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An Investigation of a Mission to Enceladus and Possible Return

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TO ENCELADUS AND A POSSIBLE RETURN

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Abstract

The 2006-2007 Capstone Senior Design team has investigated the possibility of launching a mission to Enceladus, a small inner satellite of Saturn. Recent discoveries by the Cassini mission and speculation about the presence of liquid water have heightened scientific interest in this small moon. In an effort to provide scientists with an opportunity for further study, the design team has conducted detailed analysis of high-thrust and low-thrust trajectories to send a probe to the Saturn system. Possible orbits upon arrival at Saturn were studied to determine the best options for encountering Enceladus. Research was also conducted on methods of sample collection and a possible return to Earth from the Saturn system. Through analysis of different high thrust trajectory options, an optimum launch date has been proposed for January 14, 2018, with arrival dates ranging from May 2025 through March 2026. An optimum low thrust trajectory has also been generated, with launch and arrival scheduled for December 14, 2021 and January 26, 2029, respectively. More work would be required to feather develop a complete mission architecture.

Table of Contents

ABSTRACT **3**

TABLE OF CONTENTS **4**

LIST OF FIGURES AND TABLES **5**

INTRODUCTION **7**

OBJECTIVES **8**

HIGH THRUST TRAJECTORIES **9**

Program Created and Calculations Performed 11
SAIC Trajectory Optimizer 19
Mission Analysis Environment (MAnE) 20
JAQAR Swing-by Calculator 24

LOW THRUST TRAJECTORIES **33**

SATURN SYSTEM ORBITS **36**

Sample Collection 40

POSSIBLE RETURN MISSION **41**

Titan Flyby Departure 42
Low-Thrust Return 48
Sample Return Capsule (SRC) 49

REFERENCES: **51**

APPENDIX A: PLANETARY FLYBY **53**

APPENDIX B: MANE OUTPUTS **56**

APPENDIX C: SWING-BY CALCULATOR **63**

APPENDIX D: CHEBYTOP OUTPUTS **69**

List of Figures and Tables

Figure 1: Cassini Interplanetary Trajectory Launch Date: October 6, 1997, Arrival Date: June 25, 2004 10
Figure 2: Earth to Venus - Change in Velocity vs. Transfer Time 12
Figure 3: Earth to Mars - Change in Velocity vs. Transfer Time 12
Figure 4: Earth to Jupiter - Change in Velocity vs. Transfer Time 13
Figure 5: Earth to Saturn - Change in Velocity vs. Transfer Time 13
Figure 6: Venus Flyby - Impact Parameter as a Function of Inbound Velocity 14
Figure 7: Mars Flyby - Impact Parameter as a Function of Inbound Velocity 15
Figure 8: Jupiter Flyby - Impact Parameter as a Function of Inbound Velocity 15
Figure 9: Venus Flyby - Turn Angle as a Function of Offset Distance 16
Figure 10: Mars Flyby - Turn Angle as a Function of Offset Distance 16
Figure 11: Jupiter Flyby - Turn Angle as a Function of Offset Distance 17
Figure 12: Venus Flyby - Change in Heliocentric Velocity as a Function of Offset Distance 17
Figure 13: Mars Flyby - Change in Heliocentric Velocity as a Function of Offset Distance 18
Figure 14: Jupiter Flyby - Change in Heliocentric Velocity as a Function of Offset Distance 18
Figure 15: Cassini Mission Independent Parameters Input Window 20
Figure 16: Earth to Saturn Trajectory Projection MAnE Optimum Total ΔV = 7.99 km/s, Time of Flight: 3648 days 22
Figure 17: Earth to Saturn Trajectory Projection with Additional Constraints 23
Figure 18: Earth to Saturn Trajectory Projection with Additional Constraints Best Found Trajectory with Reasonable Time of Flight 23
Figure 19: Contour Plots of the Total ΔV and the Corresponding Time of Flight 27
Figure 20: Total ΔV and the corresponding Time of Flight 28
Figure 21: Departure Window for the Earth-Jupiter-Saturn mission 29
Figure 22: Plot of the Earth-Jupiter-Saturn flight path 29
Figure 23: Contour plot of EVVEJS mission 31
Figure 24: Total ΔV and the corresponding Time of Flights 32
Figure 25: Plot of the EVVEJS mission flight path 33
Figure 26: Plot output from CHEBYTOP sample run 35

Figure 27: Insertion Delta-V versus Apoapse Radius (Periapse Radius at Enceladus' Orbit) for Various Inbound Velocities 39
Figure 28: Aerogel collection tray prior to boom insertion 41
Figure 29: Graphical Representation of the first 3 Flybys of Titan 46
Figure 30: Graphical Representation of the first four flybys of Titan 47
Figure 31: Low Thrust Sample Return Capsule Trajectory using ChebyTop 48
Figure 32: Sample Return Capsule geometry 49

Introduction

In the past forty years, significant advances have been made in the exploration of the outer realms of our solar system. NASA missions, including Pioneer and Voyager, have been sending back images and data from Jupiter, Saturn, and Neptune. The planet Saturn has been one of significant interest due to its rings, large magnetosphere, and its many moons. The Cassini-Huygens mission was launched in 1997 to gather information about Saturn and its natural satellite, Titan. Titan is one of the only satellites in our solar system with an atmosphere, and it has been speculated to be very similar to earth more than 3.8 billion years ago. The study of Titan would provide information about planetary formation and perhaps the early days of Earth. The spacecraft reached the Saturn system in June of 2004 and has continued orbiting the planet, sending back data and images.

The Cassini mission has also encountered the small inner satellite, Enceladus, during its orbit about Saturn and has provided scientists with fascinating new information. Some surfaces of Enceladus are exceptionally smooth with very few craters, indicating some recent resurfacing events. There are fissures, plains, corrugated terrain and other crustal deformations indicating that the interior of the moon may be liquid. Because the surface is bright white, it reflects almost all of the sunlight that hits it and has a surface temperature of approximately -200°C, causing scientists to believe that the interior of the planet should have frozen billions of years ago. This suggests that the natural satellite is heated by some internal source or a tidal mechanism. Scientists have also discovered an atmosphere around Enceladus, indicating gasses are being emitted. Large ice boulders have been found at the south pole, indicating recent geological activity. Further investigation showed that the south pole was significantly warmer than expected, supporting the theory of an internal heat source. The "tiger stripes" located near the southern pole were determined to be young and supplied with fresh ice. Dust or ice particles have been recorded coming from Enceladus and were determined to be the source of Saturn's E-ring. Water vapor was also discovered in the atmosphere. During Cassini's last encounter of Enceladus in November of 2005, it captured pictures of the southern pole emitting a geyser of dust and ice particles into space. This icy plume is thought to be fed by internal liquid reservoirs in the satellite.

In light of its recent discoveries, Cassini has greatly heightened interest in Enceladus. Further investigation is needed to determine more about the internal heat source and possible liquid water. The content of the plume emitted from the south pole is also of significant interest. Therefore, a new mission should be developed to launch a probe specifically to investigate Enceladus.

Objectives

The objective of the 2006-2007 senior class space design project was to design a mission solely for the purpose of investigating Saturn's inner moon, Enceladus. Complete system architecture from Earth to the Saturn system, with the use of different trajectory software, was explored. Thought was also given to a spacecraft and its dry mass, the propellant required to place the craft in orbit, and the launch vehicle required to leave Earth. Once the Saturn system was reached, an investigation of Saturn orbit insertions and maneuvers was also addressed. Additionally, methods of data collection and a possible subsequent Earth return trajectory, were also researched. A detailed record of current findings and plans were recorded in order that future teams might continue the effort of the mission's planning. The following outlines the current progress.

High Thrust Trajectories

High thrust trajectories have been the standard in interplanetary missions. In this technique, a tremendous amount of thrust is applied within a very short time, imparting a nearly instantaneous change in velocity. Although new methods for interplanetary travel are emerging, it was obvious that high-thrust maneuvers should be the first option explored to reach the Saturn system, due to successes with previous missions.

The simplest high-thrust trajectory from Earth to Saturn would be a Hohmann transfer. The Hohmann transfer is simply an elliptical path between two circular coplanar orbits, tangent to each one. This maneuver typically requires the least amount energy (and smallest ΔV) to move an object between two celestial bodies. Assuming the spacecraft begins in a parking orbit around Earth at a 400 km altitude, the change in velocity required to launch the vehicle to Saturn on a Hohmann transfer is 7.28 km/s, and the time of flight required is 6.04 years.

More complicated sequences of trajectories may be employed to reduce the overall energy consumption on interplanetary missions. For instance, a flyby of a planet can change the direction of motion of a spacecraft to give it greater momentum in the sun-centered reference frame. The Cassini mission executed numerous flybys to reach the Saturn system. Figure 1 shows the Cassini trajectory with dates of different events. Since the relative positions of Earth, Venus, Jupiter, and Saturn will not repeat this pattern within the 21st century, it would be impossible to repeat this exact mission in the time frame considered for this planned mission to Enceladus. Similar maneuvers, however, could be performed in order to reduce the need for propulsive velocity changes. The Cassini mission had a departure ΔV of approximately 4.0 km/s and a deep space propulsive burn maneuver of about 0.5 km/s, for a total ΔV of approximately

4.5 km/s required to reach the Saturn system. The total transit time between launch and arrival was 2451 days.

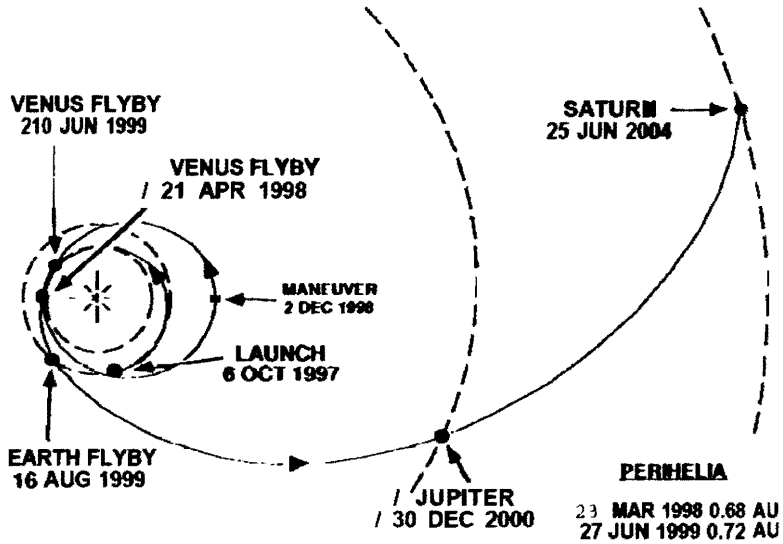

Figure 1: Cassini Interplanetary Trajectory Launch Date: October 6, 1997, Arrival Date: June 25, 2004

By comparing the simplest trajectory of a Hohmann transfer to the complex Cassini trajectory, a range of acceptable results was established. ΔV's should fall below the upper bound of the Hohmann transfer of 7.28 km/s. No lower bound was established, but results were expected to be similar to that of the Cassini mission trajectory. The Cassini mission ΔV and time of flight were therefore used as the standard throughout analysis, due to that mission's success and similarity to the objectives of this Enceladus mission.

From the evaluation of the Cassini mission and other missions to the outer planets, it was obvious a trajectory with multiple gravity assist maneuvers, possible powered flybys, and possible deep space burns was needed. To create such a trajectory, each separate event had to be aligned in such a manner that the next planet involved was in the right position so that the spacecraft would encounter it. A program capable of iterating dates and maneuvers resulting in different velocity changes was needed to piece the different legs of the mission together. Several different programs were used in the attempt of creating a high thrust trajectory capable of reaching Saturn in a reasonable amount of spent fuel. The following outlines the different programs used.

Program Created and Calculations Performed

In order to form a basis of comparison for ΔV consumption between different high-thrust trajectories, preliminary calculations were made using a patched conic approximation between the earth and various planets. In these calculations the orbits of the planets were again assumed to be circular and co-planar. A code was then implemented using Matlab to determine the variation of required ΔV-consumption versus flight times (Figures 2 through 5) for transfers to Venus, Mars, Jupiter and Saturn, respectively). These flight times were based on transfers applying varying ΔV burns, beginning with the minimum ΔV from a Hohmann transfer. Further, all initial departure conditions were input as low-earth orbits with an altitude of 400 km. From these graphs, it was evident that a flight to Venus would require a similar ΔV-consumption to that of Cassini (as expected), with a minimum possible ΔV of about 3.46 km/s. A trajectory to Mars was the only other trajectory determined to have a comparably low departure ΔV-consumption (minimum $\Delta V \approx 3.6$ km/s), since a departure to Jupiter or Saturn would require well over 4 km/s.

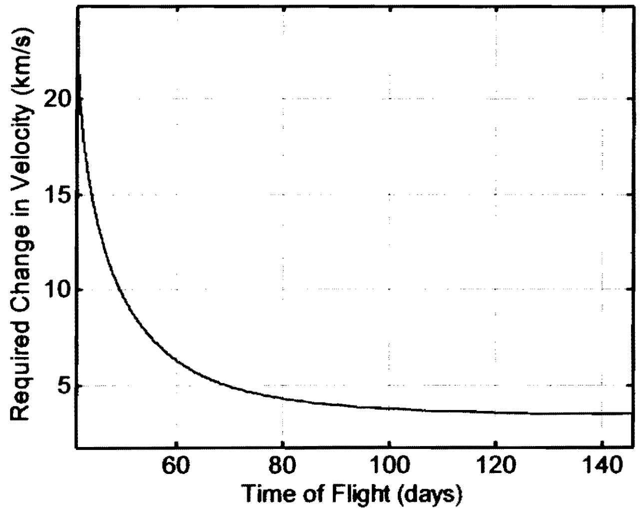

Figure 2: Earth to Venus - Change in Velocity vs. Transfer Time

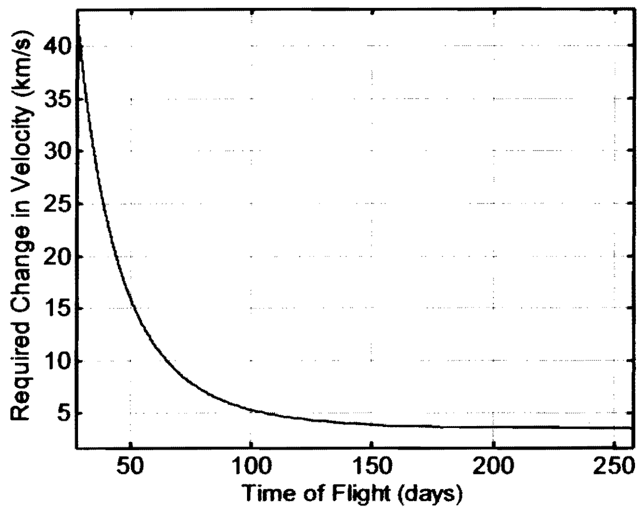

Figure 3: Earth to Mars - Change in Velocity vs. Transfer Time

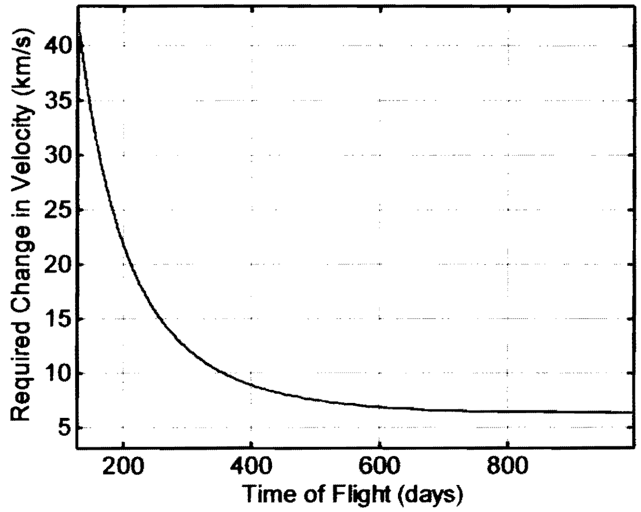

Figure 4: Earth to Jupiter - Change in Velocity vs. Transfer Time

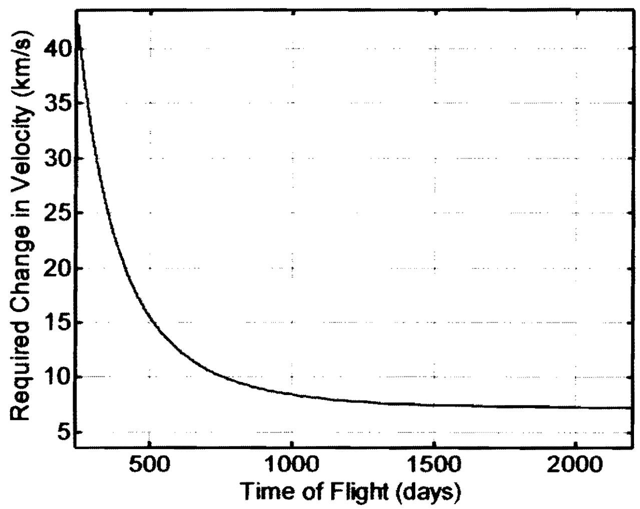

Figure 5: Earth to Saturn - Change in Velocity vs. Transfer Time

The most significant source of error in these approximations was likely the assumption that all the planets reside in the same plane. Since none of the planets considered are inclined more than 4° from the ecliptic plane[1], this would likely have contributed only minor increases in ΔV-consumption. The extra velocity change needed would be calculated as $\Delta V_i = V * \sin(\Delta i/2)$, applying the plane change at 90° true anomaly before intercept with the planet.[1] The inclination of 4° would thus have resulted in a maximum ΔV of only 3.5% times the current velocity, and would not seriously have affected the total ΔV as a function of time of flight.

Further efforts were made to determine the feasibility and effectiveness of flybys upon arrival at prospective intermediate planets (i.e. Venus, Mars and Jupiter) along a trajectory to Saturn.[2] The impact parameters for each planet were plotted against inbound V_∞ in order to determine the minimum possible offset distance, which corresponds to the maximum attainable turn angle. These have been plotted for Venus, Mars and Jupiter in Figures 6 through 8, respectively, and they have been truncated to exclude inbound velocities lower than those expected from a Hohmann transfer (i.e., velocities which are not possible).

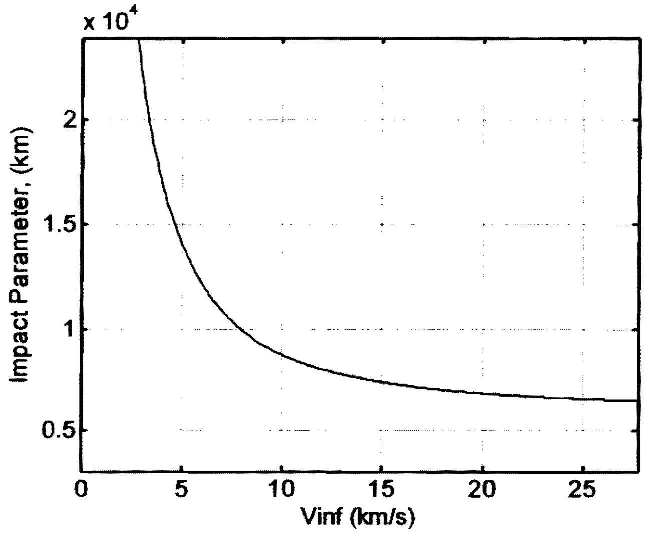

Figure 6: Venus Flyby - Impact Parameter as a Function of Inbound Velocity

[1] Bate, et al.
[2] For a detailed discussion of flybys, see Appendix A

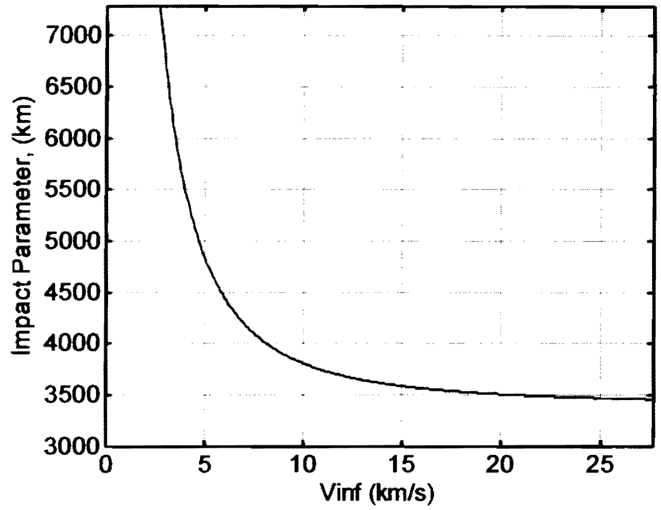

Figure 7: Mars Flyby - Impact Parameter as a Function of Inbound Velocity

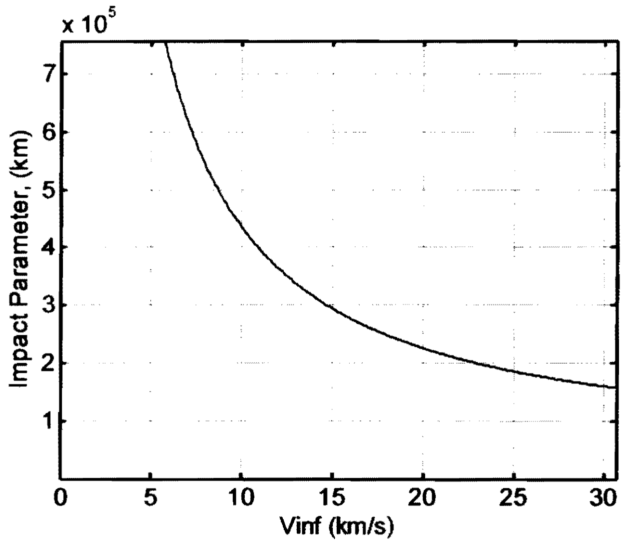

Figure 8: Jupiter Flyby - Impact Parameter as a Function of Inbound Velocity

Additionally, the turn angle and change in heliocentric speed were plotted for these flybys in Figures 9 through 14. From Figure 13, it was apparent that a flyby offset distance of less than about 13,000 km would be necessary to get an output of more than 1 km/s from a Mars flyby, and that the maximum possible increase in heliocentric velocity would be under 2.5 km/s.

Because this gravitational boost offered by Mars was so low, it was deemed impractical to use Mars as the target of a flyby in a sequence to reach Saturn.

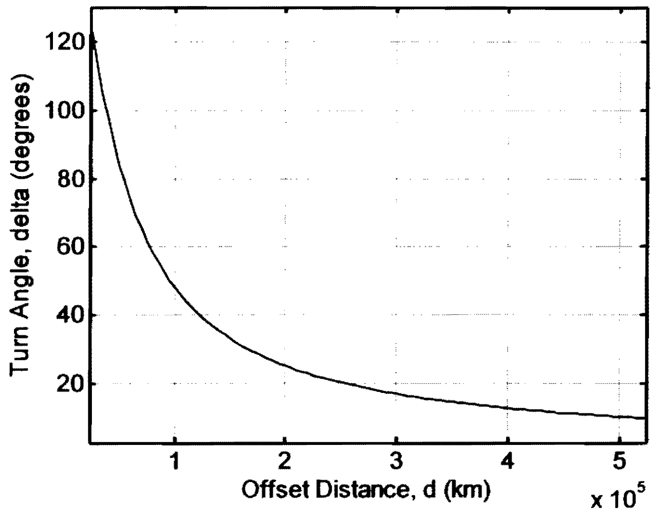

Figure 9: Venus Flyby - Turn Angle as a Function of Offset Distance

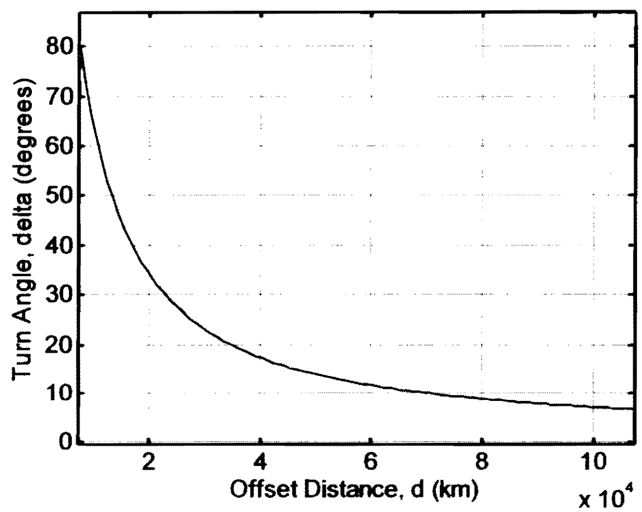

Figure 10: Mars Flyby - Turn Angle as a Function of Offset Distance

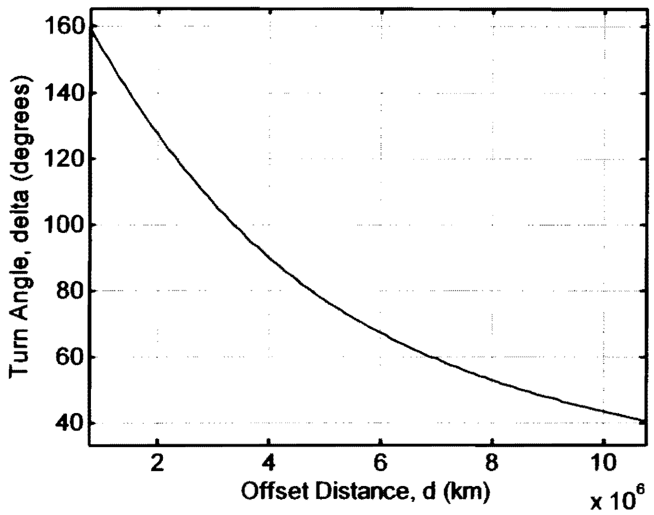

Figure 11: Jupiter Flyby - Turn Angle as a Function of Offset Distance

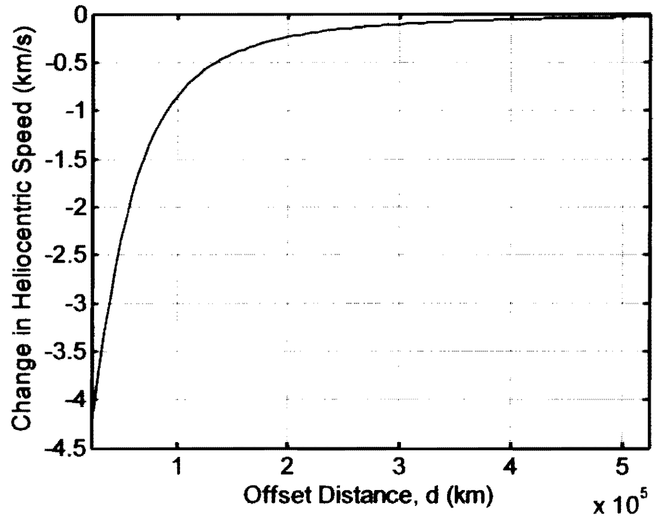

Figure 12: Venus Flyby - Change in Heliocentric Velocity as a Function of Offset Distance

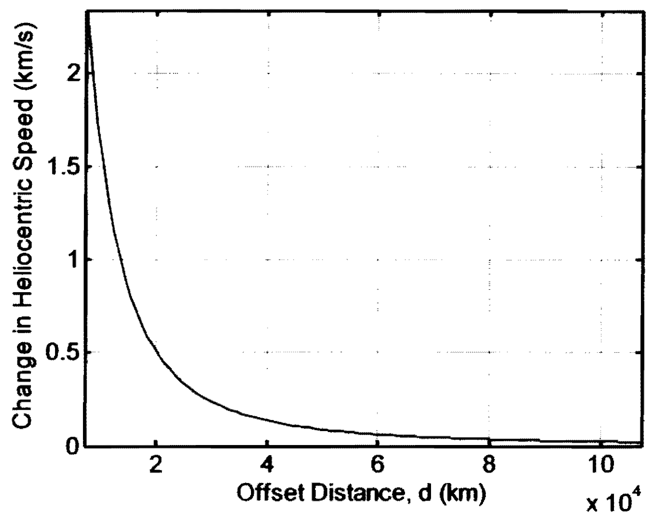

Figure 13: Mars Flyby - Change in Heliocentric Velocity as a Function of Offset Distance

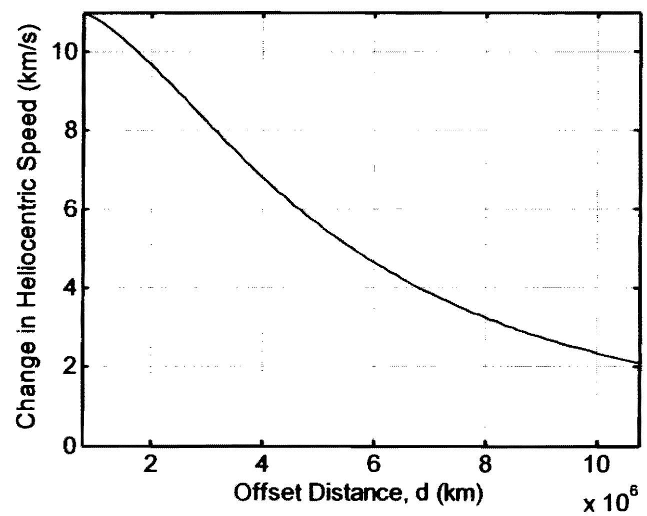

Figure 14: Jupiter Flyby - Change in Heliocentric Velocity as a Function of Offset Distance

Note that the calculations of all of these flyby properties were based on a Hohmann transfer between Earth and the respective planet, resulting in the minimum inbound V_∞. Any higher inbound V_∞ would result in a lower turn angle. Also, the flight path angle at encounter with the planet was set at 0°. This resulted in the negative changes in heliocentric velocity from the Venus flyby (since a craft would already be traveling faster than Venus upon a parallel approach). A Venus flyby, however, could increase the heliocentric speed of a spacecraft, if the craft entered Venus' sphere of influence at some flight path angle. In this case, the craft could maximize its increase in heliocentric velocity by aligning its velocity vector parallel to that of Venus through the execution a front-side flyby. Unfortunately, variable flight path angles were never taken into account in the determination of these flyby properties, as the production of this code was abandoned in favor of pre-existing, more developed orbital software packages.

SAIC Trajectory Optimizer

The first high thrust trajectory generation program used was the SAIC Trajectory Optimizer. This program came in two versions, a student version capable of 4 nodes, and a professional version, capable of 6 nodes. The professional version, however, would not run due to a maximum memory allotment that was reached during the execution of 5 or more nodes. This left only 2 nodes available for flybys, since the first node was reserved for launch from Earth and the last node for arrival at Saturn. It might have been possible to piece together two separate missions, had we needed to continue using the program, but this option was never attempted.

The SAIC program had several iterative problems. The solution often converged to include a flyby of the sun. Other supposedly optimum solutions had ΔV's in excess of 20 km/s, much greater than the desired upper bound of approximately 7.3 km/s. The professional version, which was supposed to contain additional node entry options, restrained the user from exploring multiple flyby options. The simplest trajectory options were not repeatable in the program and contained questionable ΔV values. Weighing the benefits this program had to offer against its drawbacks, the program was determined to be unsuitable for the planning of a complex mission to Saturn. The program would have been better suited for a simpler mission to Mars or Venus.

Mission Analysis Environment (MAnE)

The Mission Analysis Environment, (MAnE) is a software tool used to generate and optimize multiple-leg, heliocentric, high thrust missions. The software has the ability to optimize trajectories based on maximum net payload, minimum initial mass, total mission ΔV, or mission duration. It contains inputs for up to 10 nodes, departure and arrival orbital conditions, payload and dry mass vehicle conditions, and mathematical models and solvers.

Initial exploration of the Mission Analysis Environment included trying to recreate the Cassini mission. The software was equipped with several examples of historic trajectories, including Cassini. The input screen, shown in Figure 15, contained extremely accurate time and propulsive data for different events. An optimized trajectory was obtained quickly, yielding ΔV's that matched published Cassini data. Detailed results from this initial testing are found in Appendix B.

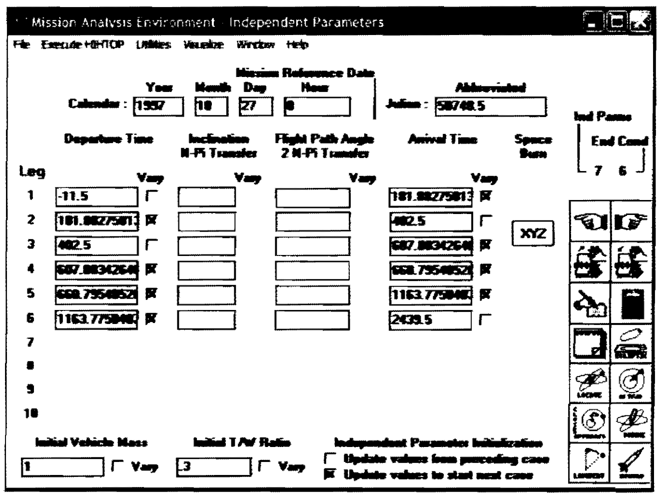

Figure 15: Cassini Mission Independent Parameters Input Window

To further investigate the iterative capability, input dates were changed and allowed to vary in order to determine if optimization would occur. Any changes to dates larger than a single day resulted in a failure to converge. This was an indication of the input precision needed to accurately utilize the software.

Several different attempts were made to generate a trajectory with possible departure dates ranging from 2015 to 2025. Initial departure was made from a 400 km parking orbit, and Saturn insertion maneuvers were executed in order to place the spacecraft in an elliptical orbit with a periapse equal to that of Enceladus' orbital distance and an apoapse equal to that of Titan's orbital distance[1]. The software lacked the parameters necessary to set a range of dates over which to search. Therefore, departure dates had to be manually changed by increments of 6 months in order to cycle through each possibility. The program contained a Personal Porkchop Plotter utility that should have assisted in the finding of a departure date. However, the utility would only work for single-leg missions. The use of Mission Analysis Environment was determined to be of little use as an initial planning tool. However, it had potential for being beneficial once initial dates for maneuvers had been determined.

Use of the Jaqar Swing-By Calculator (SBC) software, as discussed in the following section, provided a useful departure date of January 14, 2018 for a simple trajectory with one Jupiter flyby. This date was then input into MAnE for comparison. The dates of departure, inbound flyby, outbound flyby, and arrival all had the option of being allowed to vary, and all but the departure were left unfixed. Figure 16 shows the trajectory results obtained through the MAnE software. Optimization was obtained with the criteria of minimum total ΔV resulting in a total transfer time of 9.9 years and total ΔV of approximately 7.99 km/s. This transfer time found was beyond the upper bound of acceptable values, so additional parameters were imposed to analyze the effect of shorter transfer times on ΔV's.

[1] See Section on Saturn Orbits

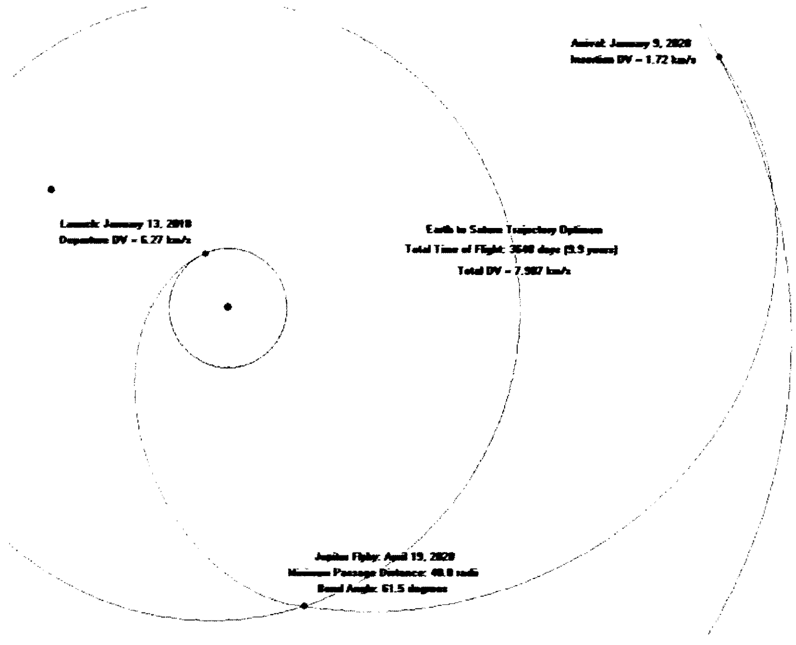

Figure 16: Earth to Saturn Trajectory Projection MAnE Optimum Total ΔV = 7.99 km/s, Time of Flight: 3648 days

The first step to achieve a better optimal solution required fixing the departure and arrival dates from Earth to Saturn. The departure date was set as the provided date achieved from SBC, and arrival was calculated from the Cassini mission length. MAnE was then allowed to converge to an optimum date for a Jupiter flyby. Next, this obtained encounter date was fixed and the arrival and departure dates were allowed to vary. This yielded more reasonable flight times and ΔV's. This method was repeated for flight times from 2400 days to 3000 days. Figures 17 and 18 show the trajectory projections with annotations containing ΔV's and dates. From the results, the expected trend of increasing flight time versus decreasing ΔV was observed. No utility was provided to generate several trajectories over a range of dates. Multiple inputs had to be tried by hand and each subsequent trajectory saved. This method proved too tedious to continue. The best trajectory found is shown in Figure 18. A detailed trajectory summary is also included in Appendix B.

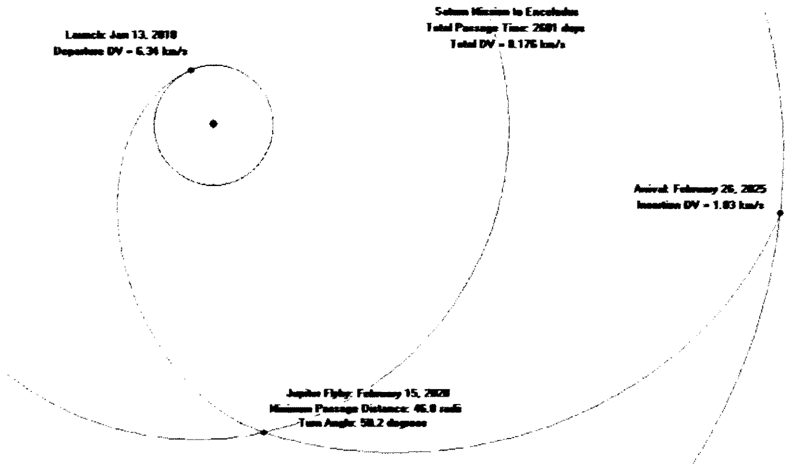

Figure 17: Earth to Saturn Trajectory Projection with Additional Constraints

Total ΔV = 8.18 km/s, Time of Flight: 2601 days

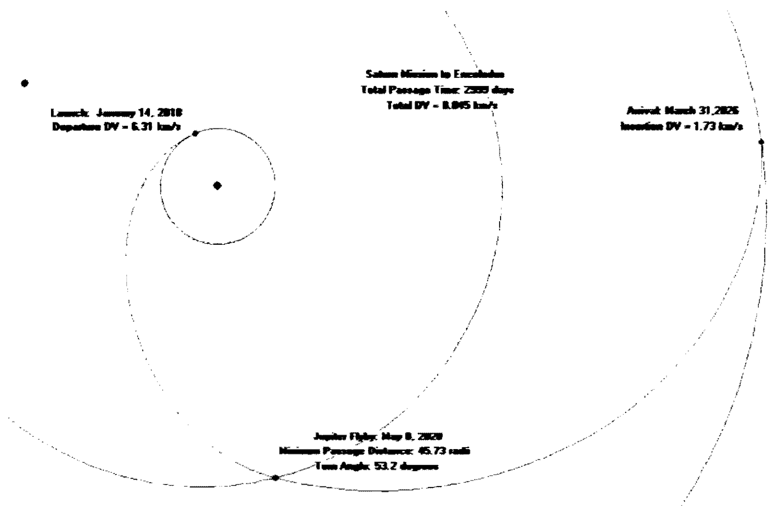

Figure 18: Earth to Saturn Trajectory Projection with Additional Constraints Best Found Trajectory with Reasonable Time of Flight

Total ΔV = 8.05 km/s, Time of Flight: 2999 days

Additional more complex trajectories were not attempted due to the lack of precise dates and maneuver data. Cassini-like trajectories were difficult to generate due to the amount of parameters that could be allowed to vary. Each leg would need significant work independently before attempting to piece an entire mission together. Tests should be conducted on Venus flybys and deep space maneuvers to create an inner planet trajectory capable of achieving larger velocities en route to Jupiter. Overall, MAnE successfully aided in the building of a single-flyby trajectory, but was not tested to its full capability, due to the required initial accuracy of inputs and time constraints.

JAQAR Swing-by Calculator

To calculate the possible high thrust trajectories, a computer program called the Swing-by Calculator (SBC) was used. The Swing-by Calculator was a software package developed by JAQAR Space Engineering to find trajectories from a departure planet to an arrival planet or heliocentric orbit via multiple swing-bys. To find an optimum departure date, SBC allowed input of a desired trajectory (including up to five planetary swing-bys) and a range of dates, with much greater variability than the ManE program. Extremely important was the ability to input a start date and a range over which to look for solutions that met the selected criteria. The program could be set to optimize total ΔV, departure C3, arrival C3 or total C3. Departure conditions could be input and, as before, the initial low-earth orbit altitude was set at 400 km. Final conditions at the arrival planet could be set either to a flyby or planetary orbit insertion. SBC also allowed for either powered or un-powered planetary swing-bys and calculations of deep-space maneuvers where necessary. Other constraints used in the calculation of different mission architectures were arrival conditions, minimum swing-by altitudes, maximum total ΔV, maximum transfer times, and maximum ΔV per swing-by. SBC also provided for several methods of calculation, the most useful of which were enumerative and differential evolution algorithms. Also, the program could be set to use the JPL DE405 ephemeris or its own analytical solution in calculating the location of the planets. Depending on what inputs, constraints and calculation methods were chosen, the calculations times varied from a few seconds to hours or days. The different settings also affected the accuracy of the data output by the program.

For the initial calculations, the SBC ephemeris and the differential evolution methods were used because they allowed the processing of large ranges of departure, swing-by and arrival dates in a reasonable calculation time. This method allowed a smaller range of departure dates meeting specific mission requirements to be found. Once this range of dates was found, powered flybys and deep space maneuvers could be added to find the minimum total ΔV. Now that a minimum ΔV was found for a given date, the range of dates was expanded again, and a new optimized minimum ΔV was calculated using the enumerative calculation and the JPL DE405 ephemeris. This process was used to yield several different possible missions, two of which were studied in greater detail because of their benefits of low ΔV's and reasonable transfer times.

The first of these two missions was a simple Earth-Jupiter-Saturn trajectory. To help find an optimal time of flight and total ΔV, a contour plot was created (Figure 19). For this mission, the minimum ΔV was found to be 8.114 km/s with a time of flight of 3057 days (Table 1). This time of flight was deemed much too long. From the contour plot and corresponding times of flight, three useful arrival dates were selected and used to determine variation in associated launch ΔV's. Figure 20 illustrates the total ΔV for the selected arrival dates with the corresponding time of flight. The figure revealed that a small increase in ΔV between selected arrival dates resulted in a large decrease in time of flight. For a ΔV increase of only 0.11 km/s, the time of flight decreased by 360 days. Consideration was also made for the possibility of altered launch dates. Figure 21 shows the increase in ΔV for the mission ranging from two weeks before to two weeks after the optimal departure date. Figure 22 represents the flight trajectory for the optimal mission with a total ΔV of 8.22 km/s and a time of flight of 2692 days, performing an un-powered swing-by of Jupiter.

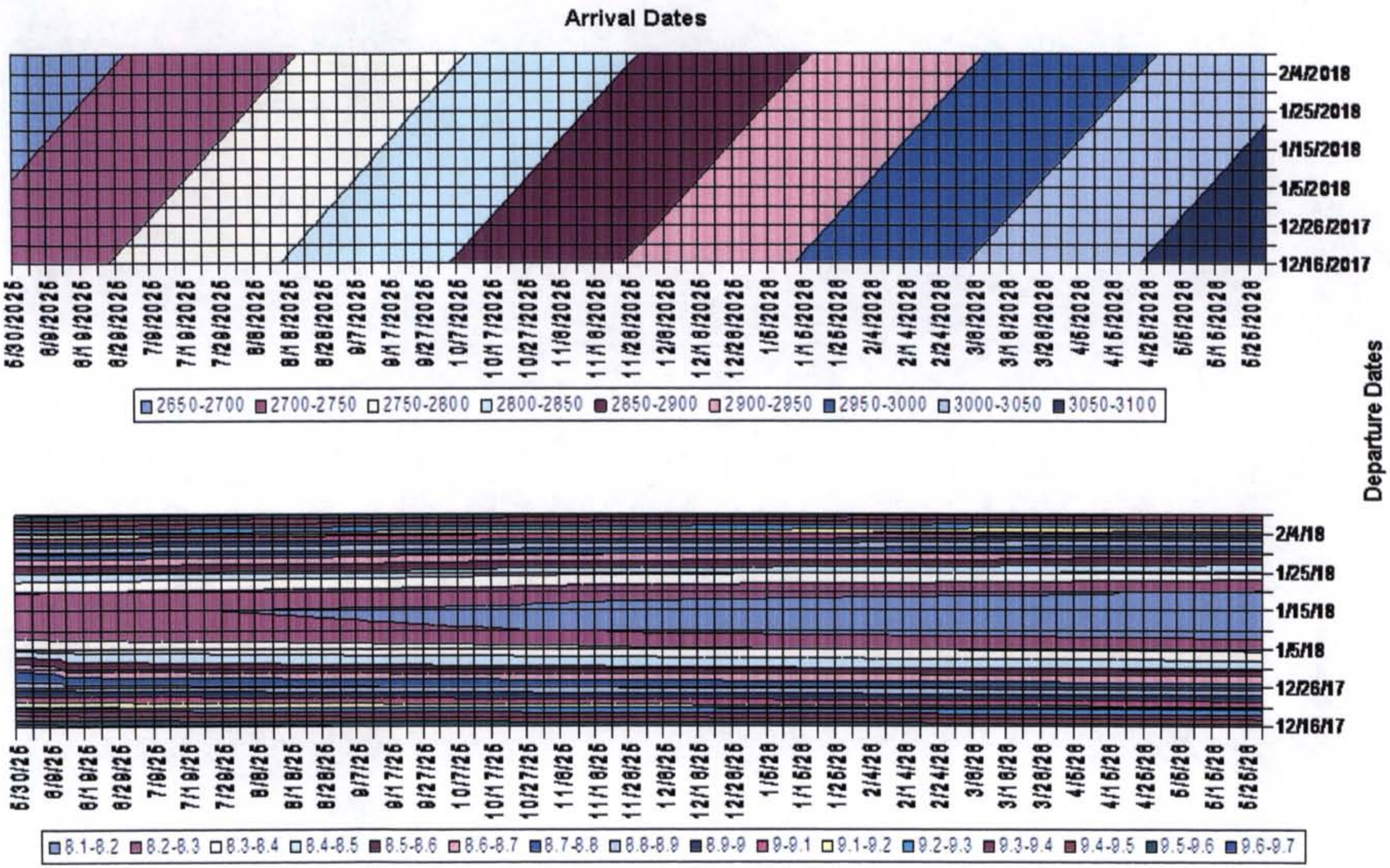

Figure 19: Contour Plots of Total ΔV and the Corresponding Time of Flight, Earth-Jupiter-Saturn Mission

Table 2: Earth-Jupiter-Saturn Mission with Minimum ΔV

Earth-Jupiter-Saturn					
	Date	ΔV (km/s)	Altitude (km)	Ra (km)	Rp (km)
Departure	1/14/2018	6.274		400	400
Swing-by	3/10/2020	0	3175403.45		
Arrival	12/31/2025	1.84		1280000	298000
Total	2908 days	8.114			

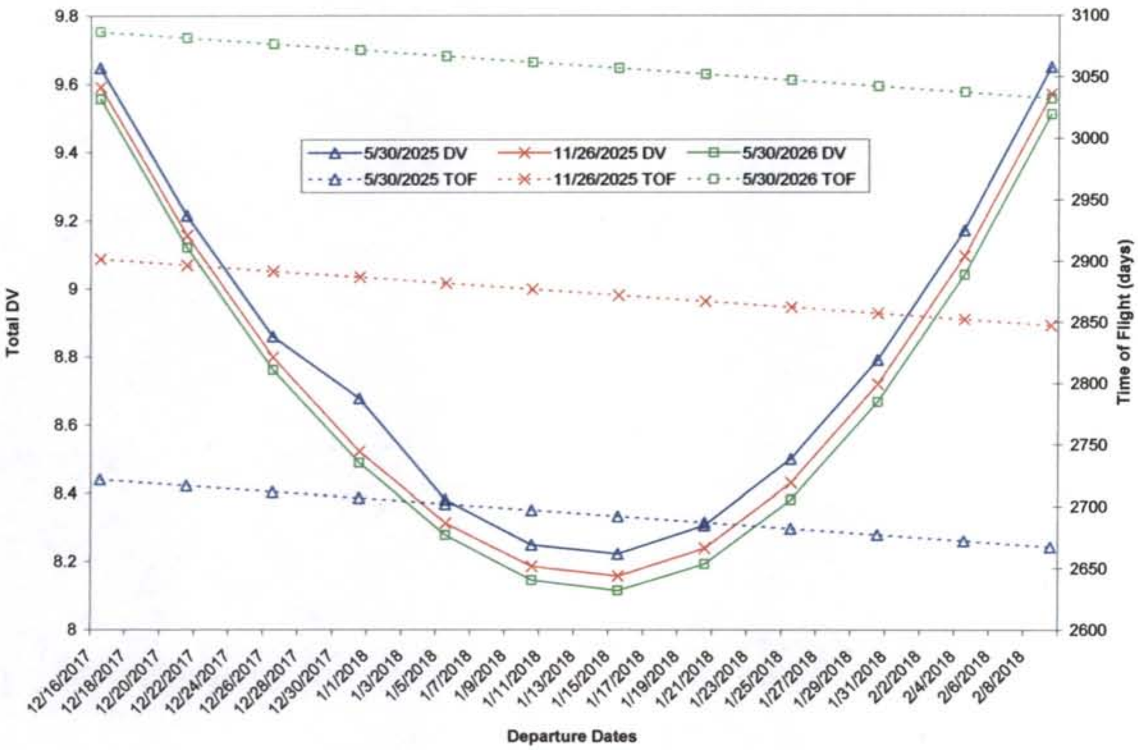

Figure 20: Total ΔV and the Corresponding Time of Flight vs. Departure Date for Three Possible Arrival Dates, Earth-Jupiter-Saturn Mission

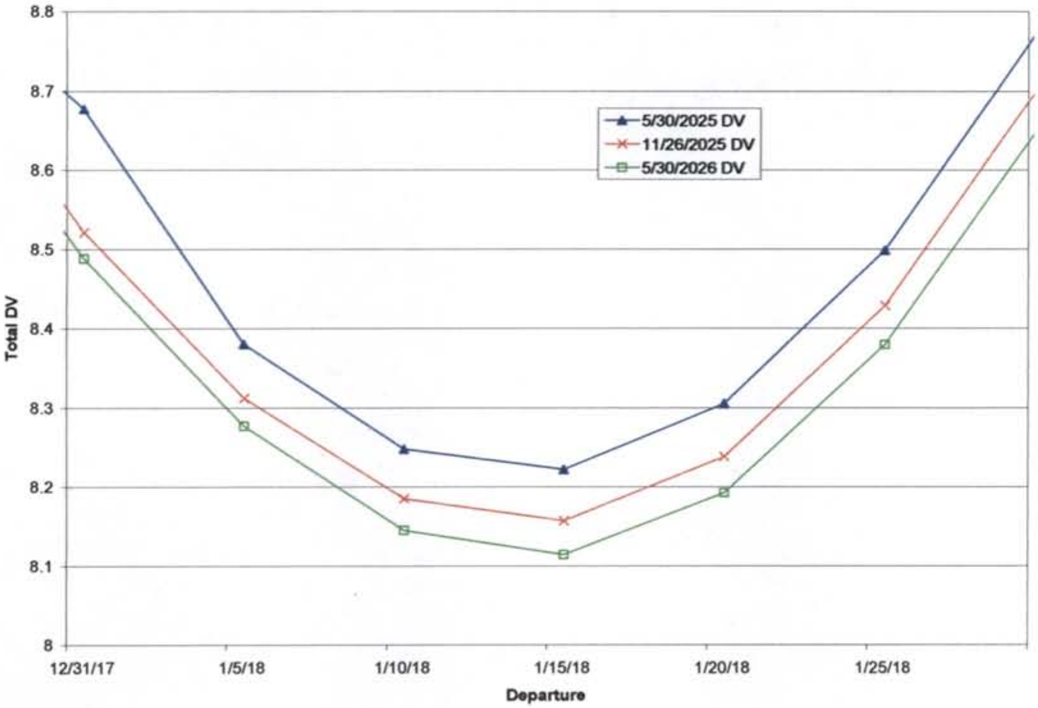

Figure 21: Departure Window for Earth-Jupiter-Saturn Mission

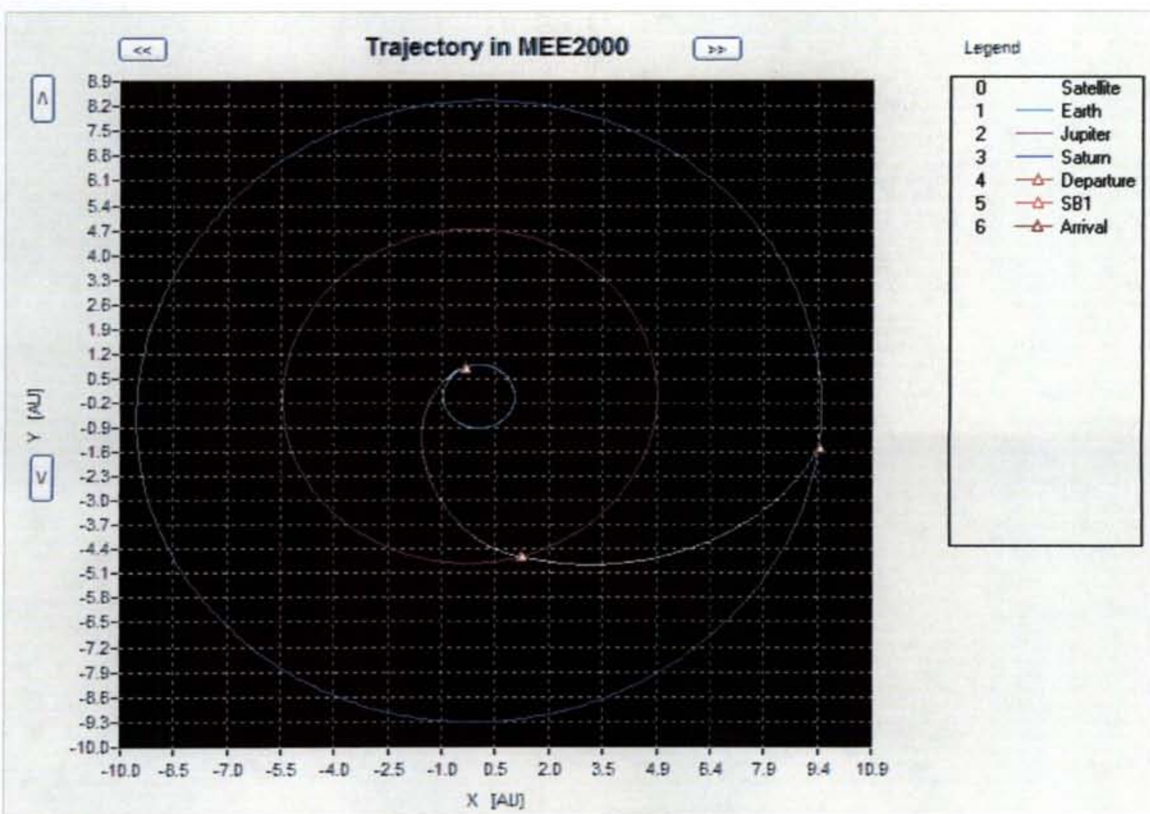

Figure 22: Plot of Earth-Jupiter-Saturn Flight Path

The second mission selected for more in-depth analysis was similar to the Cassini mission. This mission incorporated an Earth-Venus-Venus-Earth-Jupiter-Saturn trajectory. Also like Cassini, a deep space maneuver had to be added to enable the second swing-by of Venus. Since the planets would not align as favorably as during the Cassini mission, a powered flyby was required at the second passage of Venus. This allowed the minimum offset distance requirements for the planetary flybys to be met. To find a minimum total ΔV launch opportunity, a contour plot was created and is shown in Figure 23. From this plot, the departure window with a total ΔV of less than 8.5 km/s was found to lie between 8/30/2018 and 9/01/2018. Figure 24 shows the plots of the total ΔV and the corresponding time of flight for three different arrival dates. This mission had very small launch window of opportunity, unlike the Earth-Jupiter-Saturn mission. If the departure date was to be set 6 days early, the minimum total ΔV would an increase by .95 km/s. One advantage of such a mission design was the time of flight of 2500 days, which was shorter than any of the Earth-Jupiter-Saturn missions with a similar ΔV. A summary of the mission is found in Table 2, and a plot of this corresponding trajectory is given in Figure 25.

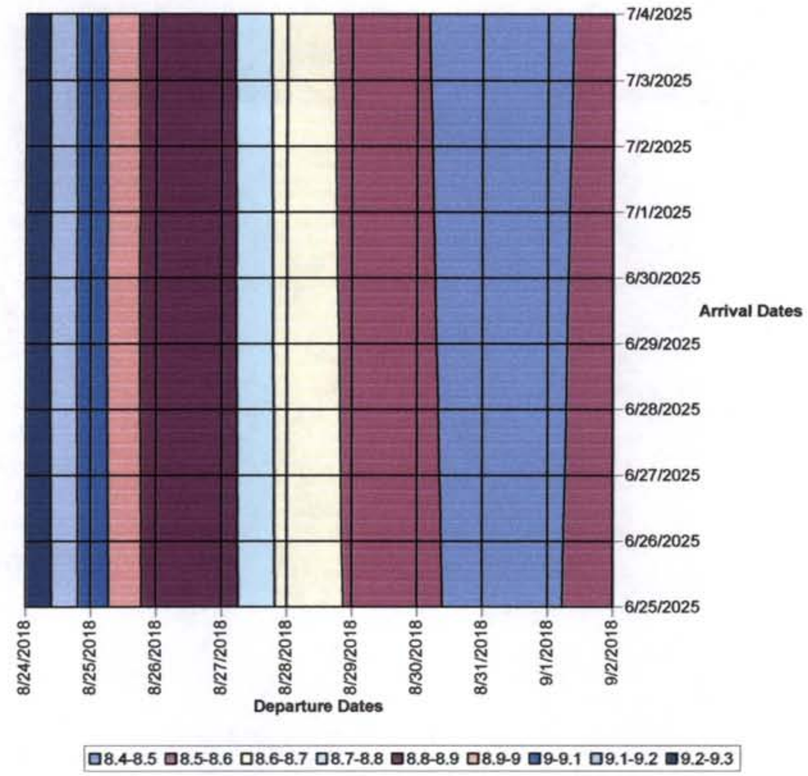

Figure 23: Contour Plot for EVVEJS Mission

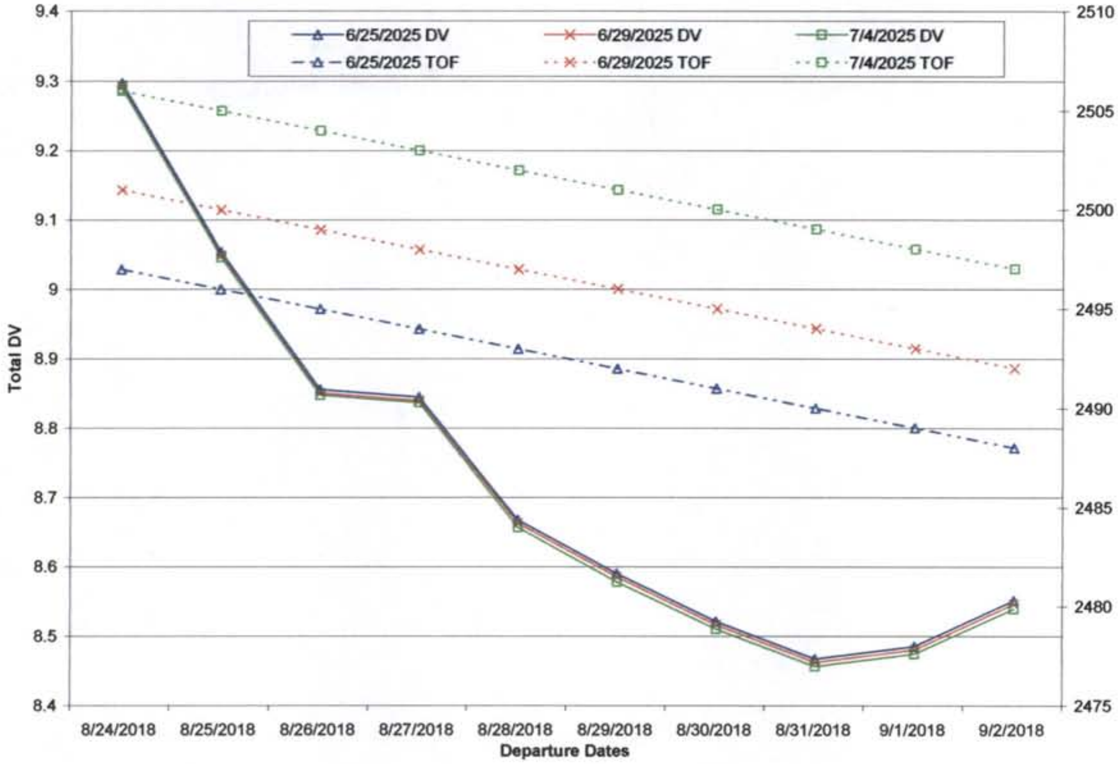

Figure 24: Total ΔV and Corresponding Time of Flights vs. Departure Date for Three Possible Arrival Dates, EVVEJS Mission

Table 3: EVVEJS Mission Details

Earth-Venus-Venus-Earth-Jupiter-Saturn		
	Event Date	ΔV (km/s)
Departure from Earth	8/31/2018	3.535
Swing-by Venus	2/8/2019	0
DSM	9/18/2019	0.655
Swing-by Venus	3/29/2020	1.68
Swing-by Earth	5/20/2020	0
Swing-by Jupiter	7/29/2021	0
Arrival at Saturn	7/4/2025	2.58
Total	2499 Days	8.45

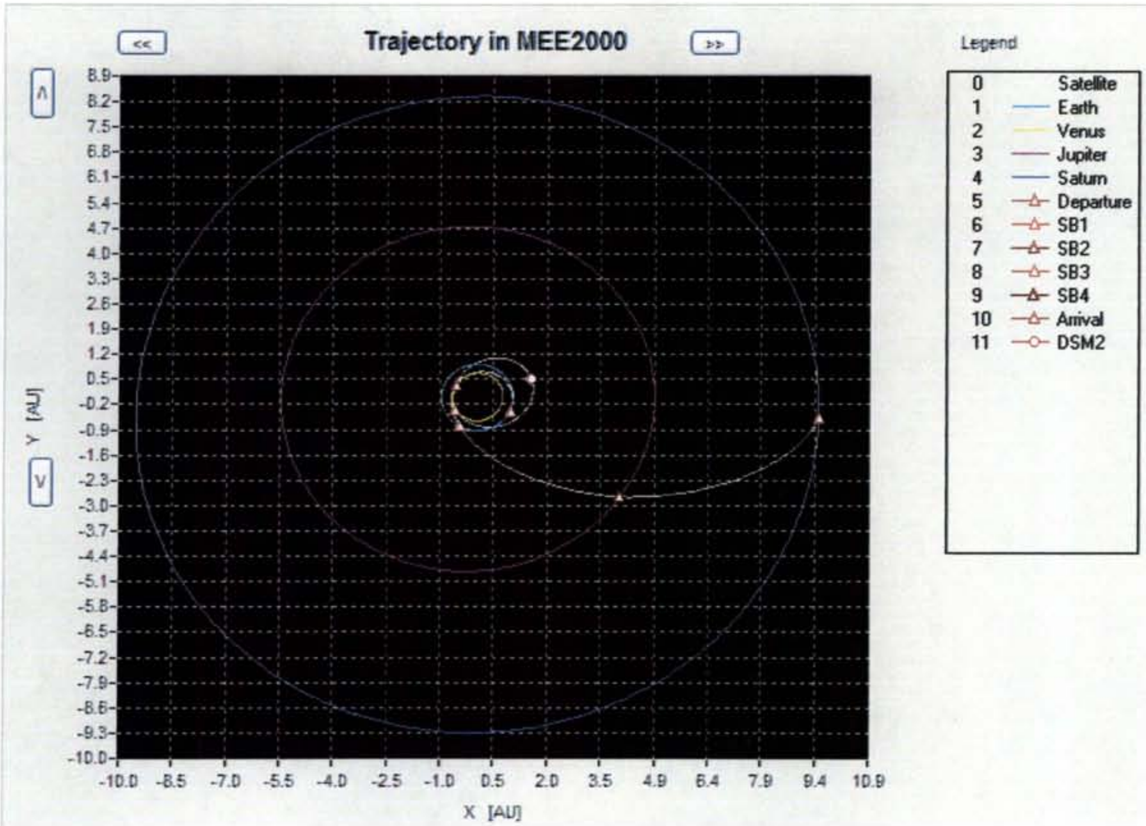

Figure 25: Plot of EVVEJS Mission Flight Path

Low Thrust Trajectories

The search for new propulsion technology has led to new systems that produce tenths of newtons of thrust instead of the traditional meganewtons. This may sound illogical, but low thrust technology is promising to power spacecraft much more efficiently than its high thrust counterparts. Low thrust engines, such as electrostatic ion thrusters, produce these very small forces over a period of weeks or months. This can accelerate crafts to the high velocities required for interplanetary travel.

NASA utilized low thrust propulsion systems in their Deep Space 1 space probe. DS-1 was used mainly as a testing platform for new technologies, such as low thrust propulsion. DS-1 utilized a NSTAR electrostatic ion thruster that had a specific impulse of 1000 to 3000 seconds and produced a maximum of 92 millinewtons of thrust over a total engine fire time of 678 days.

The Jet Propulsion Laboratory has developed a suite of programs to test and optimize these low thrust trajectories. Their Low Thrust Trajectory Tool (LTTT) contains programs that are designed to assist engineers with analyzing space travel missions that utilize low thrust propulsion systems.

To obtain the low thrust solution to the mission design CHEBYTOP was acquired from JPL and used. CHEBYTOP stands for Chebychev Trajectory Optimization Program. It was created by the Boeing Company for NASA's Ames Research Center in the late 1960s. It is currently on its tenth version.

CHEBYTOP employs chebychev polynomials to numerically solve the equations of motion without lengthy numerical integration; as such, this program was designed to be highly computationally efficient. Its strengths are that it is a quick program that requires few inputs and no initial guesses to model a mission. The drawbacks are that it is a very low fidelity model of the mission. It was intended mainly to be used for preliminary mission feasibility studies.

For our purposes, CHEBYTOP was the only low thrust program able to be acquired. MALTO, Mission Analysis Low Thrust Optimization, is JPL's premier high fidelity low thrust optimization program. An attempt was made to obtain this free program but it met resistance with the bureaucracy at the University of Tennessee.

CHEBYTOP was used to find a low thrust trajectory from Earth to Saturn. To start, several inputs were inserted into the Excel spreadsheet. These inputs included the initial orbital altitude, arrival orbital altitude, initial and arrival planets, initial mass, specific impulse of the engine, and the specific mass of the fuel.

After this, the built-in optimization program was used to optimize the departure date of the craft based on a range of dates, range of flight times, and the previous inputs. This optimization yielded an array of results that contained an optimum date for each flight time in the range of flight times. From this a flight time was chosen, and then the date for departure was input into the main CHEBYTOP program. The program then produced an output data sheet and a plot showing a visual representation of the trajectory. Appendix D contains the data sheet, and figure 26 provides the plot of the trajectory output by CHEBYTOP.

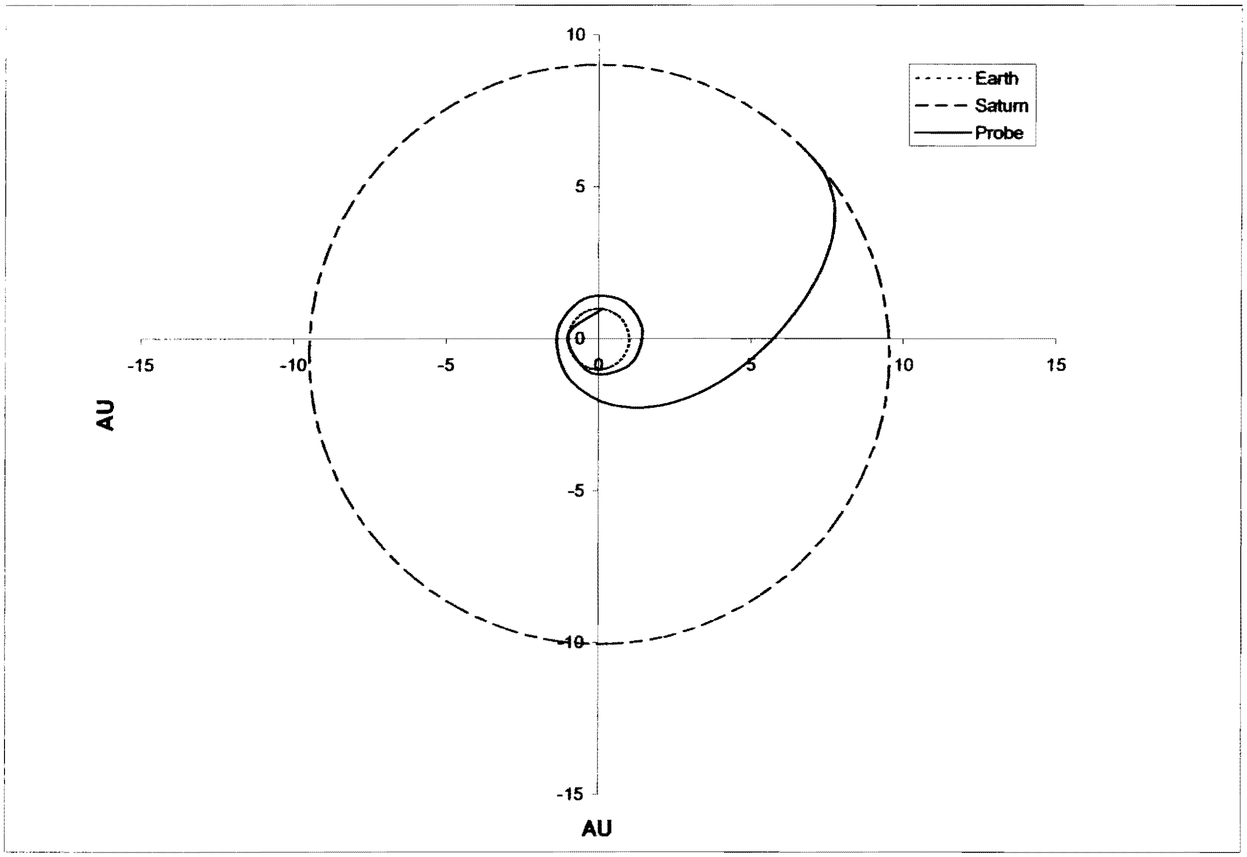

Figure 26: Plot Output from CHEBYTOP Sample Run

For Appendix D and Figure 26, the craft was sent from a 400 km low earth orbit to an orbit at the same orbital radius as Enceladus. The assumed specific impulse for the engine was 7,000 seconds. This value was most questionable, because published papers cite specific impulses ranging from 3,000 to over 75,000 seconds. It was to depart from Earth on December

14, 2021 and arrive orbiting Saturn on January 26, 2029. The initial mass of the craft was 1200 kg, and the mass of propellant used was 532.3 kg. The program was run for a range of specific impulses and specific masses in order to find a variety of optimum departure times and trip times.

Unfortunately, since CHEBYTOP is intended to be a quick first look at the mission design and no high fidelity modeling software was able to be acquired, the low thrust solution was cut in favor of the high thrust mission design made possible by the multitude of programs that optimize high thrust mission trajectories. The data gathered from CHEBYTOP gave a great place to start for future low thrust mission design, but it was not adequate for the design project at hand.

CHEBYTOP did what it was intended to do very well: it created a place to start for future work in low thrust mission design, and it also showed that this mission design was feasible. For future work in low thrust design, based on the computer programming and numerical analysis knowledge of average seniors and the amount of time with which to complete this design, the most reasonable answer to low thrust mission design would to be to obtain one of JPL's other programs in the LTTT suite. The LTTT suite contains a variety of programs that produce a high fidelity model of the intended mission design. MALTO is only the most recent software created; others include VARITOP, SEPTOP, and SEPSPOT.

Saturn System Orbits

While the majority of the analysis focused on the planning of the interplanetary trajectory, efforts were also made to determine the orbit for the spacecraft upon arrival in the Saturn system. Any trajectory delivering the spacecraft to the Saturn system would require some kind of insertion maneuver to slow it down. The Cassini mission, for instance, executed a 96-

minute burn in its Saturn insertion maneuver, achieving a velocity change of 0.626 km/s. During the burn maneuver, the spacecraft encountered its minimum distance to Saturn's center, a distance of 80,230 km. A Saturn orbit was achieved 78 minutes into the burn.

The simplest case in the planning of an Enceladus mission would be a single flyby of the satellite, with the spacecraft never actually entering into an orbit within the Saturn system. This idea was rejected, though, as inconsistent with the objective to further explore Enceladus. Next, several cases for an orbital insertion, which would allow for more prolonged encounters with Enceladus, were considered in greater detail.

One possibility was for the spacecraft to orbit Enceladus itself. This would enable the craft to keep the surface in constant view. This would also allow for the possibility of flying through the icy plume emitted near the south pole. Calculations were made to determine the feasibility of this option. The equation for the gravitational parameter of Enceladus, shown in Equation [1], gave a value of 4.7 km^3/s^2. This value was incredibly smaller than Saturn's gravitational parameter of $3.794 x 10^7$ km^3/s^2. Enceladus would thus provide little gravitational force to keep an object in orbit about itself.

$$\mu_{Enceladus} = GM_{Enceladus} \qquad \textbf{Equation 1}$$

$$\text{where } G = 6.673x10^{-20}\,\frac{km^3}{kg \cdot s^2}, \text{ and } M_{Enceladus} = 7x10^{19}\,kg$$

The radius of the sphere of influence, found by Equation [2], yielded a result of 410 km. So to orbit Enceladus within its sphere of influence, the spacecraft would have to slow to speeds less than 1 km/s, as shown in Equation [3]. Moreover, significant station-keeping maneuvers would have to be made to keep the craft in orbit. Orbiting Enceladus was therefore determined to be highly impractical.

$$R_{S.O.I} = r\left(\frac{m_2}{m_1}\right)^{0.4}$$ **Equation 2**

where $m_2 = 7x10^{19}$ kg, $m_1 = 5.6851x10^{26}$ kg, and r=238,020 km

$$V = \sqrt{\frac{2\mu_{Enceladus}}{r_1} - \frac{\mu_{Enceladus}}{a_{orbit}}}$$ **Equation 3**

Another option would be any elliptical orbit about Saturn which would offer an opportunity to encounter Enceladus. A few subsets of this type of orbit were deemed most useful and considered in greater detail. One such orbit would be an elliptical orbit with a periapse at or near Enceladus' distance from Saturn (238,000 km). The apoapse could be lowered or raised, depending on timing for intercepting Enceladus. The following special cases of this type of orbit were found to provide unique opportunities for Enceladus observation. If the apoapse were reduced to the same distance as Enceladus' orbital distance, the spacecraft could fly in a circular orbit, either just ahead of or behind Enceladus, and maintain a constant view of the surface. However, only one side of the moon would be visible at any time, unless other burns were executed to change orbits. Similarly, the spacecraft could orbit in a circular orbit either just inside or just beyond Enceladus' orbit. This would allow a significant amount of time to be spent near Enceladus, while also allowing for a slowly changing view of its surface.

Figure 27 shows the required insertion burn, in terms of ΔV consumption, for varying apoapse radii and values of inbound V_∞, with the periapse set at 238,000 km. Note that the required change in velocity increased dramatically as the apoapse approached this same distance. Thus, circular (or nearly circular) orbits near Enceladus would require much more extensive fuel burns than orbits with higher apoapses. In light of this, it was determined that the most practical option would be a more eccentric orbit with its periapse at 238,000 km and its apoapse much higher. In particular, it would be useful to position the apoapse near Titan's orbit, at 1.22×10^6 km from Saturn's center. This would allow not only for more scientific observation of Titan, but also for the possibility of using Titan to perform energy-saving maneuvers.

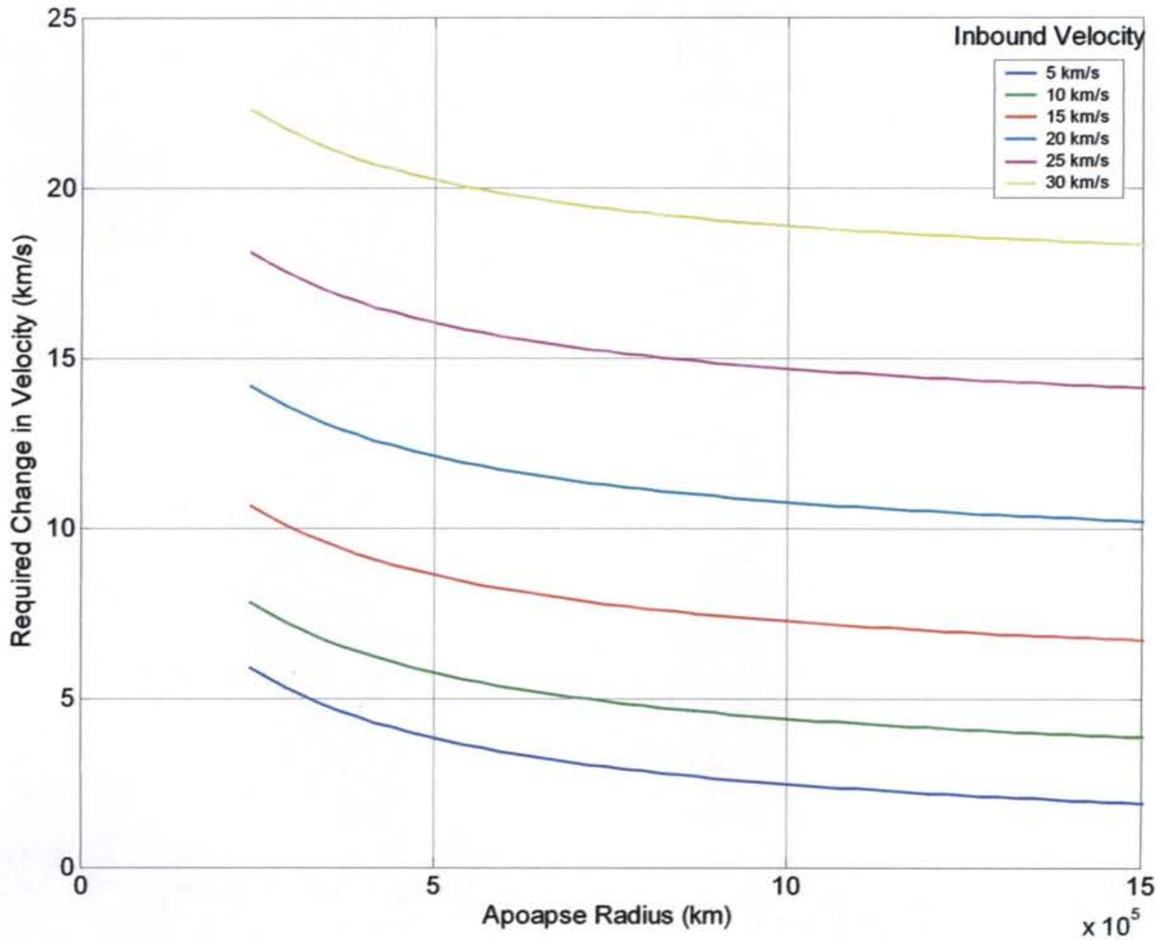

Figure 27: Insertion Delta-V versus Apoapse Radius (Periapse Radius at Enceladus' Orbit) for Various Inbound Velocities

Such options for insertion would include aerocapture and aerobraking. The aerocapture maneuver uses the atmosphere of the target body to create friction and slow a vehicle down. Titan has a planet-like atmosphere that could be used in this way. Aerobraking employs similar techniques but allows for multiple passes through the atmosphere, lowering the apoapse of the ellipse after each pass. These options should be further explored to determine if their use could lower the overall energy consumption and cost of the mission.

Additionally, more in-depth analysis would be needed to determine when and how the spacecraft would encounter Enceladus. Several flybys would be executed and would need to be planned carefully in order for the spacecraft to encounter Enceladus every time. Ephemeris data for Enceladus (and Titan) and a gravitational model incorporating Saturn's oblateness would need to be obtained to correctly determine encounter opportunities, and these would have to be linked with the Saturn orbital insertion maneuver.

Sample Collection

A sample return capsule (SRC) could be attached to the main spacecraft and would essentially hibernate utilizing a timing mechanism until the desired orbit could be attained. The probe would be activated about every 6 months to verify and perform a periodic maintenance of the instruments. Once the spacecraft had successfully maneuvered into a Saturn system orbit, the SRC would re-establish communication and begin collecting particles. Samples would be collected by extending a collection tray on a boom into the free stream debris emanating from Enceladus. The particles would be collected by impacting them on the collection tray, composed of an exotic glass called aerogel, shown in Figure 28. Aerogel is composed of silicon dioxide, can withstand the space environment, and it is extremely lightweight.[1] Once the collection

[1] JPL

process was complete, the collection tray would be retracted back into the capsule and the capsule would prepare for deployment from the spacecraft.

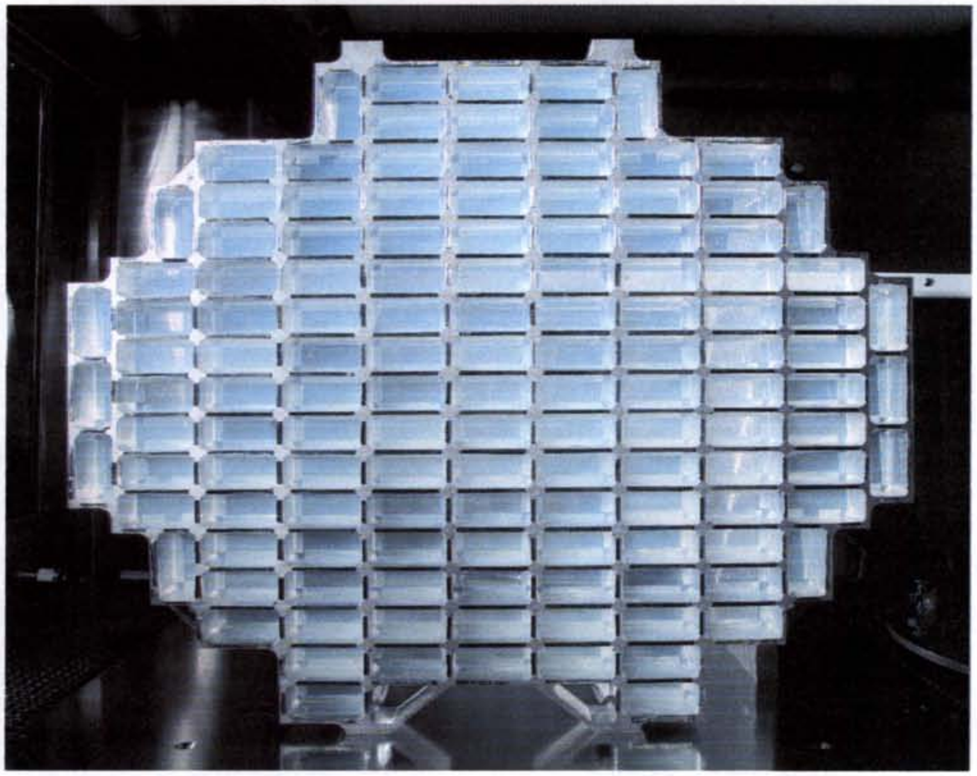

Figure 28: Aerogel Collection Tray Prior to Boom Insertion[1]

Possible Return Mission

The collection and return of samples from Enceladus' surroundings to Earth would generate significant understanding of the scientific phenomena of Enceladus' surface eruptions. In order to accomplish such a mission, a capsule containing the collected particles would have to safely transit an Earth-bound trajectory. Two design trajectories were investigated: multiple Titan flybys and a direct low thrust trajectory. A return time of 6-7 years has been estimated.

[1] JPL, "Aerogel Mystifying Blue Smoke"

Titan Flyby Departure

A question about Saturn departure was presented late in our examination of the possible mission to Enceladus. Can a probe use gravity assisted flybys of Titan, Saturn's largest satellite, to reach or surpass the velocity required to escape the Saturn system? This problem was deceivingly complex.

In order to begin the investigation of this problem, it had to be more clearly defined. It was assumed that the probe would start in an elliptical orbit whose periapse is 238,020 km from Saturn's center, or at Enceladus' orbital radius, and whose apoapse would be 1.222 million km from Saturn's center, or at Titan's orbital radius. Also this flyby departure needed to be shorter than the low-thrust departure time, which was found to be more than 700 days. Finally the flybys of Titan could be no less than 1000 km above the surface in order to remain outside the atmosphere.

What makes this problem complicated is that each flyby of Titan has to set up the next flyby within a reasonable amount of time. This reasonable amount of time was assumed to be 20 Titan revolutions, or approximately 320 days.

A method for solving this problem was developed using MATLAB. The method started off by calculating the resulting orbit from a flyby at the minimum altitude of 1000 km above Titan. This orbit would be the theoretical maximum orbit that could be achieved from a flyby and served as the upper limit for calculations. The lower limit was the current orbit that the probe was in. Somewhere between the maximum flyby and no flyby at all was a flyby that allowed for a subsequent rendezvous within the allotted 320 day limit.

After the limits were set a method of using the ratio of the orbital periods of the probe to that of Titan was created. The maximum and minimum ratios were set by the previous data. To find the optimum period ratio, a MATLAB loop was created to find the closest fraction to the maximum period ratio whose denominator, which represented the number of Titan orbits, remained below 20.

Once the optimum period ratio was calculated, it was multiplied by the orbital period of Titan to find the optimum period of the probe's orbit after the flyby. Kepler's equation (Equation 4) relates the period of any orbit to its semi-major axis.

$$a = \sqrt[3]{(\frac{T}{2\pi})^2 * \mu} \qquad \textbf{Equation 4}$$

For this reason, it was used to find the semi-major axis of the optimum orbit. The energy equation (Equation 5) could then be used to calculate the speed of the probe after flyby.

$$\frac{V^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a} \qquad \textbf{Equation 5}$$

The velocity could not yet be calculated because it requires a direction, which is the flight path angle in this case.

The flight path angle can be calculated using (Equation 6). This equation was derived from various other equations to relate flight path angle to the speed calculated previously, Titan's orbital speed, and the relative speed of the probe to Titan, V_∞.

$$\phi = a\cos(\frac{V_{Titan}^2 + V^2 - V_\infty^2}{2 * V_{Titan} * V}) \qquad \textbf{Equation 6}$$

With the velocity of the probe found, the flyby of Titan can be calculated backwards in order to find the altitude at which the probe would have to flyby Titan. This process begins with finding the turn angle, δ, achieved from the flyby. This is done by subtracting the inbound Titan approach angle from the outbound exit angle. Both of these can be found by using vector addition and subtraction to find the inbound and outbound velocities relative to Titan. The magnitude of both of these velocities is V_∞ and the angles are the inbound approach and outbound exit angles.

This turn angle will be used to calculate the eccentricity of hyperbolic flyby trajectory by using (Equation 7).

$$e = \frac{1}{\sin(\frac{\delta}{2})} \qquad \textbf{Equation 7}$$

Lastly, the eccentricity of the flyby can be used to calculate the altitude at which the craft will encounter Titan. This can be done by using (Equation 8).

$$alt = \frac{\mu * (e-1)}{V_\infty^2} \qquad \textbf{Equation 8}$$

This process was repeated and the craft continued flybys until the craft reached a total turn angle of 180 degrees. At this point the craft would have gained the maximum amount of velocity from the flybys that it could.

Table 3 shows the various parameters of each flyby and the Saturn system velocity that results from each flyby. After three flybys, the program runs into a problem that has not been resolved as of yet. The four flybys show a significant increase in the system velocity. All of this data is compiled into Table 3, along with the data for the maximum change in velocity that can be achieved from the flybys.

Table 3: Titan Flyby Values for each Flyby and Maximum

Flyby Number	Altitude of Flyby (km)	Turn Angle (degrees)	System Velocity (km/s)	Change in Velocity from initial (km/s)	Time at encounter (days)
Initial			3.182		
1	1044	35.32	3.876	0.695	3.68
2	1016	35.51	5.293	2.111	243
3	1034	35.39	6.649	3.467	259
4	1066	35.16	7.588	4.406	514
Final			7.963	4.781	

Figure 29 contains a plot of the initial orbit, the orbits of Enceladus and Titan, and the first three flyby orbits. This graphical representation is useful because it shows trends such as the fact that the semi-major axis of each orbit is rotating after each flyby. The maximum change in velocity of the probe will occur when the semi-major axis has rotated by 180 degrees.

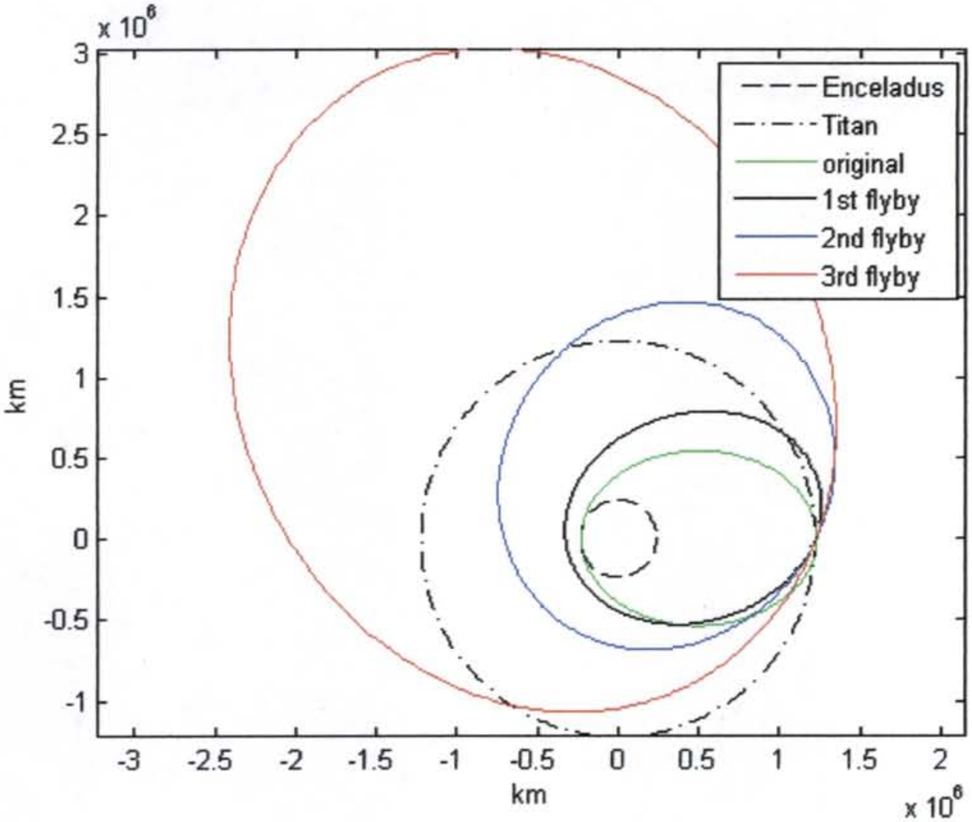

Figure 29: Graphical Representation of First 3 Flybys of Titan

Figure 30 contains the same data from Figure 29 but it also includes the fourth flyby. The fourth orbit was very large so it was left off of Figure 29 in order to view the first three flybys more closely.

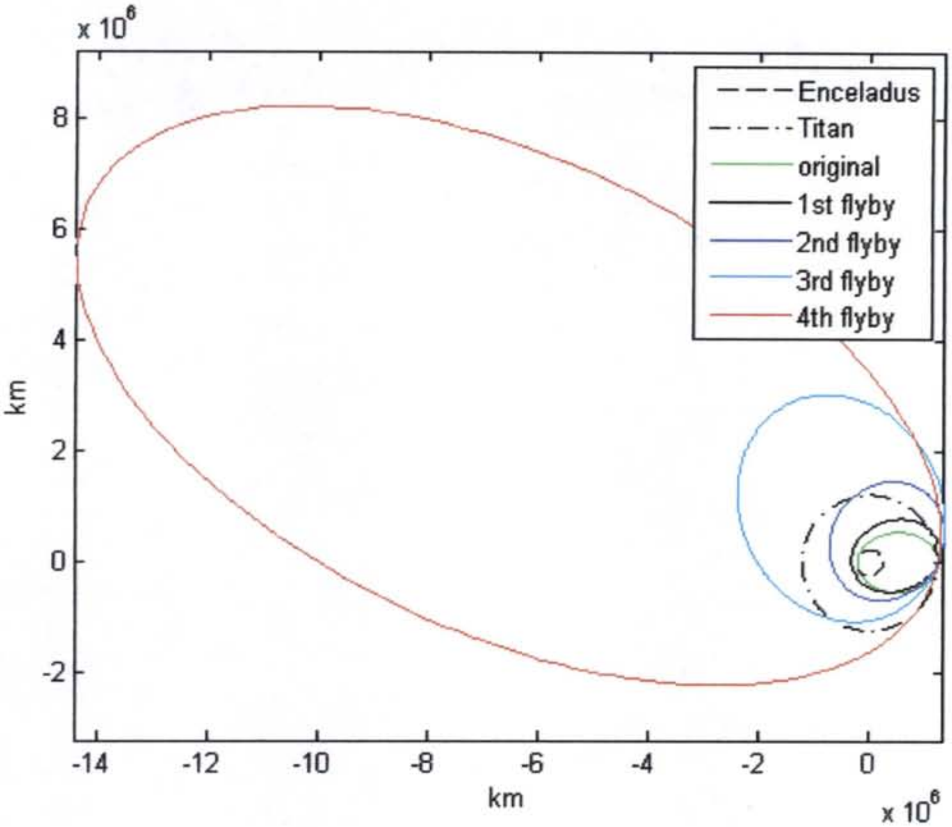

Figure 30: Graphical Representation of First Four Flybys of Titan

This project was started late in the semester, so it was not given the attention that it deserves. From the stand point of finding out if it would be feasible to used flybys of Titan to exit the Saturn system, this aspect of the project was a success because it was found that a probe can execute several flybys of Titan in order to gain the energy required. More research into the specifics of this problem should be conducted. This research should be initially focused on the fourth and fifth flybys to eject the probe from the system. Also more work should be completed to study the trajectory of the probe once it has left the system. Even though the probe would be able to escape Saturn, it would still need an additional thrust to reach even Jupiter in an effort to make it to Earth.

Low-Thrust Return

A direct low thrust trajectory was also analyzed for the sample return capsule using ChebyTop. Figure 31 shows a graphical representation of the low thrust departure trajectory from Saturn to Earth. The expected departure date provided through ChebyTop was June 1, 2030 with an expected Earth arrival date of June 9, 2036.

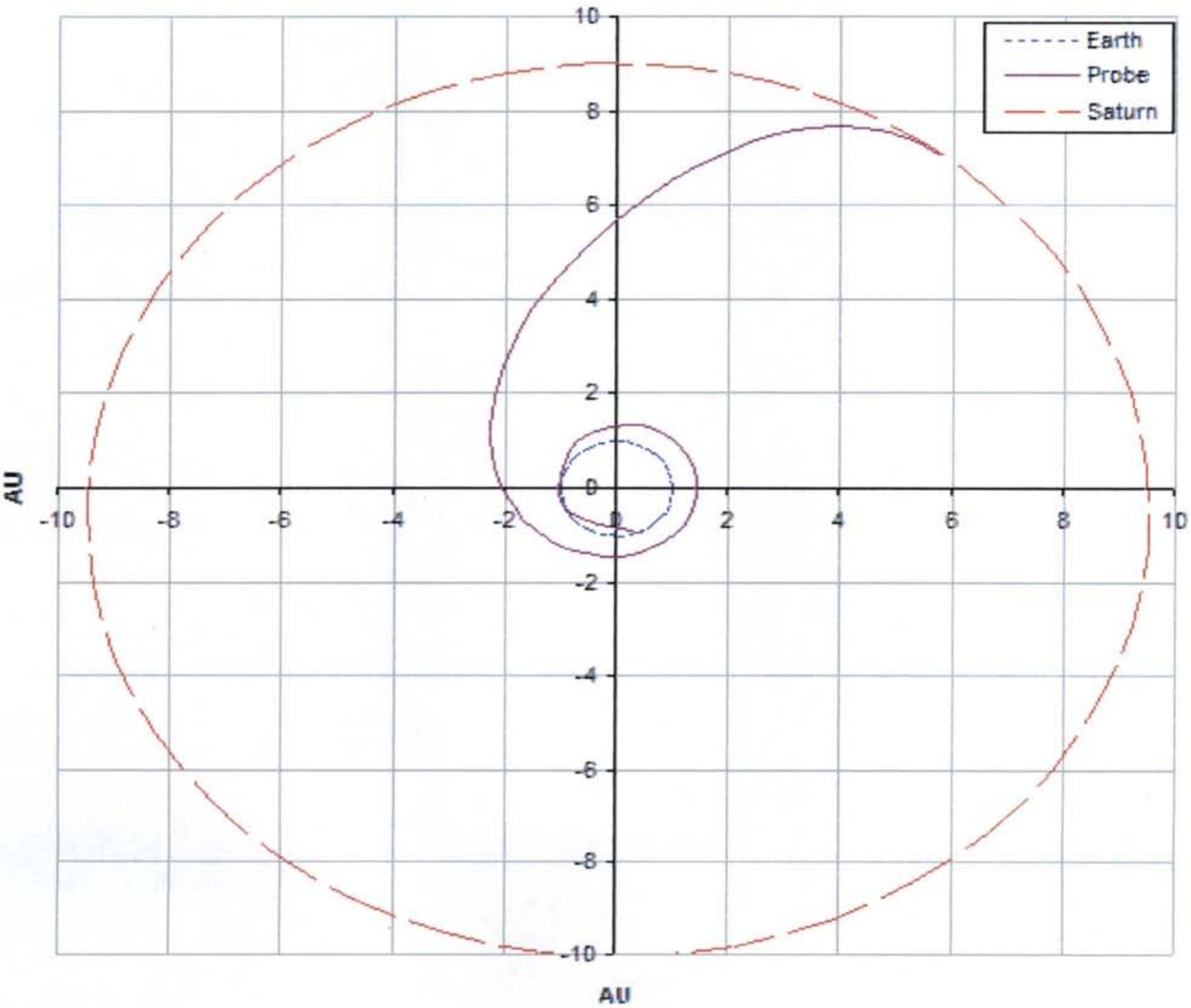

Figure 31: Low Thrust Sample Return Capsule Trajectory using ChebyTop

Sample Return Capsule (SRC)

A return capsule would need to withstand the rigors of an intense Earth-bound trajectory, re-entry, decent, and landing. An accurate description of the Earth reentry vehicle aerodynamics would be necessary. Wake-flow behavior is one of the most important issues that must be considered in the design of planetary entry vehicles. The wake-flow performance influences the payload size, placement, and shielding requirements.[1] It is known that blunt body vehicles are best suited for planetary re-entry, due to their increased drag effect, thus decreasing the heating effect by means of a shockwave. This shockwave absorbs most of the vehicle's kinetic energy that is transformed into heat as it enters the atmosphere; however, the shockwave does not absorb all of the heat generated. Therefore, other protective measures would be needed, such as an ablative heat shield. The basic geometry of a proposed Sample Return Capsule, very similar to the Stardust SRC, is shown in Figure 32.

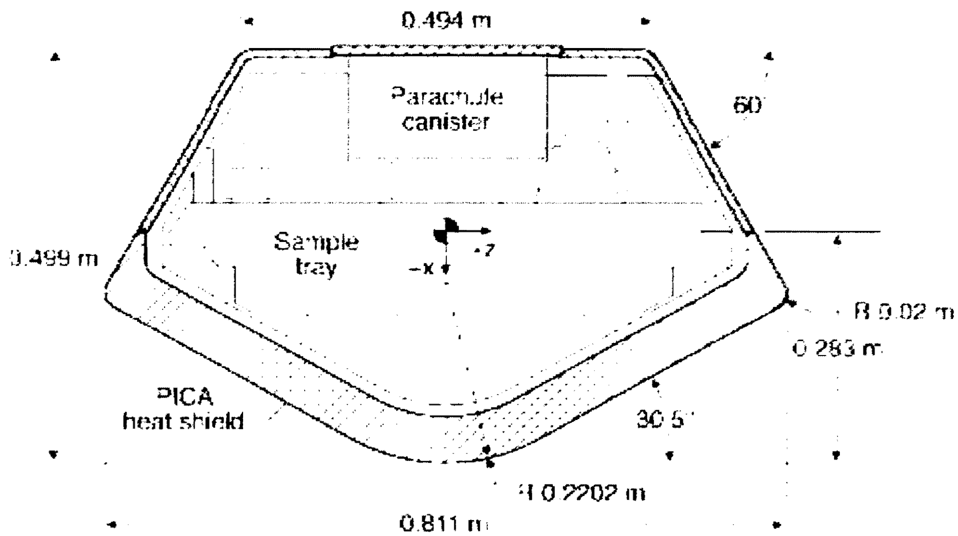

Figure 32: Sample Return Capsule Geometry[2]

[1] Hollis and Perkins
[2] Desai, et al.

The forebody of the SRC is proposed to have a 60-degree half-angle sphere-cone with an overall diameter of approximately 0.8 m and an approximate dry weight of 45.8 kg (~100 lb) without the power source.[1] The afterbody shape would have a 30 degree truncated cone. The entry velocity of the Stardust capsule was 12.6 km/s, the highest Earth entry speed yet.[2] Therefore, there is need for a forebody heat shield constructed of phenolic-impregnated carbon ablator (PICA), the same composite that was used for the Stardust SRC, due to its lightweight properties and resistance to the intense heat effects endured during re-entry. Ablative shape changes on the vehicle have been taken into account and are shown as the PICA heat shield in Figure 32. However, this figure does not depict the location of the power supply needed for the Earth-bound journey or the communication subsystem.

Options for a power supply are limited due to the distance between the probe and Earth. The Radioisotope Thermoelectric Generator (RTG) has a proven track record for expeditions of such distances and is known to exceed its life expectancy. Therefore, the RTG was chosen as the power supply for the probe. The RTG uses heat from a power source, like plutonium, to generate direct current electricity with a maximum electrical output of 300 W.[3] Power supply weights were approximated at 55.5 kg (122 lb). A communication subsystem, similar to that used in the Cassini-Huygens mission, is also proposed for the probe. Cassini was equipped with one high gain antenna and one low gain antenna. The high gain antenna was the primary source of communication, while the low gain antenna was used as a back up communicator in case of a power failure.

[1] Desai, et al.
[2] Willcockson
[3] JPL, "Expanding Frontier with Radioisotope Power Systems"

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Appendix A: Planetary Flyby

Gravitational slingshots, or gravity assisted flybys, were first used by the Mariner 10 space probe on February 5, 1974. It was on this day that the probe flew by Venus on its way to being the first spacecraft to study Mercury. Flybys are used by spacecraft in order to gain energy, and in turn velocity, in the system reference frame. They accomplish this by using the gravity from a celestial body to turn the craft's relative velocity vector in the local reference frame. In doing this the craft does not increase its velocity relative to the turning body, but it does increase its velocity in the system reference frame. Figure 1 shows a graphical explanation of how a flyby increases a craft's velocity in the system reference frame.

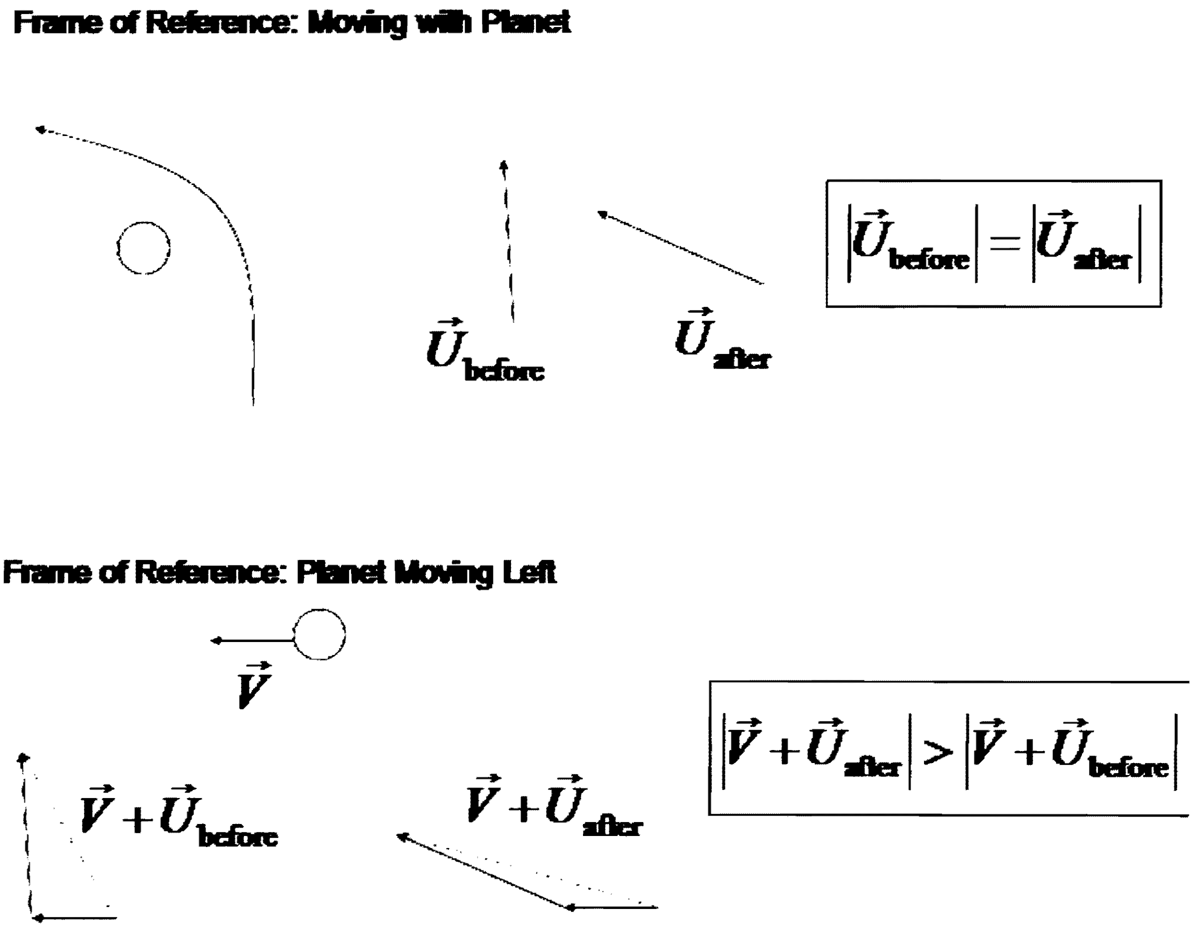

Figure 1: Graphical Explanation of Gravity Assist Flybys

The main reason for executing these maneuvers is to increase the velocity of the craft in the heliocentric reference frame without the expenditure of propellant. This is beneficial because the mass from the propellant that would be needed to increase the velocity of the craft could be used instead to store more scientific equipment or to simply decrease the overall mass of the craft.

Appendix B: ManE Outputs

Input File Name: Historic Cassini Trajectory Data

Case 1 (Converged) PERFORMANCE SUMMARY

Leg	Stay Time (days)	Depart		Arrive		Flight Time (days)
1		Earth	OCT 15, 1997, 12.0000 hours GMT Julian Date 50737.0000	Venus	APR 26, 1998, 21.2044 hours GMT Julian Date 50930.3835	193.3835
2	0.0000	Venus	APR 26, 1998, 21.2044 hours GMT Julian Date 50930.3835	Space Burn	DEC 3, 1998, 12.0000 hours GMT Julian Date 51151.0000	220.6165
3	0.0000	Space Burn	DEC 3, 1998, 12.0000 hours GMT Julian Date 51151.0000	Venus	JUN 26, 1999, 2.4363 hours GMT Julian Date 51355.6015	204.6015
4	0.0000	Venus	JUN 26, 1999, 2.4363 hours GMT Julian Date 51355.6015	Earth	AUG 18, 1999, 19.4974 hours GMT Julian Date 51409.3124	53.7109
5	0.0000	Earth	AUG 18, 1999, 19.4974 hours GMT Julian Date 51409.3124	Jupiter	JAN 3, 2001, 20.4795 hours GMT Julian Date 51913.3533	504.0409
6	0.0000	Jupiter	JAN 3, 2001, 20.4795 hours GMT Julian Date 51913.3533	Saturn	JUL 1, 2004, 12.0000 hours GMT Julian Date 53188.0000	1274.6467
					Total Duration	2451.0000

SPACECRAFT MASS SUMMARY (kg or t)

		Depart			Arrive								
Leg	Initial	Engine	Propell	Tankage	Engine	Propell	Tankage	Inert	Probes	AeroBrk	Drops	Samples	NetMass
1	1.000	0.000	0.364	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.636
2	0.636	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.636
3	0.636	0.000	0.033	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.603
4	0.603	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.603
5	0.603	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.603
6	0.603	0.000	0.000	0.000	0.000	0.426	0.000	0.000	0.000	0.000	0.000	0.000	0.178

DEPARTURE/ARRIVAL CONDITIONS (Planet Equator and Equinox of J2000)

	Depart						Arrive					
Leg	V Inf (km/s)	Decl (deg)	Rt Asc (deg)	Brn Tm (min)	Del V (km/s)	VLoss (m/s)	V Inf (km/s)	Decl (deg)	Rt Asc (deg)	Brn Tm (min)	Del V (km/s)	VLoss (m/s)
1	4.07239	-2.39	332.57	18.196	3.99312	0.000	6.02096	15.44	240.09	0.000	0.00000	0.000
2	6.02096	0.03	311.01	0.000	0.00000	0.000	18.89229	-3.93	289.00	0.000	0.00000	0.000
3	18.45626	-3.63	288.61	9.224	0.46429	0.000	9.32466	-1.45	328.45	0.000	0.00000	0.000

4	9.32466	12.50	286.97	0.000	0.00000	0.000	15.93850	-12.58	334.84	0.000	0.00000	0.000
5	15.93850	-4.41	353.82	0.000	0.00000	0.000	10.50872	-3.71	256.73	0.000	0.00000	0.000
6	10.50872	-3.77	267.70	0.000	0.00000	0.000	5.39367	15.94	39.39	35.293	10.80026	0.000

HELIOCENTRIC TRANSFER ORBIT ELEMENTS (Ecliptic and Equinox of J2000)

Leg	Semi-Axis (AU)	Eccentricity	Inclination (deg)	Asc Node (deg)	Arg Per (deg)	Departure True Anom (deg)	Arrival True Anom (deg)	Perihelion (AU)	Aphelion (AU)	Period (days)
1	0.8420379	0.200473436	1.25793	22.20242	199.13813	160.86791	56.34719	0.6732317	1.0108442	282.22529
2	1.1503013	0.367654317	3.11840	74.71831	202.92454	0.07206	178.14970	0.7273880	1.5732145	450.62557
3	1.1265137	0.396794364	3.41041	76.59841	201.40834	177.78884	321.27217	0.6795194	1.5735080	436.72009
4	1.6596144	0.564822281	1.01707	145.42484	101.90906	351.97326	78.09228	0.7222272	2.5970015	780.92478
5	4.1318537	0.789684652	0.69264	145.42421	132.79636	47.20561	151.20621	0.8689922	7.3947151	3067.72074
6	5.3142195	0.743448586	0.76892	130.35648	163.67162	135.39749	172.44274	1.3633705	9.2650684	4474.63540

SWINGBY SUMMARY (Planet Equator and Equinox of J2000)

Planet	Time (days)	Pass Dist (radii)	V Inf (km/s)	Bend Angle (deg)	Inclination (deg)	Asc Node (deg)	Arg Per (deg)	Peri ΔV (km/s)	SOI ΔV (km/s)
Venus	193.38	1.0500	6.0210	71.620	16.281	131.115	54.078	0.0000	0.0000
Venus	618.60	1.0500	9.3247	43.464	159.986	324.456	287.486	0.0000	0.0000
Earth	672.31	1.1391	15.9385	20.461	25.107	3.291	249.333	0.0000	0.0000
Jupiter	1176.35	152.1353	10.5087	10.952	3.765	356.964	175.261	0.0000	0.0000

PLANETOCENTRIC CAPTURE ORBIT ELEMENTS (Planet Equator and Equinox of J2000)

Leg	Semi-Axis (radii)	Eccentricity	Inclination (deg)	Asc Node1 (deg)	Arg Per1 (deg)	Asc Node2 (deg)	Arg Per2 (deg)	Periapse (radii)	Apoapse (radii)	Period (hours)
6	1.0000000	0.000000000	15.94444	309.38758	72.90690	309.38758	72.90690	1.0000000	1.0000000	4.19239

LAUNCH DELTA V ORIENTATION - PLANETOCENTRIC (Planet Equator and Equinox of J2000)

Altitude (km)	Inclination (deg)	Delta V (km/sec)	Declination (deg)	Rt Ascension (deg)	X Dot (km/sec)	Y Dot (km/sec)	Z Dot (km/sec)
0.00000	2.39351	3.99312	-1.46634	280.33777	0.71633	-3.92701	-0.10218

MISSION OPERATIONS

	Earth		Venus				Space Burn				Venus				Earth
	Dep	Helio	Arr		Dep	Helio	Arr		Dep	Helio	Arr		Dep	Helio	Arr
Times (days)															
Depart/Arrive	0.00		193.38		193.38		414.00		414.00		618.60		618.60		672.31
Flight/Stay		193.38		0.00		220.62		0.00		204.60		0.00		53.71	

Esc/Cap Orbits (radii)

Apoapse Distance	1.00		0.00		0.00		0.00		1.00		0.00		0.00		0.00
Periapse Distance	1.00		0.00		0.00		0.00		1.00		0.00		0.00		0.00
Spacecraft Distances (AU)															
Minimum Heliocentric		0.6732				0.7274				0.7248				0.7222	
Maximum Heliocentric		1.0108				1.5727				1.5735				1.0122	
Geocentric	0.0000		0.9151		0.9151		0.7919		0.7919		0.5904		0.5904		0.0000
Maneuvers															
Propulsion Type	Impls		None		None		None		Impls		None		None		None
Vinf (km/sec)	4.07		6.02		6.02		18.89		18.46		9.32		9.32		15.94
Eff Delta-V (km/sec)	3.99	0.00	0.00		0.00	0.00	0.00		0.46	0.00	0.00		0.00	0.00	0.00
Vel Losses (m/sec)	0.00		0.00		0.00		0.00		0.00		0.00		0.00		0.00
Propellant (kg or t)	0.36	0.00	0.00		0.00	0.00	0.00		0.03	0.00	0.00		0.00	0.00	0.00
Burn time (hr)	0.30		0.00		0.00		0.00		0.04		0.00		0.00		0.00
Thrust (lbs or klbs)	0.66		0.00		0.00		0.00		0.42		0.00		0.00		0.00
Spec Imp (sec)	900.0		0.0		0.0		0.0		900.0		0.0		0.0		0.0
Mass Changes (kg or t)															
Dry Stage Jettisoned	0.00		0.00		0.00		0.00		0.00		0.00		0.00		0.00
Probes Separated				0.00				0.00				0.00			0.00
AeroBrake Separated				0.00				0.00				0.00			0.00
Drop Mass Left				0.00				0.00				0.00			0.00
Sample Mass Added				0.00				0.00				0.00			0.00

MISSION OPERATIONS (Concluded)

	Earth			Jupiter			Saturn
	Dep	Helio	Arr		Dep	Helio	Arr
Times (days)							
Depart/Arrive	672.31		1176.35		1176.35		2451.00
Flight/Stay		504.04		0.00		1274.65	
Esc/Cap Orbits (radii)							
Apoapse Distance	0.00		0.00		0.00		1.00
Periapse Distance	0.00		0.00		0.00		1.00
Spacecraft Distances (AU)							
Minimum Heliocentric		0.6732				5.0502	
Maximum Heliocentric		1.0108				9.0376	
Geocentric	0.0000		4.2719		4.2719		10.0483
Maneuvers							
Propulsion Type	None		None		None		Impls
Vinf (km/sec)	15.94		10.51		10.51		5.39
Eff Delta-V (km/sec)	0.00	0.00	0.00		0.00	0.00	10.80
Vel Losses (m/sec)	0.00		0.00		0.00		0.00
Propellant (kg or t)	0.00	0.00	0.00		0.00	0.00	0.43
Burn time (hr)	0.00		0.00		0.00		0.59
Thrust (lbs or klbs)	0.00		0.00		0.00		0.40
Spec Imp (sec)	0.0		0.0		0.0		900.0
Mass Changes (kg or t)							
Dry Stage Jettisoned	0.00		0.00		0.00		0.00
Probes Separated				0.00			0.00
AeroBrake Separated				0.00			0.00
Drop Mass Left				0.00			0.00
Sample Mass Added				0.00			0.00

Input File Name: Optimum Solution Achieved

Case 1 (Converged)

PERFORMANCE SUMMARY

Leg	Stay Time (days)	Depart		Arrive		Flight Time (days)
1		Earth	JAN 14, 2018, 0.5617 hours GMT Julian Date 58132.5234	Jupiter	MAR 8, 2020, 8.9653 hours GMT Julian Date 58916.8736	784.3501
2	0.0000	Jupiter	MAR 8, 2020, 8.9653 hours GMT Julian Date 58916.8736	Saturn	MAR 31, 2026, 20.8198 hours GMT Julian Date 61131.3675	2214.4939
					Total Duration	2998.8441

SPACECRAFT MASS SUMMARY (kg or t)

		--------- Depart ---------			--------- Arrive ---------								
Leg	Initial	Engine	Propell	Tankage	Engine	Propell	Tankage	Inert	Probes	AeroBrk	Drops	Samples	NetMass
1	1.000	0.000	0.511	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.489
2	0.489	0.000	0.000	0.000	0.000	0.087	0.000	0.000	0.000	0.000	0.000	0.000	0.402

DEPARTURE/ARRIVAL CONDITIONS (Planet Equator and Equinox of J2000)

	---------------------- Depart -----------------------						---------------------- Arrive -----------------------					
Leg	V Inf (km/s)	Decl (deg)	Rt Asc (deg)	Brn Tm (min)	Del V (km/s)	VLoss (m/s)	V Inf (km/s)	Decl (deg)	Rt Asc (deg)	Brn Tm (min)	Del V (km/s)	VLoss (m/s)
1	8.81986	-9.62	201.80	25.539	6.31012	0.000	6.91043	4.62	89.08	0.000	0.00000	0.000
2	6.91043	-0.06	142.16	0.000	0.00000	0.000	2.77320	-22.05	300.39	8.920	1.73442	0.000

HELIOCENTRIC TRANSFER ORBIT ELEMENTS (Ecliptic and Equinox of J2000)

Leg	Semi-Axis (AU)	Eccentricity	Inclination (deg)	Asc Node (deg)	Arg Per (deg)	Departure True Anom (deg)	Arrival True Anom (deg)	Perihelion (AU)	Aphelion (AU)	Period (days)
1	3.2228397	0.694812757	0.09854	292.33465	180.97976	0.22094	168.01298	0.9835696	5.4621099	2113.27729
2	6.4650257	0.472869986	2.36707	100.87226	86.15863	94.29691	177.45219	3.4079091	9.5221423	6004.18145

SWINGBY SUMMARY (Planet Equator and Equinox of J2000)

Planet	Time (days)	Pass Dist (radii)	V Inf (km/s)	Bend Angle (deg)	Inclination (deg)	Asc Node (deg)	Arg Per (deg)	Peri ΔV (km/s)	SOI ΔV (km/s)
Jupiter	784.35	45.7287	6.9104	53.233	5.816	321.591	63.958	0.0000	0.0000

PLANETOCENTRIC CAPTURE ORBIT ELEMENTS (Planet Equator and Equinox of J2000)

Leg	Semi-Axis (radii)	Eccentricity	Inclination (deg)	Asc Node1 (deg)	Arg Per1 (deg)	Asc Node2 (deg)	Arg Per2 (deg)	Periapse (radii)	Apoapse (radii)	Period (hours)
2	12.1108848	0.673900000	22.04649	30.39052	252.54869	30.39052	252.54869	3.9493595	20.2724100	176.69548

LAUNCH DELTA V ORIENTATION - PLANETOCENTRIC (Planet Equator and Equinox of J2000)

Altitude (km)	Inclination (deg)	Delta V (km/sec)	Declination (deg)	Rt Ascension (deg)	X Dot (km/sec)	Y Dot (km/sec)	Z Dot (km/sec)
400.00000	9.61897	6.31012	-8.67430	175.97885	-6.22259	0.43743	-0.95168

MISSION OPERATIONS

	Earth			Jupiter			Saturn
	Dep	Helio	Arr		Dep	Helio	Arr
Times (days)							
Depart/Arrive	0.00		784.35		784.35		2998.84
Flight/Stay		784.35		0.00		2214.49	
Esc/Cap Orbits (radii)							
Apoapse Distance	1.06		0.00		0.00		20.27
Periapse Distance	1.06		0.00		0.00		3.95
Spacecraft Distances (AU)							
Minimum Heliocentric		0.9836				5.2038	
Maximum Heliocentric		5.2038				9.5137	
Geocentric	0.0000		5.6710		5.6710		10.5065
Maneuvers							
Propulsion Type	Impls		None		None		Impls
Vinf (km/sec)	8.82		6.91		6.91		2.77
Eff Delta-V (km/sec)	6.31	0.00	0.00		0.00	0.00	1.73
Vel Losses (m/sec)	0.00		0.00		0.00		0.00
Propellant (kg or t)	0.51	0.00	0.00		0.00	0.00	0.09
Burn time (hr)	0.43		0.00		0.00		0.15
Thrust (lbs or klbs)	0.66		0.00		0.00		0.32
Spec Imp (sec)	900.0		0.0		0.0		900.0
Mass Changes (kg or t)							
Dry Stage Jettisoned	0.00		0.00		0.00		0.00
Probes Separated				0.00			0.00
AeroBrake Separated				0.00			0.00
Drop Mass Left				0.00			0.00
Sample Mass Added				0.00			0.00

Appendix C: Swing-by Calculator

EJS mission
****
Minimum Total ΔV:
Departure planet(3): 2018-1-14
Parking Orbit: Hp = 400 km
Ha = 400 km
C3 value_Departure = 77.5282211276079 km^2/s^2
Vinf_departure_abs = 8.80501113727904 km/s
RA_departure = -158.492257552296 deg
Decl_departure = -9.59178730080377 deg
Vinf_departure = (-8.07737756184036, -3.18302457398913, -1.46715623314867)
V_departure = (-35.7824572288069, -14.4100273600393, -6.33321083782669)
ΔV = 6.30075785196506 km/s

Swingby planet(5) : 2020-3-10
Transfer time = 786 days
C3 value_SB1 = 47.460609616927 km^2/s^2
C3 value_SB2 = 49.4365849104101 km^2/s^2
Vinf_arrival_abs = 6.88916610461143 km/s
Vinf_departure_abs = 7.03111548123127 km/s
Deflection angle = 52.9967390986132 deg
Rotation azimuth = 37.7523650760399 deg
Vinf_arrival = (-4.75976115068847, -4.68392464281285, -1.69296584306192)
Vinf_departure = (1.10771769691573, -6.27704089125712, -2.96787871447533)
V_departure = (13.7658643275835, -3.19692393284965, -1.95578342261757)
ΔV = 0 km/s
Swingby altitude = 3175403.44793425 km
B-plane B*T = 5270195.49184257 km
B-plane B*R = -322008.626533531 km

Arrival planet(6) : 2025-12-31
Transfer time = 2122 days
C3 value_Planet = 8.28199773015556 km^2/s^2
Vinf_arrival_abs = 2.87784602266271 km/s
Decl_arrival = -21.8199188462148 deg
Vinf_arrival = (1.11651225885566, -2.46022438234026, -0.991309282978348)
ΔV = 1.84291710533379 km/s
B-plane magnitude = 1679004.93005755 km

Total ΔV = 8.14367495729885 km/s
Duration: 2908 days

Final orbit parameters (Planet Equatorial of Date):
Rp = 298000 km
Ra = 1280000 km
Inclination = 21.8199188462148 deg

RAAN = 193.773898938865 deg
Omega = 249.871207912394 deg
Latitude of periapsis= -20.4254458942455 deg

Vinf pl. Equ. of Date= (-2.59483797286247, 0.63610008384483, -1.069668316853)

EVVEJS mission
****
Minimum Total ΔV:
Departure planet(3): 2018-8-31
C3 value_Departure = 12.5027515440663 km^2/s^2
Vinf_departure_abs = 3.53592301161469 km/s
RA_departure = -132.916270065749 deg
Decl_departure = -49.8081362337929 deg
Vinf_departure = (-1.55381507290523, -1.6711522528466, -2.70104802087719)
V_departure = (9.23136108988409, 23.5199178540587, 8.2192265209688)

Swingby planet(2) : 2019-2-8
Transfer time = 161 days
C3 value_SB1 = 38.0194011367414 km^2/s^2
C3 value_SB2 = 38.0194011367414 km^2/s^2
Vinf_arrival_abs = 6.16598744214918 km/s
Vinf_departure_abs = 6.16598744214918 km/s
Deflection angle = 58.70494 deg
Rotation azimuth = 9.32477999999999 deg
Vinf_arrival = (-3.69263155392776, -4.70050290879761, 1.51299231593012)
Vinf_departure = (0.921140041387027, -5.65695610255171, -2.27370838382978)
V_departure = (14.03999886195, -35.0271413925982, -16.3189397539076)
Swingby altitude = 2834.87550769942 km
B-plane B*T = 14558.0732247003 km
B-plane B*R = -4348.31088233597 km

DSM ΔV : 655.291216150811 m/s
Time since last pl.: 222.07834933418 days
Time to next planet: 192.92165066582 days
DSM date : : 2019-9-18

Swingby planet(2) : 2020-3-29
Transfer time = 415 days
C3 value_SB1 = 109.925159128841 km^2/s^2
C3 value_SB2 = 161.434623757939 km^2/s^2
Vinf_arrival_abs = 10.484519976081 km/s
Vinf_departure_abs = 12.7056925729351 km/s
Deflection angle = 31.3915962745795 deg
Rotation azimuth = 236.88075608547 deg
Vinf_arrival = (5.30233393193194, -8.09137761183808, -4.04227935028861)
Vinf_departure = (-0.219677052264537, -11.242253166889, -5.91592000302753)
V_departure = (-19.5855861328824, -38.5409225688867, -16.9736464400955)
ΔV = 1.68114680213345 km/s
Swingby altitude = 463.345757308762 km
B-plane B*T = -8942.65126871613 km
B-plane B*R = -993.961217602066 km

Swingby planet(3) : 2020-5-20
Transfer time = 52 days
C3 value_SB1 = 376.487795121225 km^2/s^2
C3 value_SB2 = 375.923233976413 km^2/s^2
Vinf_arrival_abs = 19.4032934091413 km/s
Vinf_departure_abs = 19.3887398759283 km/s
Deflection angle = 7.95376299766942 deg
Rotation azimuth = 67.9870107030332 deg
Vinf_arrival = (-6.92095118284302, -15.8240202033864, -8.84243260922844)
Vinf_departure = (-4.67335689380215, -17.0348452961528, -7.99356084956539)
V_departure = (20.5551258338969, -30.9547137713745, -14.0285566618999)
Swingby altitude = 7839.45269096373 km
B-plane B*T = 13493.0278727615 km
B-plane B*R = 7084.02061506475 km

Swingby planet(5) : 2021-7-29
Transfer time = 435 days
C3 value_SB1 = 180.69039338596 km^2/s^2
C3 value_SB2 = 179.38837580428 km^2/s^2
Vinf_arrival_abs = 13.4421126831298 km/s
Vinf_departure_abs = 13.3935945811526 km/s
Deflection angle = 2.01260488799583 deg
Rotation azimuth = 4.32561272459728 deg
Vinf_arrival = (5.94317378449932, -11.0706163255886, -4.77603736635421)
Vinf_departure = (6.20308732849998, -10.7574724317224, -5.01865223740561)
V_departure = (13.5455207778823, -0.276501871609328, -0.70494064428926)
Swingby altitude = 39299480.5267213 km
B-plane B*T = 27684117.8328464 km
B-plane B*R = -28963336.4819596 km

Arrival planet(6) : 2025-7-4
Transfer time = 1436 days
C3 value_Planet = 36.5622527870701 km^2/s^2
Vinf_arrival_abs = 6.04667286919593 km/s
Decl_arrival = -25.4606109392843 deg
Vinf_arrival = (1.42848652466682, -5.38859013826525, -2.34195972589324)
ΔV = 2.5833547607915 km/s
B-plane magnitude = 828219.822438188 km

Total ΔV = 8.45571579069045 km/s
Duration: 2499 days

Final orbit parameters (Planet Equatorial of Date):
Rp = 290000 km
Ra = 1360000 km

Inclination = 90 deg
RAAN = 113.095408497982 or 293.095408497982 deg
Omega = 166.865350616113 or 295.944128737544 deg
Latitude of periapsis= 13.1346493838873 or -64.0558712624559 deg

Vinf pl. Equ. of Date= (-5.02186926171895, 2.1415331243478, -2.5994071983038)

Appendix D: Chebytop Outputs

Mission to Saturn   Feb 27,2007 9:45:33

Departure Planet:	Earth	2459563.452	14-Dec-21	vhl=	0.000	km/s	constrained
Arrival Planet:	Saturn	2462163.452	26-Jan-29	vhp=	0.000	km/s	constrained

Variable thrust performance index jv= 3.340 w/kg   Departure position optimized, Arrival position optimized

Variable thrust trajectory units are au and years   Clock angle reference is north

time	x	y	z	t	theta	phi	mag a	p/p0	cone	clock
0.00	0.1221	0.9767	0.0000	0.9843	82.875	-0.003	0.5993	1.0000	70.49	86.19
90.00	-1.0061	0.0918	0.0009	1.0103	174.785	0.053	1.1427	1.0000	86.35	89.18
180.00	-0.2612	-1.1194	0.0009	1.1495	256.865	0.044	1.0382	1.0000	102.66	92.25
270.00	1.0283	-0.8507	-0.0024	1.3345	320.400	-0.103	0.5981	1.0000	111.65	94.47
360.00	1.4280	0.2885	-0.0059	1.4569	371.421	-0.231	0.3532	1.0000	90.94	90.95
450.00	0.7636	1.2537	-0.0053	1.4680	418.657	-0.208	0.5873	1.0000	66.11	86.13
540.00	-0.4734	1.3102	0.0010	1.3931	469.865	0.041	1.0944	1.0000	71.54	86.98
630.00	-1.3414	0.2651	0.0092	1.3674	528.820	0.385	1.5453	1.0000	85.60	88.98
720.00	-1.1007	-1.1118	0.0112	1.5646	585.287	0.409	1.5787	1.0000	101.29	91.03
810.00	-0.1178	-1.9662	0.0050	1.9698	626.571	0.145	1.3347	1.0000	114.30	93.00
900.00	1.0114	-2.2538	-0.0067	2.4703	654.168	-0.156	1.0659	1.0000	124.30	95.02
990.00	2.0567	-2.1871	-0.0222	3.0023	673.239	-0.423	0.8446	1.0000	132.34	97.25
1080.00	2.9741	-1.9207	-0.0407	3.5406	687.146	-0.659	0.6696	1.0000	138.83	99.80
1170.00	3.7715	-1.5418	-0.0620	4.0750	697.765	-0.872	0.5292	1.0000	143.53	102.61
1260.00	4.4660	-1.0993	-0.0857	4.6001	706.172	-1.067	0.4113	1.0000	146.23	105.50
1350.00	5.0735	-0.6206	-0.1113	5.1125	713.026	-1.247	0.3088	1.0000	146.14	107.93
1440.00	5.6063	-0.1224	-0.1384	5.6093	718.749	-1.414	0.2190	1.0000	141.12	108.96
1530.00	6.0731	0.3854	-0.1667	6.0876	723.632	-1.569	0.1461	1.0000	124.99	107.70
1620.00	6.4798	0.8966	-0.1955	6.5445	727.878	-1.712	0.1158	1.0000	88.52	104.13
1710.00	6.8298	1.4069	-0.2244	6.9768	731.640	-1.843	0.1532	1.0000	53.97	99.44
1800.00	7.1246	1.9135	-0.2527	7.3814	735.034	-1.962	0.2280	1.0000	38.71	95.05
1890.00	7.3641	2.4143	-0.2800	7.7548	738.152	-2.069	0.3163	1.0000	32.65	91.67
1980.00	7.5471	2.9075	-0.3055	8.0935	741.069	-2.163	0.4114	1.0000	30.23	89.32
2070.00	7.6712	3.3915	-0.3286	8.3939	743.850	-2.244	0.5112	1.0000	29.46	87.77
2160.00	7.7334	3.8648	-0.3488	8.6524	746.554	-2.310	0.6152	1.0000	29.54	86.77
2250.00	7.7297	4.3257	-0.3653	8.8653	749.232	-2.362	0.7232	1.0000	30.14	86.14
2340.00	7.6555	4.7724	-0.3775	9.0291	751.939	-2.396	0.8350	1.0000	31.08	85.75
2430.00	7.5058	5.2028	-0.3848	9.1408	754.729	-2.412	0.9508	1.0000	32.31	85.52
2520.00	7.2750	5.6144	-0.3863	9.1977	757.659	-2.407	1.0707	1.0000	33.81	85.38
2600.00	6.9969	5.9622	-0.3824	9.2006	760.435	-2.382	1.1807	1.0000	35.37	85.31

tl	0.0	ta	2600.0	tend	2600.0	vhl	0.000	vhp	0.000	jv	3.340
the	677.560	pe	6.206	sap	0.000	a0	128.08	is	7000.00	eff	0.85000
alpha	40.00	jc	4.004	tp	2751.92	m0	1200.0	mf	667.7	mp	532.3
mps	248.2	mn	366.2	mt	53.2	ms	0.0	tf	3711.68	mna	0.0
morb	0.00	mnet	366.20	mpr	0.00	mi	0.00	delv	0.00	pj	5.275
m0/pj	227.480	eqi dv	40.246	mp/pj	100.912	m0/pe	193.358	mf/pe	107.583	mp/pe	85.775
m/m0	0.55639										

Escape Spiral	time	585.96	m0	1200.0	m	1086.7	mp	113.3
Capture Spiral	time	525.73	m0	769.4	m	667.7	mp	101.7

Off:	320.4	1187.6	2600.0
On:	0	403.7	2064.2

Figure 1: Output data from Sample CHEBYTOP Run