Neptune Aerocapture Mission and Spacecraft Design Overview

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A detailed Neptune aerocapture systems analysis and spacecraft design study was performed as part of NASA's In-Space Propulsion Program. The primary objective was to assess the feasibility of a spacecraft point design for a Neptune/Triton science mission that uses aerocapture as the Neptune orbit insertion mechanism. This paper provides an overview of the science, mission and spacecraft design resulting from that study. The estimated delivered wet mass allocation to Neptune orbit was ~924 kg. The aerocapture entry system, comprised of aeroshell and post-aerocapture orbit correction propellant, was ~1223 kg, for a total atmospheric entry mass allocation of ~2174 kg. The aeroshell used was a 2.88 m long flattened ellipsled with a lift to drag ratio of 0.8. A Delta-IV Heavy launch vehicle combined with a 30kW solar electric propulsion (SEP) stage and a Venus/Jupiter gravity assist are were used to get the spacecraft to Neptune in 10.25 years. The SEP stage and Orbiter both have 35% dry mass margins ((allocation – CBE)/allocation) and the overall launch stack has an additional ~10% unallocated reserve. The feasibility of the mission requires the solution of two key technical challenges: improvement in aerothermodynamic computational tools for Neptune; and development of thermal protection material manufacturing processes for the increased thickness needed for aerocapture. Several other component technologies were identified as being able to provide significant performance improvements including: radioiostopic power generation, solar cells and array structure, low mass/power science instruments, and small stowed volume/large aperture deployable Ka-Band antennas.

Nomenclature

ACS	=	Articulation and Attitude Control System
Alloc	=	Allocation
AU	=	Astronomical Unit
CBE	=	Current Best Estimate
C&DS	=	Command and Data System
Cont	=	Contingency: (MEV – CBE) / CBE
CM	=	Center of Mass
DSN	=	Deep Space Network
dV	=	Delta Velocity
EMI/EMC	=	Electromotive Interference / Electromotive Compatibility
EOL	=	End of Life
EOM	=	End of Mission
Gbits	=	Gigabits
HGA	=	High Gain Antenna
IR	=	Infrared
JPL	=	Jet Propulsion Laboratory
kg	=	kilograms
km	=	kilometers

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kW = kilowatt
L/D = Lift over Drag
LGA = Low Gain Antenna
LHP = Loop Heat Pipe
LV = Launch Vehicle
Marg = Margin: (Alloc – MEV) / MEV
MEV = Maximum Expected Value
MMRTG = Multi-Mission Radioisotpic Thermal Generator
PPU = Power Processor Unit
RF = Radio Frequency
RTG = Radioisotopic Thermal Generator
SEP = Solar Electric Propulsion
SSPA = Solid State Power Amplifier
TCM = Trajectory Correction Maneuver
TPS = Thermal Protection System
TRL = Technology Readiness Level
TWTA = Traveling Wave Tube Amplifier
UHF = Ultra High Frequency
UV = Ultraviolet
W = Watts

I. Introduction

Aerocapture is being investigated as a means for interplanetary orbit insertion by NASA's In Space Propulsion Program. A systems analysis and spacecraft point design study was performed in Fiscal Year 2003 based on a reference mission to Neptune. The purpose of this study was to quantify the feasibility and performance of an aerocapture system to insert a spacecraft into a scientifically useful orbit about Neptune that includes regular flybys of Neptune's moon Triton. This paper is one of eleven papers (Ref 1-10) associated with this Neptune aerocapture study; see Ref 1 for an overview of the entire Neptune aerocapture study. The multi-center Neptune Aerocapture team is largely the same team that perform a similar mission study for a Titan Aerocapture in 2002[11,12].

Of the twelve study papers, this paper discusses the science objectives and the resulting spacecraft configuration. Disciplines addressed in the other papers (Ref 2-10) include: trajectory design[2], deep space navigation[3], Neptune atmospheric models[4], aerodynamics and stability analysis[5], aerocapture guidance[6] and performance analysis[7], Neptune radiative and convective heating environments[8], TPS[9], and structural design/analysis[10].

II. Science

Science objectives for a Neptune/Triton mission were selected to yield a payload that representatively stressed spacecraft accommodation issues. The payload used for the study does not constitute a fully vetted recommendation for a Neptune/Triton mission, but instead represents a typical range of requirements such a mission might expect.

A. Science Objectives & Measurements

Science objectives were composed from various sources including:

- 2002 Solar System Exploration Decadal Survey
- Science community White Papers that contributed to the SSEDS and Individual scientists that were on the SSEDS Panels
- NASA SSE and SEC Roadmaps, 2003

The science objectives used for this study were (listed in no particular order):

1. Global imaging spectrometry of Neptune, Triton, other satellites, and the rings, at UV, visible, and IR wavelengths; repeated as needed for time variability
2. Global microwave radiometry at Neptune to infer temperature and pressure as a function of altitude
3. Measure the low-order gravity fields of Neptune satellites; at Triton, measure higher-order harmonics to infer gross interior structure
4. Measure Neptune's magnetic field at low altitudes, with sufficient accuracy and spatial and temporal coverage to map its generation region and determine its temporal variability
5. Measure abundances of key atmospheric constituents as a function of depth (Neptune)
6. Measure atmospheric temperature and pressure as a function of depth (Neptune)

7. Measure winds as a function of depth (Neptune)
8. Microwave radiometry to infer temperature and pressure as a function of depth (Neptune)
9. Measure the energetic charged particle environment
10. Measure the plasma wave environment

B. Science Instruments

Table 1 presents 1) the instrument suite selected to achieve the science objectives, and 2) the spacecraft accommodation considerations for the instruments. The mass and power for many of these instruments would be considered aggressive for a mission in formulation today. Instrument development would be required to achieve the payload capability for the total mass/power allocation presented.

Table 1. Science Instruments

Instrument	**Spatial Res** **Neptune (km)**	**Triton (km)**	**Global Coverage (Neptune & Triton)**	**# of channels/ spectra/ wavelens/ samples**	**Bits per channel per spatial resolution**	**Total Data (Gbits)**	**Mass (kg)**	**Avg Power (W)**
Visible Imager (low-res)	100	10	100%	7	10	28.03	5.2	3
(high-res)	10	1	10%	7	10			
Infrared Imaging Spectrometer (low-res)	1000	100	100%	1024	8	32.88	5.85	10
(high-res)	100	10	10%	1024	8			
Thermal IR Imaging Spectrometer (low-res)	5000	500	100%	50	8	0.06	4.2	1
(high-res)	500	50	10%	50	8			
Ultra Violet Imaging Spectrometer (low-res)	1000	100	100%	1024	8	32.88	2.0	3
(high-res)	100	10	10%	1024	8			
	Integrate Time (s)		Duration (s)					
Microwave Radiometer	0.1		43200	6	12	0.62	5.2	10
Magnetometer	0.1		2592000	3	24	3.73	0.8	1
UVIS Occultation Port	0.01		36000	512	8	29.50	1.3	3
Mass Spectrometer	60.0		72000	1	50000	0.12	6.5	6
Charged Particle Detector			Allocations			1	7.8	2
Plasma Wave Spectrometer						22	5.2	10

III. Mission overview

Certain aspects of the mission were assumed as ground rules. Other aspects of the mission were open to system trades and/or inherited from other outer planet mission studies performed at JPL.

A. Ground Rules

Several ground rules and assumptions were set to bound the study. These items were not subject to any system trades analysis.

- The TRL 6 cutoff date shall be no later than 2010.
- The Neptune atmospheric probe will be a "black box" with a 124 kg launch mass allocation (each).
- The orbiter shall perform an aerocapture for Neptune orbit insertion.
- The Earth to Neptune propulsion system shall be SEP.
- The orbiter shall accommodate the instruments in Table 1 for their intended science purpose.

B. Earth to Neptune Trajectory

The Earth to Neptune SEP trajectory selected for this study, shown in Figure 1, was the result of an extensive analysis of Chemical / SEP cruise and chemical/aerocapture Neptune insertion options (Ref 2). This trajectory provides a Neptune atmospheric entry velocity of 29 km/s.

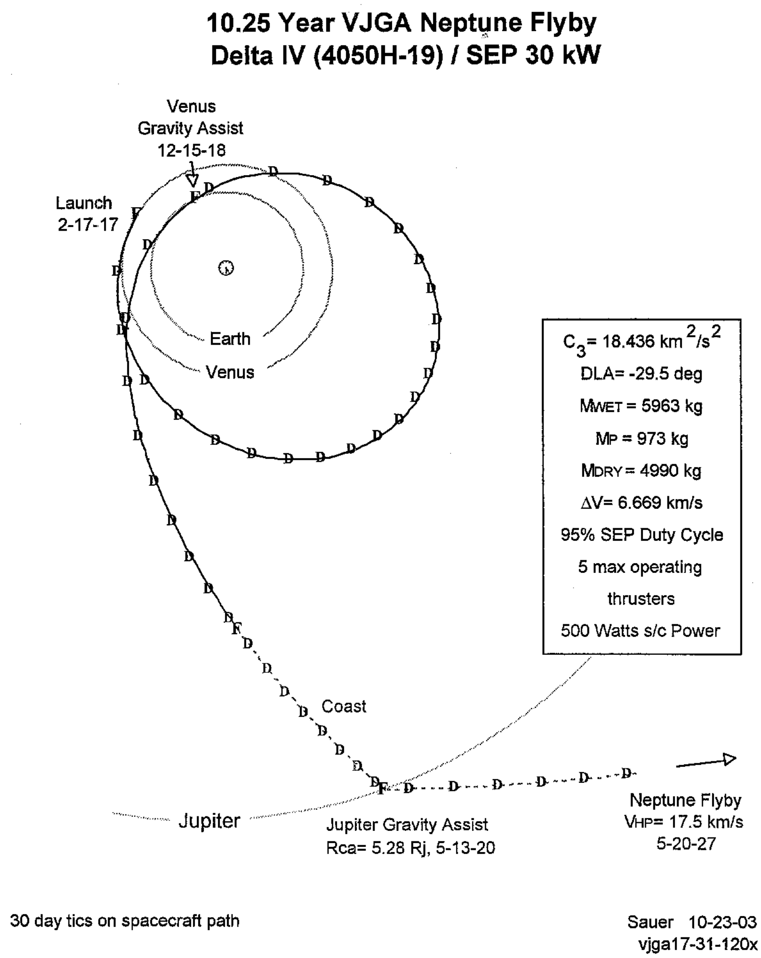

Figure 1. Earth to Neptune Trajectory

C. Mission Timeline

The mission timeline is listed below. For the "Time" column, 'L' = Launch, 'A' = Orbiter atmospheric interface, 'y' = years, 'd' = days, 'h' = hours, and 'm' = minutes.

Time	Event
L+0	Launch, SEP burn start
L+10m	Venus flyby
L+32m	SEP burn out, solar array jettison
A-135d	TCM 1
A-121d	Probe 1 Release TCM (2)
A-120d	Probe 1 Release and separate
A-114d	Probe 2 Release TCM (3)
A-113d	Probe 2 Release and separate
A-105d	Atmospheric Interface Target TCM (4)
A-60d	TCM 5
A-10d	TCM 6
A-36h	TCM 7 (if needed)
A-3.5h	Probe 1 entry
A-2h	Probe 1 end of life (EOL)
A-2h	Probe 2 entry
A-30m	Probe 2 EOL
A-30m	Jettison non-aero external components
A-29m	Align for aerocapture interface
A+38h	Jettison aeroshell
A+77h	Periapsis raise burn
A+2y	End of Mission

The probes are released 90 minutes apart to allow the first probe to reach its end of mission before the second probe enters the atmosphere.

The Neptune approach trajectory and subsequent aerocapture flight is retrograde to Neptune (157 deg inclination) to match Triton's orbit plane and motion. The primary deceleration pulse of aerocapture lasts less than 10 minutes, during which the Orbiter is modulating its trajectory with bank angle using bi-propellant thrusters. The aeroshell is jettisoned soon after atmospheric exit to limit thermal soak back from the heatshield to the orbiter. The orbiter then performs a TCM at the first post-aerocapture apoapsis (~430000km) to raise the periapsis out of the atmosphere and up to the desired science altitude (~4000 km). The final desired science orbit is 4000x488000km @ 157 deg inclination (Triton's orbit is ~circular at 330000km altitude). The science orbit provides a Triton flyby every 3 orbits (~11.75 days).

IV. Mission System Description

The Neptune Orbiter Flight System was the primary focus of this study; Ground Data and Mission Operations Systems were out of scope of this study. The launch configuration is shown in Figure 2. The stack fits easily inside the Delta IV fairing and looks slightly off-center geometrically because of the center of mass alignment within the aeroshell for proper aeroshell angle of attack and stability. The launch system mass summary is shown in Table 2. The SEP cruise, post-SEP cruise, and probe communications relay configurations are shown in Figures 3-5 respectively.

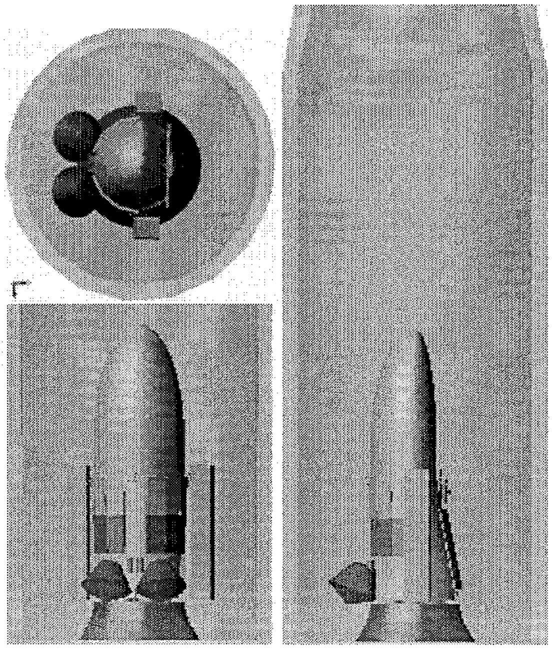

Figure 2. Delta IV H Launch Configuration

Table 2. Launch Mass Summary

Mass in kg	CBE	Cont	MEV	Marg	Alloc
Launch Capability					**5964**
Launch Reserve				10.7%	576
Launch Wet Alloc					**5388**
SEP LV Adapter	48	30.0%	62	12.2%	70
Xenon	928	13.0%	1048	0.0%	1048
SEP Dry Mass	1134	29.5%	1468	20.0%	1762
Cruise Hydrazine			111		111
Cruise Probes	159	30.0%	207	20.0%	249
A/C Entry Alloc					**2147**
A/C Aeroshell/TPS	681	30.0%	885	20.0%	1062
A/C ACS Prop			22		22
A/C Peri Raise Prop			139		139
Orbit Wet Alloc					**924**
Orbit Prop			124		124
Orbit Dry Mass	524	27.3%	667	20.0%	**801**

CBE = Current Best Estimate
Cont = Contingency = (MEV-CBE)/CBE
MEV = Maximum Expected Value
Marg = Margin = (Alloc-MEV)/MEV
Alloc = Allocation

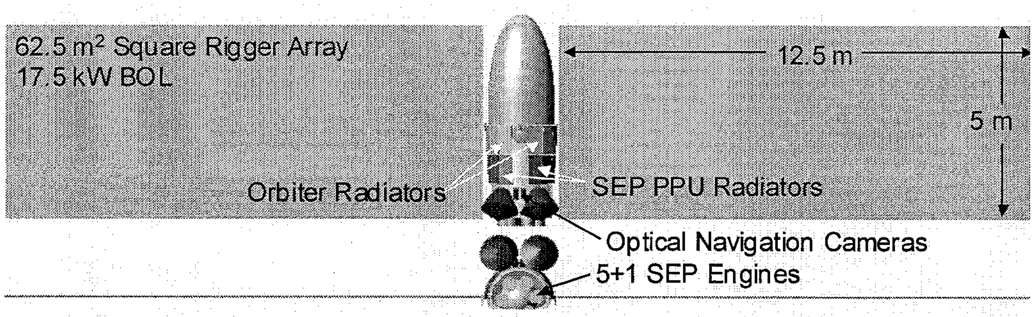

Figure 3. SEP Cruise Configuration

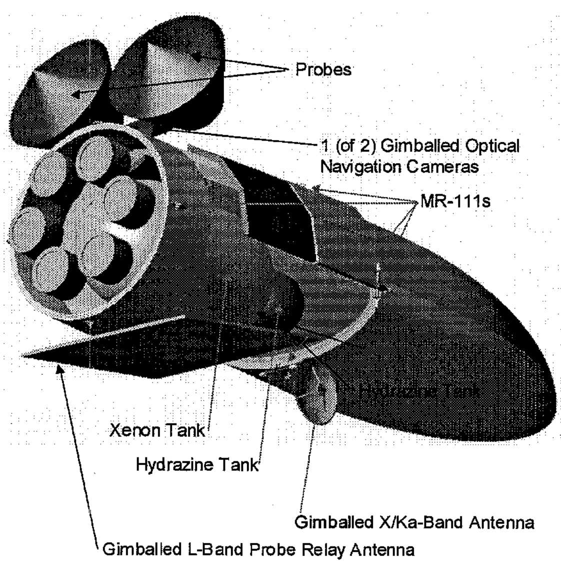

Figure 4. Post SEP Cruise Configuration

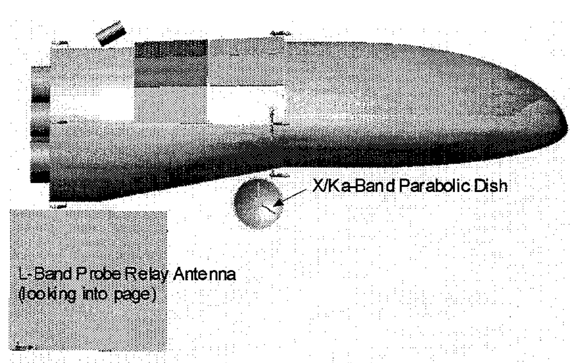

Figure 5. Probe Comm Relay Configuration

A. Key Mission System Trades

Several trades associated with how the orbiter interacts with the rest of mission system are worth mentioning. These trades do not represent a complete trade space for the Neptune mission.

1. Number of Probes and Probe Delivery

The Probes could be delivered before or after aerocapture. The volume constraints of the aeroshell led to the probes being carried outside the aeroshell, thus delivery prior to aerocapture. Three probes were desired at the start of the study. Although three probes could probably be accommodated on the launch vehicle, the relay telecom solution could not converge because the first probe would be too far away to provide a meaningful data rate. Three probes at one hour each versus two probes at 1.5 hours each could be a reasonable solution, but the probes would require higher ballistic coefficients which could complicate packaging.

2. Probe Lifetime

Because the probes are released prior to aerocapture, and the orbiter reaches Neptune with a speed of 29 km/s, probe lifetime equates directly to telecomm relay distance for the probe. 90 minutes was selected, along with the largest non-deployable L-Band antenna configuration, to achieve relay for two probes relayed in sequence.

3. Neptune/Triton Science Orbit

The desire to insert into an orbit that resonates with Triton's led to an aerocapture exit apoapsis that was so high (488,000 km) that the margin between aerocapture to that apoapsis and escape from the Neptune gravity well was too close to the performance capability of the aerocapture system. To achieve a comfortable aerocapture performance margin, the aerocapture exit apoapsis altitude was lowered to 430,000 km and propellant was added to the orbiter to allow the orbiter to raise the apoapsis to the desired science orbit.

V. SEP Stage Design

The SEP stage is designed to provide mission functionality from launch through Probe 2 EOL. The SEP stage relies on the orbiter for its flight computer and attitude control (reaction wheels). Structural mass was parametrically scaled against the orbiter/aeroshell mass (payload), Xenon mass, and other primary components. The SEP stage dry mass summary is shown in Table 3.

The "Flt" column of Table 3 specifies the number of line items in the detailed mass list for the respective subsystem. Articulation and Att Control includes gimbals and actuators for SEP thrusters, cameras, and antennas. Attitude control for the system is provided by the reaction wheels in the orbiter. The hydrazine propulsion system provides TCM capability and momentum de-saturation. For Telecomm, pre-aerocapture antennas and L-Band Probe relay radios are included; all X/Ka band radio equipment resides in the orbiter. The Power system is mostly solar arrays. For Thermal, all the SEP PPU radiators have louvers.

Table 3. SEP Stage Dry Mass Summary

Mass in kg	Flt	CBE	Cont	MEV
SEP Dry Mass	**193**	**1133.8**	**29.5%**	**1468.3**
Articulation & Att Control	16	46.4	25.2%	58.1
Telecom	11	16.4	30.0%	21.3
Power	5	319.2	29.7%	413.8
Propulsion	25	288.5	30.0%	375.1
Structure	32	310.8	30.0%	404.0
Cabling	28	69.1	30.0%	89.8
Hydrazine Propulsion	50	19.0	17.9%	22.4
Thermal	26	64.5	30.0%	83.9

VI. Orbiter Design

The Orbiter Flight System (OFS) is a single fault tolerant system, except for structure, dual stage RTGs, and antennas, with dual string block redundant avionics and selective cross strapping. The OFS is a 3-axis stabilized spacecraft. The OFS has two primary configurations: aerocapture and orbital science. Figures 6, 7 and 8 illustrate the different configurations. Although all mechanism mass was include in the mass lists, Figures 6, 7, and 8 do not include some mechanism detail such as the HGA two axis gimbal assembly and the instrument platform two axis gimbal assembly. Table 4 summarizes the aerocapture system mass, defined for this paper to include the aeroshell structure, TPS, and propellant required for bank angle control and post aerocapture orbit adjust to reach the initial science orbit. Table 5 summarizes the orbiter post aerocapture dry mass.

The aerocapture system is 57% of the total entry mass. This include the TPS and aeroshell structure (~48% mass fraction) and enough propellant to perform bank angle control of the aeroshell during aerocapture and perform orbit adjust maneuvers post-aerocapture to achieve the desired science orbit of 4,000 x 488,000 km. Twenty-four 66N SCAT Bi-Prop engines are used to provide up to 7.5 deg/sec^2 acceleration for bank angle control. The bi-prop engines are positioned and balanced to provide spacecraft torques about the velocity vector (40 deg angle of attack) even with one jet failed.

Table 4. Aerocapture System Mass

Mass in kg	Flt	CBE	Cont	MEV
Aerocapture System	**34**	**841.6**	**24%**	**1045.9**
Dry Mass	**34**	**681.0**	**30%**	**885.2**
TPS: Nose	1	204.0	30%	265.1
TPS: Windward	1	292.5	30%	380.3
TPS: Leeside Nose	1	0.6	30%	0.7
TPS: Leeside & Base	1	58.1	30%	75.5
TPS Adhesive	1	6.3	30%	8.2
Upper Structure	1	42.8	30%	55.6
Lower Structure	1	44.9	30%	58.4
Base Structure	1	7.2	30%	9.3
Separation Springs & H/W	26	24.7	30%	32.1
Propellant Mass		**160.7**		**160.7**
Bank Angle Control		21.9		21.9
4000x430000 dV (m/s)	360	115.5		115.5
4000x488000 dV (m/s)	78	23.3		23.3

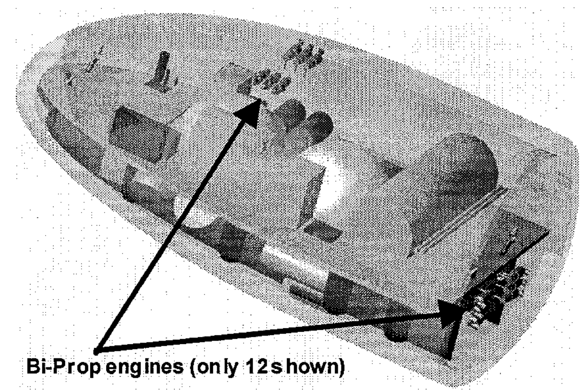

Figure 6. Aerocapture Configuration

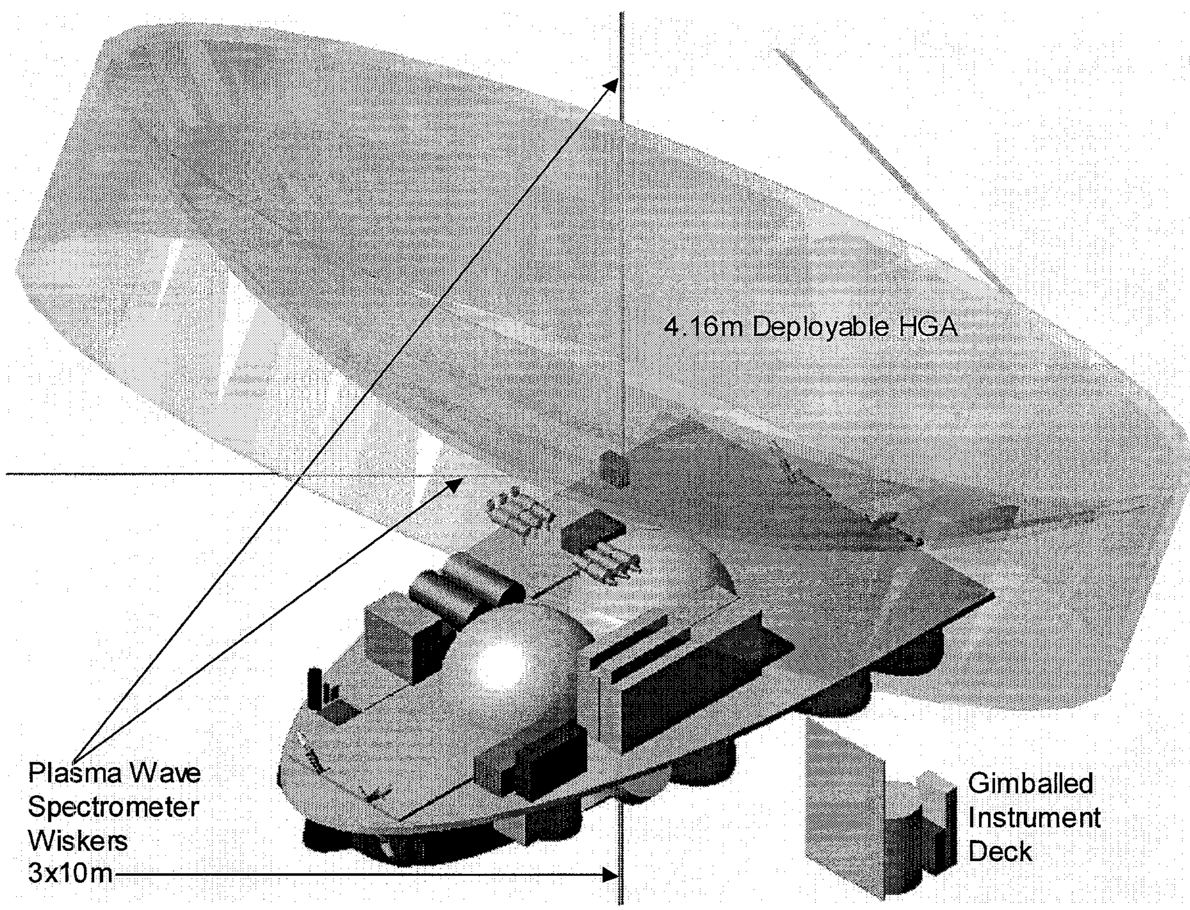

Figure 7. Orbit Configuration Isometric

Table 6 summarizes the orbiter power modes. The modes listed are not all the modes identified in the study, just the ones that stress the system. Heater power in all phases is minimal because of an aggressive assumption that the MMRTG excess heat, ~2600W, can be distributed across the spacecraft well enough to not require the heater power typical for deep space missions. The available power listed is the power output of two dual stage MMRTGs (14% efficiency) with 1.5% output degradation per year. It is assumed that once the telecom and instrument components have been turned on, that they are never turned completely off, but rather are placed in a low power standby mode when not in use.

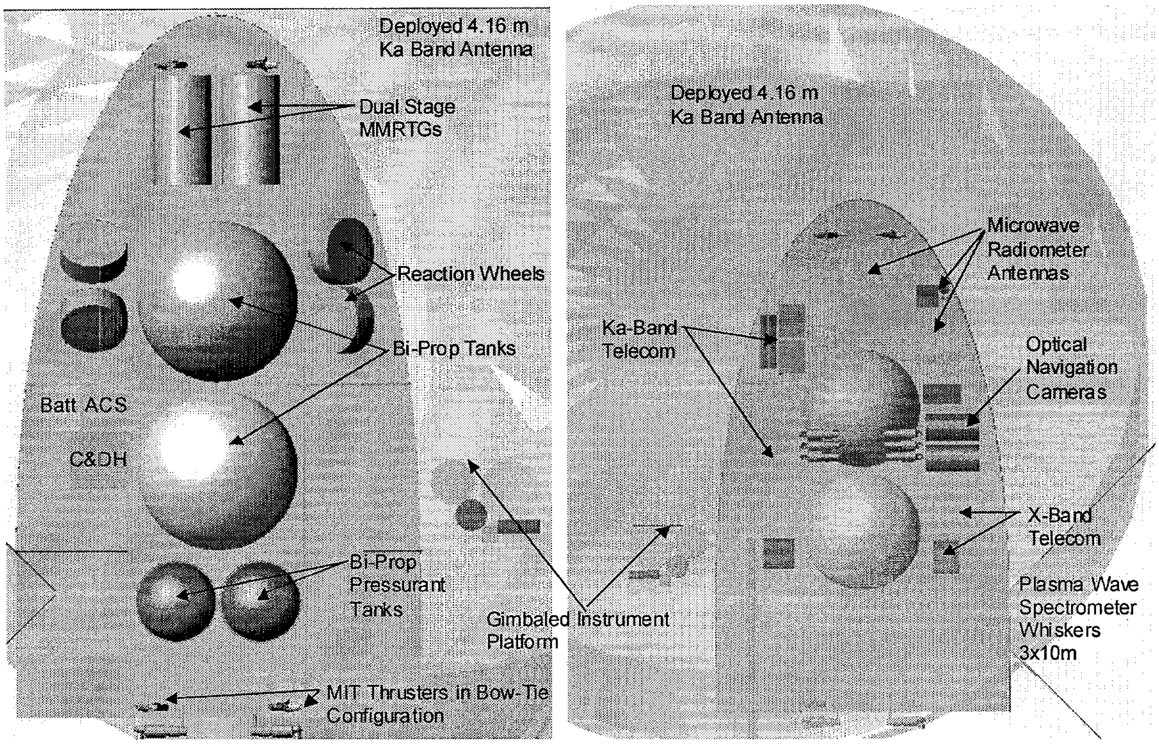

Figure 8. Orbit Configuration, Top/Bottom Views

Table 5. Orbiter Dry Mass Summary

Mass in kg	Flt	CBE	Cont	MEV
Orbiter Dry Mass	**236**	**524.2**	**27.3%**	**667.5**
Instruments	20	33.8	30.0%	43.9
Articulation & Att Control	28	53.7	12.0%	60.2
Command & Data Handling	32	34.7	27.0%	44.0
Power	4	64.7	30.0%	84.1
Telecom	23	41.9	28.5%	53.8
Structure	10	96.6	30.0%	125.5
Dual Mode Bi-Propulsion	118	104.9	27.5%	133.7
Thermal	1	54.0	30.0%	70.2
Harness	1	40.0	30.0%	52.0

Table 6. Orbiter Average Power (W)

	Mission Phase			
Component	**Relay**	**Aerocap**	**Science**	**Comm**
Instruments	0	0	30	24
ACS	88	69	78	78
C&DS	24	38	24	24
Telecom	90	13	13	38
Propulsion	0	90	50	50
Thermal	20	0	20	20
CBE Totals	222	210	214	234
MEV (20%)	**266**	**252**	**257**	**280**
Mission Year	10	10	12	12
Available Power (W)	**357**	**357**	**344**	**344**
Margin	34.2%	41.9%	34.1%	22.8%

B. Key Flight System Trades

1. Aeroshell Shape and Size

A discussion of the various aeroshell shapes and sizes analyzed for this study is extensive out of scope for this paper, but are discussed in detail in Ref 5. The desired aeroshell L/D, ballistic coefficient, and stability effect shape, size, and center of mass placement for the aeroshell; all of which affect volumetric and mass packaging efficiency of the internal components. For this study, a 0.8 L/D flattened ellipsled was selected.

Although the flattened ellipsled has the best volumetric efficiency (volume/surface area), the center of mass (CM) constraints associated with the desired angle of attack (40 deg, tends to want CM close to mid point between nose and tail) and stability (tends to want CM to be lower than the widest part of aeroshell) yield a large volume in the upper back region of the aeroshell that cannot be effectively utilized because of the need to offset any mass at the back of the aeroshell with mass in the nose. Although the 0.8 L/D flattened ellipsled works for this study, there may be opportunity to reduce overall vehicle mass by using some of the other alternative shapes examined.

2. Probe Relay

The communications frequency (L-Band chosen), probe lifetime (total vertical descent as function of ballistic coefficient), and probe latitude / longitude placement (concurrent vs separate relay, antenna pointing error) all drive the relay antenna(s) design. The current L-Band design is a compromise between UHF (good visibility through atmosphere, but antenna too large) and X-Band (terrible visibility through atmosphere, but reasonable antenna size). Because the probes are not tracked after release, the communications link has to accommodate the expected pointing error over the duration of the probe lifetime (+/- 3 deg).

Deployable UHF or L-Band antennas could be used to increase useable surface area on the SEP stage and overall data capability.

3. Orbiter Structure

The primary structure of the orbiter had to accommodate launch modes and stresses, aerocapture stresses, and mechanical constraints associated with separating from the LV, the SEP stage, and aeroshell. It is possible that a series of smaller decks oriented in the horizontal plane with respect to the launch configuration might provide a better foundation for efficient spacecraft packaging.

C. Subsystem Descriptions

Orbiter subsystems will be discussed in order of overall system impact. In general, subsystems discussed first drive the system design more than those discussed last.

1. Aerocapture System

The aerocapture system is defined as the TPS, the underlying aeroshell structure, and the propellant required to achieve the desired science orbit. The aerocapture system structure, TPS and their associated aero-thermal design basis are described in more detail in Ref 8-10. In summary, the aerothermal, aerodynamic, guidance, control analysis, and structural mass converged on a 2.88m flattened ellipsled with a L/D of 0.8.

The TPS was separated into 4 constant thickness zones to save mass: nose, windward side, forward leeside, and aft leeside (which includes the baseplate). For aeroshell jettison after aerocapture, the nose and windward zones were considered a single unit. The base plate would be jettisoned first, and the leeside and windward sides would be jettisoned concurrently.

24 bi-prop engines were used for bank angle control. This large number of engines allows discrete levels of torque to provide a range of small impulse corrections to large accelerations (7.5 deg/sec2). The engine configuration also provides redundancy for one engine out capability. These engines remain with the Orbiter after the aeroshell is jettisoned to eliminate the need for a separate aerocapture propulsion system.

Approximately 161 kg of propellant is required for bank angle control during aerocapture, periapsis raise maneuver at the first apoapsis after aerocapture, and an apoapsis raise maneuver to place the orbiter in an orbit properly phased for a Triton flyby resonance.

2. Telecom

The Orbiter telecom system includes X, Ka, and L-Band components. The L-Band components are required for atmospheric probe relay; L-Band is capable of penetrating Neptune's atmosphere at the desired science altitudes. The X and Ka bands are used for Earth Communication, X for safe mode and Ka for science data return. Table 7 summarizes the driving data return links. The L-Band antenna is approximately 1.6 m x 1.6 m with a mass of about 8 kg. This antenna is attached to the SEP stage with a single axis gimbal to point to the probes during probe entry.

The Ka link uses a 35 W TWTA in combination with a deployable 4.16m HGA (59.6 dB) and is capable of returning ~270 Gbits of data assuming 4 hours of comm. per day to a 70m DSN station. This provides ample margin to the 151 Gbits of planned science data. X-Band link uses a 15 W SSPA with a small dish antenna (32.2 dB) to provide Earth communication up to SEP stage separation. The L-Band probe relay link uses a 20W SSPA with a 25.1 dB antenna up to 324 Kbits total data return from the first probe and up to 648 Kbits total from the second probe.

Table 7. Orbiter Telecommunications Links

Probe to Orbiter Relay					Probe			Orbiter	
Mission Phase	Freq	Max Range	Data	Excess	Power	Ant	Off	Ant	Off
Probe 1 Relay	L-Band	255000 (km)	80	3	20	0	80	25.1	3
Probe 2 Relay	L-Band	133000 (km)	120	4	20	0	80	25.1	3

Orbiter to Earth					Orbiter			DSN
Probe Relay	Ka-Band	31	400	3	35	41.6	0.12	70M
Launch	X-Band	0.25	12.5	3	15	2	45	70M
Cruise	X-Band	31	100	3	15	32.2	0.36	70M
Orbit Science	Ka-Band	31	25600	3	35	59.6	0.01	70M

3. Power

Two dual stage Mutli-Mission Radioisotopic Thermal Generator (MMRTG) units, generating 3000W thermal, were selected for the Orbiter power source. The single stage MMRTG is planned for TRL 6 by 2006, laboratory tests currently have the dual stage technology at TRL 2-3 and funding profiles plan TRL 6 by 2010. The expected performance of the dual stage MMRTG is approximately 14% efficiency with 1.5% degradation per year providing 420W at beginning of life and 344W at end of life.

Secondary batteries are included to help during peak periods with a typical assortment of battery charge controllers, power switching, and power conversion electronics.

4. ACS

The Orbiter is a 3-axis controlled spacecraft that uses reaction wheels for attitude pointing and hydrazine thrusters for attitude maneuvers and reaction wheel de-saturation. The primary optical instrument deck is on a two axis gimbal as is the deployed HGA. Sun sensors are not utilized because of there no time critical sun point requirement (MMRTG power source). Star trackers are utilized in an orthogonal mount configuration to provide better attitude knowledge. Both the star trackers and the IMU have aggressive mass and powered consistent with low TRL units that are funded for achieving TRL 6 in the next decade. The current configuration could easily handle the mass of conventional units.

5. Propulsion

The propulsion system is dual mode bi-prop system with a re-pressurization system. Hydrazine thrusters are used for short and infrequent pulse ACS duties; bi-prop engines are used for longer duration impulses such as aerocapture bank angle control and orbit adjust maneuvers. Thruster configurations are designed to accommodate a single thruster failure by removing an entire thruster string from the system.

6. C&DH

The C&DH system is based on a RAD750, 3U Compact PCI implementation that is currently offered in various forms by multiple vendors. Although some of the cards required may not yet exist in 3U format, it is assumed that they will by time of mission implementation, or can be developed with little difficulty.

7. Thermal

The mission design presents several challenges for the thermal design:

1. The MMRTGs together generate ~2600W of waste heat.
2. The MMRTGs are enclosed in an aeroshell designed to keep heat from getting in (making it harder to get heat out).
3. The radiator system has to be designed to work before, during, and after aerocapture
4. Inside the aeroshell, the system will experience solar distance of 0.7 AU (Venus) to 31 AU.

Titanium and water loop heat pipes (LHP) running to hot radiators mounted on the SEP stage were chosen to solve the problem of getting the heat out of the aeroshell. Aluminum and ammonia loop heat pipes were also added to transport Orbiter electronics heat out of the aeroshell. A second set of titanium / water LHP carries MMRTG heat directly to the hydrazine tank.

From the previous year's Titan aerocapture study (Ref 12), a ~30 node lumped mass model of a spacecraft in an aeroshell using these LHP concepts was constructed to compute temperature distributions during the key mission phases for various design options. The computational model results confirmed that all of the key avionics and propulsion components were maintained well within prescribed operating temperatures during both the cruise to Saturn and after orbit insertion when the aeroshell was jettisoned and the orbiter exposed to the cold space environment at Titan. Although this analysis was not updated for the Neptune spacecraft, the results of the Titan study provide confidence that a similar design can do the job.

8. Structure

The structure is discussed in more detail in Ref 10. In general, the Orbiter primary structural design was driven by:

1. LV frequencies and loads.
2. Aerocapture loads.
3. Aeroshell separation planes.
4. Desired aeroshell center of mass.

VII. NEW TECHNOLOGY DEVELOPMENT

Other than aerothermal analysis tools and TPS manufacturing techniques, there were no enabling technologies identified to implement the flight system. The In Space Propulsion Program is continuing funding to improve the aerothermal analysis tools. The dual stage MMRTGs, deployable Ka band HGA, and deployable square rigger solar arrays are strongly enhancing technologies, but these technologies are already independently funded for development and there are other options to implement the mission if these technologies do not become a reality.

VIII. RECOMMENDED ADDITIONAL ANALYSIS

Many questions and trades consistent with continued Phase A/B efforts were identified by the study team. A summary of these issues is presented below along with a general classification of the issue as a lien, or opportunity, or either.

- **Launch Vehicle**: There is so much unallocated margin on the Delta-IV Heavy that an Atlas V 551 could be feasible, especially if an Earth gravity assist is considered (opportunity).
- **Science Instruments**: 1) Develop conceptual designs for instruments and verify TRL, mass, power, volume estimates (Lien). 2) Verify optical, and radiative fields of view for all instrument (especially thermal radiative for instrument on gimbaled deck, and RF for microwave radiometer antennas).
- **Power**: 1) Develop detailed power modes and profiles (either). 2) Verify 2 MMRTGs are adequate for full mission (Lein). 3) Verify EMI/EMC compatibility for component configuration (either).
- **Thermal**: Verify MMRTG heat can be effectively routed to other spacecraft components to eliminate need for heaters (Lein).
- **Telecom**: 1) Add LGA/MGA for Earth acquisition prior to high bandwidth links (Lein).
- **Aeroshell**: 1) Re-investigate best shape for volumetric efficiency including center of mass location (opportunity). 2) Verify heating and TPS for new ballistic coefficient (either).
- **Cost**: Generate cost estimate for complete flight system (lien).

IX. CONCLUSIONS

The study demonstrates general technical feasibility for a Neptune Orbiter flight system designed to use aerocapture as the orbit insertion mechanism. Many liens exist against the conceptual design presented, but opportunities and large launch mass margins balance the liens. Technology readiness for the flight system is good with all major components currently being funded to achieve TRL 6 in the next decade.

X. ACKNOWLEDGEMENTS

Personnel (authors for Ref 1-10) from Langley Research Center, Johnson Space Center, Ames Research Center, Marshall Space Flight Center, and the Jet Propulsion Laboratory were instrumental in determining system level requirements and subsystem capabilities for trade and mission performance analysis.

Discipline experts at the Jet Propulsion Laboratory including Dave Hansen (Telecom), John Huang (Antennas), Ray Baker (Propulsion), and Bill Nesmith (MMRTGs) all provided valuable input into the subsystem conceptual designs represented in this paper.

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