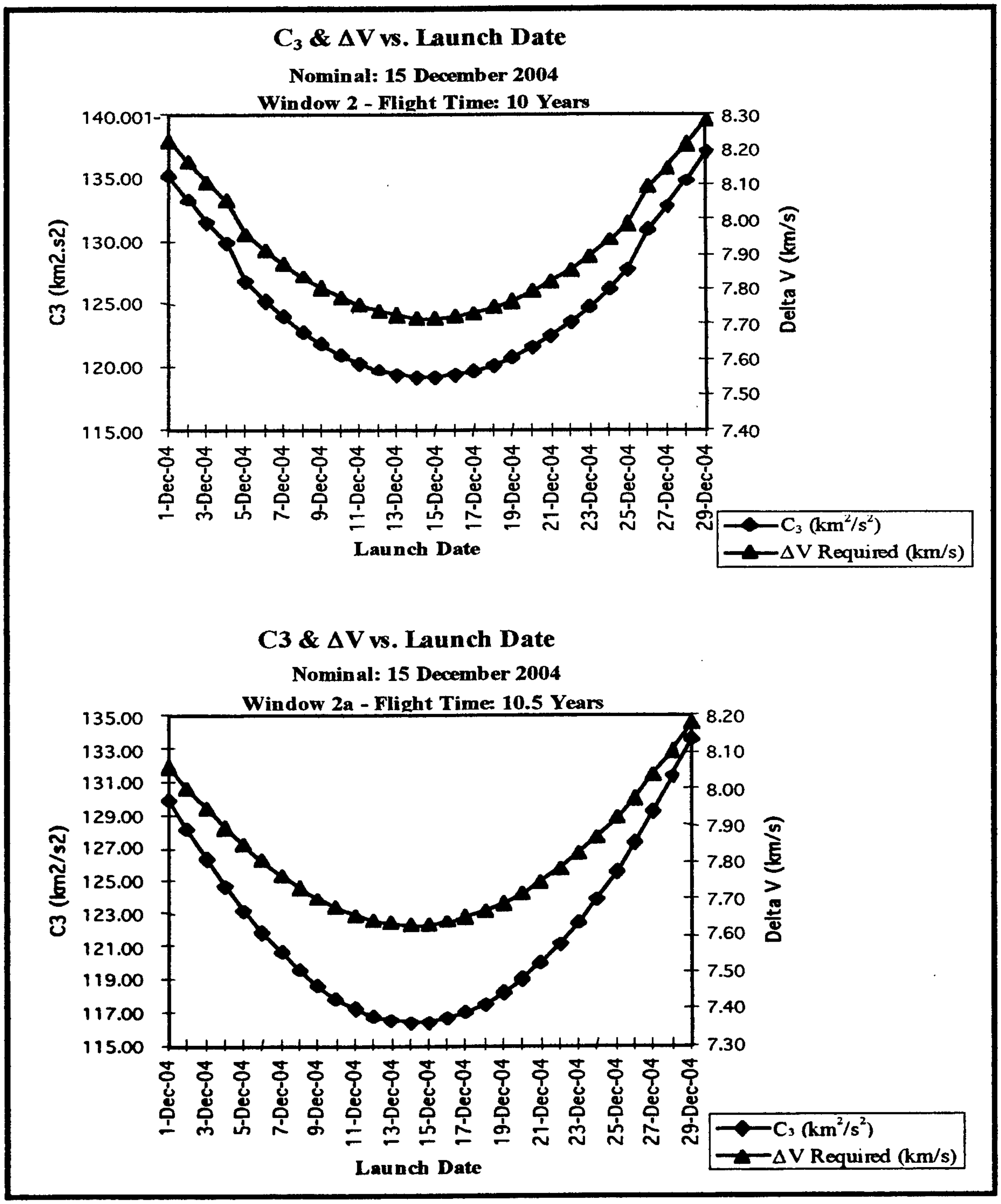

Figure 9: C_3 and ΔV for Periods 2 and 2a

Most references for launch vehicles, rocket motors, and upper stages list the ΔV capabilities, with a payload mass fraction of 0.5, meaning that the payload and initial rocket mass values are the same. The ΔV required for dates that are near nominal are relatively high, and a single medium-sized rocket motor may not provide enough thrust for the injection demand. In these cases, two motors may have to be stacked, or a launch vehicle with an integrated upper stage may be considered. For this JGA analysis, both possibilities will be investigated, plus the option of using a larger launch vehicle. The latter will probably not be a viable option due to launch costs.

3. Jupiter Flyby Constraints

The most critical constraint at the Jupiter control point was the altitude at the closest point of approach, the periapsis altitude. While a lower altitude would undoubtedly provide a greater energy and velocity increase from the gravity assist, there are a couple factors directly affecting the spacecraft that must be considered. First, Jupiter has a very strong magnetic field that could seriously damage the spacecraft's delicate circuitry and jeopardize the mission. If the spacecraft were to fly very close to the planet to gain a large energy boost, it will fly through more of the magnetic field and be exposed to it for a longer period of time. To alleviate this problem the spacecraft must fly at a distance that would decrease the effect of the field. However, the spacecraft would not gain as large an energy and velocity boost from the JGA, and hence require a greater C_3 at Earth departure in order to still fly by Pluto within the desired time frame.

Another important consideration is the velocity at which the spacecraft will be traveling as it performs the JGA. At the closest point of the JGA, the spacecraft will be traveling at its fastest speed, with an increasing acceleration approaching that point. The spacecraft must be structurally sound and be able to withstand the g-forces it encounters during the flyby. The spacecraft may have to endure additional structural testing on Earth that would definitely increase the overall cost of the spacecraft.

The general guideline expressed at JPL is that the JGA radius (measured from the center of Jupiter) should not be any closer than the orbital radius of Jupiter's moon Io, which is approximately 421,600 km, or 5.8 Jupiter radii. This altitude will serve to keep the spacecraft sufficiently out of the worst part of Jupiter's magnetic field. While the magnetic field is an important and potentially problematic consideration, it is not an absolute constraint. *Galileo* did venture below Io's radius just before its Jupiter orbit insertion point, passing by the planet at 4.0 Jupiter radii, or 285,968 km altitude (D'Amario, 1995, p. 1389). Another "Ice and Fire Preproject" mission, *Europa Orbiter*, will fly about as close to Jupiter's surface as possible without being a probe, at an altitude of 1428 km, only 1.02 Jupiter radii from the planet's center (Sweetser, 1997, p. 7).

Figure 10 displays graphs of the periapsis altitude for each launch date in all four periods. When conducting the preliminary analysis for the JGA using MIDAS, there was no indication that the periapsis altitude would be questionable on any of the periods. However, during the CATO optimization refinement, it was determined that all of the launch dates in Period 1, and approximately half of those in Periods 1a and 2a would result in JGA flyby altitudes lower than Io's orbital altitude. Since the last daily file in Period 1a indicated that the periapsis altitude was still above the specified value, the period was extended, to include those dates that result in satisfactory flyby altitudes. Therefore, as indicated in Figures 8 and 10, Period 1a begins on 31 October 2003 and ends on 2 December 2003, a five day extension of the original period. For all of the following analyses, this extended period will be used for Period 1a. The parameter values for the extension days are also included in Appendix H.

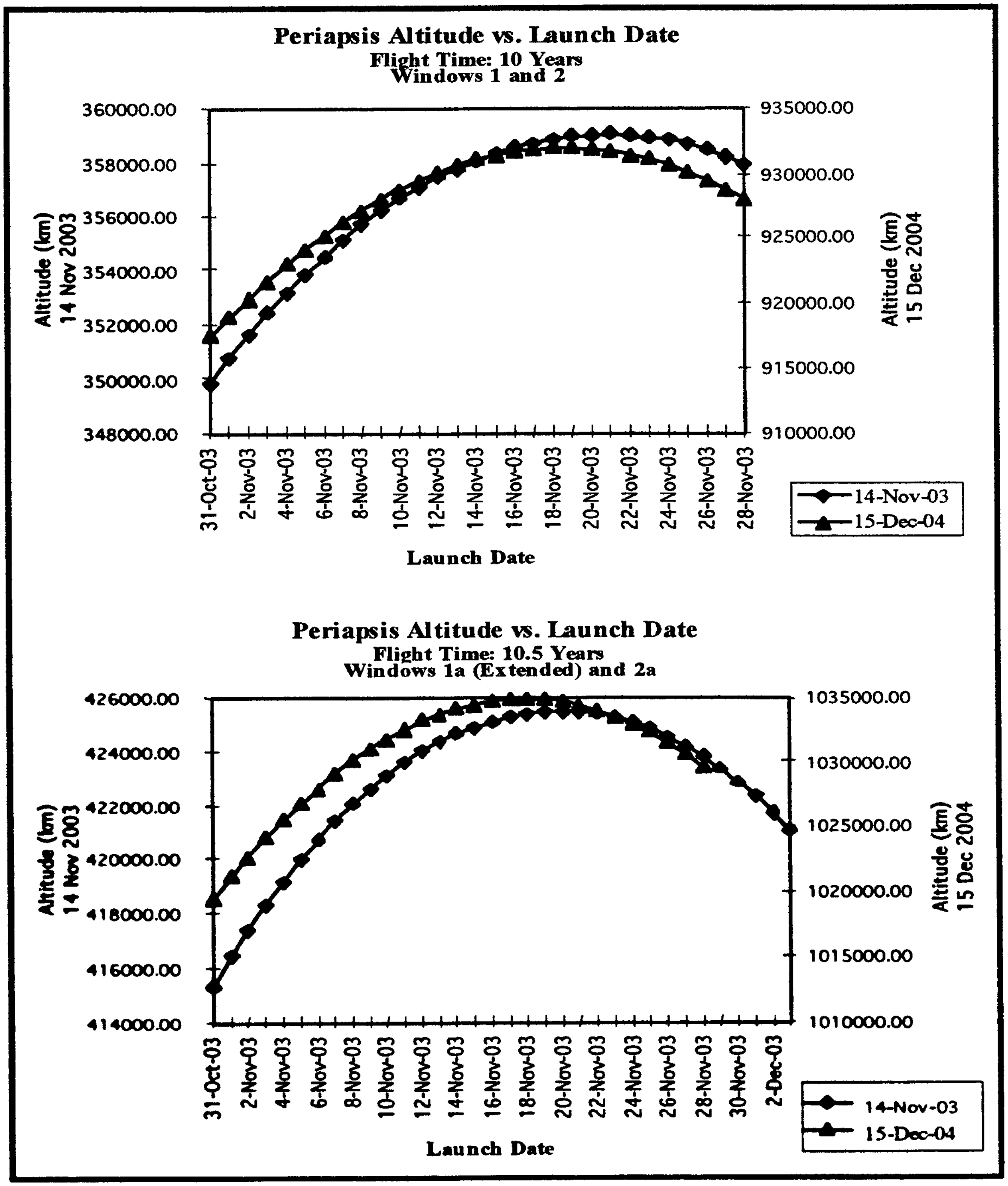

Figure 10: Periapsis Altitude Comparison for all Periods

V. ANALYSIS OF POTENTIAL LAUNCH VEHICLES

This section contains discussions and analyses for potential launch vehicles for the PKE. The analyses will include all four launch periods, to determine the optimum vehicle for each possible launch date. The calculations contained in this section and the corresponding appendices are all specific to the PKE's mass. For example, a ΔV given for a specific rocket motor may not be the maximum amount available from that motor, but it is the maximum available given the PKE's mass and size. To perform the interplanetary injection, only multi-staged launch vehicles capable of GTO, elliptical, or interplanetary orbits were considered. In most cases, the upper stage of the launch vehicle does not provide sufficient ΔV to inject the payload into the desired orbit with the desired velocity. For these cases, an additional rocket motor is necessary to augment the upper stage's capability. To maintain low launch and integration costs, the chosen rocket should be as small as possible while supplying the requisite ΔV. Accounting for size, wet mass, and dry mass, the following rocket motors were examined in this analysis: STAR 27, STAR 30BP, STAR 48B, and STAR 63F. Appendix I contains a more detailed table of motor characteristics, with applicable equations. The majority of this analysis was performed using Microsoft Excel, except where indicated.

STAR Motor Type	STAR 27	STAR 30BP	STAR 48B	STAR 63F
Length (m)	1.2376	1.5062	2.0320	2.7089
Diameter (m)	0.6934	0.7620	1.2446	1.5900
Ignition Mass (kg)[1]	361.20	544.90	2143.00	4591.67
Burn-Out Mass (kg)[1]	24.22	34.75	118.61	291.80
Average Thrust (N)	25,700	26,500	68,400	105,000
Effective I_{SP} (s)	288.0	292.0	292.1	297.1
Burn Time (s)	34.4	54.0	84.1	120.0
Payload Mass Fraction[2]	0.31	0.23	0.07	0.03
ΔV (km/s)	2.94	3.68	6.05	6.86
g-Force Before Burn	5.03	3.83	3.03	2.25
g-Force After Burn	14.22	13.87	25.03	23.69

(1) Includes remote Safe and Arm (S&A)/Explosive Transfer Assembly (ETA) weight.

(2) Calculated using PKE payload mass of 160 kg.

Table 4: STAR Rocket Motor Characteristics (Barnett and Farless, 1996, p. 27)

A. DELTA 7925

The Delta 7925 is a three-stage Medium Launch Vehicle (MLV) and the largest of the long line of Delta rockets, capable of carrying a maximum payload of 1840 kg into geosynchronous transfer orbit (GTO). It is one of the two Delta II rockets in current production. The four-digit designator "7925" indicates that the rocket has an RS-27A engine (Extra Extended Long Tank), nine augmentation solid rocket graphite epoxy motors (GEMs) surrounding the base, an AJ10-118K aerojet, and a PAM-D derivative upper stage motor (STAR 48B). The primary mission of the Delta 7925 is to place payloads into GTO or polar elliptical orbit, but can also provide enough thrust to boost a small payload into an interplanetary trajectory. Launched from either Vandenberg Air Force Base, CA, or Cape Canaveral Air Force Station, FL, the estimated launch costs are $45-50 million (Isakowitz, 1995, p. 233).

Use of a Delta 7925 to launch the PKE would change the orbit injection procedure slightly. For an interplanetary trajectory, the Delta does not place the payload in an initial parking orbit. Instead, it directly injects the payload into the trajectory from the launch path. JPL would have to accommodate this by having the Delta 7925 launched just prior to the designated Earth Departure time given by CATO, and ensure that the third stage and any additional inertial upper stages (IUS) were timed to the given event epoch.

The third stage of the Delta 7925, the STAR 48B motor, injects the payload into a GTO or elliptical trajectory. As listed in Appendix H, minimum ΔV for all periods required to transfer out of Earth orbit is 7.21 km/s. However, the STAR 48B provides only 6.86 km/s, therefore, an additional rocket motor is necessary to augment the STAR 48B's capabilities. To perform this analysis, the additional STAR rocket motor was added to the PKE, increasing the total payload mass that the third stage has to carry. The procedure began by choosing a STAR motor that is of reasonable mass for the Delta 7925, less than 1680 kg (PKE mass subtracted from the maximum capable to GTO, 1840 kg). In this case, the STAR 27 and STAR 30BP are acceptable. The motor's wet mass is added to the PKE mass to yield the total mass that the third stage must carry. From this total mass, the necessary C_3 is be obtained using "C_3 vs. Injected Mass" graphs found in the Delta II Payload Planners Guide (1993, p. 2-23). As it was more useful to have a graph of "Injected Mass vs. C_3," the program DeltaGraph™ 4.0 was used to plot several key points in the curve and generate a sixth order polynomial equation to model that curve. This equation was then used in Excel to obtain the C_3 necessary to inject the given mass into the desired trajectory. From the C_3, the corresponding ΔV is obtained and added to the ΔV from the additional STAR motor. This results in a total ΔV that the Delta third stage plus an additional STAR motor can accomplish. By comparing this number with the ΔV's required for every daily file, a profile is generated that determines which launch dates the Delta 7925 in combination with an additional STAR motor can be used for launch. For days where the necessary ΔV is above this threshold, a different combination is necessary: either a larger STAR motor, or a larger launch vehicle. Table 6 in Appendix I contains the detailed information pertaining to the STAR rocket motors. The calculations used to determine the Delta 7925 performance profile are contained in the *Delta 7925 Calculations* section, Tables 7 and 8, of Appendix I. Figure 11 compares the ΔV requirements for each day all four periods with that obtainable from the Delta 7925 in combination with a STAR 27 and 30BP. For the launch periods in 2003, either Delta combination will provide sufficient thrust to the payload. In 2004, the Delta 7925 can be used only during a small portion of each period, as indicated on the graph. For the outlying points, a larger launch vehicle must be considered.

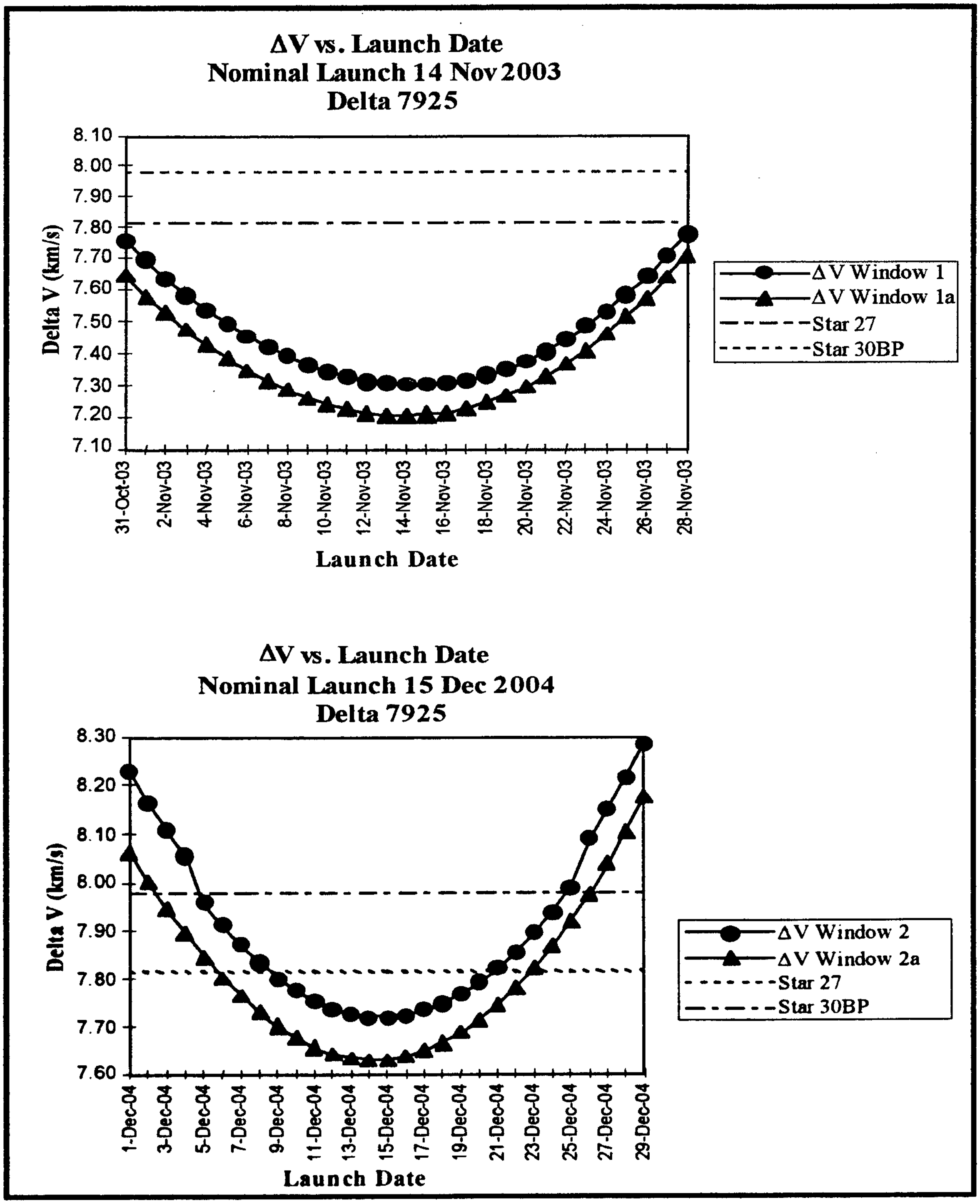

Figure 11: Delta 7925 ΔV Comparisons

B. ATLAS II

The Atlas family of medium lift launch vehicles provide a greater lift capability that the Delta 7925. The smallest of those currently in production and utilization, Atlas I, can put a payload of 2255 kg into GTO. The primary missions of the Atlas I and its follow-ons, Atlas II, Atlas IIA and Atlas IIAS, are to place payloads into low Earth orbit and GTO. Launched from either Vandenberg or Cape Canaveral, the launch costs for the Atlas I are $65-75 million, $75-85 million for the Atlas II, $80-90 million for the Atlas IIA, and $95-105 million for the Atlas IIAS (Isakowitz, 1995, p. 207). There are three versions of the Atlas upper stage, the Centaur I (RL 10A-3-3A), Centaur II (RL 10A-3-3A, slightly larger size), and Centaur IIA (RL 10A-4N).

The analysis procedure for the Atlas Centaur differs slightly from that of the Delta combinations in that the Centaur performs a burn that injects the payload into a circular parking orbit, then restarts to boost the payload into GTO. This analysis includes the Centaur II and Centaur IIA, but the numbers for the Centaur II are comparable for the Centaur I. The launch profile is developed by comparing the total ΔV available from both Centaur burns to the ΔV required for the interplanetary injection.

The method for determining the available ΔV is the same for all launch vehicles that place their payloads into a circular parking orbit before performing Earth-escape maneuvers. The upper stage is considered separately from the rest of the launch vehicle, using the standard Rocket Equation.

$$\Delta V = \frac{g_0 I_{sp} \ln\left(\frac{M_i}{M_f}\right)}{1000}, \text{ km/s}$$

$$g_0 = 9.81 \text{ m/s}$$

The *Atlas Calculations* section in Appendix I lists the necessary input values and resulting ΔV's calculated for the two Centaur upper stages listed in Table 9. These calculations also accounted for gravity losses, by reducing the ΔV available from the upper stage by 5%. The boost available by the Centaur rockets exceeds the required ΔV that is listed in each daily file. Even the least capable combination, Atlas I with a Centaur I upper stage, will provide sufficient ΔV to place the PKE into the proper interplanetary trajectory, without the need for an additional STAR motor attached to the spacecraft. Figure 12 compares the ΔV available from the two Centaur stages to the requirements for each day of all four periods. The daily file data is taken from Appendix H, and the Centaur data is from Table 9 in Appendix I.

C. SOYUZ/MOLNIYA

The Soyuz is Russia's most commonly used medium-lift launch vehicle that can be launch from either the Plesetsk Cosmodrome or the Baikonur Cosmodrome at Tyuratam. Its primary mission is placing spacecraft into LEO, and has launched manned Soyuz spacecraft, unmanned Progress resupply vessels, and Kosmos observation satellites. The Molniya launch vehicle is a Soyuz with a third stage that boosts the payload into the highly elliptical orbit that bears the same name or sun synchronous orbits. The Soyuz by itself can place a 7000 kg payload into LEO, while the Molniya can put 1800 kg into a circular sun synchronous orbit or 2000 into a Molniya orbit. With a relatively short launch preparation time when compared with US launch vehicles, the estimated launch costs are $12-25 million (Isakowitz, 1995, p. 164).

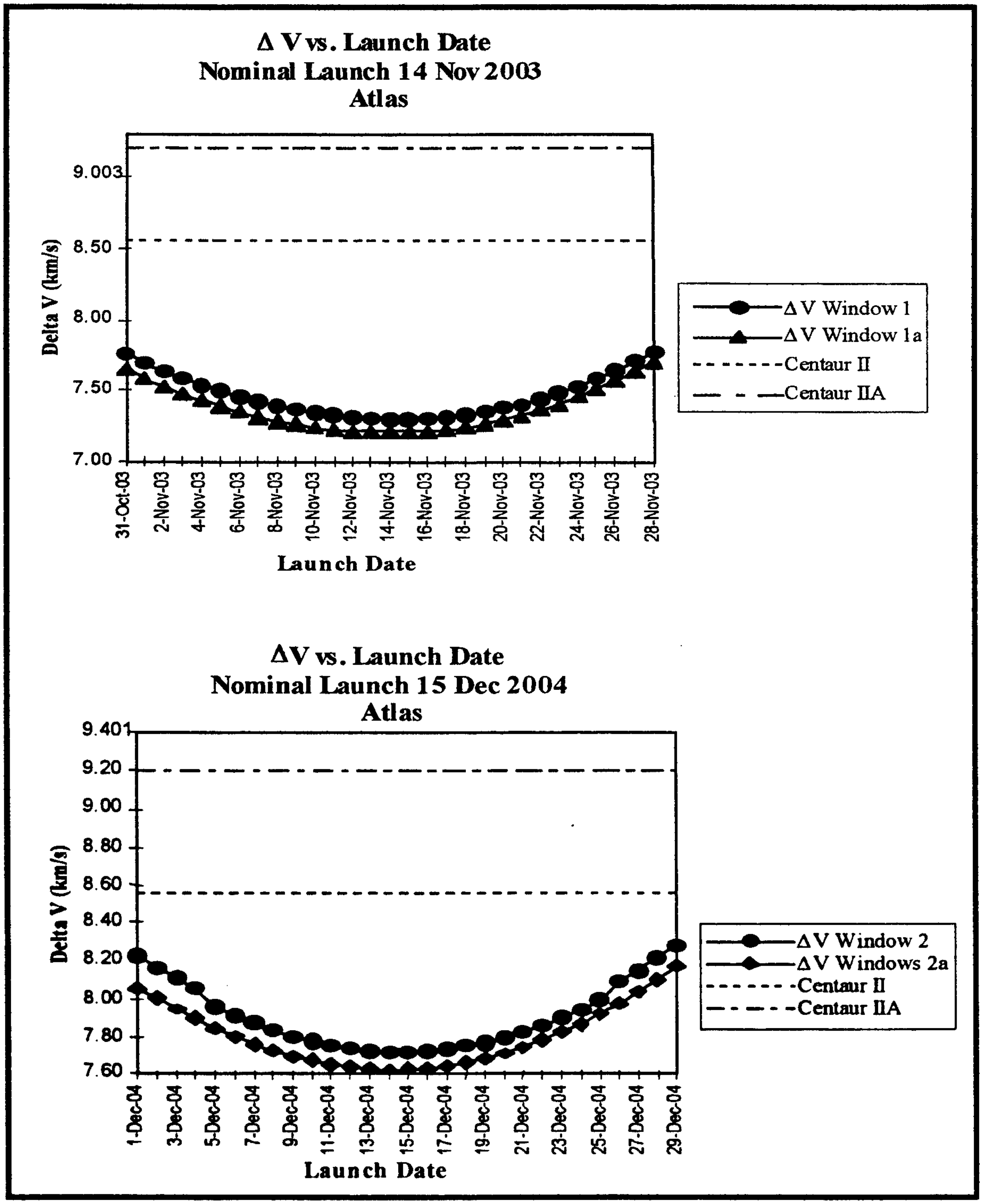

Figure 12: Atlas Centaur ΔV Comparison

The analysis procedure for the Molniya upper stage, Block L, is similar to that of the Atlas Centaur. The ΔV attainable by the Block L is calculated using the same equation listed in the previous section for the Atlas, and compared against that required by all four periods. If the upper stage alone does not provide enough energy, then a STAR Rocket motor must be added to the payload to provide the needed ΔV. Of the four STAR motors considered for this analysis, only the STAR 63F was too massive for the Block L to carry. The Molniya can place approximately 2500 kg into the desired parking orbit of 185 km altitude. The total mass of the PKE plus the STAR 63F is over 4500 kg. The *Molniya Calculations* section of Appendix I displays all of the calculations for the Block L (Table 10) and the Block L plus the remaining three STAR motors (Table 11).

The Block L can only provide 4.50 km/s of ΔV, which is not nearly enough for any of the launch periods (see Appendix H). Even with the addition of the STAR 27, there is still not enough boost to reach the desired Earth escape velocity. The STAR 30BP can accommodate a small portion of the launch dates in Period 1a, and only the nominal date in Period 1. The STAR 48B in combination with the Block L can provide sufficient ΔV for all of the launch periods. Figure 13 compares the ΔV available from all three STAR motor combinations with that required for each launch date.

D. PROTON

The Proton is the next in Russia's line of medium-lift launch vehicles. There are two versions currently in production, the D-1 and D-1-e. While the D-1 is limited to LEO missions, the D-1-e has a fourth stage that allows it to place payloads in LEO, GTO, GEO, or interplanetary orbits. Approximately 90% of all Proton launches have been the D-1-e, and can only be launched from the Baikonur Cosmodrome in Tyuratam (Isakowitz, 1995, p. 145). The minimum performance for the D-1-e is the placement of a 2200 kg payload into geostationary orbit. The launch vehicle is capable of launching up to 4000 kg into a Mars or lunar transfer orbit. The estimated launch costs for both Proton vehicles is $50-70 million (Isakowitz, 1995, p. 147).

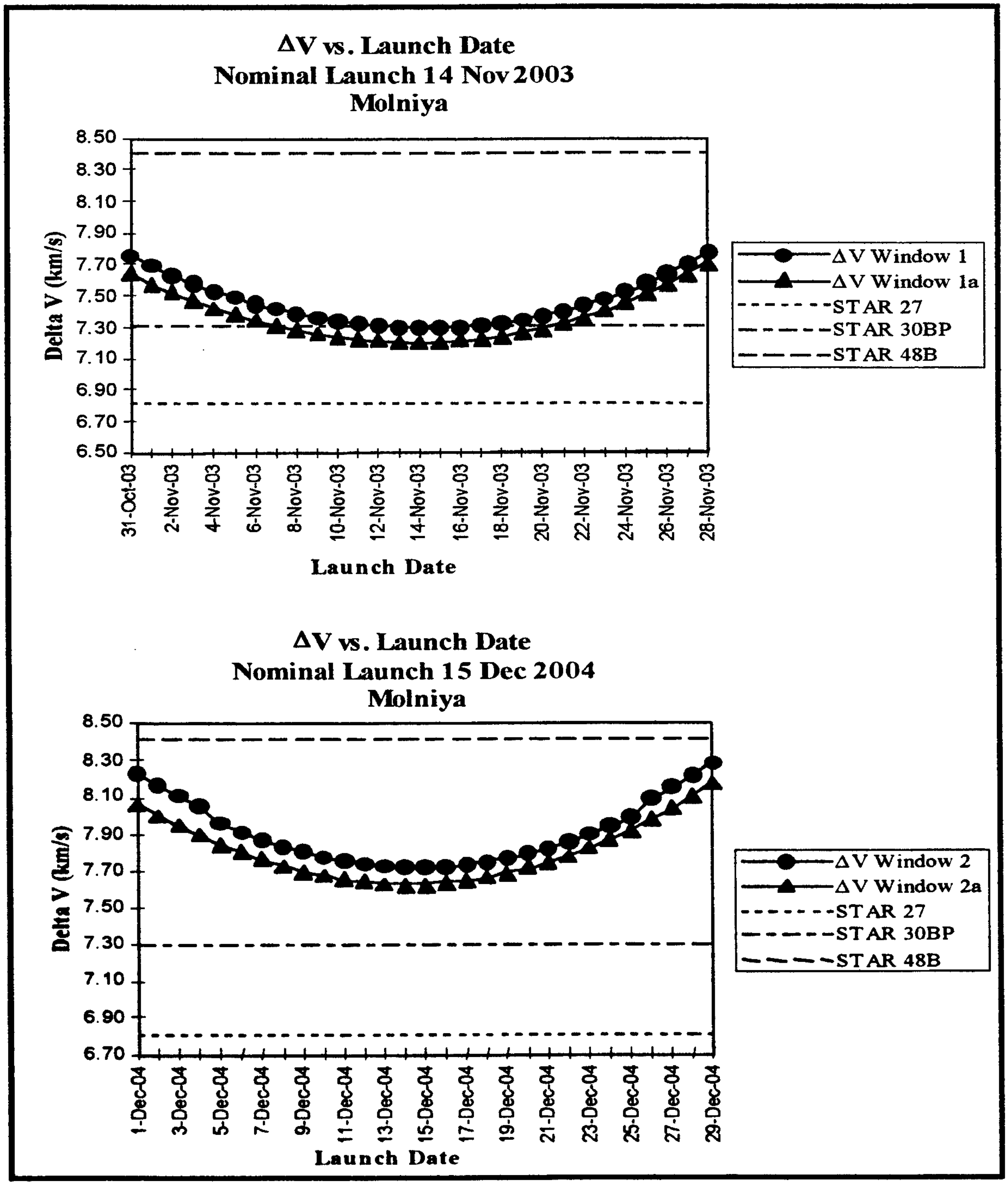

Figure 13: Molniya Block L ΔV Comparisons

The Proton's upper stage, the Block D, is about four times larger than the Molniya's Block L upper stage, but provides only 25% more average thrust. Like the Block L, the Proton's Block D does not provide enough ΔV to ferry the payload to interplanetary orbit, contributing only 5.95 km/s of ΔV. The analyses for the Block D (Table 12) and the Block D with STAR motors (Table 13) are included in the *Proton Calculations* section of Appendix I, and are similar to that of the Molniya. An additional STAR motor must be attached to the payload to compensate for the shortcomings of the Block D. All but the STAR 63F can be accommodated by the Block D's fairing and load-bearing capabilities. The STAR 63F plus the PKE payload equal a mass of over 4700 kg, which is just beyond the Proton D-1-e's performance range (Isakowitz, 1995, p. 151). As shown in Figure 14, the STAR 27, 30BP, and 48B all provide ample thrust to put the PKE in the proper interplanetary trajectory on any of the subject launch dates.

E. ARIANE 4 AND ARIANE 5

The workhorses of the European Space Agency (ESA), the Ariane rockets were developed and tested by a consortium of European engineers in the mid to late 1980's. The launch site, located at Kourou, French Guyana, near the equator, is ideal for ESA-sponsored and cooperative payload launches because of the accessibility to all inclinations and the lack of launch-hindering activity such as hurricanes, earthquakes, and high population densities. The Ariane 4 is the main medium-lift capable launch vehicle and has six variations to suit a wide variety of needs. The Ariane 5 is the follow-on version that has a slightly higher vehicle performance than the Ariane 4. The Ariane 4 and 5 primary missions are to place payloads into LEO, GTO, and polar orbits. The ESA has early-on the Ariane development process designed a launch structure that would increase the Ariane's capability and marketability by launching two PAM-D payloads simultaneously on the same vehicle. The Sylda, Systeme de Lancement Double Ariane, allows the Ariane 5 to launch a dual payload totaling almost 6000 kg into GTO. The increases size and capacity of the Ariane 5 is reflected in its estimated launch costs, about $120 million for a dual launch. The smallest version of the Ariane 4, the AR40, costs only \$45-60 million, while the largest, the AR44L, costs \$90-100 million (Isakowitz, 1995, p. 34).

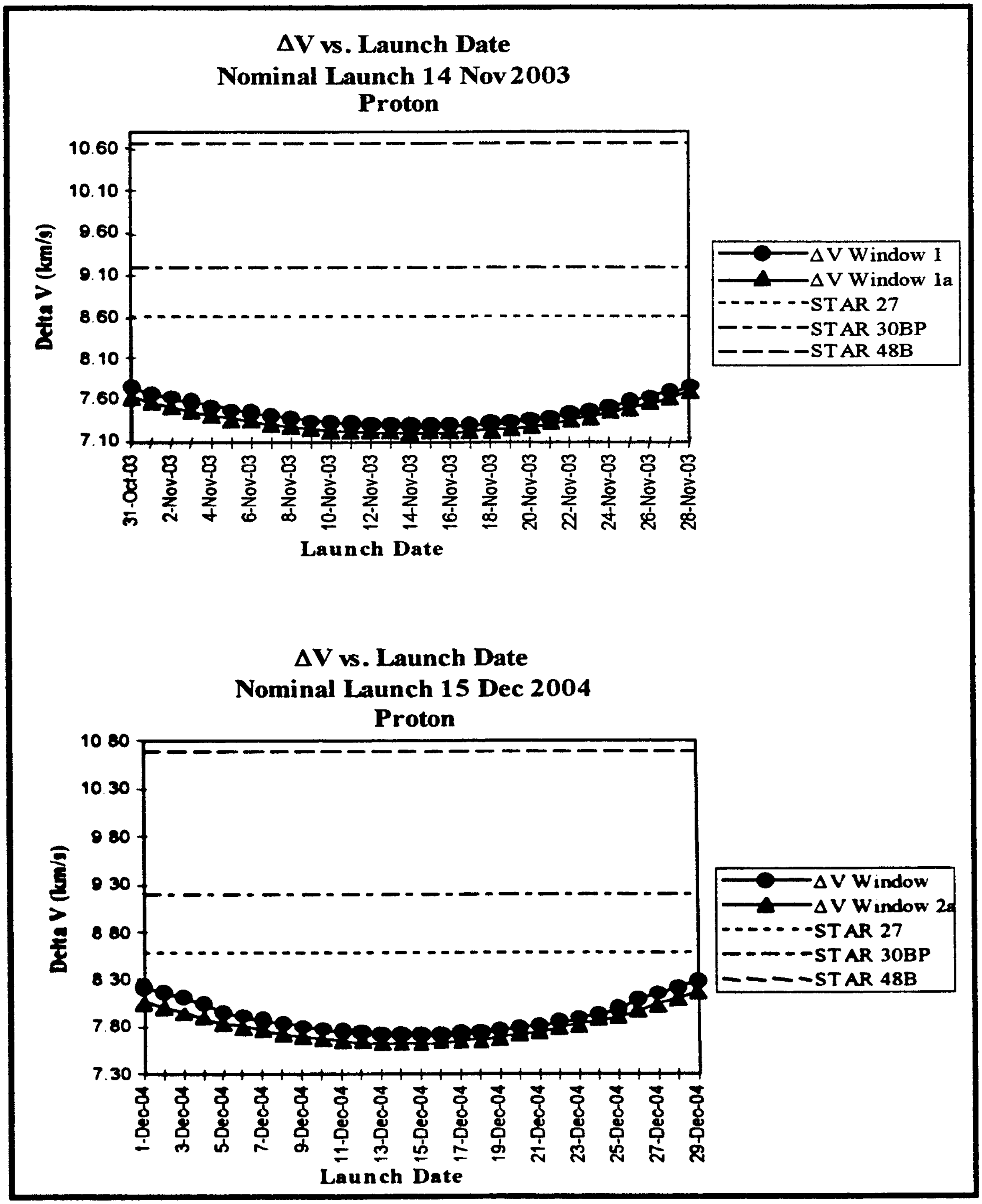

Figure 14: Proton Block D ΔV Comparisons

The analysis for the Ariane 4 upper stage, HM7B, revealed that though the overall launch vehicle itself provided less capability than the larger Ariane 5, the HM7B was able to provide sufficient ΔV (9.27 km/s) for interplanetary injection during all launch dates. The Ariane 5's upper stage, the L9, is slightly smaller than the HM7B, and provides less thrust, and thus is not able to place the PKE into the desired interplanetary orbit on it's own. The Ariane 5 is able to carry a larger payload, however, because its second stage, or core stage, is significantly larger and more powerful than the Ariane 4's corresponding stage, supplying 43% more average thrust, allowing the payload to be placed into a higher altitude orbit. The ΔV achievable by the Ariane 5's L9 is 6.66 km/s, slightly less than that required on the nominal launch dates, therefore, a STAR motor is required to provide the extra boost. Table 14 of section *Ariane Calculations* located within Appendix I details the characteristics of each upper stage, while Table 15 of that same section lists the calculations for the L9 combined with the STAR rocket motors. While no additional STAR motors are required for the Ariane 4, all four specified STAR motors are compatible with the Ariane 5. Figures 15 and 16 compare the required ΔV with that provided by the upper stages.

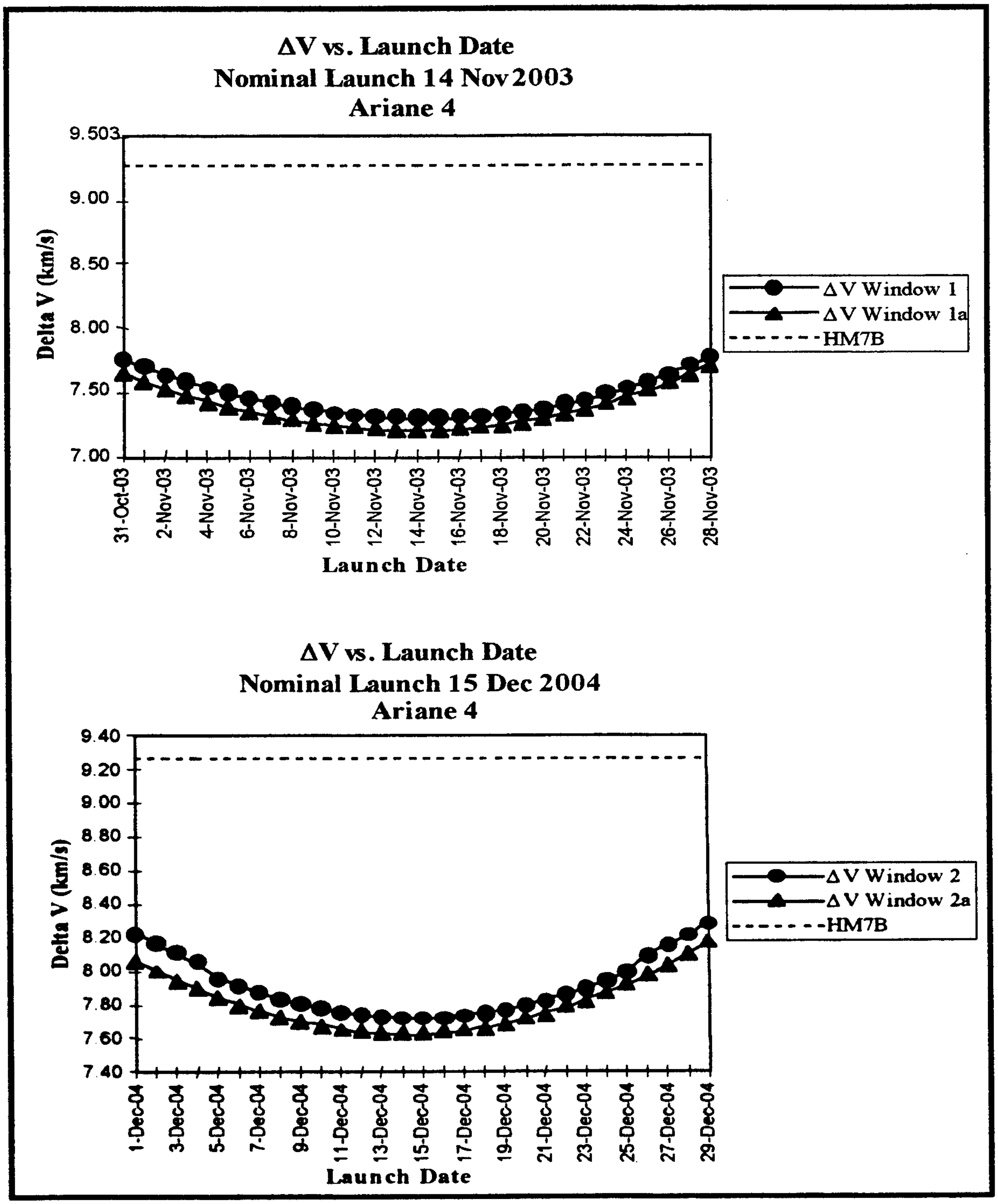

Figure 15: Ariane 4 HM7B ΔV Comparisons

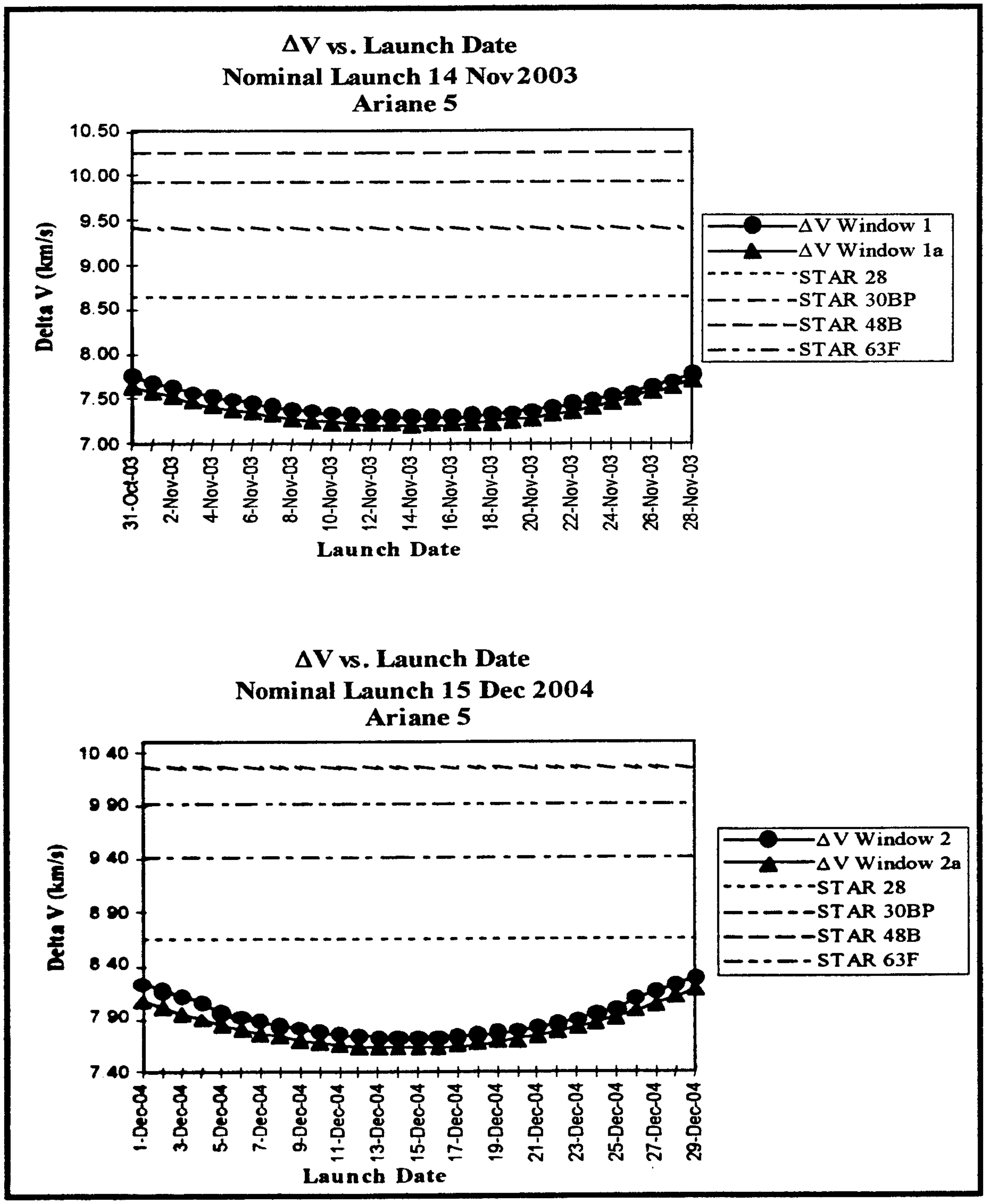

Figure 16: Ariane 5 L9 ΔV Comparisons

VI. SUMMARY AND RECOMMENDATIONS

A. COST COMPARISON

One of the most important factors in any space mission development is cost. Among other things, the launch vehicle and associated support costs can occupy a large portion of the program's budget. In this JGA analysis, many combinations of launch vehicles and additional rocket motors were examined, providing a cost spectrum that ranges from the low cost to the expensive. Table 5 summarizes the estimated costs for the launch vehicles and STAR rocket motors* investigated in the previous chapter. These costs must then be coupled with their performance during all of the launch periods, to determine feasibility. In addition, the size of the STAR motor will incur additional costs; generally, the larger the motor and the more fuel it carries, the more expensive. Procuring a small inexpensive launch vehicle for the PKE may be good for the budget, but not if it can only be used on the nominal launch dates. In some cases, it may be more prudent to use a slightly larger, more expensive vehicle if it will allow more flexibility in the chosen launch date, or if a contingency arises and the nominal launch date must be scrapped. For this analysis, launch cost and STAR motor costs will be factors, but the launch vehicle recommendation will be based primarily on the performance, as described in the following sections. The cost of a STAR motor is so small in comparison to even the smallest of the given launch vehicles that it will not drive the launch vehicle decision.

B. 14 NOVEMBER 2003 OPPORTUNITY

Figure 17 displays a summary of the viable launch vehicles and STAR motor additions over the span of the 2003 launch period. Based solely on the cost information from Table 5, the recommended launch vehicle is a Molniya plus a STAR 30BP on the

* STAR rocket motor cost figures from current catalog provided by Mr. Michael Kramer, Program Manager, Thiokol, Inc., via telephone. These figures are in FY98 dollars. STAR 63F cost is only a rough estimate due to unresolved issues involving non-recurring costs.

payload. Although this defines the shortest launch period, it is the least expensive option. For a ten year flight time, the launch period is five days long, which does allow for sufficient planning, however, the meteorological conditions of the launch site must also be considered to determine if this period is actually long enough to account for contingencies. By conducting a statistical analysis using historical launch data a probability of successful launch can be established. JPL can use this study to decide if this launch vehicle combination is cost effective. However, the logistics and associated costs of planning and executing a launch off foreign soil may totally negate the cost savings offered by the launch vehicle. In addition, the legality of launching a US government payload from a non-US vehicle must be considered, and may preclude the use of the Molniya. If this is not feasible, then the recommended launch vehicle is the Delta 7925 with a STAR 27. This option is more expensive, but provides a full launch period and alleviates the need for a statistical launch analysis and additional logistics planning, which in turn reduces the overall planning and associated costs.

Launch Vehicle/Rocket Motor	**Estimated Launch Costs ($ million)**
Molniya	12 - 25
Delta 7925	45 - 50
Ariane 4 (AR40)	45 - 60
Proton	50 - 70
Ariane 4 (AR42P)	60 - 75
Ariane 4 (AR42L) Atlas II	75 - 85
Atlas IIA	80 - 90
Ariane 4 (AR44P) Ariane 4 (AR44LP)	80 - 95
Ariane 4 (AR44L)	90 - 110
Atlas IIAS	95 - 105
Ariane 5	120
STAR 27	1.130
STAR 30BP	1.145
STAR 48B	1.670
STAR 63F	3.5

Table 5: Launch Vehicle and STAR Rocket Motor Cost Comparison

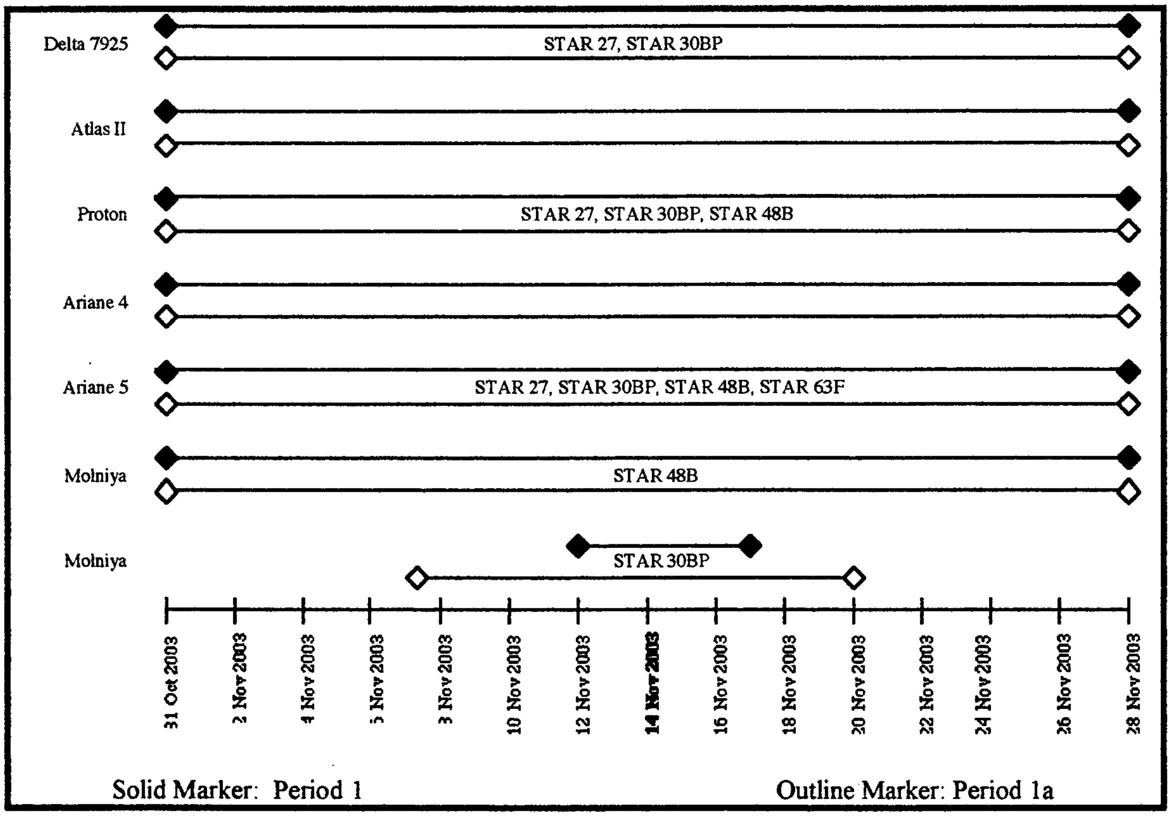

Figure 17: Launch Vehicle Summary for Periods 1 and 1a

C. 15 DECEMBER 2004 OPPORTUNITY

Figure 18 shows a summary of the viable launch vehicles plus STAR motors analyzed over the span of the 2004 launch period opportunity. There are two major differences between this summary and that for the 2003 opportunity. First, there is only one STAR motor option for the Molniya, the STAR 48B. Secondly, there are still two options for the Delta 7925, but they do not span the entire launch period. This is because the C_3 required at each 2004 launch date is slightly higher than the 2003 launch dates. Based only on cost information contained in Table 5 combined with the information in Figure 18, the recommended launch vehicle is the Molniya with a STAR 48B rocket motor, as this is the least expensive option. However, as stated in the previous section, the logistics of launching from a foreign site may negate the vehicle savings, and the issue of legality based on US foreign policy would have to be carefully considered. With this in mind, the recommended launch vehicle then becomes a Delta 7925 with a STAR 27 motor attached to the payload. This affords a generous twelve day launch period for a ten year flight time, seventeen days for a 10.5 year flight time. The Delta 7925 is slightly more expensive than the Molniya, which would allow the entire twenty nine day launch period. The costs of the STAR 27, 30BP, and 48B rocket motor are relatively equal, so their costs do not significantly impact the cost comparison (Table 5). Use of the STAR 30BP with the Delta 7925 would extend the available launch period, but provides more capability than is required. The slight extra cost may not be justified since the STAR 27 does provide a sizable period.

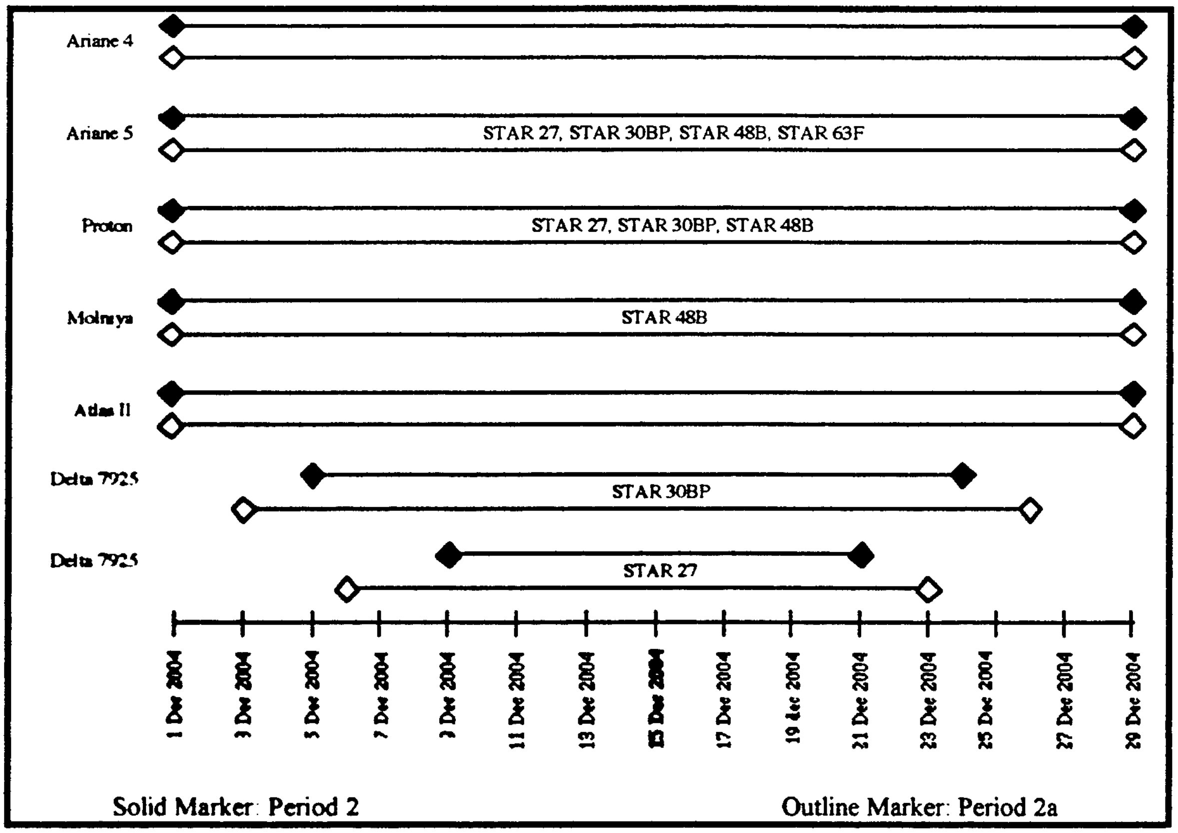

Figure 18: Launch Vehicle Summary for Periods 2 and 2a

D. POSSIBILITIES FOR ADDITIONAL STUDY

This JGA analysis provides the Ice and Fire development team a preliminary study in determining the most feasible launch period and launch vehicle for the PKE. The majority of the data for the launch period analysis was produced by MIDAS and CATO, and have furnished accurate information pertaining to the trajectory to Jupiter and Pluto. The launch vehicle analysis was conducted using information from various sources to determine the capabilities of the vehicles in question. Factors not accounted for in this analysis include gravity loss (except the Atlas analysis, since these effect could be approximated with a standard loss figure) and other finite burn effects. The calculations were performed on a strictly theoretical basis, using orbital mechanics and the rocket equation. Further definition of this path to Pluto can be conducted to include these effects and produce more detailed results, including not only the viable launch vehicles but also launch site information and vehicle integration. Additional analysis must also be conducted for the launch into the Earth parking orbit, to ensure the orbit is of the correct inclination and timed correctly for the Earth departure hyperbola to Jupiter. Lastly, NASA is interested in further lowering overall costs by launching two or more missions on one launch vehicle. A study that examines the possibility of launching the PKE with the Ice and Fire preprojects Europa Orbiter and Solar Probe, or future Discovery missions to Jupiter and beyond could prove invaluable to the progress of NASA's next generation of smaller, cheaper, more advanced exploratory spacecraft.

APPENDIX A: DESCRIPTION OF SOME MIDAS INPUT VARIABLES†

Name	Default	Description
altb	0	Altitude constraint for first body
body	blank	Name(s) of intermediate body(ies)
bulsi	blank	Name of target body
c3	0	Constrained departure energy, C_3, km^2/s^2
dvpl	blank	Magnitude of deterministic post launch ΔV, km/sec
head	blank	Title or description of trajectory
jdate	10^{10}	Departure time
jdl	1990,1,1	Epoch calendar date: year, month, day, hours, minutes, seconds
kadp	0	Launch adapter mass proportionality coefficient
minyn	blank	Increment limits, minimum, delta, maximum
nda	2	Type flag for target body
ndb	2	Type flag for first body
nlv	9	Launch vehicle identifier
poi	90	Parking orbit inclination, degrees
rcb	blank	Periapsis distance for unpowered swingby of first intermediate body, km/radii
shota	blank	Name of departure body
tend	10^{10}	Flight time, days
tm1	10^{10}	Time of first deep space maneuver
tpb	10^{10}	Arrival time at first body
varyi	blank	Independent search variables
varyn	blank	Parameter study variable
vlist	blank	List of saved variables

† Reference (Sauer, 1991, pp. 7-14) contains extended list of input variables, but does not contain definitions for all variables used.

APPENDIX B: MIDAS COMMAND LINE OPTIONS

Letter Designation	Description
A	Flag to allow maneuver addition for converged trajectories with large gradients
B	Flag to save a trajectory file of spacecraft state vectors at intervals defined by DELPO in an unformatted or binary form
C	Flag to print ongoing search information on screen
D	Flag for a post mortem dump of common blocks
E	Flag for a condensed output file with additional data for perihelion and aphelion
F	Flag to save a trajectory file of spacecraft state vectors at intervals defined by DELPO in a formatted format
G	Flag to grab additional deep space maneuvers when required
H	Flag to dump some additional initialization data
I	Flag for detailed dump of search variables
J	Flag to allow unpowered swingby trajectories to be targeted with a deep space maneuver when the swingby altitude is above the minimum constrained altitude, equivalent to setting the variable array NVB positive
[‡]L	Flag for a brief appended one line trajectory summary
M	Flag for post mortem dump of deep space and flyby maneuver scans
N	Flag to prohibit optimization and search
O	Flag to prohibit trajectory updates, equivalent to setting NIMP=0
P	Flag to calculate finite difference gradients
Q	Flag to dump initialization variables at each trajectory update
S	Flag for a detailed trajectory output file
T	flag for a trace of trajectory variables

[‡] No explanation is offered in the reference (Sauer, 1991, pp. 5-6) as to why there is no option listed for the letter "K," and likewise for the letters "R," "W," and "Y."

U	Flag to allow more than one deep space maneuver to be added to a converged trajectory, equivalent to setting NIMP=2
V	Flag for a condensed trajectory output file, same format as with option "E" except without aphelion and perihelion printout
X	Flag to inhibit addition of maneuvers
Z	Flag to trace search for internal body targeting

APPENDIX C: MIDAS OUTPUT FILES

e59_03a.out

```
MIDAS    ver 16.07    6/6/96

      Jet Propulsion Laboratory
   California Institute of Technology
         SUN FORTRAN 1.4

 head='2000 V-M-V to Outer Solar System'
 shota='Earth',
 bulsi='Pluto',
 body='Jupiter'
 jdl=2004,1,1.0,ndb=2,ALTB=71400,poi=30,nda=1,
 varyi='jdate','tpb','tm1',
 nlv=28
 lvcont=.1
 kadp=.05
 tpb=500.,
 tend=9.
 varyn='tend',minyn=9.,.5,13.
 vlist='+jdate','tend','c3','dvpl','rcb','mnet'
 choice=0
 jdate=0, $
 $end

 Input File is: e59_03a.inp
 Output File is: e59_03a.out


 Start Time:    Nov 18, 1996    09:23:59


2000 V-M-V to Outer Solar System            Nov 18, 1996    09:23:59

                        Heliocentric Ephemeris in AU and Days
                         Earth Ecliptic and Equinox of 2000
                                  Minimum Delta-V
                       launch vehicle: Delta II (7925)/Star-30C

                      Trajectory 3.01   Search converged in 9 iterations

              Minimum Delta-V= 7.2925633 km/s      Gradient Norm= .96142330E-09

   nv=  2  0   tend=   10.000

   jdate          tpb
 -47.395692     518.33830
 .38456932E-08  .66786854E-16

  veq 7.2925633     vhl 10.301086     vhp 14.876826     dvt 7.2925633     dvmt       .0
dvpl       0.
  minj 223.59497    mwet 212.94759      mp        .0     mdry 212.94759    mnet 212.94759
mi       0.
  dvl 7.2925633     dvb        .0     dva 13.965921
```

adate 3605.1043 tend 10.000000 rcb 426579.93 poi 30.000000
pta -.56163633E-03 ptend -.56163633E-03 prcb .0 ppoi 0.

Heliocentric State of Ephemeris Bodies and Body Centered Equatorial Parameters

Departure Parameters: Earth (2,0) Nov 14, 2003 14:30:12
2452958.104

Heliocentric Ecliptic of 2000

x .61244933 y .77701072 z -.75507728E-05 xd -.13791928E-01 yd .10585731E-01 zd -.84823088E-07
radi .98936335 theta 51.754390 phi -.43727859E-03 radot -.22401297E-03 vthe .17384614E-01 vphi -.86532744E-07
sma 1.0000002 ecc .16707428E-01 inc .52206064E-03 lan 174.86672 apf 288.08696 tru 308.80070

Body Centered Ecliptic of 2000

vhxo -7.8640565 vhyo 6.4476751 vhzo 1.6420941 vhro .19564507 vhto 10.167478 vhpo 1.6420956

Body Centered Planet Equator and Equinox of Date

c3 106.11237 dla 23.262594 rla 146.26503 vhxo -7.8701191 vhyo 5.2556596 vhzo 4.0683706
poi 23.262594 pon 56.265031 thec 7.2925633 phic .0 lani .0 apfi 0.
phao 91.088265

Arrival Parameters: Pluto (2,0) Nov 14, 2013 2:30:12
2456610.604

Heliocentric Ecliptic of 2000

x 6.2413082 y -31.774693 z 1.6080028 xd .31481899E-02 yd -.23129918E-04 zd -.90928555E-03
radi 32.421763 theta 281.11278 phi 2.8428308 radot .58360910E-03 vthe .30847023E-02 vphi -.93938653E-03
sma 39.373641 ecc .24880330 inc 17.167128 lan 110.36286 apf 114.35845 tru 55.968120

Body Centered Ecliptic of 2000

vhxi 1.2764511 vhyi -14.643679 vhzi 2.2920062 vhri 14.710818 vhti -1.5699180 vhpi 1.5643277

Body Centered Planet Equator and Equinox of Date

vhp 14.876826 dap 8.8626047 rap 274.98173 vhxi 1.2764511 vhyi -14.643679 vhzi 2.2920062
phai 8.5674329

Gravity Assist Parameters: Jupiter (2,0) Unpowered Jun 2, 2005 8:07:09
2453523.838

Heliocentric Ecliptic of 2000

x -5.1795559 y -1.7054025 z .12285535 xd .22712858E-02 yd -.68167255E-02 zd -.23179111E-04
radi 5.4544743 theta 198.22447 phi 1.2906264 radot -.26004151E-04 vthe .71851121E-02 vphi -.22599132E-04
sma 5.2028030 ecc .48507139E-01 inc 1.3031449 lan 100.27496 apf 273.87601 tru 184.07147

Body Centered Ecliptic of 2000

vhxi -11.279934 vhyi 1.6919657 vhzi .20302850E-01 vhxo -2.0504614 vhyo -11.153877 vhzo 1.2193043
vhri 10.182843 vhti -5.1347868 vhpi -.20910627 vhro 5.4619611 vhto 9.9531202 vhpo 1.0965586

Body Centered Planet Equator and Equinox of Date
Reference plane for B plane angle is Planet Orbit Plane

vhxi 9.4301527 vhyi 6.4153476 vhzi .12488610 vhxo -6.0496837 vhyo 9.5279896 vhzo 1.6488791
vhi 11.406141 vho 11.406141 bend 88.113944 dvb .0 beta1 5.7164254 beta2 181.32530
rca 426579.93 thec 348.63074 phic -5.5130154 inci 8.3195196 lani 29.933261 apfi 318.39672
alfa1 .15550446 alfa2 .0 bt 1000583.7 br 107577.25 phai 26.779010 phao 118.61107

Heliocentric Trajectory

Departure: Earth time: .000 days Nov 14, 2003 14:30:12 2452958.104
x0 .61244933 y0 .77701072 z0 -.75507728E-05 xd0 -.18333800E-01 yd0 .14309575E-01 zd0 .94830385E-03
radi .98936335 theta 51.754390 phi -.43727859E-03 radot -.11101849E-03 vthe .23256824E-01 vphi .94830301E-03
sma 5.2470747 ecc .81144956 inc 2.3349568 lan 51.765114 apf .59932154 tru 359.38995
elx0 -.52129163 ely0 .42740271 elz0 .10885093 eldx0 -.49513398E-02 eldy0 -.68450555E-02 eldz0 .28228820E-02
x -5.1795559 y -1.7054025 z .12285535 xd -.42434209E-02 yd -.58395335E-02 zd -.11453233E-04
radi 5.4544743 theta 198.22447 phi 1.2906264 radot .58550800E-02 vthe .42195245E-02 vphi -.14336811E-03
sma 5.2470747 ecc .81144956 inc 2.3349568 lan 51.765114 apf .59932154 tru 145.83811
elx -.15378382 ely .23067241E-01 elz .27679683E-03 eldx .14206925E-03 eldy .23298263E-03 eldz -.62643621E-03

Gravity Assist: Jupiter time: 565.734 days Jun 2, 2005 8:07:09 2453523.838
x0 -5.1795559 y0 -1.7054025 z0 .12285535 xd0 .10870453E-02 yd0 -.13258628E-01 zd0 .68102807E-03
radi 5.4544743 theta 198.22447 phi 1.2906264 radot .31285423E-02 vthe .12933520E-01 vphi .61071643E-03
sma -4.2926805 ecc 2.2195914 inc 2.9955433 lan 172.72333 apf 5.8737399 tru 19.657855
elx0 -.27954756E-01 ely0 -.15206524 elz0 .16623261E-01 eldx0 .11101389E-03 eldy0 .88738176E-04 eldz0 -.29911308E-05
x 6.2413082 y -31.774693 z 1.6080028 xd .38854021E-02 yd -.84805620E-02 zd .41445880E-03
radi 32.421763 theta 281.11278 phi 2.8428308 radot .90798177E-02 vthe .21779987E-02 vphi -.35911690E-04
sma -4.2926805 ecc 2.2195914 inc 2.9955433 lan 172.72333 apf 5.8737399 tru 102.49227
elx .0 ely .0 elz .0 eldx .97586845E-06 eldy .37593819E-04 eldz -.53970949E-05
Arrival: Pluto time: 3652.500 days Nov 14, 2013 2:30:12 2456610.604

hca 146.45 82.83
rn .00 .00

Solution time= 0.24 sec   Cumulative run time= 1.07 sec
Step= .100E-01

e59_04.out

```
MIDAS     ver 16.07     6/6/96

      Jet Propulsion Laboratory
   California Institute of Technology
          SUN FORTRAN 1.4

 head='2000 V-M-V to Outer Solar System'
 shota='Earth',
  bulsi='Pluto',
 body='Jupiter'
 jdl=2005,1,1.0,ndb=2,ALTB=357000,poi=30,nda=1,
 varyi='jdate','tpb','tm1',
 tpb=500.,
 tend=10.
 choice=0
 jdate=0, $
 $end

  Input File is: e59_04.inp
 Output File is: e59_04.out

 Start Time:    Nov 26, 1996    09:24:47

2000 V-M-V to Outer Solar System                Nov 26, 1996    09:24:47

                         Heliocentric Ephemeris in AU and Days
                          Earth Ecliptic and Equinox of 2000
                                 Minimum Delta-V

                     Trajectory 1.02    Search converged in 10 iterations

                 Minimum Delta-V= 7.7191002 km/s        Gradient Norm= .11455092E-11

   jdate          tpb
 -16.109188      482.62147
 -.45820368E-11  .21250363E-16

  veq 7.7191002      vhl 10.916377      vhp 13.700356      dvt 7.7191002      dvmt        .0
dvpl        0.
  dvl 7.7191002      dvb        .0      dva 12.791996

 adate 3636.3908      tend 10.000000      rcb 996174.73      poi 30.000000
  pta -.53267415E-03  ptend -.53267415E-03  prcb       .0      ppoi       0.

            Heliocentric State of Ephemeris Bodies and Body Centered Equatorial Parameters

  Departure Parameters: Earth (2,0)                              Dec 15, 2004    21:22:46
2453355.391
                         Heliocentric Ecliptic of 2000
```

x .99533658E-01 y .97910634 z -.11213629E-04 xd -.17396401E-01 yd .16755529E-02 zd -.12715642E-08
radi .98415252 theta 84.195386 phi -.65283950E-03 radot -.92447992E-04 vthe .17476661E-01 vphi -.23249349E-08
sma 1.0000002 ecc .16707007E-01 inc .65288399E-03 lan 174.86430 apf 288.09263 tru 341.23846

Body Centered Ecliptic of 2000

vhxo -10.788820 vhyo 1.1558253 vhzo 1.1969605 vhro .58742696E-01 vhto 10.850397 vhpo 1.1969612

Body Centered Planet Equator and Equinox of Date

c3 119.16729 dla 8.1775225 rla 176.96340 vhxo -10.790209 vhyo .57240276 vhzo 1.5527523
poi 8.1775225 pon 86.963400 thec 7.7191002 phic .0 lani .0 apfi 0.
phao 90.308319

Arrival Parameters: Pluto (2,0) Dec 16, 2014 9:22:46 2457007.891

Heliocentric Ecliptic of 2000

x 7.4875149 y -31.762265 z 1.2457432 xd .31247698E-02 yd .85293617E-04 zd -.91415760E-03
radi 32.656642 theta 283.26452 phi 2.1861755 radot .59861748E-03 vthe .30609750E-02 vphi -.93767537E-03
sma 39.373641 ecc .24880330 inc 17.167128 lan 110.36286 apf 114.35845 tru 58.215829

Body Centered Ecliptic of 2000

vhxi .88828211 vhyi -13.503898 vhzi 2.1343505 vhri 13.419146 vhti -2.2338451 vhpi 1.6236361

Body Centered Planet Equator and Equinox of Date

vhp 13.700356 dap 8.9624984 rap 273.76348 vhxi .88828211 vhyi -13.503898 vhzi 2.1343505
phai 11.628729

Gravity Assist Parameters: Jupiter (2,0) Unpowered Apr 28, 2006 14:54:54 2453854.121

Heliocentric Ecliptic of 2000

x -3.9447745 y -3.7157727 z .10337066 xd .50858521E-02 yd -.51417879E-02 zd -.92977156E-04
radi 5.4202304 theta 223.28773 phi 1.0927694 radot -.17830150E-03 vthe .72299862E-02 vphi -.89593015E-04
sma 5.2028030 ecc .48508772E-01 inc 1.3031249 lan 100.27328 apf 273.87649 tru 209.13119

Body Centered Ecliptic of 2000

vhxi -12.902623 vhyi -5.2124342 vhzi .23825236 vhxo -5.0551218 vhyo -12.928253 vhzo 1.0049536
vhri 12.968228 vhti -5.0526114 vhpi -.90698408E-02 vhro 12.561025 vhto 5.9446215 vhpo .76553813

Body Centered Planet Equator and Equinox of Date
Reference plane for B plane angle is Planet Orbit Plane

vhxi 5.9374537 vhyi 12.572714 vhzi .61433791 vhxo -5.0512309 vhyo 12.876719 vhzo 1.5424306

vhi 13.917755 vho 13.917755 bend 46.697670 dvb .0 beta1 5.0314908
beta2 183.67028
rca 996174.73 thec 358.41530 phic -4.8258457 inci 6.8437727 lani 43.119890
apfi 315.09057
alfa1 .18489200 alfa2 .0 bt 1509816.4 br 125899.31 phai 21.286647
phao 154.49105

Heliocentric Trajectory

Departure: Earth time: .000 days Dec 15, 2004 21:22:46 2453355.391
x0 .99533658E-01 y0 .97910634 z0 -.11213629E-04 xd0 -.23627466E-01 yd0
.23430979E-02 zd0 .69130127E-03
radi .98415252 theta 84.195386 phi -.65283950E-03 radot -.58521246E-04 vthe
.23743290E-01 vphi .69130061E-03
sma 7.9697898 ecc .87651542 inc 1.6677311 lan 84.217808 apf .27977418
tru 359.69779
elx0 -.69696116 ely0 .74666680E-01 elz0 .77324025E-01 eldx0 -.69003804E-03 eldy0 -
.85232773E-02 eldz0 .15656737E-02
x -3.9447745 y -3.7157727 z .10337066 xd -.23660362E-02 yd -.81522205E-
02 zd .44625096E-04
radi 5.4202304 theta 223.28773 phi 1.0927694 radot .73114771E-02 vthe .43118589E-
02 vphi -.94831287E-04
sma 7.9697898 ecc .87651542 inc 1.6677311 lan 84.217808 apf .27977418
tru 138.77813
elx -.17140635 ely -.69245172E-01 elz .31650904E-02 eldx .50609041E-04 eldy
.37090934E-03 eldz -.38186768E-03

Gravity Assist: Jupiter time: 498.731 days Apr 28, 2006 14:54:54 2453854.121
x0 -3.9447745 y0 -3.7157727 z0 .10337066 xd0 .21662750E-02 yd0 -
.12608479E-01 zd0 .48743209E-03
radi 5.4202304 theta 223.28773 phi 1.0927694 radot .70762972E-02 vthe .10663292E-
01 vphi .35254225E-03
sma -5.4081458 ecc 1.7577506 inc 2.1861762 lan 193.30941 apf 338.11432
tru 51.882059
elx0 -.67155337E-01 ely0 -.17174684 elz0 .13350419E-01 eldx0 .96979938E-04 eldy0
.16884637E-03 eldz0 -.41561798E-05
x 7.4875149 y -31.762265 z 1.2457432 xd .36377957E-02 yd -.77138599E-
02 zd .31853297E-03
radi 32.656642 theta 283.26452 phi 2.1861755 radot .83488230E-02 vthe .17708214E-
02 vphi .52934112E-07
sma -5.4081458 ecc 1.7577506 inc 2.1861762 lan 193.30941 apf 338.11432
tru 111.84082
elx .0 ely .0 elz .0 eldx .19304321E-04 eldy .40056430E-04 eldz -
.41717189E-05
Arrival: Pluto time: 3652.500 days Dec 16, 2014 9:22:46 2457007.891

hca 139.08 59.96
rn .00 .00

Solution time= 0.21 sec Cumulative run time= 0.64 sec
Step= .100E-01

e59 04a.out

MIDAS ver 16.07 6/6/96

Jet Propulsion Laboratory

California Institute of Technology
SUN FORTRAN 1.4

```
head='2000 V-M-V to Outer Solar System'
shota='Earth',
 bulsi='Pluto',
 body='Jupiter'
 jdl=2005,1,1.0,ndb=2,ALTB=71400,poi=30,nda=1,
 varyi='jdate','tpb','tm1',
 tpb=500.,
 nlv=28
 lvcont=.1
 kadp=.05
 tend=8.
 varyn='tend',minyn=8.,.5,13.
 vlist='+jdate','tend','c3','dvpl','rcb','mnet'
 choice=0
 jdate=0, $
 $end
```

Input File is: e59_04a.inp
Output File is: e59_04a.out

Start Time: Nov 26, 1996 09:23:22

2000 V-M-V to Outer Solar System Nov 26, 1996 09:23:22

Heliocentric Ephemeris in AU and Days
Earth Ecliptic and Equinox of 2000
Minimum Delta-V
launch vehicle: Delta II (7925)/Star-30C

Trajectory 5.01 Search converged in 9 iterations

Minimum Delta-V= 7.7192212 km/s Gradient Norm= .26425675E-08

```
 nv=  4  0   tend=   10.000

 jdate         tpb
-16.109186     482.62147
 .10570270E-07 -.43368087E-18

 veq 7.7192212      vhl 10.916377     vhp 13.700356      dvt 7.7192212      dvmt        .0
dvpl       0.
 minj 184.37989     mwet 175.59990      mp        .0     mdry 175.59990     mnet 175.59990
mi      0.
 dvl 7.7192212      dvb        .0      dva 12.791996

adate 3636.3908     tend 10.000000     rcb 996174.73      poi 30.000000
 pta -.53155568E-03 ptend -.53155568E-03 prcb       .0    ppoi       0.
```

Heliocentric State of Ephemeris Bodies and Body Centered Equatorial Parameters

Departure Parameters: Earth (2,0) Dec 15, 2004 21:22:46
2453355.391

Heliocentric Ecliptic of 2000

x .99533621E-01 y .97910634 z -.11213629E-04 xd -.17396401E-01 yd .16755522E-02 zd -.12715567E-08
radi .98415252 theta 84.195388 phi -.65283950E-03 radot -.92447982E-04 vthe .17476661E-01 vphi -.23249274E-08
sma 1.0000002 ecc .16707007E-01 inc .65288399E-03 lan 174.86430 apf 288.09263 tru 341.23846

Body Centered Ecliptic of 2000

vhxo -10.788820 vhyo 1.1558256 vhzo 1.1969605 vhro .58743391E-01 vhto 10.850397 vhpo 1.1969611

Body Centered Planet Equator and Equinox of Date

c3 119.16729 dla 8.1775228 rla 176.96340 vhxo -10.790209 vhyo .57240304 vhzo 1.5527524
poi 8.1775228 pon 86.963398 thec 7.7192212 phic .0 lani .0 apfi 0.
phao 90.308323

Arrival Parameters: Pluto (2,0) Dec 16, 2014 9:22:46 2457007.891

Heliocentric Ecliptic of 2000

x 7.4875149 y -31.762265 z 1.2457432 xd .31247698E-02 yd .85293617E-04 zd -.91415760E-03
radi 32.656642 theta 283.26452 phi 2.1861755 radot .59861748E-03 vthe .30609750E-02 vphi -.93767537E-03
sma 39.373641 ecc .24880330 inc 17.167128 lan 110.36286 apf 114.35845 tru 58.215829

Body Centered Ecliptic of 2000

vhxi .88828211 vhyi -13.503898 vhzi 2.1343505 vhri 13.419146 vhti -2.2338451 vhpi 1.6236361

Body Centered Planet Equator and Equinox of Date

vhp 13.700356 dap 8.9624984 rap 273.76348 vhxi .88828211 vhyi -13.503898 vhzi 2.1343505
phai 11.628729

Gravity Assist Parameters: Jupiter (2,0) Unpowered Apr 28, 2006 14:54:54 2453854.121

Heliocentric Ecliptic of 2000

x -3.9447744 y -3.7157727 z .10337066 xd .50858521E-02 yd -.51417879E-02 zd -.92977156E-04
radi 5.4202304 theta 223.28773 phi 1.0927694 radot -.17830150E-03 vthe .72299862E-02 vphi -.89593015E-04
sma 5.2028030 ecc .48508772E-01 inc 1.3031249 lan 100.27328 apf 273.87649 tru 209.13119

Body Centered Ecliptic of 2000

vhxi -12.902623 vhyi -5.2124342 vhzi .23825237 vhxo -5.0551218 vhyo -12.928253 vhzo 1.0049536
vhri 12.968228 vhti -5.0526114 vhpi -.90698282E-02 vhro 12.561025 vhto 5.9446215 vhpo .76553813

Body Centered Planet Equator and Equinox of Date
Reference plane for B plane angle is Planet Orbit Plane

vhxi 5.9374543 vhyi 12.572714 vhzi .61433777 vhxo -5.0512302 vhyo 12.876719 vhzo 1.5424306

vhi 13.917755 vho 13.917755 bend 46.697670 dvb .0 beta1 5.0314908
beta2 183.67028
rca 996174.73 thec 358.41530 phic -4.8258469 inci 6.8437735 lani 43.119894
apfi 315.09056
alfa1 .18450377 alfa2 .0 bt 1509816.4 br 125899.31 phai 21.286647
phao 154.49105

Heliocentric Trajectory

Departure: Earth time: .000 days Dec 15, 2004 21:22:46 2453355.391
x0 .99533621E-01 y0 .97910634 z0 -.11213629E-04 xd0 -.23627466E-01 yd0
.23430974E-02 zd0 .69130124E-03
radi .98415252 theta 84.195388 phi -.65283950E-03 radot -.58520834E-04 vthe
.23743290E-01 vphi .69130058E-03
sma 7.9697898 ecc .87651542 inc 1.6677311 lan 84.217810 apf .27977205
tru 359.69780
elx0 -.69549777 ely0 .74509922E-01 elz0 .77161666E-01 eldx0 -.68858849E-03 eldy0 -
.85053815E-02 eldz0 .15623861E-02
x -3.9447744 y -3.7157727 z .10337066 xd -.23660362E-02 yd -.81522205E-
02 zd .44625103E-04
radi 5.4202304 theta 223.28773 phi 1.0927694 radot .73114771E-02 vthe .43118589E-
02 vphi -.94831280E-04
sma 7.9697898 ecc .87651542 inc 1.6677311 lan 84.217810 apf .27977205
tru 138.77813
elx -.17104645 ely -.69099776E-01 elz .31584448E-02 eldx .50502803E-04 eldy
.37013046E-03 eldz -.38106585E-03

Gravity Assist: Jupiter time: 498.731 days Apr 28, 2006 14:54:54 2453854.121
x0 -3.9447744 y0 -3.7157727 z0 .10337066 xd0 .21662750E-02 yd0 -
.12608479E-01 zd0 .48743209E-03
radi 5.4202304 theta 223.28773 phi 1.0927694 radot .70762972E-02 vthe .10663292E-
01 vphi .35254225E-03
sma -5.4081458 ecc 1.7577506 inc 2.1861762 lan 193.30941 apf 338.11432
tru 51.882059
elx0 -.67014329E-01 ely0 -.17138622 elz0 .13322387E-01 eldx0 .96776307E-04 eldy0
.16849184E-03 eldz0 -.41474529E-05
x 7.4875149 y -31.762265 z 1.2457432 xd .36377957E-02 yd -.77138599E-
02 zd .31853297E-03
radi 32.656642 theta 283.26452 phi 2.1861755 radot .83488230E-02 vthe .17708214E-
02 vphi .52934103E-07
sma -5.4081458 ecc 1.7577506 inc 2.1861762 lan 193.30941 apf 338.11432
tru 111.84082
elx .0 ely .0 elz .0 eldx .19263787E-04 eldy .39972322E-04 eldz -
.41629594E-05
Arrival: Pluto time: 3652.500 days Dec 16, 2014 9:22:46 2457007.891

hca 139.08 59.96
rn .00 .00

Solution time= 0.21 sec Cumulative run time= 1.60 sec
Step= .100E-01

e59 04b.out

MIDAS ver 16.07 6/6/96

Jet Propulsion Laboratory

California Institute of Technology
SUN FORTRAN 1.4

```
head='2000 V-M-V to Outer Solar System'
shota='Earth',
 bulsi='Pluto',
 body='Jupiter'
 jdl=2005,1,1.0,ndb=2,ALTB=35700,poi=30,nda=1,
 varyi='jdate','tpb','tm1',
 tpb=500.,
 tend=10.
 varyn='tend',minyn=10.,-.5,7.
 vlist='+jdate','tend','c3','dvmt','rcb'
choice=0
 jdate=0, $
 $end
```

Input File is: e59_04b.inp
Output File is: e59_04b.out

Start Time: Nov 26, 1996 09:24:00

2000 V-M-V to Outer Solar System Nov 26, 1996 09:24:00

Heliocentric Ephemeris in AU and Days
Earth Ecliptic and Equinox of 2000
Minimum Delta-V

Trajectory 1.02 Search converged in 10 iterations

Minimum Delta-V= 7.7191002 km/s Gradient Norm= .11454559E-11

nv= 0 0 tend= 10.000

jdate	tpb
-16.109188	482.62147
-.45818235E-11	-.86736174E-17

veq 7.7191002 vhl 10.916377 vhp 13.700356 dvt 7.7191002 dvmt .0
dvpl 0.
dvl 7.7191002 dvb .0 dva 12.791996

adate 3636.3908 tend 10.000000 rcb 996174.73 poi 30.000000
pta -.53267415E-03 ptend -.53267415E-03 prcb .0 ppoi 0.

Heliocentric State of Ephemeris Bodies and Body Centered Equatorial Parameters

Departure Parameters: Earth (2,0) Dec 15, 2004 21:22:46
2453355.391

Heliocentric Ecliptic of 2000

x .99533658E-01 y .97910634 z -.11213629E-04 xd -.17396401E-01 yd
.16755529E-02 zd -.12715642E-08
radi .98415252 theta 84.195386 phi -.65283950E-03 radot -.92447992E-04 vthe
.17476661E-01 vphi -.23249349E-08
sma 1.0000002 ecc .16707007E-01 inc .65288399E-03 lan 174.86430 apf 288.09263
tru 341.23846

Body Centered Ecliptic of 2000

vhxo -10.788820 vhyo 1.1558253 vhzo 1.1969605 vhro .58742696E-01 vhto 10.850397 vhpo 1.1969612

Body Centered Planet Equator and Equinox of Date

c3 119.16729 dla 8.1775225 rla 176.96340 vhxo -10.790209 vhyo .57240276 vhzo 1.5527523
poi 8.1775225 pon 86.963400 thec 7.7191002 phic .0 lani .0 apfi 0.
phao 90.308319

Arrival Parameters: Pluto (2,0) Dec 16, 2014 9:22:46 2457007.891

Heliocentric Ecliptic of 2000

x 7.4875149 y -31.762265 z 1.2457432 xd .31247698E-02 yd .85293617E-04 zd -.91415760E-03
radi 32.656642 theta 283.26452 phi 2.1861755 radot .59861748E-03 vthe .30609750E-02 vphi -.93767537E-03
sma 39.373641 ecc .24880330 inc 17.167128 lan 110.36286 apf 114.35845 tru 58.215829

Body Centered Ecliptic of 2000

vhxi .88828211 vhyi -13.503898 vhzi 2.1343505 vhri 13.419146 vhti -2.2338451 vhpi 1.6236361

Body Centered Planet Equator and Equinox of Date

vhp 13.700356 dap 8.9624984 rap 273.76348 vhxi .88828211 vhyi -13.503898 vhzi 2.1343505
phai 11.628729

Gravity Assist Parameters: Jupiter (2,0) Unpowered Apr 28, 2006 14:54:54 2453854.121

Heliocentric Ecliptic of 2000

x -3.9447745 y -3.7157727 z .10337066 xd .50858521E-02 yd -.51417879E-02 zd -.92977156E-04
radi 5.4202304 theta 223.28773 phi 1.0927694 radot -.17830150E-03 vthe .72299862E-02 vphi -.89593015E-04
sma 5.2028030 ecc .48508772E-01 inc 1.3031249 lan 100.27328 apf 273.87649 tru 209.13119

Body Centered Ecliptic of 2000

vhxi -12.902623 vhyi -5.2124342 vhzi .23825236 vhxo -5.0551218 vhyo -12.928253 vhzo 1.0049536
vhri 12.968228 vhti -5.0526114 vhpi -.90698408E-02 vhro 12.561025 vhto 5.9446215 vhpo .76553813

Body Centered Planet Equator and Equinox of Date
Reference plane for B plane angle is Planet Orbit Plane

vhxi 5.9374537 vhyi 12.572714 vhzi .61433791 vhxo -5.0512309 vhyo 12.876719 vhzo 1.5424306
vhi 13.917755 vho 13.917755 bend 46.697670 dvb .0 beta1 5.0314908 beta2 183.67028
rca 996174.73 thec 358.41530 phic -4.8258457 inci 6.8437727 lani 43.119890 apfi 315.09057
alfa1 .18489200 alfa2 .0 bt 1509816.4 br 125899.31 phai 21.286647
phao 154.49105

Heliocentric Trajectory

Departure: Earth time: .000 days Dec 15, 2004 21:22:46 2453355.391
x0 .99533658E-01 y0 .97910634 z0 -.11213629E-04 xd0 -.23627466E-01 yd0 .23430979E-02 zd0 .69130127E-03
radi .98415252 theta 84.195386 phi -.65283950E-03 radot -.58521246E-04 vthe .23743290E-01 vphi .69130061E-03
sma 7.9697898 ecc .87651542 inc 1.6677311 lan 84.217808 apf .27977418 tru 359.69779
elx0 -.69696116 ely0 .74666680E-01 elz0 .77324025E-01 eldx0 -.69003804E-03 eldy0 -.85232773E-02 eldz0 .15656737E-02
x -3.9447745 y -3.7157727 z .10337066 xd -.23660362E-02 yd -.81522205E-02 zd .44625096E-04
radi 5.4202304 theta 223.28773 phi 1.0927694 radot .73114771E-02 vthe .43118589E-02 vphi -.94831287E-04
sma 7.9697898 ecc .87651542 inc 1.6677311 lan 84.217808 apf .27977418 tru 138.77813
elx -.17140635 ely -.69245172E-01 elz .31650904E-02 eldx .50609041E-04 eldy .37090934E-03 eldz -.38186768E-03

Gravity Assist: Jupiter time: 498.731 days Apr 28, 2006 14:54:54 2453854.121
x0 -3.9447745 y0 -3.7157727 z0 .10337066 xd0 .21662750E-02 yd0 -.12608479E-01 zd0 .48743209E-03
radi 5.4202304 theta 223.28773 phi 1.0927694 radot .70762972E-02 vthe .10663292E-01 vphi .35254225E-03
sma -5.4081458 ecc 1.7577506 inc 2.1861762 lan 193.30941 apf 338.11432 tru 51.882059
elx0 -.67155337E-01 ely0 -.17174684 elz0 .13350419E-01 eldx0 .96979938E-04 eldy0 .16884637E-03 eldz0 -.41561798E-05
x 7.4875149 y -31.762265 z 1.2457432 xd .36377957E-02 yd -.77138599E-02 zd .31853297E-03
radi 32.656642 theta 283.26452 phi 2.1861755 radot .83488230E-02 vthe .17708214E-02 vphi .52934112E-07
sma -5.4081458 ecc 1.7577506 inc 2.1861762 lan 193.30941 apf 338.11432 tru 111.84082
elx .0 ely .0 elz .0 eldx .19304321E-04 eldy .40056430E-04 eldz -.41717189E-05
Arrival: Pluto time: 3652.500 days Dec 16, 2014 9:22:46 2457007.891

hca 139.08 59.96
rn .00 .00

Solution time= 0.22 sec Cumulative run time= 0.64 sec
Step= .100E-01

APPENDIX D: CATO INTERACTIVE COMMANDS

Command	**Description**
file *filename*	Obtain input from file *filename* (non-interactive mode)
interactive	Obtain input from terminal (interactive mode)
expert	Do not show this menu before each command prompt
novice	Show this menu before each command prompt
menu	Show this menu once (doe not change expert/novice mode)
comment	Entire line is a comment which will be ignored
echo	Entire line is a comment which will be echoed
odecontrol	Specify gravitating bodies and integration controls
spacecraft	Specify spacecraft model parameters
+CP *controlpoint*	Add a control point labeled *controlpoint*
:CP *controlpoint*	Modify the control point with label *controlpoint*
-CP *controlpoint*	Delete the control point with label *controlpoint*
+BP *breakpoint*	Add a breakpoint labeled *breakpoint*
:BP *breakpoint*	Modify the breakpoint with label *breakpoint*
-BP *breakpoint*	Delete the breakpoint with label *breakpoint*
engines	Define engine(s) for use with finite burn maneuvers
+MV *breakpoint, type*	Add a maneuver at breakpoint labeled *breakpoint* of type named *type*
:MV *breakpoint*	Modify the maneuver at the breakpoint labeled *breakpoint*

-MV *breakpoint*	Delete the maneuver at the breakpoint labeled *breakpoint*
costmodel *breakpoint*	Specify cost model for the breakpoint labeled *breakpoint*
istate	Specify initial state for the trajectory
fstate	Specify final state for the trajectory
show *a, b, c, d*	Show breakpoint/control point variables specified by *a* (label of a trajectory feature point), *b* (label of a second feature point), *c* (feature point type qualifier: either BP or CP), *d* (other qualifier: ACTIVE, ONBOARD, or STATES)
autoshow *a, b, c, d*	Set/unset autoshow mode (arguments same as *show* command)
optimize	Begin optimization
continue *number, trajectory, fraction*	Take *number* of steps; *trajectory* file, step *fraction*
checkpoint *name*	Save current status in checkpoint/restart file *name*
restart *name*	Restart from checkpoint/restart file *name*
new	Discard current optimization problem; begin a new one
quit	Exit CATO immediately

APPENDIX E: MASL BODY NUMBER SYSTEM

Body Name	Body Number
Sun	0[§]
Solar System Barycenter	99
Mercury	1
Mercury Barycenter	9901
Venus	2
Venus Barycenter	9902
Earth	3
Moon	103
Earth-Moon Barycenter	9903
Mars	4
Phobos	104
Deimos	204
Mars Barycenter	9904
Jupiter	5
Io	105
Europa	205
Ganymede	305
Callisto	405
Amalthea	505
Himalia	605
Elara	705
Pasiphae	805
Sinope	905
Lysithea	1005
Carme	1105
Ananke	1205

[§] In CATO, the *&ode GravBods* command must be terminated with a **0**, therefore, the value of **10** is substituted for the sun (Bright and Byrnes, 1996, p. 4-8).

Leda	1305
Thebe	1405
Adrastea	1505
Metis	1605
Jupiter Barycenter	9905
Saturn	6
Mimas	106
Enceladus	206
Tethys	306
Dione	406
Rhea	506
Titan	606
Hyperion	706
Iapetus	806
Phoebe	906
Janus	1006
Epimetheus	1106
Saturn Barycenter	9906
Uranus	7
Ariel	107
Umbriel	207
Titania	307
Oberon	407
Miranda	507
Uranus Barycenter	9907
Neptune	8
Triton	108
Nereid	208
Neptune Barycenter	9908
Pluto	9
Charon	109

Pluto Barycenter 9909

Spacecraft: They are assigned negative numbers at the beginning of a project. The numbers *-650* to *-699* are allocated to MASL for this purpose.

Asteroids and Comets: They are assigned a number, allowing MASL software to gain access to the ephemeris for that particular body. Comets numbers are between *2000001* and *2999999*; asteroids are numbered from *1000001* to *1999999* if previously numbered and *3000001* to *3999999* if unnumbered.

APPENDIX F: CATO INPUT FILES

14 NOVEMBER 2003 NOMINAL LAUNCH

```
COM CATO input file using MIDAS e59_03a.out for 14 Nov 2003 launch
COM Michelle D. Reyes
ODECONTROL
 &ode GravBods(1:7) = 3, 103, 5, 9, 109, 10, 0
/
SPACECRAFT
 &spa area =        1.77,
    massDef =       639,
    GR =            1.5
/
COM Earth Departure
+BP Earth
 &bp jdMin =        2452957.3,
   jd =             2452957.4,
   jdMax =          2452957.5,
   djd =            0.001,
   fixedJd =        T,
   noMvr =          .TRUE,
   cBody =          3
/
COM Departure Hyperbola
+CP DepHyp
 &cp bodyU =        3,
   iOrbU =          110223,
   Umin =  2452958.100, 6563.0, -15.0, -600.0, 101.00000, -90.000000, 140.00000,
   U =     2452958.104, 6563.1,   0.0,    0.0, 106.11237,  23.262594, 146.26503,
   Umax =  2452958.200, 6563.2,  15.0,  600.0, 111.00000,  90.000000, 151.00000,
   DuFinite =     1e-4,   0.10,  1e-4,    2.0,      1e-5,       1e-4,      1e-4,
   FixedU =          T,      T,     T,      T,         F,          F,         F
/
COM Going to Jupiter
+BP JupTraj
 &bp jdMin =        2453249.4,
   jd =             2453249.5,
   jdMax =          2453249.6,
   djd =            0.001,
   fixedJd =        T,
   cbody =          0
```

```
/
COM Jupiter Gravity Assist Flyby
+CP JGA
 &cp bodyU =      5,
   iOrbU =        23,
   Umin = 2453500.80, 325000.00,  0.00000, -600, 10.0000, -90.00000, 30.0000,
   U =    2453523.84, 426579.93,  5.71643,    0, 11.4061,  0.627345, 34.2275,
   Umax = 2453550.00, 900000.00, 11.00000,  600, 12.0000,  90.00000, 36.0000,
   DuFinite =   1e-4,       1.0,     1e-4,  1.0,    1e-5,      1e-4,    1e-4,
   FixedU =        F,         F,        F,    T,       F,         F,       F
/
COM Going to Pluto
+BP PluTraj
 &bp jdMin =      2455075.4,
   jd =           2455075.5,
   jdMax =        2455075.6,
   djd =          0.001,
   fixedJd =      T,
   cbody =        0
/
COM Arrival Hyperbola
+CP ArrHyp
 &cp bodyU =      9,
   iOrbU =        30083,
   Umin =  2456606.5, -3725, -12100, -3000, 14.5000, -90.00000, 269.00000,
   U =     2456606.9, -3700, -12000,     0, 14.8768,   8.86260, 274.98176,
   Umax =  2456607.5, -3675, -11975,  3000, 15.1000,  90.00000, 280.00000,
   DuFinite =   1e-4,  1e-4,   1e-4,    10,    1e-5,      1e-4,      1e-4,
   FixedU =        T,     T,      T,     T,       F,         F,         F
/
COM Pluto Arrival and Beyond
+BP Pluto
 &bp jdMin =      2456609.0,
   jd =           2456610.0,
   jdMax =        2456615.0,
   djd =          0.001,
   fixedJd =      T,
   noMvr =        .TRUE,
   cbody =        9
/
INTERACTIVE
```

15 DECEMBER 2004 NOMINAL LAUNCH

```
COM CATO input file using MIDAS e59_04.out for 15 Dec 2004 launch
COM Michelle D. Reyes
ODECONTROL
 &ode GravBods(1:7) = 3, 103, 5, 9, 109, 10, 0
/
SPACECRAFT
 &spa area =      1.77,
    massDef =     639,
    GR =          1.5
/
COM Earth Departure
+BP Earth
 &bp jdMin =      2453355.3909996,
   jd =           2453355.3909997,
   jdMax =        2453355.3909998,
   djd =          0.001,
   fixedJd =      T,
   noMvr =        .TRUE,
   cBody =        3
/
COM Departure Hyperbola
+CP DepHyp
 &cp bodyU =      3,
   iOrbU =        110223,
   Umin =  2453355.391, 6563.0, -15.0, 600.0, 100.000, 180.0000, 180.0000,
   U =     2453355.391, 6563.1,   0.0,   0.0, 119.167,   8.1775, 176.9634,
   Umax =  2453355.400, 6563.2,  15.0, 600.0, 130.000, 180.0000, 360.0000,
   DuFinite =      1e-4,   0.10,  1e-4,   2.0,    1e-5,     1e-4,     1e-4,
   FixedU =           T,      T,     T,     T,       F,        F,        F
/
COM Going to Jupiter
+BP JupTraj
 &bp jdMin =      2453605.4,
   jd =           2453605.5,
   jdMax =        2453605.6,
   djd =          0.001,
   fixedJd =      T,
   cbody =        0
/
COM Jupiter Gravity Assist Flyby
+CP JGA
```

```
 &cp bodyU =         5,
   iOrbU =           23,
   Umin = 2453607.00,   425000.00,   0.0000,  -600,   8.000,  -90.0000,  -180.000,
   U =    2453854.12,   996174.73,   5.0315,     0,  13.918,    2.5299,    64.721,
   Umax = 2453875.00,  1500000.00,  11.0000,   600,  18.000,   90.0000,   180.000,
   DuFinite =   1e-4,         1.0,     1e-4,   1.0,    1e-5,      1e-4,      1e-4,
   FixedU =        F,           F,        F,     T,       F,         F,         F
/
COM Going to Pluto
+BP PluTraj
 &bp jdMin =         2453875.4,
   jd =              2453875.5,
   jdMax =           2453875.6,
   djd =             0.001,
   fixedJd =         T,
   cbody =           0
/
COM Arrival Hyperbola
+CP ArrHyp
 &cp bodyU =         9,
   iOrbU =           30083,
   Umin =  2457007.500,  -3725,  -12100,  -3000,   8.0000,  -90.00000,  180.0000,
   U =     2457007.891,  -3700,  -12000,      0,  13.7004,    8.96250,  273.7635,
   Umax =  2457008.000,  -3675,  -11975,   3000,  18.1000,   90.00000,  360.0000,
   DuFinite =     1e-4,   1e-4,    1e-4,   10.0,     1e-5,       1e-4,      1e-4,
   FixedU =          T,      T,       T,      T,        F,          F,         F
/
COM Pluto Arrival and Beyond
+BP Pluto
 &bp jdMin =         2457009.0,
   jd =              2457010.0,
   jdMax =           2457011.0,
   djd =             0.001,
   fixedJd =         T,
   noMvr =           .TRUE,
   cbody =           9
/
INTERACTIVE
```

APPENDIX G: CATO OUTPUT FILE

14 NOVEMBER 2003

BREAKPOINT : Earth		Fid JD: 2452958.1039997		
	Lower	Value	Upper |	{ NO MVR }
Julian Date	2452958.1039996	2452958.1039997	2452958.1039998	day

CONTROL POINT: DepHyp		Fid JD: 2452958.1040000		
PARAMETER SET: Hyperbolic Asymptote		equator and equinox of epoch	WRT:	Earth
	Lower	Value	Upper	
Julian Date	2452958.1039999	2452958.1040000	2452958.2000000	day
Periapsis radius	6563.0000000000	6563.1000000000	6563.2000000000	km
B Plane Angle	-15.00000000000	0.0000000000000	15.000000000000	deg
Time wrt Periapsis	-600.0000000000	0.0000000000000	600.00000000000	s
* C3 (energy)	101.00000000000	106.34861165096	111.00000000000	km2/s2
* Decl V-infinity	-90.00000000000	23.125476672141	90.000000000000	deg
* Rt Asc V-infinity	140.00000000000	146.57404794729	151.00000000000	deg

BREAKPOINT : JupTraj		Fid JD: 2453249.5000000		
	Lower	Value	Upper | R V	{ NO MVR }
Julian Date	2453249.4000000	2453249.5000000	2453249.6000000	day

CONTROL POINT: JGA		Fid JD: 2453523.8380000		
PARAMETER SET: Hyperbolic Asymptote		equator and equinox of epoch	WRT:	Jupiter
	Lower	Value	Upper	
* Julian Date	2453300.0000000	2453518.8914051	2453524.0000000	day
* Periapsis Altitude	320000.00000000	357697.41777897	900000.00000000	km
* B Plane Angle	0.0000000000000	8.4109121398453	11.000000000000	deg
Time wrt Periapsis	-600.0000000000	0.0000000000000	600.00000000000	s
* V-infinity	10.000000000000	11.404344396689	12.000000000000	km/s
* Decl V-infinity	-90.00000000000	0.6517761239788	90.000000000000	deg
* Rt Asc V-infinity	30.000000000000	34.731119541816	36.000000000000	deg

BREAKPOINT : PluTraj		Fid JD: 2453540.1000000		
	Lower	Value	Upper | R V	{ NO MVR }
Julian Date	2453540.0000000	2453540.1000000	2453540.2000000	day

CONTROL POINT: ArrHyp		Fid JD: 2456606.9000000		
PARAMETER SET: Hyperbolic Asymptote		Earth ecliptic and equinox of J2000	WRT:	Pluto
	Lower	Value	Upper	
Julian Date	2456606.5000000	2456606.9000000	2456607.5000000	day
B dot T	-3725.000000000	-3700.000000000	-3675.000000000	km
B dot R	-12100.00000000	-12000.00000000	-11975.00000000	km
Time wrt Periapsis	-3000.000000000	0.0000000000000	3000.0000000000	s
* V-infinity	14.500000000000	14.919291415128	15.100000000000	km/s
* Decl V-infinity	-90.00000000000	8.8438720029606	90.000000000000	deg
* Rt Asc V-infinity	269.00000000000	274.77505649502	280.00000000000	deg

BREAKPOINT : Pluto		Fid JD: 2456610.0000000		
	Lower	Value	Upper |	{ NO MVR }
Julian Date	2456609.0000000	2456610.0000000	2456615.0000000	day

15 DECEMBER 2004

BREAKPOINT : Earth — Fid JD: 2453355.3909997

	Lower	Value	Upper	| R V { NO MVR }
Julian Date	2453355.3909996	2453355.3909997	2453355.3909998	day

CONTROL POINT: DepHyp — Fid JD: 2453355.3910000

PARAMETER SET: Hyperbolic Asymptote — equator and equinox of epoch — WRT: Earth

	Lower	Value	Upper	
Julian Date	2453355.3909999	2453355.3910000	2453355.4000000	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)	100.00000000000	119.18640926400	130.00000000000	km2/s2
* Decl V-infinity	-180.00000000000	8.1012491206984	180.00000000000	deg
* Rt Asc V-infinity	-180.00000000000	177.02857151898	360.00000000000	deg

BREAKPOINT : JupTraj — Fid JD: 2453605.5000000

	Lower	Value	Upper	| R F MAGNITUDE
Julian Date	2453605.4000000	2453605.5000000	2453605.6000000	day

CONTROL POINT: JGA — Fid JD: 2453854.1210000

PARAMETER SET: Hyperbolic Asymptote — equator and equinox of epoch — WRT: Jupiter

	Lower	Value	Upper	
* Julian Date	2453607.0000000	2453851.1143771	2453875.0000000	day
* Periapsis Altitude	425000.00000000	931119.74814780	1500000.0000000	km
* B Plane Angle	0.0000000000000	6.5016802270471	11.000000000000	deg
Time wrt Periapsis	-600.00000000000	0.0000000000000	600.00000000000	s
* V-infinity	8.0000000000000	13.939182583954	18.000000000000	km/s
* Decl V-infinity	-90.000000000000	2.5288449161530	90.000000000000	deg
* Rt Asc V-infinity	-180.00000000000	64.827506321751	180.00000000000	deg

BREAKPOINT : PluTraj — Fid JD: 2453875.5000000

	Lower	Value	Upper	| R F MAGNITUDE
Julian Date	2453875.4000000	2453875.5000000	2453875.6000000	day

CONTROL POINT: ArrHyp — Fid JD: 2457007.8910000

PARAMETER SET: Hyperbolic Asymptote — Earth ecliptic and equinox of J2000 — WRT: Pluto

	Lower	Value	Upper	
Julian Date	2457007.5000000	2457007.8910000	2457008.0000000	day
B dot T	-3725.0000000000	-3700.0000000000	-3675.0000000000	km
B dot R	-12100.000000000	-12000.000000000	-11975.000000000	km
Time wrt Periapsis	-3000.0000000000	0.0000000000000	3000.0000000000	s
* V-infinity	8.0000000000000	13.731089217497	18.100000000000	km/s
* Decl V-infinity	-90.000000000000	9.0284132243914	90.000000000000	deg
* Rt Asc V-infinity	180.00000000000	273.55459293015	360.00000000000	deg

BREAKPOINT : Pluto — Fid JD: 2457010.0000000

	Lower	Value	Upper	| R V { NO MVR }
Julian Date	2457009.0000000	2457010.0000000	2457011.0000000	day

APPENDIX H: NOMINAL LAUNCH PERIOD SUMMARY

PERIOD 1: 14 NOVEMBER 2003, 10 YEAR FLIGHT TIME

Earth Departure Hyperbola Control Point					
Launch Date (Calendar)	Launch Date (Julian)	C_3 (km^2/s^2)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)	ΔV Required (km/s)
31-Oct-03	2452944.10	120.39	24.67	157.06	7.76
1-Nov-03	2452945.10	118.44	24.38	156.34	7.70
2-Nov-03	2452946.10	116.66	24.12	155.62	7.64
3-Nov-03	2452947.10	115.04	23.91	154.89	7.59
4-Nov-03	2452948.10	113.57	23.73	154.17	7.54
5-Nov-03	2452949.10	112.24	23.58	153.44	7.49
6-Nov-03	2452950.10	111.06	23.45	152.71	7.46
7-Nov-03	2452951.10	110.00	23.35	151.97	7.42
8-Nov-03	2452952.10	109.09	23.28	151.22	7.39
9-Nov-03	2452953.10	108.30	23.22	150.46	7.36
10-Nov-03	2452954.10	107.64	23.17	149.69	7.34
11-Nov-03	2452955.10	107.12	23.14	148.92	7.33
12-Nov-03	2452956.10	106.72	23.13	148.14	7.31
13-Nov-03	2452957.10	106.47	23.12	147.36	7.30
14-Nov-03	**2452958.10**	**106.35**	**23.13**	**146.57**	**7.30**
15-Nov-03	2452959.10	106.37	23.15	145.82	7.30
16-Nov-03	2452960.10	106.45	23.29	144.83	7.30
17-Nov-03	2452961.10	106.78	23.22	144.06	7.31
18-Nov-03	2452962.10	107.23	23.24	143.26	7.33
19-Nov-03	2452963.10	107.84	23.27	142.44	7.35
20-Nov-03	2452964.10	108.60	23.31	141.63	7.37
21-Nov-03	2452965.10	109.53	23.35	140.81	7.41
22-Nov-03	2452966.10	110.64	23.38	140.00	7.44
23-Nov-03	2452967.10	111.92	23.42	139.19	7.48
24-Nov-03	2452968.10	113.38	23.46	138.40	7.53
25-Nov-03	2452969.10	115.02	23.49	137.63	7.58
26-Nov-03	2452970.10	116.84	23.52	136.88	7.64
27-Nov-03	2452971.10	118.83	23.54	136.14	7.71
28-Nov-03	2452972.10	121.00	23.56	135.43	7.78

Jupiter Gravity Assist Control Point						
JGA Date (Calendar)	JGA Date (Julian)	Periapsis Altitude (km)	B Plane Angle (degrees)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
25-May-05	2453515.53	349871.99	8.45	11.38	-0.44	33.88
25-May-05	2453515.79	350778.55	9.45	11.39	-0.32	33.94
25-May-05	2453516.05	351629.04	8.45	11.39	-0.21	34.00
25-May-05	2453516.31	352426.56	8.44	11.39	-0.11	34.05
26-May-05	2453516.57	353173.59	8.44	11.39	-0.01	34.10
26-May-05	2453516.82	353872.08	7.44	11.39	0.07	34.14
26-May-05	2453517.07	354523.60	8.43	11.40	0.15	34.19
26-May-05	2453517.31	355129.39	8.43	11.40	0.23	34.23
27-May-05	2453517.55	355690.44	8.43	11.40	0.30	34.26
27-May-05	2453517.79	356207.50	8.43	11.40	0.36	34.30
27-May-05	2453518.02	356681.17	8.42	11.40	0.43	34.33
27-May-05	2453518.24	357111.89	8.42	11.40	0.48	34.36
27-May-05	2453518.47	357499.98	8.42	11.41	0.54	34.38
28-May-05	2453518.68	357795.03	8.42	11.41	0.59	34.40
28-May-05	**2453518.89**	**358098.57**	**8.41**	**11.41**	**0.64**	**34.42**
28-May-05	2453519.10	358359.44	8.41	11.14	0.69	34.44
28-May-05	2453519.30	358581.12	8.41	11.41	0.73	34.45
29-May-05	2453519.50	358757.50	8.41	11.41	0.77	34.46
29-May-05	2453519.69	358891.99	8.40	11.42	0.81	34.47
29-May-05	2453519.88	358984.13	8.40	11.42	0.85	34.47
29-May-05	2453520.06	359033.67	8.40	11.42	0.89	34.47
29-May-05	2453520.24	359040.44	8.40	11.42	0.92	34.47
29-May-05	2453520.42	359004.24	8.40	11.42	0.96	34.46
30-May-05	2453520.59	358973.03	8.39	11.42	0.99	34.46
30-May-05	2453520.75	358850.24	8.39	11.42	1.02	34.45
30-May-05	2453520.92	358684.13	8.39	11.43	1.05	34.43
30-May-05	2453521.08	358474.66	8.39	11.43	1.08	34.41
30-May-05	2453521.23	358221.90	8.39	11.43	1.11	34.39
30-May-05	2453521.39	357925.56	8.39	11.43	1.13	34.37

Pluto Arrival Hyperbola Control Point				
Arrival Date (Calendar)	Arrival Date (Julian)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.91	8.84	274.78
10-Nov-13	2456606.90	14.92	8.84	274.78
10-Nov-13	2456606.90	14.92	8.84	274.78
10-Nov-13	2456606.90	14.92	8.84	274.78
10-Nov-13	2456606.90	14.92	8.84	274.78
10-Nov-13	2456606.90	14.92	8.84	274.78
10-Nov-13	2456606.90	14.92	8.84	274.78
10-Nov-13	**2456606.90**	**14.92**	**8.84**	**274.78**
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.93	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.92	8.84	274.77
10-Nov-13	2456606.90	14.93	8.84	274.77
10-Nov-13	2456606.90	14.93	8.84	274.77
10-Nov-13	2456606.90	14.93	8.84	274.77

ΔV (km/s) = sqrt (C3 + 2*μ/r_{park}) - sqrt(μ/r_{park})

r_{park} = 6563.14 km

μ = 398600 km^3/s^2

PERIOD 1a: 14 NOVEMBER 2004, 10.5 YEAR FLIGHT TIME

Earth Departure Hyperbola Control Point					
Launch Date (Calendar)	Launch Date (Julian)	C_3 (km^2/s^2)	Declination V_∞ (deg)	Rt. Ascension (deg)	ΔV Required (km/s)
31-Oct-03	2452944.10	116.93	25.91	156.70	7.65
1-Nov-03	2452945.10	114.98	25.55	155.92	7.58
2-Nov-03	2452946.10	113.21	25.23	155.14	7.53
3-Nov-03	2452947.10	111.62	24.96	154.37	7.47
4-Nov-03	2452948.10	110.19	24.73	153.60	7.43
5-Nov-03	2452949.10	108.91	24.53	152.83	7.38
6-Nov-03	2452950.10	107.77	24.36	152.06	7.35
7-Nov-03	2452951.10	106.78	24.23	151.28	7.31
8-Nov-03	2452952.10	105.92	24.11	150.50	7.29
9-Nov-03	2452953.10	105.19	24.02	149.71	7.26
10-Nov-03	2452954.10	104.59	23.95	148.91	7.24
11-Nov-03	2452955.10	104.13	23.89	148.11	7.23
12-Nov-03	2452956.10	103.80	23.85	147.31	7.22
13-Nov-03	2452957.10	103.61	23.82	146.50	7.21
14-Nov-03	**2452958.10**	**103.55**	**23.79**	**145.71**	**7.21**
15-Nov-03	2452959.10	103.64	23.80	144.94	7.21
16-Nov-03	2452960.10	103.79	23.89	143.93	7.22
17-Nov-03	2452961.10	104.17	23.82	143.14	7.23
18-Nov-03	2452962.10	104.69	23.82	142.33	7.25
19-Nov-03	2452963.10	105.35	23.83	141.50	7.27
20-Nov-03	2452964.10	106.17	23.85	140.68	7.29
21-Nov-03	2452965.10	107.16	23.86	139.86	7.33
22-Nov-03	2452966.10	108.32	23.89	139.03	7.37
23-Nov-03	2452967.10	109.65	23.90	138.23	7.41
24-Nov-03	2452968.10	111.16	23.92	137.44	7.46
25-Nov-03	2452969.10	112.85	23.94	136.67	7.51
26-Nov-03	2452970.10	114.71	23.95	135.92	7.58
27-Nov-03	2452971.10	116.75	23.95	135.19	7.64
28-Nov-03	2452972.10	118.95	23.96	134.49	7.71
29-Nov-03	2452973.10	121.32	23.96	133.81	7.79
30-Nov-03	2452974.10	123.86	23.95	133.16	7.87
1-Dec-03	2452975.10	126.55	23.94	132.53	7.96
2-Dec-03	2452976.10	129.41	23.93	131.93	8.05
3-Dec-03	2452977.10	132.44	23.91	131.35	8.14

Jupiter Gravity Assist Control Point						
JGA Date (Calendar)	JGA Date (Julian)	Periapsis Altitude (km)	B Plane Angle (degrees)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
9-Jun-05	2453531.22	415365.74	8.06	10.89	-0.68	33.64
9-Jun-05	2453531.46	416428.53	8.05	10.89	-0.54	33.70
9-Jun-05	2453531.70	417418.42	8.05	10.90	-0.41	33.75
10-Jun-05	2453531.94	418340.44	8.04	10.90	-0.30	33.81
10-Jun-05	2453532.18	419198.50	8.03	10.90	-0.19	33.85
10-Jun-05	2453532.41	419995.72	8.03	10.90	-0.09	33.90
11-Jun-05	2453532.65	420734.58	8.02	10.90	0.00	33.94
11-Jun-05	2453532.88	421417.03	8.02	10.90	0.08	33.98
11-Jun-05	2453533.10	422044.63	8.01	10.91	0.16	34.02
11-Jun-05	2453533.32	422618.60	8.01	10.91	0.23	34.05
12-Jun-05	2453533.54	423139.92	8.00	10.91	0.30	34.08
12-Jun-05	2453533.75	423609.31	8.00	10.91	0.37	34.11
12-Jun-05	2453533.96	424027.32	7.99	10.91	0.43	34.13
12-Jun-05	2453534.17	424394.35	7.99	10.92	0.49	34.15
12-Jun-05	**2453534.37**	**424650.30**	**7.98**	**10.92**	**0.54**	**34.17**
13-Jun-05	2453534.56	424915.71	7.98	10.92	0.59	34.18
13-Jun-05	2453534.75	425134.46	7.97	10.92	0.64	34.19
13-Jun-05	2453534.94	425299.27	7.97	10.92	0.69	34.20
13-Jun-05	2453535.12	425414.14	7.97	10.92	0.73	34.20
13-Jun-05	2453535.30	425478.63	7.96	10.92	0.77	34.21
13-Jun-05	2453535.48	425492.54	7.96	10.93	0.81	34.20
14-Jun-05	2453535.65	425455.67	7.96	10.93	0.85	34.20
14-Jun-05	2453535.82	425425.70	7.95	10.93	0.89	34.19
14-Jun-05	2453535.98	425286.52	7.95	10.93	0.92	34.18
14-Jun-05	2453536.14	425096.16	7.95	10.93	0.96	34.17
14-Jun-05	2453536.30	424854.55	7.95	10.93	0.99	34.15
14-Jun-05	2453536.45	424561.68	7.94	10.93	1.02	34.13
15-Jun-05	2453536.60	424217.54	7.94	10.94	1.05	34.10
15-Jun-05	2453536.75	423822.15	7.94	10.94	1.08	34.07
15-Jun-05	2453536.89	423375.53	7.94	10.94	1.11	34.04
15-Jun-05	2453537.04	422877.70	7.93	10.94	1.13	34.00
15-Jun-05	2453537.18	422328.69	7.93	10.94	1.16	33.97
15-Jun-05	2453537.32	421728.48	7.93	10.94	1.18	33.92
15-Jun-05	2453537.46	421077.09	7.93	10.94	1.21	33.88

Pluto Arrival Hyperbola Control Point				
Arrival Date (Calendar)	Arrival Date (Julian)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
10-May-14	2456787.90	14.01	8.96	274.77
10-May-14	2456787.90	14.01	8.96	274.77
10-May-14	2456787.90	14.01	8.96	274.77
10-May-14	2456787.90	14.01	8.96	274.77
10-May-14	2456787.90	14.01	8.96	274.77
10-May-14	2456787.90	14.01	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	**2456787.90**	**14.02**	**8.96**	**274.77**
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.77
10-May-14	2456787.90	14.02	8.96	274.76
10-May-14	2456787.90	14.02	8.96	274.76
10-May-14	2456787.90	14.02	8.96	274.76
10-May-14	2456787.90	14.02	8.96	274.76
10-May-14	2456787.90	14.02	8.96	274.76
10-May-14	2456787.90	14.02	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76
10-May-14	2456787.90	14.03	8.96	274.76

PERIOD 2: 15 DECEMBER 2004, 10 YEAR FLIGHT TIME

Earth Departure Hyperbola Control Point					
Launch Date (Calendar)	Launch Date (Julian)	C_3 (km^2/s^2)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)	ΔV Required (km/s)
1-Dec-04	2453341.39	135.24	6.82	185.31	8.23
2-Dec-04	2453342.39	133.29	6.81	184.81	8.17
3-Dec-04	2453343.39	131.47	6.81	184.30	8.11
4-Dec-04	2453344.39	129.78	6.83	183.78	8.06
5-Dec-04	2453345.39	126.70	7.01	183.57	7.96
6-Dec-04	2453346.39	125.25	6.98	182.90	7.91
7-Dec-04	2453347.39	123.96	7.04	182.32	7.87
8-Dec-04	2453348.39	122.81	7.13	181.72	7.84
9-Dec-04	2453349.39	121.81	7.23	181.09	7.80
10-Dec-04	2453350.39	120.96	7.35	180.45	7.78
11-Dec-04	2453351.39	120.27	7.49	179.78	7.75
12-Dec-04	2453352.39	119.75	7.63	179.11	7.74
13-Dec-04	2453353.39	119.39	7.78	178.42	7.73
14-Dec-04	2453354.39	119.21	7.94	177.73	7.72
15-Dec-04	**2453355.39**	**119.19**	**8.10**	**177.03**	**7.72**
16-Dec-04	2453356.39	119.33	8.27	176.32	7.72
17-Dec-04	2453357.39	119.63	8.44	175.62	7.73
18-Dec-04	2453358.39	120.09	8.61	174.91	7.75
19-Dec-04	2453359.39	120.71	8.78	174.21	7.77
20-Dec-04	2453360.39	121.48	8.95	173.50	7.79
21-Dec-04	2453361.39	122.41	9.12	172.80	7.82
22-Dec-04	2453362.39	123.49	9.29	172.11	7.86
23-Dec-04	2453363.39	124.73	9.46	171.42	7.90
24-Dec-04	2453364.39	126.13	9.62	170.74	7.94
25-Dec-04	2453365.39	127.69	9.79	170.07	7.99
26-Dec-04	2453366.39	130.90	9.95	169.08	8.09
27-Dec-04	2453367.39	132.79	10.11	168.43	8.15
28-Dec-04	2453368.39	134.85	10.26	167.79	8.22
29-Dec-04	2453369.39	137.08	10.41	167.17	8.29

Jupiter Gravity Assist Control Point						
JGA Date (Calendar)	JGA Date (Julian)	Periapsis Altitude (km)	B Plane Angle (degrees)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
22-Apr-06	2453847.69	917387.47	6.93	13.89	2.04	64.29
22-Apr-06	2453847.97	918897.13	6.89	13.89	2.09	64.34
22-Apr-06	2453848.25	920325.96	6.85	13.90	2.14	64.39
23-Apr-06	2453848.52	921650.33	6.81	13.90	2.18	64.44
23-Apr-06	2453848.78	922918.57	6.78	13.91	2.22	64.49
23-Apr-06	2453849.04	924109.29	6.74	13.91	2.26	64.54
23-Apr-06	2453849.29	925216.17	6.71	13.91	2.29	64.58
24-Apr-06	2453849.54	926242.03	6.68	13.92	2.33	64.62
24-Apr-06	2453849.78	927186.18	6.65	13.92	2.36	64.66
24-Apr-06	2453850.02	928047.93	6.63	13.92	2.39	64.69
24-Apr-06	2453850.25	928826.60	6.50	13.93	2.42	64.72
24-Apr-06	2453850.47	929521.53	6.57	13.93	2.45	64.75
25-Apr-06	2453850.69	930154.59	6.55	13.93	2.48	64.78
25-Apr-06	2453850.90	930679.96	6.52	13.94	2.50	64.81
25-Apr-06	**2453851.11**	**931119.75**	**6.50**	**13.94**	**2.53**	**64.83**
25-Apr-06	2453851.32	931473.38	6.48	13.94	2.55	64.85
26-Apr-06	2453851.52	931740.30	6.46	13.95	2.58	64.86
26-Apr-06	2453851.71	931919.93	6.44	13.95	2.60	64.88
26-Apr-06	2453851.90	932011.70	6.42	13.95	2.62	64.89
26-Apr-06	2453852.09	932014.98	6.40	13.95	2.64	64.89
26-Apr-06	2453852.27	931929.14	6.38	13.96	2.66	64.90
26-Apr-06	2453852.45	931753.51	6.36	13.96	2.68	64.90
27-Apr-06	2453852.62	931487.41	6.34	13.96	2.70	64.91
27-Apr-06	2453852.79	931130.14	6.32	13.96	2.72	64.90
27-Apr-06	2453852.96	930680.97	6.31	13.97	2.73	64.89
27-Apr-06	2453853.12	930139.54	6.29	13.97	2.75	64.88
27-Apr-06	2453853.28	929504.39	6.27	13.97	2.77	64.87
27-Apr-06	2453853.44	938775.10	6.26	13.97	2.78	64.85
28-Apr-06	2453853.59	927950.86	6.24	13.98	2.80	64.84

Pluto Arrival Hyperbola Control Point				
Arrival Date (Calendar)	Arrival Date (Julian)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
16-Dec-14	2457007.89	13.72	9.03	273.58
16-Dec-14	2457007.89	13.72	9.03	273.58
16-Dec-14	2457007.89	13.72	9.03	273.57
16-Dec-14	2457007.89	13.72	9.03	273.57
16-Dec-14	2457007.89	13.72	9.03	273.57
16-Dec-14	2457007.89	13.72	9.03	273.57
16-Dec-14	2457007.89	13.73	9.03	273.57
16-Dec-14	2457007.89	13.73	9.03	273.57
16-Dec-14	2457007.89	13.73	9.03	273.56
16-Dec-14	2457007.89	13.73	9.03	273.56
16-Dec-14	2457007.89	13.73	9.03	273.56
16-Dec-14	2457007.89	13.73	9.03	273.56
16-Dec-14	2457007.89	13.73	9.03	273.56
16-Dec-14	2457007.89	13.73	9.03	273.56
16-Dec-14	**2457007.89**	**13.73**	**9.03**	**273.55**
16-Dec-14	2457007.89	13.73	9.03	273.55
16-Dec-14	2457007.89	13.73	9.03	273.55
16-Dec-14	2457007.89	13.73	9.03	273.55
16-Dec-14	2457007.89	13.73	9.03	273.55
16-Dec-14	2457007.89	13.73	9.03	273.55
16-Dec-14	2457007.89	13.73	9.03	273.55
16-Dec-14	2457007.89	13.74	9.03	273.55
16-Dec-14	2457007.89	13.74	9.03	273.54
16-Dec-14	2457007.89	13.74	9.03	273.54
16-Dec-14	2457007.89	13.74	9.03	273.54
16-Dec-14	2457007.89	13.74	9.03	273.54
16-Dec-14	2457007.89	13.74	9.03	273.54
16-Dec-14	2457007.89	13.74	9.03	273.54
16-Dec-14	2457007.89	13.74	9.03	273.54

ΔV (km/s) = sqrt (C3 + 2*μ/r_{park}) - sqrt(μ/r_{park})

r_{park} = 6563.14 km

μ = 398600 km^3/s^2

PERIOD 2a: 15 DECEMBER 2004, 10.5 YEAR FLIGHT TIME

Earth Departure Hyperbola Control Point					
Launch Date (Calendar)	Launch Date (Julian)	C_3 (km^2/s^2)	Declination V_∞ (deg)	Rt.Ascension V_∞ (deg)	ΔV Required (km/s)
1-Dec-04	2453341.39	129.96	7.29	185.12	8.06
2-Dec-04	2453342.39	128.09	7.27	184.61	8.00
3-Dec-04	2453343.39	126.34	7.26	184.08	7.95
4-Dec-04	2453344.39	124.73	7.28	183.56	7.90
5-Dec-04	2453345.39	123.21	7.46	182.98	7.85
6-Dec-04	2453346.39	121.84	7.42	182.32	7.81
7-Dec-04	2453347.39	120.63	7.47	181.72	7.77
8-Dec-04	2453348.39	119.55	7.56	181.10	7.73
9-Dec-04	2453349.39	118.62	7.66	180.46	7.70
10-Dec-04	2453350.39	117.85	7.78	179.80	7.68
11-Dec-04	2453351.39	117.23	7.91	179.12	7.66
12-Dec-04	2453352.39	116.78	8.05	178.44	7.64
13-Dec-04	2453353.39	116.50	8.19	177.74	7.63
14-Dec-04	2453354.39	116.38	8.35	177.03	7.63
15-Dec-04	**2453355.39**	**116.43**	**8.51**	**176.32**	**7.63**
16-Dec-04	2453356.39	116.63	8.67	175.61	7.64
17-Dec-04	2453357.39	117.00	8.83	174.90	7.65
18-Dec-04	2453358.39	117.52	9.00	174.19	7.67
19-Dec-04	2453359.39	118.20	9.16	173.47	7.69
20-Dec-04	2453360.39	119.03	9.33	172.77	7.71
21-Dec-04	2453361.39	120.01	9.49	172.07	7.75
22-Dec-04	2453362.39	121.15	9.66	171.37	7.78
23-Dec-04	2453363.39	122.44	9.82	170.68	7.82
24-Dec-04	2453364.39	123.89	9.98	170.00	7.87
25-Dec-04	2453365.39	125.50	10.14	169.32	7.92
26-Dec-04	2453366.39	127.27	10.29	168.66	7.98
27-Dec-04	2453367.39	129.20	10.45	168.01	8.04
28-Dec-04	2453368.39	131.30	10.59	167.37	8.11
29-Dec-04	2453369.39	133.58	10.73	166.75	8.18

Jupiter Gravity Assist Control Point						
JGA Date (Calendar)	JGA Date (Julian)	Periapsis Altitude (km)	B Plane Angle (degrees)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
29-Apr-06	2453856.38	1019477.68	6.49	13.53	2.00	64.25
30-Apr-06	2453856.65	1021134.18	6.44	13.53	2.05	64.31
30-Apr-06	2453856.92	1022697.68	6.39	13.53	2.10	64.36
30-Apr-06	2453857.18	1024168.32	6.35	13.54	2.15	64.41
30-Apr-06	2453857.44	1025548.12	6.31	13.54	2.19	64.46
1-May-06	2453857.69	1026837.58	6.28	13.54	2.23	64.50
1-May-06	2453857.94	1028031.93	6.24	13.55	2.27	64.54
1-May-06	2453858.18	1029133.63	6.21	13.55	2.31	64.58
1-May-06	2453858.42	1030141.94	6.17	13.55	2.34	64.62
2-May-06	2453858.65	1031056.16	6.14	13.56	2.37	64.65
2-May-06	2453858.87	1031875.55	6.11	13.56	2.41	64.68
2-May-06	2453859.09	1032599.40	6.08	13.56	2.43	64.71
2-May-06	2453859.30	1033227.00	6.06	13.57	2.46	64.74
3-May-06	2453859.51	1033757.69	6.03	13.57	2.49	64.76
3-May-06	**2453859.72**	**1034190.86**	**6.01**	**13.57**	**2.52**	**64.78**
3-May-06	2453859.91	1034525.90	5.98	13.58	2.54	64.80
3-May-06	2453860.11	1034762.21	5.96	13.58	2.57	64.81
3-May-06	2453860.30	1034899.18	5.93	13.58	2.59	64.82
3-May-06	2453860.48	1034936.17	5.91	13.58	2.61	64.83
4-May-06	2453860.67	1034872.53	5.89	13.59	2.63	64.84
4-May-06	2453860.84	1034707.56	5.87	13.59	2.65	64.84
4-May-06	2453861.02	1034440.54	5.85	13.59	2.67	64.84
4-May-06	2453861.18	1034070.73	5.83	13.59	2.69	64.84
4-May-06	2453861.35	1033597.38	5.81	13.60	2.71	64.84
5-May-06	2453861.51	1033019.71	5.79	13.60	2.73	64.83
5-May-06	2453861.67	1032336.91	5.77	13.60	2.75	64.82
5-May-06	2453861.82	1031548.18	5.76	13.60	2.76	64.80
5-May-06	2453861.98	1030652.70	5.74	13.61	2.78	64.78
5-May-06	2453862.12	1029649.58	5.72	13.61	2.79	64.76

Pluto Arrival Hyperbola Control Point				
Arrival Date (Calendar)	Arrival Date (Julian)	V_∞ (km/s)	Declination V_∞ (deg)	Rt. Ascension V_∞ (deg)
16-Jun-14	2457189.89	12.91	9.04	273.45
16-Jun-14	2457189.89	12.91	9.04	273.45
16-Jun-14	2457189.89	12.91	9.04	273.45
16-Jun-14	2457189.89	12.91	9.04	273.45
16-Jun-14	2457189.89	12.91	9.04	273.45
16-Jun-14	2457189.89	12.91	9.04	273.44
16-Jun-14	2457189.89	12.91	9.04	273.44
16-Jun-14	2457189.89	12.91	9.04	273.44
16-Jun-14	2457189.89	12.91	9.04	273.44
16-Jun-14	2457189.89	12.91	9.04	273.44
16-Jun-14	2457189.89	12.91	9.04	273.43
16-Jun-14	2457189.89	12.91	9.04	273.43
16-Jun-14	2457189.89	12.91	9.04	273.43
16-Jun-14	2457189.89	12.92	9.04	273.43
16-Jun-14	**2457189.89**	**12.92**	**9.04**	**273.43**
16-Jun-14	2457189.89	12.92	9.04	273.43
16-Jun-14	2457189.89	12.92	9.04	273.43
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.42
16-Jun-14	2457189.89	12.92	9.04	273.41
16-Jun-14	2457189.89	12.92	9.04	273.41
16-Jun-14	2457189.89	12.92	9.04	273.41
16-Jun-14	2457189.89	12.92	9.04	273.41

APPENDIX I: DETAILED MOTOR CHARACTERISTICS AND CALCULATIONS SUMMARY

Motor Type	Star 27	Star 30BP	Star 48B	Star 63F	Equation or Variable
Ignition Mass (kg)[**]	361.20	544.90	2143.00	4591.67	M_{ign}
Module Mass at Burnout (kg)	24.22	34.75	118.61	291.80	M_{bo}
Payload Mass Fraction	0.31	0.23	0.07	0.03	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass (kg)	336.98	510.15	2024.39	4299.87	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn (kg)	521.20	704.90	2303.00	4751.67	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout (kg)	184.22	194.75	278.61	451.80	$M_f=M_i-M_p$
Effective Specific Impulse (s)	288.00	292.00	292.10	297.10	I_{sp}
Average Thrust (N)	25700	26500	68400	105000	T
ΔV (km/s)	2.94	3.68	6.05	6.86	$\Delta V=g_0 I_{sp}[\ln(M_i/M_f)]/1000$
Acceleration Before Burn (m/s^2)	49.31	37.59	29.70	22.10	$a_i=T/M_i$
G-Force Before Burn	5.03	3.83	3.03	2.25	$G_i=a_i/g_0$
Acceleration After Burn (m/s^2)	139.51	136.07	245.50	232.40	$a_f=T/M_f$
G-Force After Burn	14.22	13.87	25.03	23.69	$G_f=a_f/g_0$

M_{PKE}=160 kg
g_0=9.81 m/s^2

Table 6: STAR Rocket Motor Calculations

DELTA 7925 CALCULATIONS

Because of the mass restriction on the Delta to GTO, only two STAR motors were considered, the STAR 27 and 30BP. The following Excel tables, Tables 7 and 8, illustrate the steps taken to determine the maximum ΔV obtainable from each launch combination.

Select motor from the following list: Star 27, Star 30BP	**Star 27**	
Fuel load:	100	%
Mass of Motor Wet:	361.2	kg
Mass of Spacecraft + Adapter:	160.00	kg
Total Mass for Delta (motor + spacecraft + adapter): (Plus 10% for LV Performance)	573.32	kg
C_3 From Delta 7925:	39.24	km^2/s^2
ΔV From Delta 7925:	4.88	km/s
ΔV From Motor:	2.94	km/s
Total ΔV:	**7.82**	**km/s**

Table 7: Delta 7925 With Star 27 Motor

[**] Ignition Mass, Module Mass at Burnout, Effective Specific Impulse, and Average Thrust from Barnett and Farless, 1996, p. 28.

Select motor from the following list: Star 27, Star 30BP	**Star 30BP**	
Fuel load:	100	%
Mass of Motor Wet:	544.9	kg
Mass of Spacecraft + Adapter:	160.00	kg
Total Mass for Delta (motor + spacecraft + adapter): (Plus 10% for LV Performance)	775.39	kg
C_3 From Delta 7925:	24.73	km^2/s^2
ΔV From Delta 7925:	4.30	km/s
ΔV From Motor:	3.68	km/s
Total ΔV:	**7.98**	**km/s**

Table 8: Delta 7925 With Star 30BP Motor

To calculate the "C_3 From Delta 7925," the curve-fitted graph shown in Figure 19 was used, and the equation plugged into the spreadsheet, the x = "Total Mass for Delta". Curve fitting was executed on DeltaGraph 4.0, because this feature was found to be inadequate on Excel.

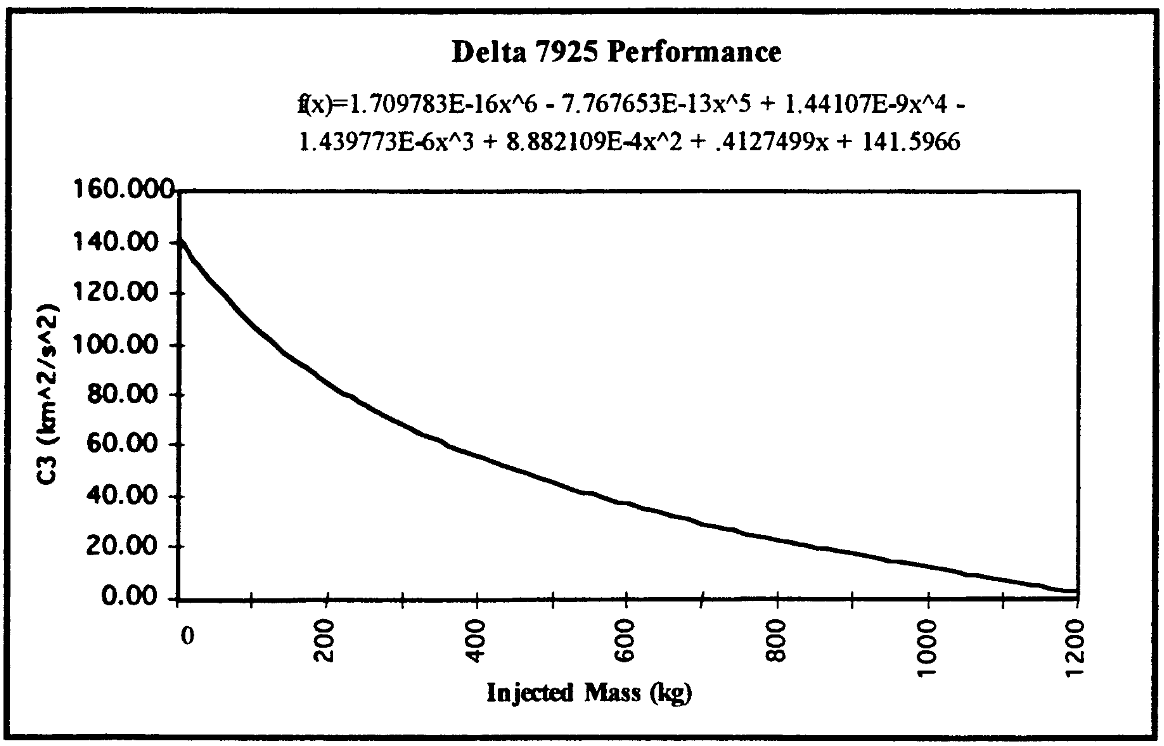

Figure 19: Delta 7925 Performance Capability

The "ΔV From Delta 7925" required to achieve a given interplanetary injection energy (C_3), is calculated by:

$$\Delta V = \sqrt{\left(C_3 + \frac{2\mu}{r_{park}}\right)} - \sqrt{\left(\frac{\mu}{r_{park}}\right)}, \text{ km/s}$$

In this, case r_{park} is actually the altitude at which the third stage fires, 6563.1 km. "ΔV From Motor" is the from Table 6 at the beginning of this appendix. "Total ΔV" is the sum of the ΔV's from the motor and the Delta 7925.

ATLAS CALCULATIONS

Motor Type	Centaur-II (RL10A-3)	Centaur-IIA (RL 10A-4)	Equation or Variable
Ignition Mass (kg)	15626.00	18484.00	M_{ign}
Module Mass at Burnout (kg)	1837.00	1928.00	M_{bo}
Payload Mass Fraction	0.01	0.01	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass (kg)	13789.00	16556.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn (kg)	15786.00	18644.00	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout (kg)	1997.00	2088.00	$M_f=M_i-M_p$
Effective Specific Impulse (s)	444.40	451.00	I_{sp}
Average Thrust (N)	146800	198400	T
ΔV (km/s)	8.56	9.20	$\Delta V=g_0 I_{sp}[\ln(M_i/ M_f)]/1000$
Acceleration Before Burn (m/s^2)	9.30	10.64	$a_i=T/M_i$
G-Force Before Burn	0.95	1.08	$G_i=a_i/g_0$
Acceleration After Burn (m/s^2)	73.51	95.02	$a_f=T/M_f$
G-Force After Burn	7.49	9.69	$G_f=a_f/g_0$

Table 9: Atlas Centaur Rocket Calculations

MOLNIYA CALCULATIONS

Motor Type	Block L (Molniya)	Equation or Variable
Ignition Mass (kg)	4500.00	M_{ign}
Module Mass at Burnout (kg)	1050.00	M_{bo}
Payload Mass Fraction	0.03	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass (kg)	3450.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn (kg)	4660.00	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout (kg)	1210.00	$M_f=M_i-M_p$
Effective Specific Impulse (s)	340.00	I_{sp}
Average Thrust (N)	66700.00	T
ΔV (km/s)	4.50	$\Delta V=g_0 I_{sp}[\ln(M_i/ M_f)]/1000$
Acceleration Before Burn (m/s^2)	14.31	$a_i=T/M_i$
G-Force Before Burn	1.46	$G_i=a_i/g_0$
Acceleration After Burn (m/s^2)	55.12	$a_f=T/M_f$
G-Force After Burn	5.62	$G_f=a_f/g_0$

Table 10: Molniya Block L Rocket Calculations

Upper Stage	STAR 27	STAR 30BP	STAR 48B	Equations and Variables
Effective Payload Mass, M_{pl} (kg)	521.20	704.90	2303.00	$M_{pl}=M_{PKE}+M_{ign}$
Ignition Mass, M_{ign} (kg)	4500.00	4500.00	4500.00	M_{ign}
Module Mass at Burnout, M_{bo} (kg)	1050.00	1050.00	1050.00	M_{bo}
Payload Mass Fraction	0.10	0.14	0.34	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass, M_p (kg)	3450.00	3450.00	3450.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn, M_i (kg)	5021.20	5204.90	6803.00	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout, M_f (kg)	1571.20	1754.90	3353.00	$M_f=M_i-M_p$
Effective Specific Impulse (s)	340.00	340.00	340.00	I_{sp}
Average Thrust, T (N)	66700.00	66700.00	66700.00	T
ΔV (km/s)	3.88	3.63	2.36	$\Delta V=g_0I_{sp}[\ln(M_i/M_f)]$
Acceleration Before Burn (m/s^2)	13.28	12.81	9.80	$a_i=T/M_i$
G-Force Before Burn	1.35	1.31	1.00	$G_i=a_i/g_0$
Acceleration at End of Burn (m/s^2)	42.45	38.01	19.89	$a_f=T/M_f$
G-Force After Burn	4.33	3.87	2.03	$G_f=a_f/g_0$
Total ΔV for Interplanetary (km/s)	6.81	7.31	8.41	$\Delta V_{upper\ stage}+\Delta V_{STAR}$

Table 11: Molniya Block L With STAR Motors

PROTON CALCULATIONS

Motor Type	Block D (Proton)	Equation or Variable
Ignition Mass (kg)	18400.00	M_{ign}
Module Mass at Burnout (kg)	3300.00	M_{bo}
Payload Mass Fraction	0.01	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass (kg)	15100.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn (kg)	18560.00	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout (kg)	3460.00	$M_f=M_i-M_p$
Effective Specific Impulse (s)	361.00	I_{sp}
Average Thrust (N)	83500.00	T
ΔV (km/s)	5.95	$\Delta V=g_0I_{sp}[\ln(M_i/\ M_f)]/1000$
Acceleration Before Burn (m/s^2)	4.50	$a_i=T/M_i$
G-Force Before Burn	0.46	$G_i=a_i/g_0$
Acceleration After Burn (m/s^2)	24.13	$a_f=T/M_f$
G-Force After Burn	2.46	$G_f=a_f/g_0$

Table 12: Proton Block D Rocket Calculations

Upper Stage	STAR 27	STAR 30BP	STAR 48B	Equations and Variables
Effective Payload Mass, M_{pl} (kg)	521.20	704.90	2303.00	$M_{pl}=M_{PKE}+M_{ign}$
Ignition Mass, M_{ign} (kg)	18400.00	18400.00	18400.00	M_{ign}
Module Mass at Burnout, M_{bo} (kg)	3300.00	3300.00	3300.00	M_{bo}
Payload Mass Fraction	0.03	0.04	0.11	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass, M_p (kg)	15100.00	15100.00	15100.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn, M_i (kg)	18921.20	19104.90	20703.00	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout, M_f (kg)	3821.20	4004.90	5603.00	$M_f=M_i-M_p$
Effective Specific Impulse (s)	361.00	361.00	361.00	I_{sp}
Average Thrust, T (N)	83500.00	83500.00	83500.00	T
ΔV (km/s)	5.67	5.53	4.63	$\Delta V=g_0 I_{sp}[\ln(M_i/M_f)]$
Acceleration Before Burn (m/s^2)	4.41	4.37	4.03	$a_i=T/M_i$
G-Force Before Burn	0.45	0.45	0.41	$G_i=a_i/g_0$
Acceleration at End of Burn (m/s^2)	21.85	20.85	14.90	$a_f=T/M_f$
G-Force After Burn	2.23	2.13	1.52	$G_f=a_f/g_0$
Total ΔV for Interplanetary (km/s)	8.60	9.22	10.68	$\Delta V_{upper\ stage}+\Delta V_{STAR}$

Table 13: Proton Block D With Star Motors

ARIANE CALCULATIONS

Motor Type	HM7B (Ariane 4)	L9 (Ariane 5)	Equation or Variable
Ignition Mass (kg)	12100.00	10900.00	M_{ign}
Module Mass at Burnout (kg)	1300.00	1200.00	M_{bo}
Payload Mass Fraction	0.01	0.01	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass (kg)	10800.00	9700.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn (kg)	12260.00	11060.00	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout (kg)	1460.00	1360.00	$M_f=M_i-M_p$
Effective Specific Impulse (s)	444.20	324.00	I_{sp}
Average Thrust (N)	62700.00	27500.00	T
ΔV (km/s)	9.27	6.66	$\Delta V=g_0 I_{sp}[\ln(M_i/ M_f)]/1000$
Acceleration Before Burn (m/s^2)	5.11	2.49	$a_i=T/M_i$
G-Force Before Burn	0.52	0.25	$G_i=a_i/g_0$
Acceleration After Burn (m/s^2)	42.95	20.22	$a_f=T/M_f$
G-Force After Burn	4.38	2.06	$G_f=a_f/g_0$

Table 14: Ariane Upper Stage Rocket Calculations

Upper Stage	STAR 27	STAR 30BP	STAR 48B	STAR 63F	Equations and Variables
Effective Payload Mass, M_{pl} (kg)	521.20	704.90	2303.00	4751.67	$M_{pl}=M_{PKE}+M_{ign}$
Ignition Mass, M_{ign} (kg)	10900.00	10900.00	10900.00	10900.00	M_{ign}
Module Mass at Burnout, M_{bo} (kg)	1200.00	1200.00	1200.00	1200.00	M_{bo}
Payload Mass Fraction	0.05	0.06	0.17	0.30	$PMF=M_{PKE}/(M_{PKE}+M_{ign})$
Propellant Mass, M_p (kg)	9700.00	9700.00	9700.00	9700.00	$M_p=M_{ign}-M_{bo}$
Initial Mass Before Burn, M_i (kg)	11421.20	11604.90	13203.00	15651.67	$M_i=M_{PKE}+M_{ign}$
Final Mass at Burnout, M_f (kg)	1721.20	1904.90	3503.00	5951.67	$M_f=M_i-M_p$
Effective Specific Impulse (s)	324.00	324.00	324.00	324.00	I_{sp}
Average Thrust, T (N)	27500.00	27500.00	27500.00	27500.00	T
ΔV (km/s)	5.71	5.74	4.22	3.07	$\Delta V=g_0 I_{sp}[\ln(M_i/M_f)]$
Acceleration Before Burn (m/s^2)	2.41	2.37	2.08	1.76	$a_i=T/M_i$
G-Force Before Burn	0.25	0.24	0.21	0.18	$G_i=a_i/g_0$
Acceleration at End of Burn (m/s^2)	15.98	14.44	7.85	4.62	$a_f=T/M_f$
G-Force After Burn	1.63	1.47	0.80	0.47	$G_f=a_f/g_0$
Total ΔV for Interplanetary (km/s)	8.65	9.43	10.27	9.93	$\Delta V_{upper\ stage}+\Delta V_{STAR}$

Table 15: Ariane 5 L9 With Star Motors

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