Mission Trades for Aerocapture at Neptune

Muriel A. Noca [1] and Robert W. Bailey.[2]
Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California 91109, USA

A detailed Neptune aerocapture systems analysis and spacecraft design study was performed to improve our understanding of the techonology requirement for such a hard mission. The primary objective was to engineer a point design based on blunt body aeroshell technology and quantitatively assess feasibility and performance. This paper reviews the launch vehicle, propulsion, and trajectory options to reach Neptune in the 2015-2020 time frame using aerocapture and all-propulsive vehicles. It establishes the range of entry conditions that would be consistent with delivering a ~ 1900 kg total entry vehicle maximum expected mass to Neptune including a ~ 790 kg orbiter maximum expected mass to the science orbit. Two Neptune probes would be also be delivered prior to the aerocapture maneuver. Results show that inertial entry velocities in the range of 28 to 30 km/s are to be expected for chemical and solar electric propulsion options with several gravity assists (combinations of Venus, Earth and Jupiter gravity assists). Trip times range from approximately 10-11 years for aerocapture orbiters to 15 years for all-propulsive vehicles. This paper shows that the use of aerocapture enables this mission given the payload to deliver around Neptune compared to an all-propulsive orbit insertion approach. However, an all-propulsive chemical insertion option is possible for lower payload masses than the one needed for this science mission. Both approaches require a Delta IV heavy class launch vehicle.

I. Introduction

As part of the NASA In-Space Propulsion Program, aerocapture was investigated as an option for orbit insertion around Neptune. This study involved several NASA centers and had for objective to conceptually design an aerocapture system for a generic orbiter with atmospheric probes mission. This paper provides an overview of the mission trades performed during this study. The main objectives of the mission trades were to:

1) Identify potential mission architecture and trajectories for a launch circa 2015-2020, which meant to identify launch vehicle options, launch opportunities and sensitivities, and potential trajectories using chemical ballistic propulsion and solar electric propulsion (SEP);
2) Understand the sensitivities in flight time and Neptune atmosphere's inertial entry velocities;
3) Provide a baseline trajectory.

The level of analysis for the mission trades varied from relatively detailed, in the case of the aerocapture system and trajectory optimization, to more parametric in the case of the chemical system design. The approach was to survey as much as possible the trajectory trade space, both for chemical with multiple gravity assists and for SEP with a wide range of flight times and various gravity assist options. Once the trajectories were compiled, the delivered mass at Neptune was calculated given the maximum performances of representative launch vehicles. This delivered mass was then compared to the actual mass needed for an aerocapture vehicle and for a chemical insertion vehicle, thus quantifying the benefits of aerocapture.

[1] Senior Engineer, Deep Space Mission Architecture Group, MS 301-180, 4800 Oak Grove Dr., Pasadena, CA 91109
[2] Senior Engineer, Flight System Group, MS 301-180, 4800 Oak Grove Dr., Pasadena, CA 91109, AIAA Member

This paper first summarizes the transportation architectures considered. It then describes the launch and transit options for ballistic chemical trajectories and SEP trajectories. These trajectories provide the range of inertial entry velocities to expect. Both aerocapture and chemical insertion are then discussed, including details about the systems and trajectories. Based on the systems and orbiter designs, Earth-Neptune trajectories can be picked and traded against. This paper finally shows the overall architecture trade results.

These trades do not represent all possible options for mission design or mission architecture, but they do span the range of likely options.

II. Transportation Architectures

The science objectives and basic spacecraft concept for this Neptune mission were based on previous studies performed internally at the Jet Propulsion Laboratory.[1] The mission includes two Neptune atmospheric probes and a Neptune Orbiter.[2] The atmospheric probes were considered here as black boxes (207 kg maximum expected value total), and only the navigation aspects of carrying these probes were taken into account.[3] The probes perform a direct entry a few hours before the Orbiter's insertion. The desired science orbit around Neptune is a 4,000 x 488,000 km altitude at 157 degree inclination to match Triton's orbit plane and motion (Triton's orbit is almost circular at 330,000 km radius from Neptune). The science orbit provides a fly-by of Triton every 3 orbits (~12 days).

To understand the sensitivities in aerocapture entry conditions into Neptune's atmosphere and understand the benefits of aerocapture, it was necessary to perform a trade study of the various and most probable transportation options to Neptune in the 2015-20 launch time frame. The transportation architectures considered from Earth to the final science orbit were the following:

1. **Option 1A**: Chemical ballistic transit trajectory to Neptune with a chemical insertion at Neptune.
2. **Option 1B**: Chemical ballistic transit trajectory to Neptune with aerocapture.
3. **Option 2A**: SEP transit trajectory to Neptune with a chemical insertion at Neptune.
4. **Option 2B**: SEP transit trajectory to Neptune with aerocapture.

The Orbiter was designed to perform aerocapture and modified in the trades when a chemical insertion was performed instead. As will be discussed, the baseline concept uses SEP to reach Neptune. To make the comparison between aerocapture and chemical insertion easier, and as the orbiter was designed to accommodate the ΔV for apoapsis raise, the targeted apoapsis altitude for the chemical insertion was 430,000 km (consistent with the target aerocapture altitude). The aerocapture entry is assumed to start at 1000 km altitude (25,766 km radius). The aerocapture pass is retrograde to match Triton's orbit. The chemical insertion burn is assumed to be performed at 4000 km altitude (28,766 km radius).

All four combinations of transportation were evaluated and will be described. The technique of aerobraking was not included in this trade study because the need for a highly elliptical science orbit precludes the multi-pass orbit circularization strategy that makes aerobraking useful. The next two sections will describe the transit trajectories, followed by a description of the orbit insertions.

III. Earth to Neptune Chemical Ballistic Trajectories

Ballistic direct trajectories as well as gravity assist trajectories were computed for a launch period between 2012 and 2019. The gravity assists (GA) include several combinations of Venus (V), Earth (E), Jupiter (J) and Saturn (S) fly-bys. Figure 1 shows the launch C3 for each of the trajectories surveyed. This list of trajectories does not represent all possible options of gravity assist nor launch date, and a more thorough survey should be done to complement this analysis.

As can be seen in Figure 1, the direct trajectories require a very high launch C3, which implies a relatively small launch mass. The Jupiter gravity assist (JGA) and Jupiter Saturn gravity assist also require a large launch C3 and thus offer poor performances. For all practical purposes, these trajectory options were deleted from the trade space.

Note that some of gravity assist trajectories require a deep space maneuver, which ΔV can be significant. Figure 2 shows the corresponding deep space maneuver ΔV required for each of the trajectories.

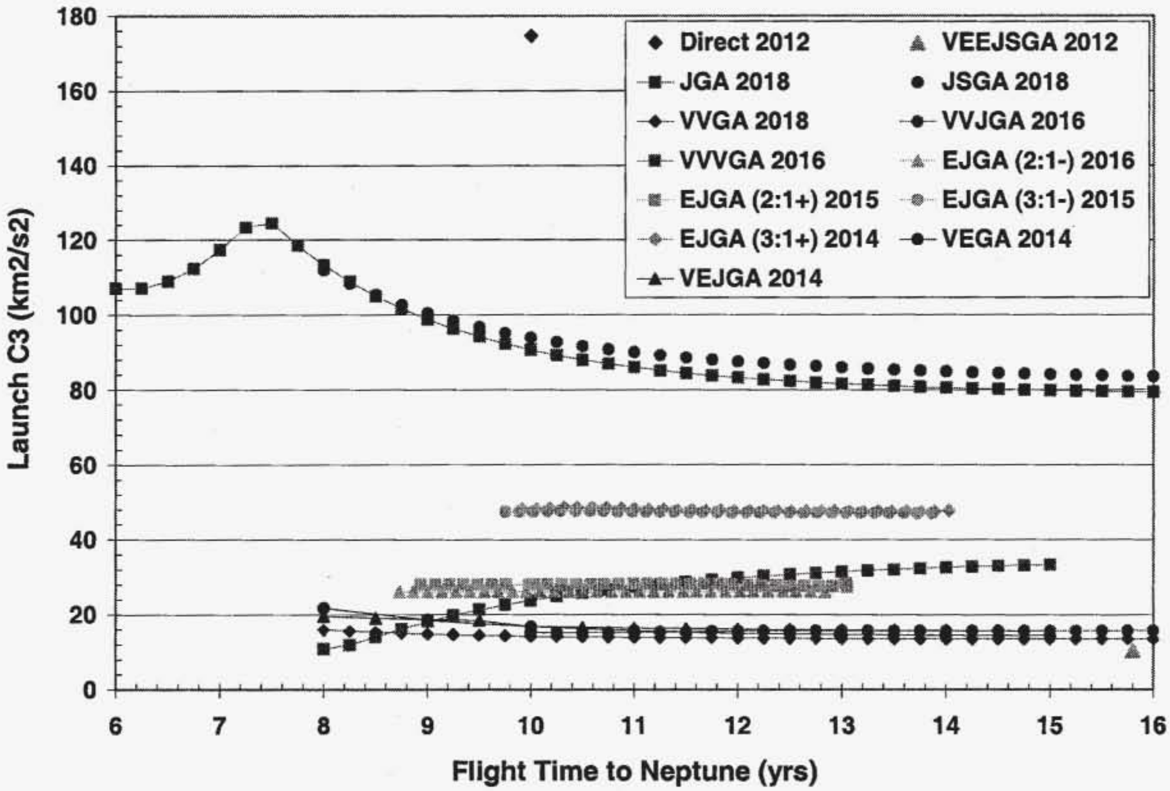

Figure 1: Launch C3 for various direct and GA trajectories to Neptune (with associated launch dates).

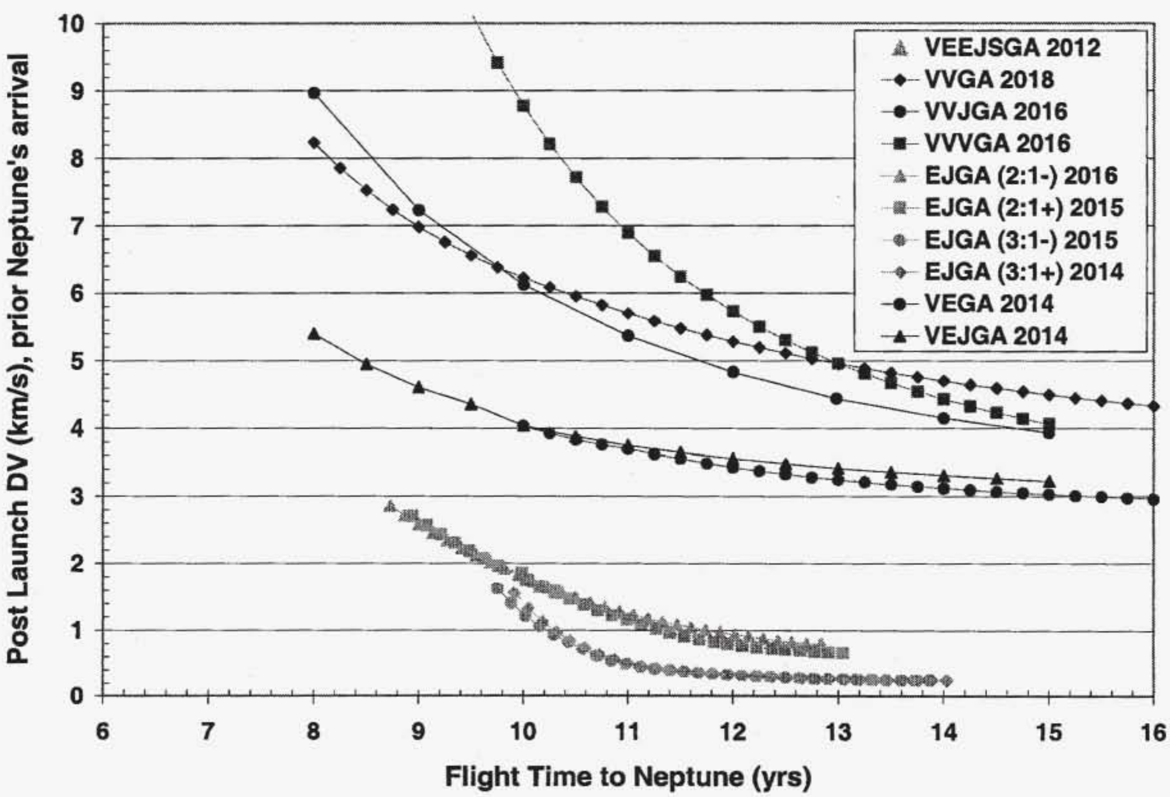

Figure 2: Corresponding post-launch ΔV for the gravity assist trajectories to Neptune.

Although the Venus only gravity assists offer a low launch C3, the post launch ΔV required for these options is quite high. The best performing options are the Venus or Earth combined with a Jupiter gravity assist. The launch date availability for these is restrained to 2014 – 2016. The use of a Jupiter gravity assist requires proper positioning of the planets, which only happens every ~ 10 years for Jupiter. Again, gravity assist with different launch dates and lower post launch ΔV may exist with launch C3 close to or lower than the ones presented here.

The maximum launch injected mass was derived from Figure 1 given the launch C3 for a Delta 4450, Atlas 551 and a Delta IV Heavy. These launch vehicles were picked as representative of a range of launch vehicle performance. The launch vehicle data was provided by the NASA KSC Launch Support Group[4]. Out of the injected mass provided by the launch vehicle, one needs to subtract the mass of propellant and chemical propulsion system needed to perform the post-launch ΔV. To do that we assumed a chemical Isp of 325 sec and a propulsion system dry mass equal to 20% of the propellant mass (approximately equivalent to a propulsion dry mass fraction of 16% of the chemical stage). Figure 3 then shows the net delivered mass (injected mass minus chemical propellant and propulsion dry mass to perform the post-launch ΔV) at Neptune's arrival (not inserted into Neptune's orbit).

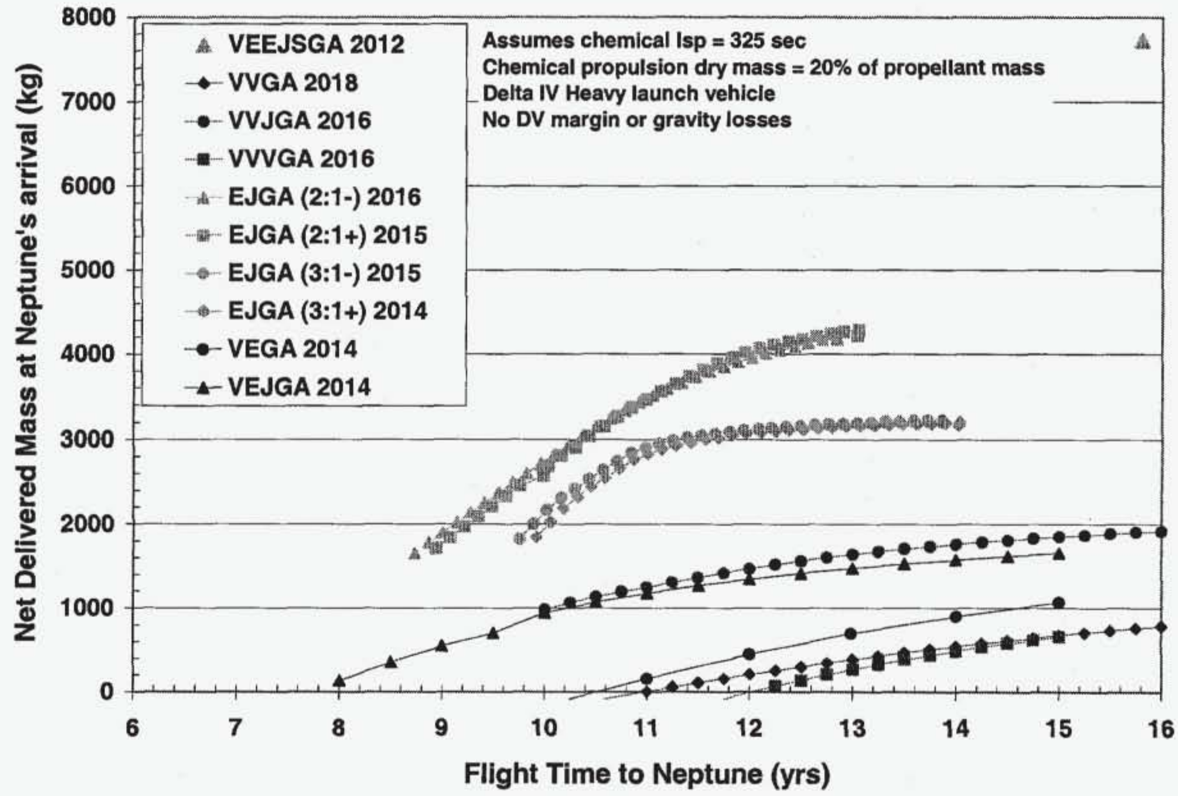

Figure 3: Net delivered mass at Neptune's arrival. Launch on a Delta IV Heavy.

Figure 3 shows that the EJGAs have the best potential for delivering significant masses around Neptune. The VEEJSGA is unique and very dependent on launch date, but shows that there could be trajectories performing even better than the EJGAs for a longer flight time.

The corresponding inertial entry velocities at a 1000 km altitude are provided in Figure 4. With both Figures 3 and 4, and assuming the EJGAs provide sufficient mass at Neptune's arrival, flight times of 10 to 12 years to Neptune imply inertial entry velocities in the range of 28 – 31 km/s. This information will help select a baseline inertial entry velocity for a detailed aerocapture design, as will be discussed later.

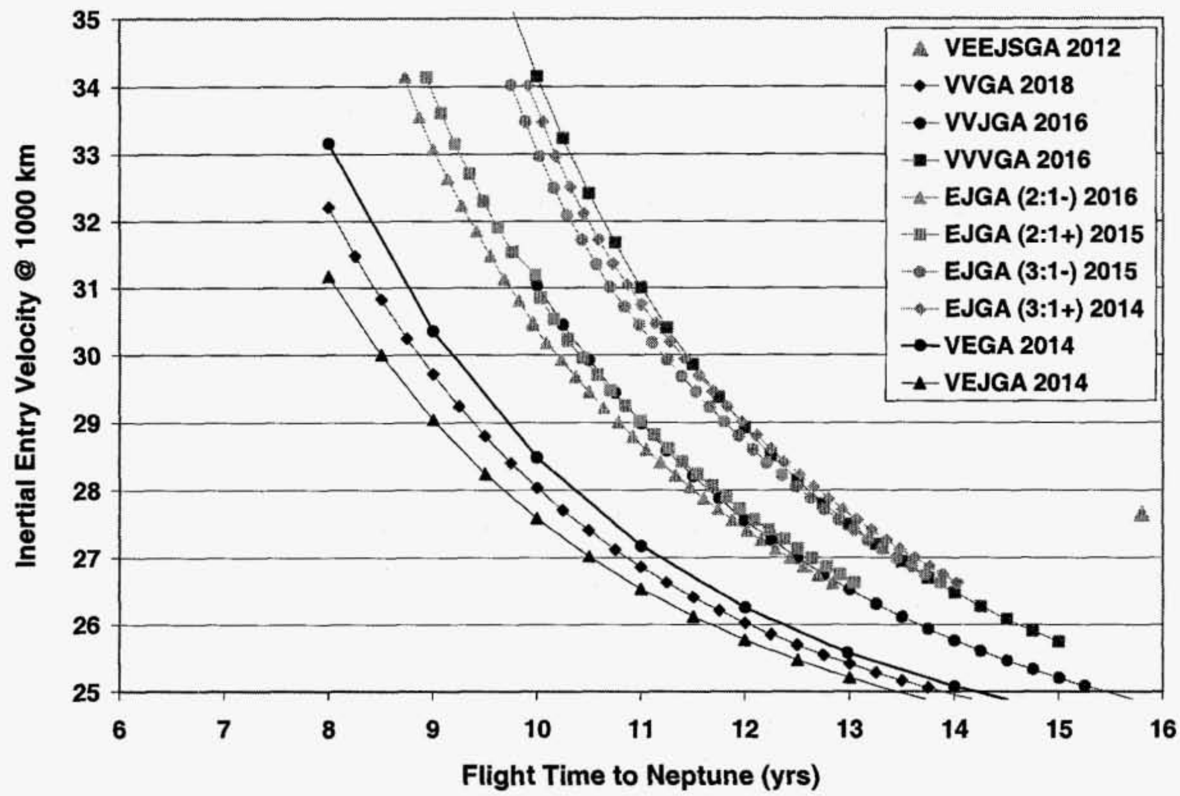

Figure 4: Inertial entry velocity for the ballistic gravity assist trajectories.

IV. Earth to Neptune Solar Electric Propulsion Trajectories

As for the chemical ballistic trajectories, an extensive database of gravity assists Solar Electric Propulsion (SEP) trajectories on various launch vehicles was built, as they clearly provided better delivered mass for equivalent flight times compared to direct (no gravity assist) SEP trajectories. These trajectories served the purpose of evaluating the sensitivities in launch date, number of thrusters, power levels and inertial entry velocities.

The SEP low-thrust trajectory optimization were run with a code named SEPTOP for Solar Electric Propulsion Trajectory Optimization Program, which is based on the calculus of variations. This code optimizes two body interplanetary trajectories and can model discrete numbers of operating Xenon thrusters throughout the trajectory. The trajectories allowed for a coast time duty cycle of 5% to simulate times when the spacecraft is not thrusting due to housekeeping activities, and assumed a constant 250 W from the solar arrays for the spacecraft. A 10% launch vehicle margin was assumed.

A. Solar Electric Propulsion System Assumptions

The ion thruster used to calculate the SEP trajectories is the NEXT engine. The characteristics of the NEXT engine technology can be found in many references.[5,6] Table 1 shows the projected performances of the NEXT engine. The high-Isp profile of the engine was used to calculate the trajectories.

The ion propulsion system (IPS) was designed more as a propulsion module than just thrusters and power processing units. Figure 5 shows a simplified block diagram of a typical single string ion propulsion system (IPS). To that basic configuration was added redundancy, structural and thermal considerations. Figure 5 also shows an example of what the IPS module designed here could look like.

The number of thrusters and PPUs was calculated on the basis of power requirements and thruster propellant throughput. The system architecture followed a conventional approach with parallel strings of PPUs and thrusters. Each PPU drives one thruster but is cross-strapped to two engines. One spare ion engine, one spare PPU and DCIU were also included for single-fault tolerance. Each thruster was gimbaled separately. The PPUs were assumed to be 95% efficient.

The solar arrays were sized based on a projection of the AEC-Able Square Rigger array capability. Since this array technology scales with power from ~ 1 kW up to ~ 30 kW, it was used as a representative potential technology for SEP applications. The specific mass was assumed to be 130 W/kg. A 14% degradation factor was applied to the array Beginning-of-Life (BOL) power to account for various degradation phenomena. Also, in order to support power demand during launch, a primary battery was used prior to solar array deployment.

The tank mass fraction was assumed to be 2.5% for Xenon when stored as a supercritical gas (~2000 psia). Furthermore, a 10% propellant contingency was added to the deterministic propellant mass to account for flow rate characterization, residuals, attitude control and margin.

Since the system masses are function of mainly power level, launch mass and propellant mass, each trajectory was uniquely considered and had a system mass associated with it. The component and subsystem sizing assumptions are given in Table 2. To be consistent with the JPL Team X conceptual design guidelines at the time of the study, 30% mass contingency was applied to all spacecraft subsystems. These masses represent the Maximum Expected Value mass (MEV).

For the purpose of the trade studies, the mass model of Table 2 was used. However, the final baseline design and the final table comparing all options (section VII) included a better definition of the system. This system was about 35 kg heavier than the model used here. Most of the difference resides in a larger thermal and structural mass.

Table 1: High-level NEXT thruster characteristics as compared to the flown DS1 ion engine NSTAR.

	NSTAR DS1	NEXT
Max. thruster processed power (kW)	0.5 - 2.3	1.2 - 6.2
Engine diameter (cm)	30	40
Maximum Isp (sec)	3100	3900
Xe throughput per engine (kg)	130	250
Thruster mass (kg)	8.3	12
Power Processing Unit mass (kg)	11.9	21

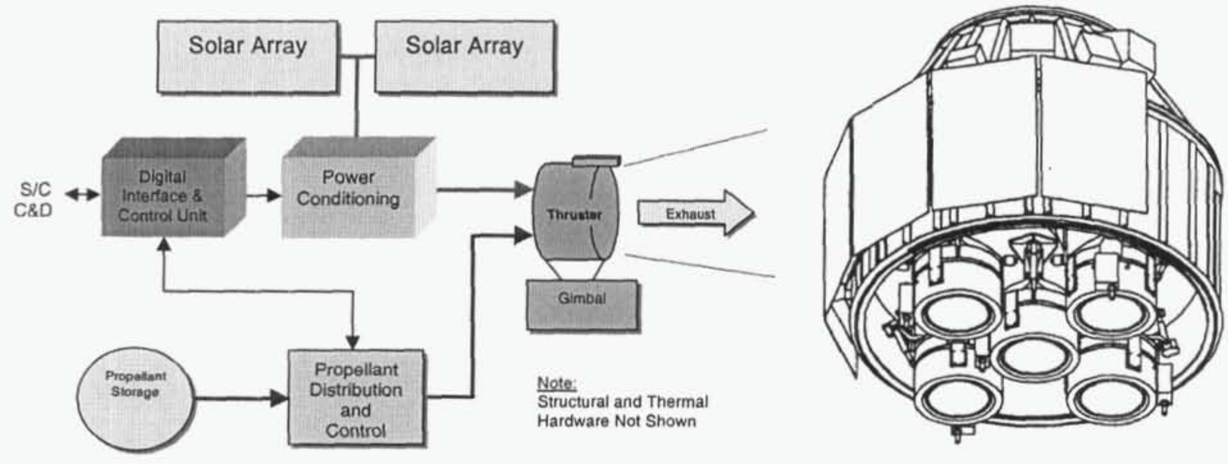

Figure 5: Ion propulsion module block diagram and conceptual configuration for system sizing

Table 2: Example of a 30-kW ion propulsion system mass model (includes 30% contingency).

Subsystem		Maximum Expected Mass (kg)
Not scaled with propellant mass:	- Propulsion	263
	- Power	415
	- Cabling	90
	- Thermal	69
	- Telecom+ACS +electronics	102
	- Structure	390
Scaled with propellant mass:	- Tank	2.5%
	- Tank structure	4%
	- Thermal	1%

B. SEP trajectory performances and inertial entry velocity characterization

As previously mentioned, the SEP trajectories considered use one to several gravity assist, because the mass needed for the orbiter and orbiter insertion system was beyond what a direct SEP trajectory could provide. With the appropriate thruster model, trajectories were run for several power levels, launch vehicle types and launch years. Results are in terms of net delivered mass. The net delivered mass is defined as the spacecraft mass minus the dry mass of the ion propulsion system and minus the propellant mass used for the SEP transfer. Therefore the net delivered mass is everything on the spacecraft that isn't SEP propellant or part of the ion propulsion module. Each trajectories were optimized for maximum delivered dry mass at destination (the launch C3 and thrusting profile were optimized). The gravity assist opportunities were also optimized for a given launch year. Figure 6 shows results of net delivered mass as a function of flight time to Neptune (arrival at Neptune's vicinity, not the final science orbit around Neptune), launch vehicle, power level and launch years for the NEXT thruster system.

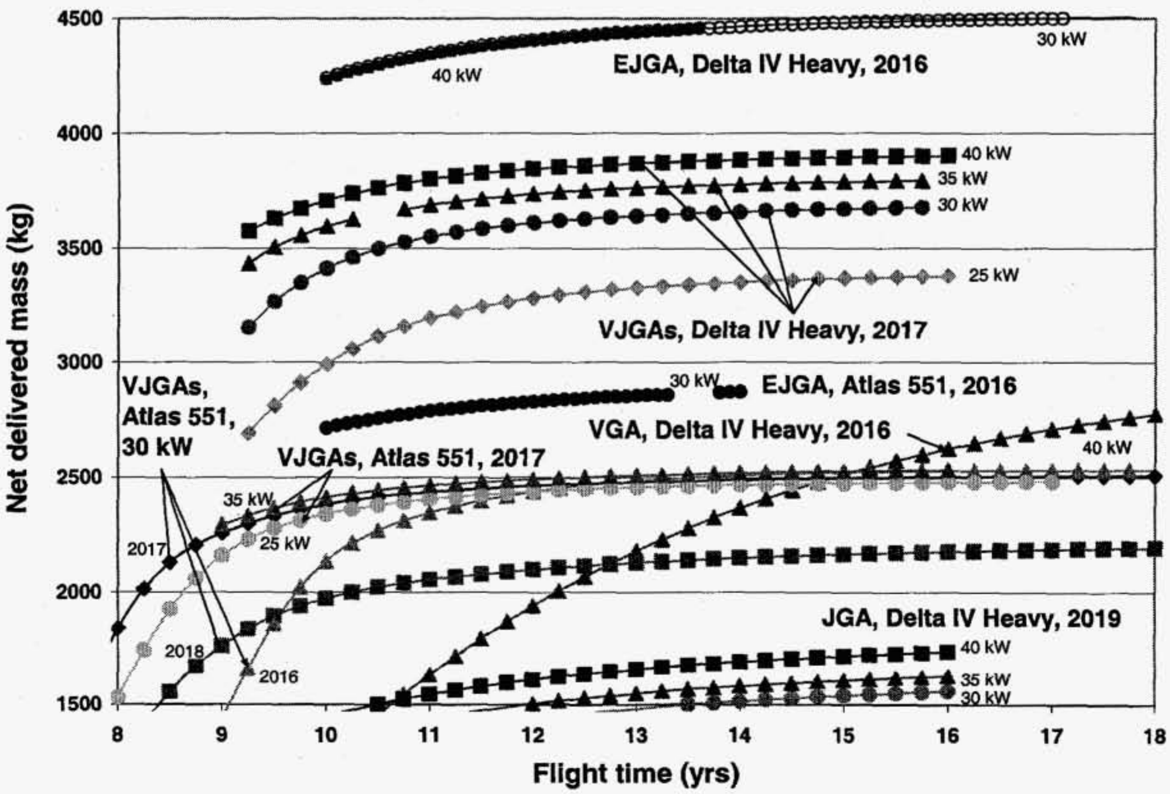

Figure 6: Net delivered mass at Neptune's arrival. SEP trajectories.

As can be seen in Figure 6, the net delivered mass is a strong function of launch year, gravity assist combination and power level. However, it is believed that the cases shown here are quite representative of the range of performance that can be expected from an SEP system. Not all of the trajectories run are shown in this figure as some provided performances below 1900 kg, and thus were judged insufficient.

Figure 7, 8 and 9 show the corresponding launch C3, radius of Jupiter fly-by and inertial entry velocities at 1000 km altitude at Neptune, respectively. The radius of the Jupiter fly-by was unconstrained and thus optimized in this study. For the VJGAs and EJGAs of particular interest, the fly-by occurs typically between the orbits of Io and Europa, so significant radiation can be expected.

As Figure 9 shows, the inertial entry velocity increases is very dependent on flight time, launch date, and gravity assist. However, the inertial entry velocity is only weakly dependent on SEP power for a given launch date and thruster technology. Thus choosing a flight time range will determine a range of inertial entry velocities. Flight times between 10 to 12 years offer the most "net delivered mass" benefit and result in entry velocities less than 30 km/s for most launch opportunities.

The weak sensitivity to the SEP power level in inertial entry velocity is mostly due to the fact that over the range of power looked at, the trajectory optimization code is trying to follow the same optimum acceleration path. Thus for high power level, it will optimize the trajectory at lower launch C3, thus injecting more mass. The acceleration, which is proportional to the power level to mass ratio will be roughly the same as a low power, large C3, low launch mass case. Since it will follow almost the same trajectory profile, the arrival hyperbolic velocity will only vary slightly. This is the case for a fixed flight time and launch date.

The launch window to perform a given gravity assist is about one month. The sensitivity in propellant mass for that window is included in the 10% propellant margin.

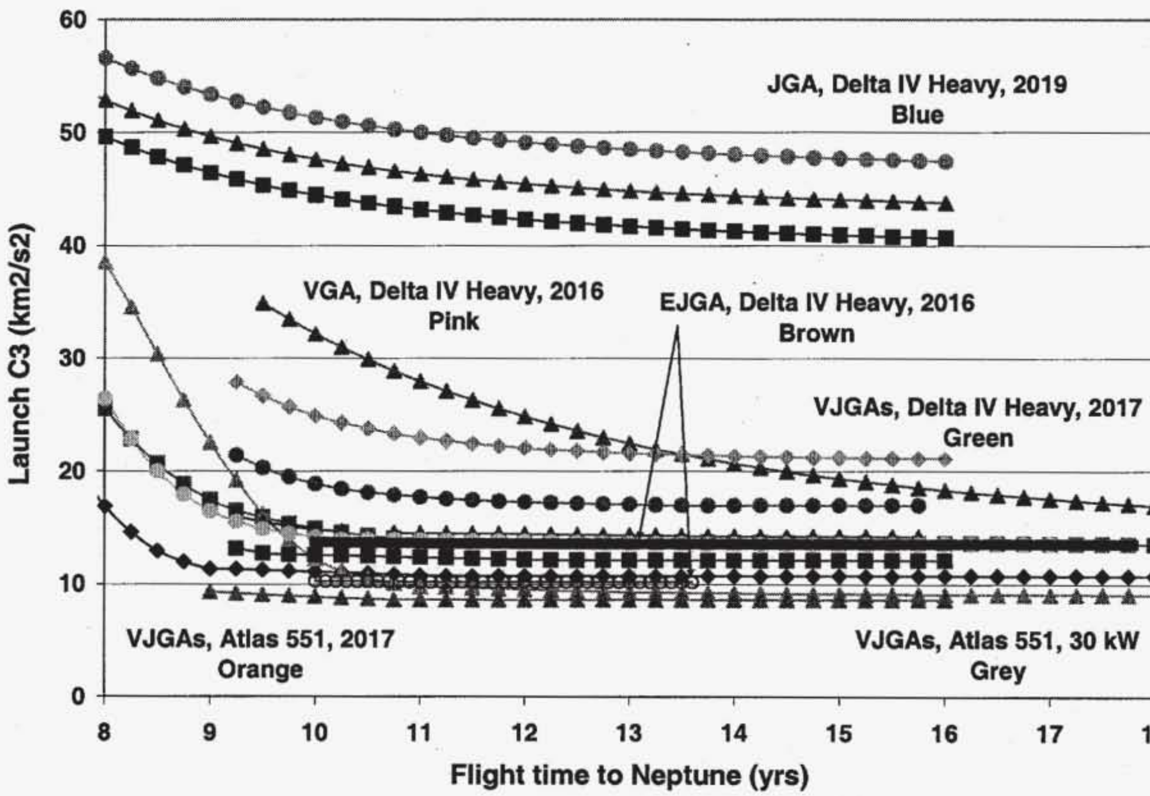

Figure 7: Launch C3 for the SEP trajectories.

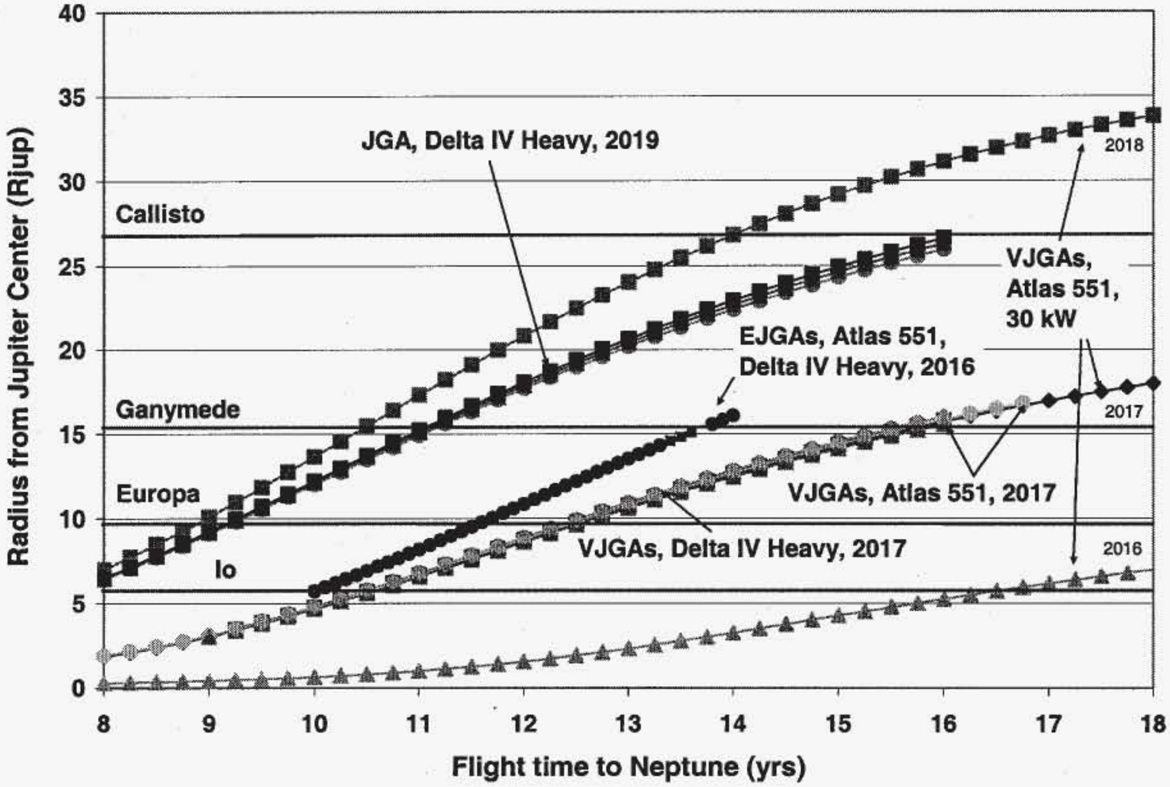

Figure 8: Jupiter gravity assist fly-by distance for the SEP trajectories.

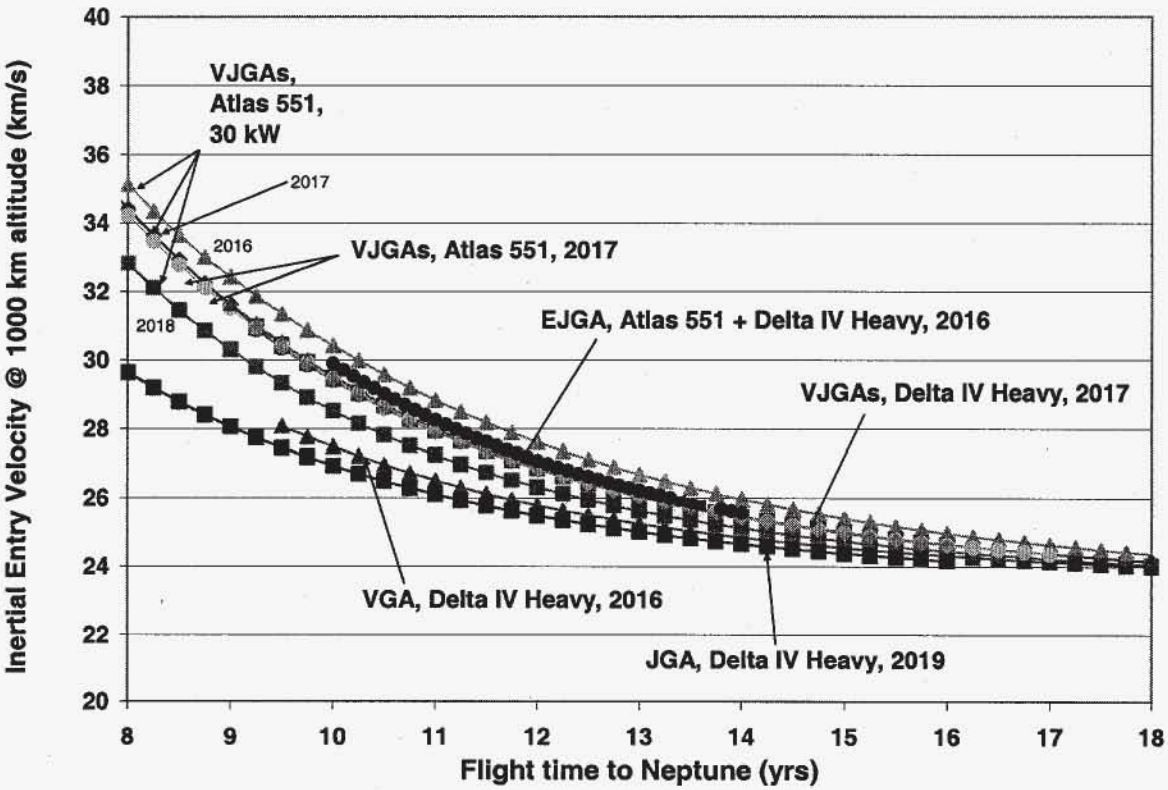

Figure 9: Inertial entry velocity for the SEP trajectories.

V. Aerocapture System and Insertion (Options 1B and 2B)

A. Selection of the inertial entry velocity

The selection of the chemical ballistic or SEP trajectory Earth to Neptune transit trajectory depends mostly on the performance (net delivered mass) provided by the trajectory and is an iterative process since the mass desired at Neptune's arrival depends on the design of the aerocapture system. In both chemical ballistic and SEP cases, an entry velocity of 29 km/s represents the best compromise between short flight times and high net delivered masses. Although somewhat arbitrary, it was felt that the aerocapture design would not change significantly for inertial entry velocities between 28 and 30 km/s.

B. Baseline trajectory

The baseline trajectory for the design of the aeroshell and other components of the aerocapture system was selected based on the following criteria:

1. The mission architecture should use the smallest launch vehicle possible to reduce cost;
2. The trajectory performance should provide adequate system mass margin (30%) for maximum expected mass, and adequate system reserves (> 10%);
3. The trajectory should provide a Neptune inertial entry velocity close to 29 km/s.

At the time of selection, the SEP trajectories were providing better performance than chemical ballistic option. It was also felt that a Venus Jupiter gravity assist would be sufficient performance wise. Thus the trajectory selected featured (see Figure 10):

Launch vehicle:	Delta IV Heavy (5 m fairing)
Flight time:	10.25 years
Launch date:	2/17/2017
Arrival date:	5/20/2027
Gravity Assist:	Venus Jupiter
Launch C3:	18.4 km^2/s^2
Launch mass:	5964 kg
Propellant mass:	973 kg deterministic
Vhyp @ Neptune:	17.5 km/s
$V_{entry_{inertial}}$:	29 km/s @ 1000 km
Thrusters:	5 maximum operating NEXT
SEP power level:	30 kW (1 AU EOL)

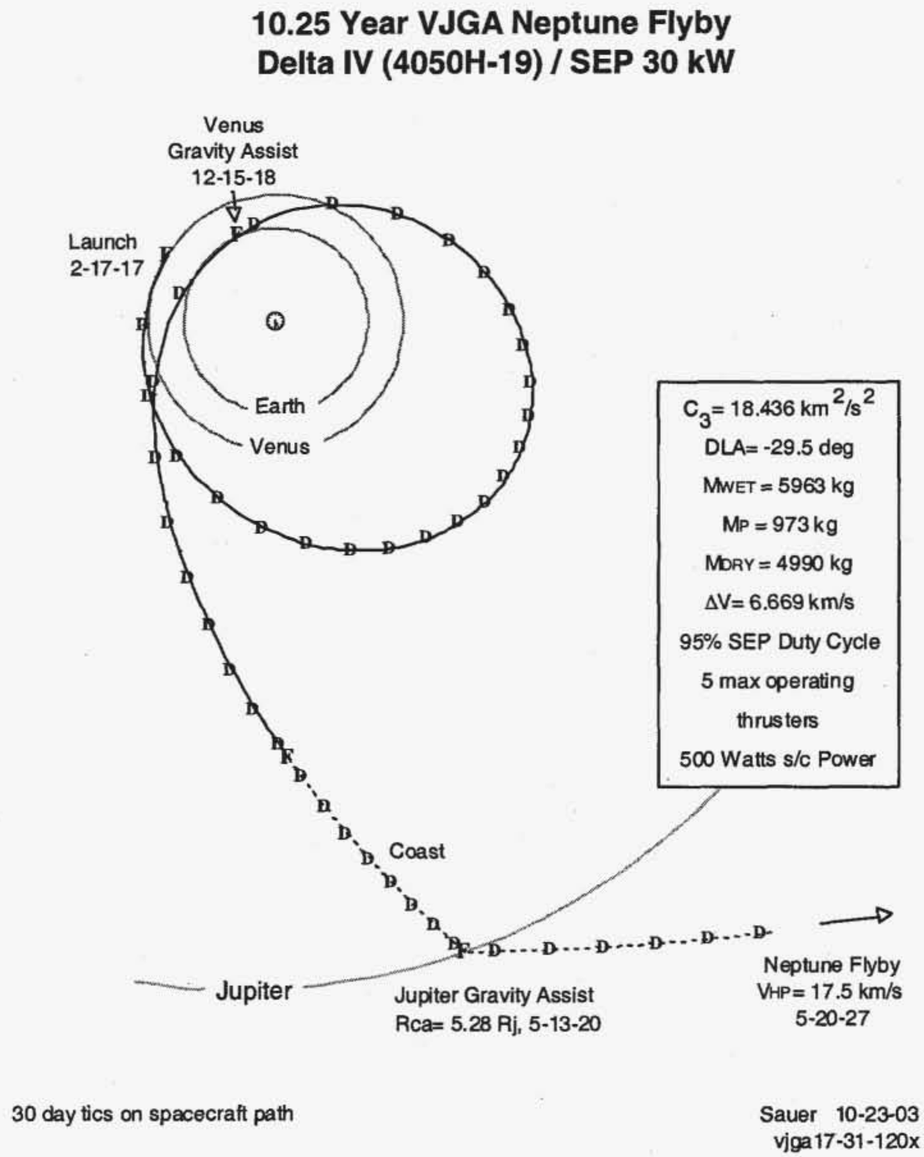

Figure 10: SEP baseline trajectory to Neptune.

C. Aerocapture system and Neptune orbiter description

The aerocapture system and the orbiter are described in detail in reference [2] and [7]. They have been designed for the baseline trajectory, which was chosen to match an inertial entry velocity of 29 km/s. The aerocapture system and the aeroshell shape were subject to an intensive trade related to the aerothermal and aerodynamic properties, control authority issues and volumetric efficiency. The design converged on a 2.88 m ellipsled with a L/D of 0.8. Figures 11 and 12 show the SEP cruise configuration and the post-SEP and aerocapture configuration respectively.

The thermal protection system was optimized to reduce mass. The aeroshell is jettisoned after aerocapture, and a dual mode bi-propellant system was used to raise the pariapsis. A ΔV of 438 m/s was used to size the propellant tank and loading. This ΔV includes the maneuvers to be performed during the aerocapture pass, a periapsis raise maneuver at first apoapsis and and apoapsis raise maneuver to place the orbiter in phase for Triton fly-bys.

Table 3 summarizes the mass breakdown for the aerocapture system As can be seen, the total dry mass of the aeroshell system is 1119 kg (maximum expected mass) for a total entry maximum expected dry mass of 1911 kg (~ 58.6% aerocapture system entry dry mass fraction). Other aerocapture-related hardware was ejected before entry. This hardware is also summarized in Table 3.

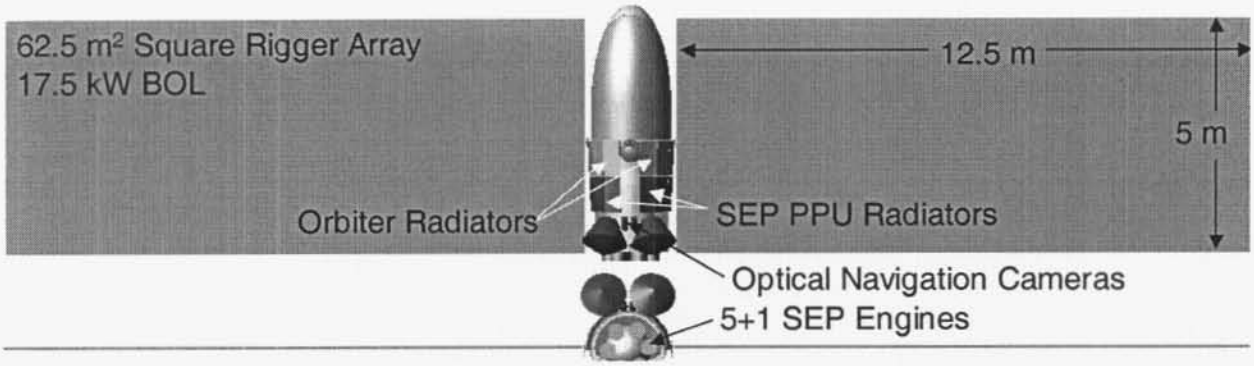

Figure 11: Neptune orbiter SEP cruise configuration.

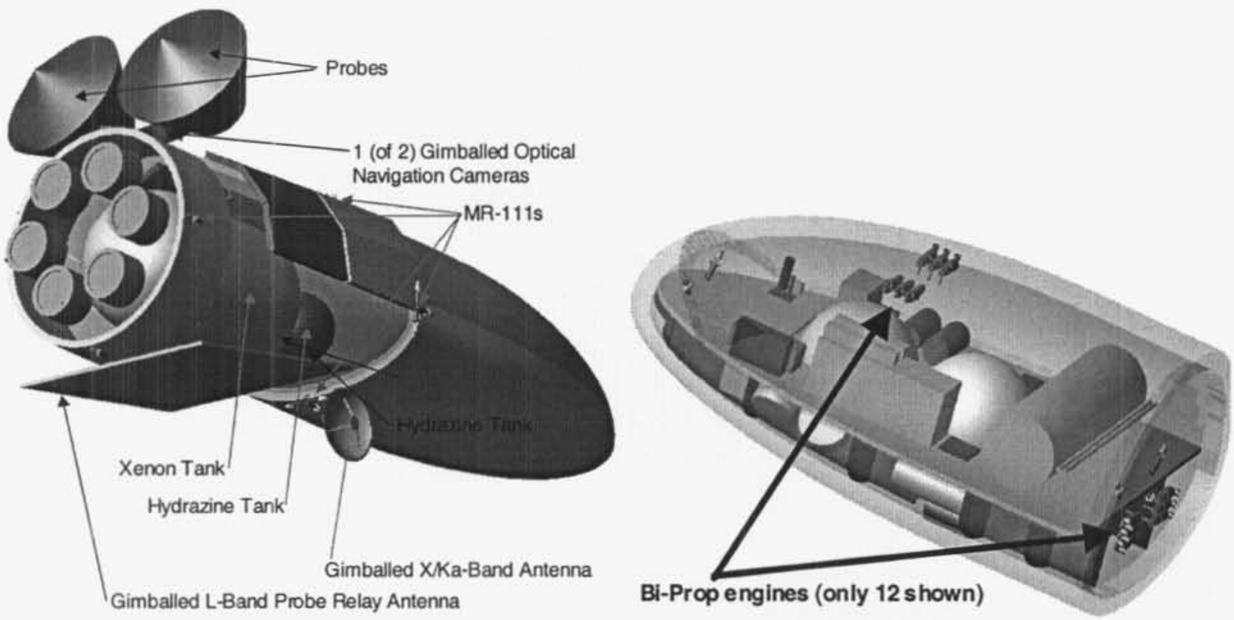

Figure 12: Neptune orbiter post-SEP and aerocapture configuration.

Table 3: Aerocapture system mass breakdown (includes 30% contingency).

Subsystem	Maximum Expected Mass (kg)
Mass that entered the atmosphere:	
- Heatshield, backshell and structure	957
- Hydrazine propellant	162
Aerocapture mass jettisoned prior entry:	
- 2 entry probes and ~ 100 kg of hydrazine	207 + 111 = 318

VI. Chemical System and Insertion (Options 1A and 2A)

A. Selection of the transit trajectory and chemical insertion

The selection of the chemical ballistic or SEP trajectory Earth to Neptune transit trajectory depends on the performance (net delivered mass) provided by the trajectory but also very strongly on the Neptune Orbit Insertion (NOI) burn. Unlike aerocapture, the chemical insertion will be very dependent on the arrival velocity (Vinfinity). For the chemical ballistic transfer trajectories, a balance between the post-launch ΔV and the NOI ΔV needs to be taken into account to minimize the sum of both ΔVs.

Given the chemical ballistic gravity assist trajectories provided in Figures 1 and 2 and their arrival velocity at Neptune, one can calculate the NOI ΔV at 4000 km altitude. Figure 13 provides such ΔV. A similar plot could be done for the SEP transit trajectories.

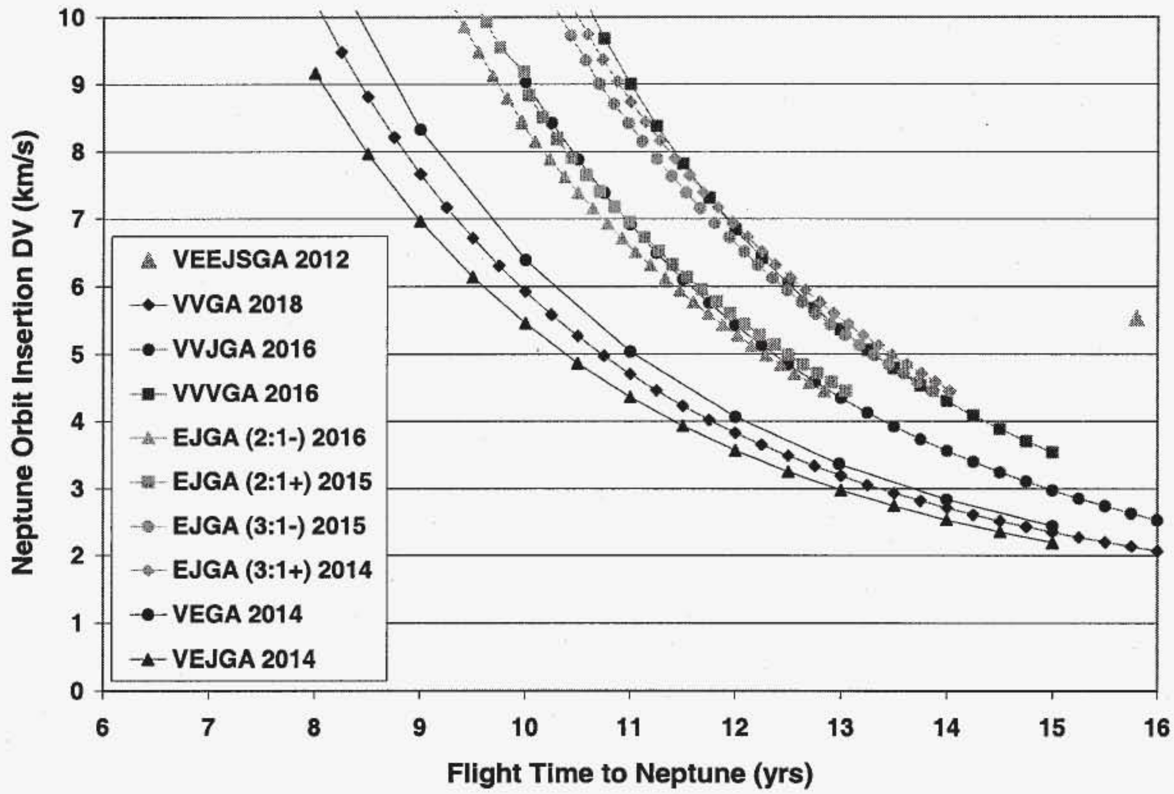

Figure 13: Neptune orbit insertion ΔV for the chemical ballistic transfer trajectories.

Since the NOI ΔV are large (which motivates the development of aerocapture), we assumed a dedicated chemical stage for that maneuver. In the case of chemical transit trajectories, we combined both ΔVs in a single stage (might provide conservative results).

B. Chemical propulsion system

To perform the chemical deep space maneuvers or insertion burns, a generic bi-propellant system was assumed, staged as necessary to accommodate for the large ΔVs. The dry mass for this system is summarized in Table 4. The specific impulse of the chemical system was assumed at 325 sec. In addition, 5% of the deterministic propellant mass was held as propellant contingency for maneuver clean-ups. Some of the structural mass depends on the dry mass at the beginning of the ΔV phase (Mi).

Table 4: Chemical propulsion system mass breakdown (includes 30% contingency).

Subsystem	Maximum Expected Mass (kg)
Not scaled with propellant mass:	
- Propulsion	19.5
- Thermal	16.5
- Telecom+electronics	2.3
- Structure	71.5 + 4% of initial mass (Mi)
Scaled with propellant mass:	
- Tank	5%
- Tank structure	4%
- Thermal	1%

Given these assumptions, the net delivered mass in a 4000 x 430,000 km altitude science orbit could be calculated. Figure 14 shows the net delivered mass for the various chemical ballistic trajectories. This figure represents option 1A, an all chemical propulsion to the science orbit. The "best" case for an all propulsive option is a Venus-Earth-Jupiter gravity assist that can deliver around 500 kg in 15 years. The Earth-Jupiter gravity assists may offer better performances but no data was generated that would confirm the slope for longer flight times.

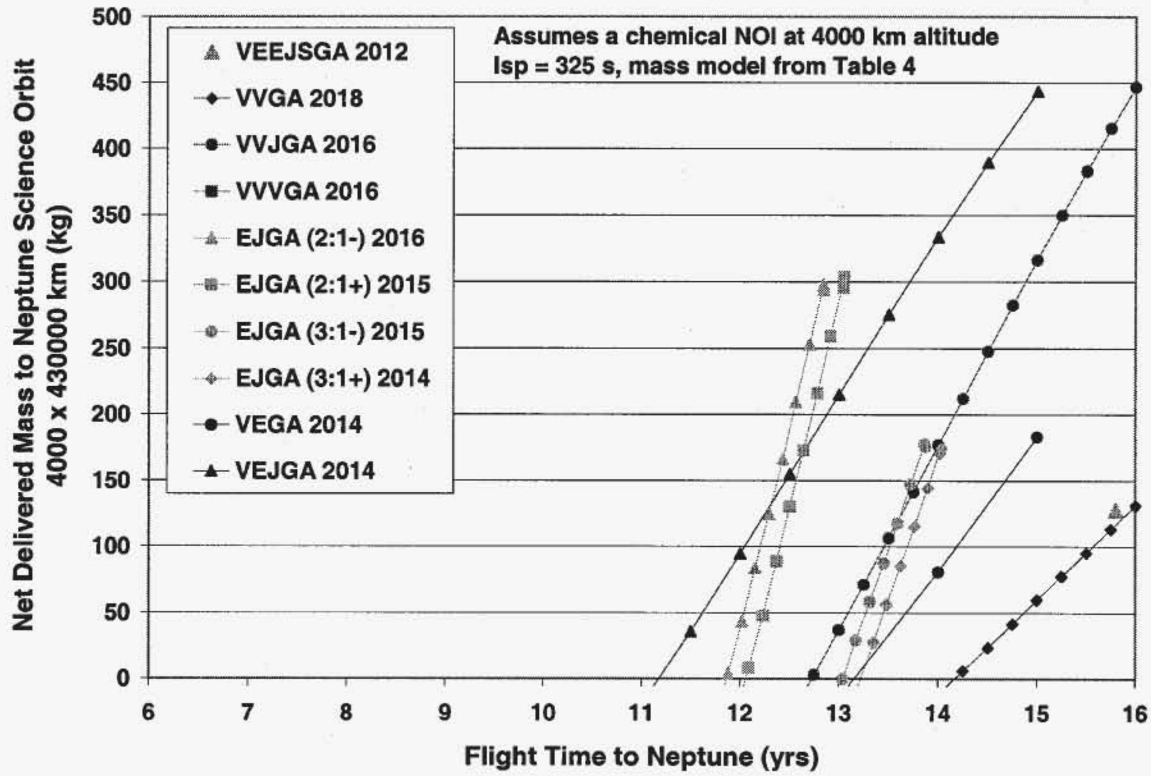

Figure 14: Net delivered mass in Neptune science orbit. Option A1 all chemical.

VII. Mission Architecture Trade Results

The overall mission architecture trade results are summarized in Table 5. This table shows first the type of launch vehicle followed by the gravity assist type, the transit propulsion system and the Neptune capture system. It assumes that the full capability of the launch vehicle is used and calculates the payload surplus or deficit mass compared to the mass required at Saturn before insertion. The points picked in this table represent cases for each architecture option with an inertial entry velocity of 29 km/s, thus reducing the number of trajectory options.

In the case of chemical insertion, the orbiter to lander interface is assumed not to be jettisoned and thus is included in the payload in Neptune's orbit mass. The detailed mass breakdown can be found in [2].

The payload surplus or deficit mass is the mass above or below the necessary mass to deliver the atmospheric probes and orbiter around Neptune (system margin needs to be above 15-20%). Table 5 clearly show the advantages of aerocapture, which in every case looked at provided more payload reserve and shorter flight times than for a chemical insertion burn. However, they also show that it is possible to deliver sufficient payload mass (low margin) with a chemical insertion system. Here again, the penalty will be flight time.

Table 5: Architecture trades summary table.

Launch Vehicle	Delta IV H							Atlas 551	
Gravity Assist	VEJGA	EJGA				VJGA		EJGA	
Earth to Neptune Prop System	Chem	Chem		SEP		SEP		Chem	SEP
NOI Prop System	Chem	Aero	Aero	Chem	Aero	Chem	Aero	Aero	Aero
Option	A1	A2	A2	B1	B2	B1	B2	A2	B2
							[7]		
Cruise Time to Neptune (yrs)	15.0	10.8	11.8	15.0	10.5	15.0	10.3	11.8	10.5
Launch Year	2014	2016	2014	2016	2016	2017	2017	2014	2016
Launch C3 (km2/sec2)	15.6	26.0	47.3	13.5	13.6	17.0	18.4	47.3	9.1
SEP Power (kW, EOL)				30	30	30	30		30
Inertial Entry Velocity (km/s)		29	29		29		29	29	29
Neptune Cruise Chem DV (km/s)[1]	3429	1413	357					357	
NOI Chem DV (km/s)[1]	2300			2871		2781			
	[9]			[6]		[6]		[6]	
Launch Capability	**7012**	**5695**	**3550**	**6543**	**6532**	**6130**	**5964**	**2630**	**4850**
Propellant Mass[2,3]	4158	2040	376	655	809	1025	1070	279	713
LV to Prop Module Adapter	62	62	62	62	62	62	62	62	62
Prop Module Dry Mass	806	542	289	1437	1449	1465	1468	243	1441
Chem Prop Mod to Payload Adapter	40	40	40					40	
Pre-NOI Separated Mass[10]	318	318	318	318	318	318	318	318	318
Pre-NOI Net Delivered Mass	**1628**	**2694**	**2464**	**4071**	**3895**	**3260**	**3046**	**1688**	**2315**
Aerocapture System[4]		1119	1119		1119		1119	1119	1119
NOI Chem Propellant Mass[8]	966			2417		1898			
NOI Chem Dry Mass	280			487		413			
Payload in Neptune Orbit	792	792	792	792	792	792	792	792	792
System Margin = LV-MEV	**(409)**	**783**	**553**	**375**	**1984**	**157**	**1135**	**(223)**	**404**
System Margin % = (LV-MEV)/MEV	**-5.5%**	**15.9%**	**18.5%**	**6.1%**	**43.6%**	**2.6%**	**23.5%**	**-7.8%**	**9.1%**

MEV: Maximum Expected Value = best estimate + 30% contingency

Assumptions and Notes:

All masses are MEV mass listed in kg

[1] Includes 5% DV contingency

[2] Chem Propellant mass calculated using "Launch Capability" as system total mass; Chem Isp = 325 sec

[3] SEP Propellant mass calculated using "Launch Capability" as system total mass; includes 10% prop mass contingency

[4] Aerocapture System Mass: aeroshell structure, TPS, and DV to achieve 28766x488,000 km orbit

[6] Propellant mass and Prop Module Dry Mass for SEP / Chem options includes propellant and dry mass for both SEP and chemical stages

[7] Neptune Aerocapture Study Reference Mission

[8] Chem Propellant mass calculated using "Pre-NOI Net Delivered Mass" as Initial mass; Chem Isp = 325

[9] Total Cruise+NOI DV split equally between two stages; I.e. Cruise delta-V is staged

[10] Includes Probes and ~100kg of cruise hydrazine

VIII. Conclusions

This paper summarizes the transit trajectory options for a Neptune orbiter mission and derives the range of entry conditions for the aerocapture maneuver inside Neptune's atmosphere. This survey shows that inertial entry velocities in the range of 28 - 30 km/s are to be expected. This range offers the best combination of highest delivered mass to Neptune's orbit and lowest entry heating. The study chose to baseline an inertial entry velocity of 29 km/s for the detailed design of the aerocapture system, and the corresponding SEP trajectory is provided.

This paper also summarizes the mission transportation trades performed during the study to show the benefits of aerocapture. The study shows that aerocapture as an orbit insertion option provides more delivered mass in every launch vehicle and gravity assist case looked at than chemical insertion, and shorter flight time (typically by 4-5 years). However, all chemical or SEP with chemical insertion cases exist that would deliver about 450 kg in Neptune's orbit with a Delta IV Heavy with flight times around 15 years. The baseline trajectory case for this study is an SEP aerocapture case on a Delta IV heavy with a flight time of 10.2 years delivering a 790 kg orbiter.

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