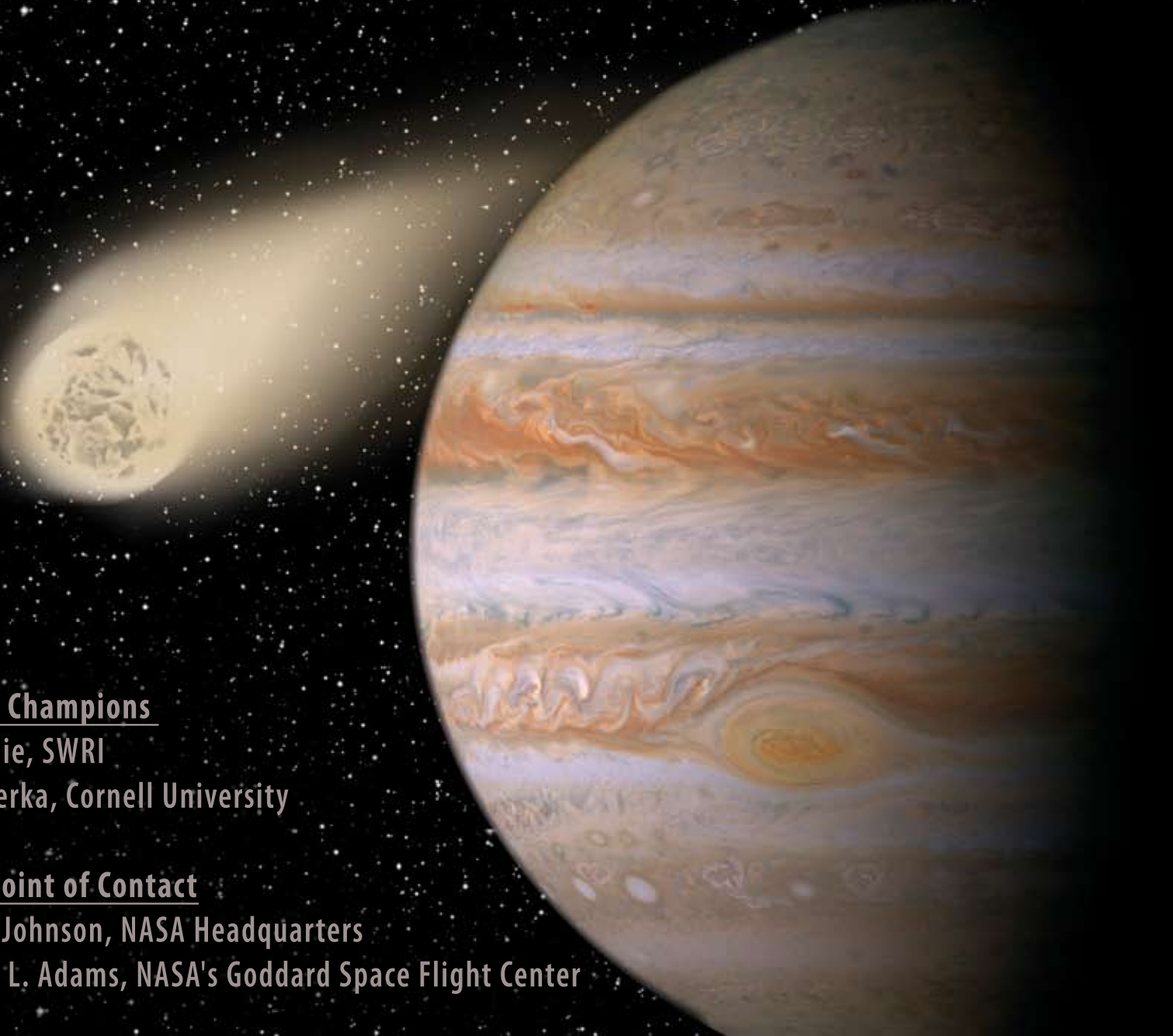


CHIRON

Orbiter Mission

Mission Concept Study Report to the NRC Planetary Science Decadal Survey, Primitive Bodies Panel



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Chiron Orbiter Mission Study

Final Report

Presented to the Planetary Decadal Survey Steering
Committee and Primitive Bodies Panel

4 May 2010

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Contents

Chiron Orbiter Mission Executive Summary.....	1
1.0 Scientific Objectives.....	1
2.0 Chiron Orbiter Mission Concept.....	2
2.1 Overview	2
2.2 Concept Maturity Level.....	3
2.3 Preliminary Concepts	3
2.4 Final Concept Selection	4
3.0 Architecture Options.....	5
3.1 Final Option 1 – All Chemical Propulsion.....	6
3.2 Final Options 2 and 3 (2019 and 2023) – Solar Electric and Chemical Propulsion.....	6
3.3 Final Option 4: 6 ASRG Radioisotope Electric Propulsion	7
3.4 Final Option 5: 2 HP ASRG Radioisotope Electric Propulsion.....	7
4.0 Chiron Orbiter Mission Elements.....	8
4.1 Instrument Payload.....	8
4.2 Trajectories	8
4.3 Spacecraft Subsystems.....	9
5.0 Concept of Operations and Mission Design.....	11
5.1 Data Downlink/Uplink/Tracking.....	11
5.2 Ground System.....	12
5.3 Risk	12
6.0 Schedule.....	12
6.1 Chiron Orbiter Mission Schedule.....	12
6.2 Technology Development Plan	12
7.0 Mission Life-Cycle Costs.....	14
7.1 Costing Methodology and Basis of Estimate	14
7.2 Cost Estimate	14
8.0 Recommendation for Future Concepts	15
8.1 Optical Communication.....	15
8.2 Liquid Oxygen Propulsion for the Inter-Planetary Spacecraft.....	15
9.0 Conclusion	16
Acknowledgements	17
Appendices	
Appendix A - Propellant Mass Parametric Study for Preliminary Concept Options A, B & C	
Appendix B - Radioisotope Electric Propulsion Trade Summary	
Appendix C - References	
Appendix D - Acronym List	

CHIRON ORBITER MISSION

Executive Summary

The National Research Council's Planetary Science Decadal Survey: Primitive Bodies Panel has commissioned NASA's Goddard Space Flight Center (GSFC) with a rapid architecture study conducted under NASA Headquarters leadership. The purpose of the study is to determine whether a mission to orbit the Centaur Object Chiron can be achieved within a New Frontiers cost range with a limited supply of Advanced Stirling Radioisotope Generators (ASRGs). Guidelines include a launch between 2015 and 2025, a limit of two ASRGs for the mission, and remain within a New Frontiers cost range (FY15, \$800 M excluding launch vehicle). At the request of the Primitive Bodies Panel, radioisotope electric propulsion options using more than two ASRGs were also analyzed. The Chiron Orbiter Study was conducted by GSFC's Architecture Design Laboratory (ADL) in partnership with the NASA Glenn Research Center's (GRC) Collaborative Modeling for Parametric Assessment of Space Systems (COMPASS) design center.

Chiron is the first object discovered in its class, known as Centaurs. There is one specific behavior Chiron exhibits that sets it apart from all other known Centaurs: it is known to sporadically display cometary behavior. The length of the observational history for Chiron is an important consideration. The observations demonstrate that Chiron can show activity at any point in its orbit (whether at perihelion or at aphelion). The activity is not continuous at a level that can be monitored from the Earth. Outburst phases are seen to last for many years at a time while apparently quiescent phases can last just as long. The scientific goal of the mission is to study this object and its activity over time. While a fly-by mission to observe Chiron is less complex than placing a satellite in Chiron orbit, it does not meet the science objective to observe and measure sporadic cometary behavior. The time-scales of variation

suggest that an in-depth investigation would not be possible without an orbiting platform that is in place for at least a few years. The notional instrumentation chosen for this mission study reflect a general need for basic remote sensing observations ranging from imaging, spectroscopic, in situ sampling, and gravity measurements. The driving challenge for this mission is getting into Chiron orbit. It remains for future studies to quantify the optimal payload that should be carried but it is clear this will need to be a relatively small spacecraft for its class.

This study focuses mainly on the propulsion and trajectories needed to orbit Chiron. While satellite subsystems are discussed, the ability to deliver sufficient mass to Chiron for scientific observation is the main study theme. Five options that describe the type of propulsion system needed to place a satellite into orbit around Chiron are presented in this report. Table 1 summarizes the study results and shows that a mission to orbit Chiron is feasible. The chemical and solar electric/chemical propulsion systems fall within the New Frontiers cost range; however, delivered mass to Chiron of these systems will require a reduction in science instruments or a larger launch vehicle. The radioisotope propulsion systems deliver sufficient mass to achieve all the science objectives; however, these systems fall outside of the New Frontiers cost range.

Finally, while it is difficult to deliver a sufficient mass to Chiron in a New Frontiers cost range; other Centaur objects are closer to Earth and a satellite of sufficient mass could be delivered to these objects as New Frontiers missions. The study team examined the Centaur objects Okyrhoe and Echeclus and determined orbit parameters to these Centaurs. These results are presented in Table 2.

1.0 Scientific Objectives

Centaurs are in orbits that lead to non-trivial interactions with one or more Jovian planets but have perihelia larger than Jupiter's orbit. These interactions render all Centaur orbits unstable

Table 1: Final options for Chiron Orbiter

Option	Propulsion Type	Launch date	Transit time (years)	Total Delivered Mass (kg)	Instrument Mass (kg)	Cost Bin
1	All Chemical	2/2019	13	342	56	New Frontiers
2	Solar Electric/Chemical	3/2019	13	528	56	New Frontiers
3	Solar Electric/Chemical	4/2023	13	576	56	New Frontiers
4	Radioisotope (6 ASRGs)	5/2024	13	831	72	Flagship
5	Radioisotope (2 High Power ASRGs)	5/2025	11	737	76	Flagship

Table 2: Orbit Trajectories for Other Centaur Objects

Centaur	Launch date	C3 (km ² /s ²)	Cruise (years)	VHP (km/s)	Note
Okyrhoe	2019	84	9 yr	3 m/s	Delivery of sufficient mass w/Chemical propulsion possible
Echeclus	2022	88	11 yr	3.8 m/s	Similar properties to Chiron transit

meaning that any of these objects have only recently been placed there. These objects are of importance to the current dynamical structure of the solar system and are tools for understanding their source population in much the same way that we use comets to trace back to their source. In principle, any Centaur is an important window into this group of objects and thus their source regions. This study was specifically targeted at an in-depth exploration of Chiron even though there are other Centaurs that are easier to get to. There is one specific behavior Chiron exhibits that sets it apart from all other known Centaurs: it is known to sporadically display cometary behavior. Furthermore, this activity cannot be driven by water-ice sublimation as is the case for normal cometary activity. It may well be true that other Centaurs could do the same thing but so far only Chiron has been caught in the act. The length of the observational history for Chiron is also an important consideration. These observations tell us that Chiron can show activity at any point in its orbit (whether at perihelion or at aphelion). Also, we know that activity is not continuous at a level that can be monitored from the Earth. Outburst phases are seen to last for many years at a time while apparently quiescent phases can last just as long.

The Chiron Orbiter Mission studies the atmosphere, surface, and interior of the Centaur Object Chiron and monitors the object's nature and level of outgassing activity. The primary scientific goal is to assess the feasibility of an orbiter with an 80 kg science payload capable of orbiting Chiron at distances needed to achieve 10 meter im-

aging resolution. The notional science payload is made up of the following: multispectral imager, IR mapping spectrometer, ion and neutral mass spectrometer, UV spectrometer, thermal mapper, and radio science. The target launch date for the Chiron Orbiter Mission is between 2015 and 2025. Table 3 shows Chiron Orbiter scientific goals and objectives in priority order.

2.0 Chiron Orbiter Mission Concept

2.1 Overview

Due to the time limits of a rapid architecture study, the Chiron orbiter team focused mainly on the biggest challenge of how to get to Chiron. Chiron is in a heliocentric orbit with an aphelion outside of Uranus' orbit and a perihelion inside of Saturn's orbit. Figure 1 shows Chiron's orbit. Various propulsion types for getting to Chiron are examined. These include; all chemical propulsion, solar electric/chemical propulsion, and radioisotope electric propulsion.

A five step process is followed for the Chiron orbiter study:

1. To characterize the trade space, five initial architecture options are formulated based on mission parameters and propulsion type. These are named preliminary Options A-E. The preliminary Options are described in Section 2.3. The total spacecraft mass delivered to Chiron is determined for each option.
2. Using the preliminary option results to narrow the trade space, five final architecture options are developed and described in Section 3.0.

Table 3: Chiron Orbiter Mission Scientific Goals and Objectives

Priority #	Goals	Objectives
1	Observe current geologic state and composition of surface and infer past evolution and relative importance of surface processes.	A. Near global imaging of surface at 10 meter resolution, including albedo and color maps. B. Spectral mapping of surface in the 1 to 5 micron range. C. Thermal mapping of surface.
2	Observe and measure the sporadic outgassing activity and determine the composition of outgassed volatiles.	A. Monitor outgassing through imaging to determine locations, nature of, and rate of activity of active areas. B. Obtain remote sensing measurements of transient atmosphere through IR and UV spectroscopy. C. Obtain INMS in situ measurements of emitted gases.
3	Characterize bulk properties and interior structure.	A. Determine mass and internal mass distribution from radio tracking. B. Determine size, shape and volume from imaging. C. Determine and monitor rotation state through repeated imaging of landmarks.

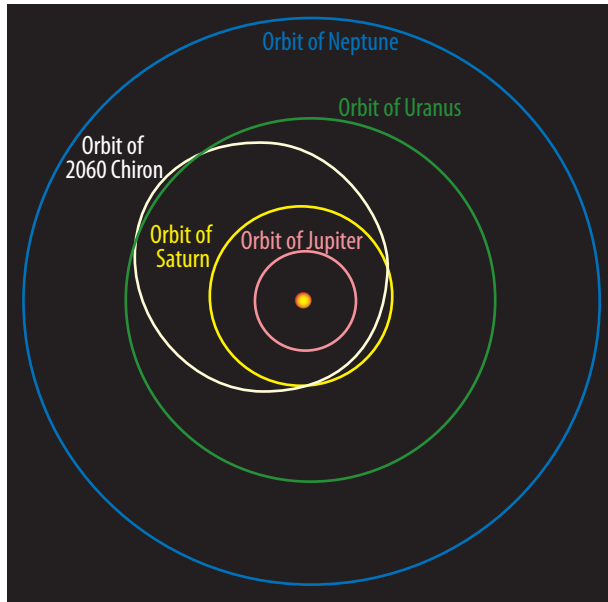


Figure 1: Chiron's orbit around the sun

These are named final Options 1–5. The total delivered mass of each final option is determined.

3. After the total delivered mass is determined, the subsystem mass for each final option is derived using a parametric mass allocation tool.
4. A strawman spacecraft is developed that meets mission requirements. The strawman spacecraft provides subsystem masses without regard for a specific propulsion type and is described in Section 4.0.
5. The parametric subsystem masses for each final option are compared to the strawman subsystem masses to determine the feasibility and merit of each final option.

Launch occurs between 2019 and 2025, depending on propulsion option. After initial spacecraft checkout, the satellite begins its cruise phase. The cruise phase duration is between 11 and 13 years and includes planetary flybys for some of the options. After the cruise phase, the satellite enters into orbit around Chiron and begins a 3 year observation period. During the observation period, data is transmitted to Earth.

A primary constraint for this study is the requirement that no more than two ASRGs are available for the mission due to Plutonium availability. This limits the power available at Chiron to approximately 268 Watts [see ASRG Functional Description 912C002144, October 2009, Figure 4.2-1]. An Atlas V 551 is baselined as the launch vehicle. Additionally, the Primitive Bodies Panel requested that the study team evaluate

configurations with an unconstrained number of ASRGs to bound the “best possible” limit of the trade space. One of these unconstrained ASRG configurations uses High Power (HP) ASRGs. The HP ASRGs are in development and will require additional resources to raise them to a Technical Readiness Level of 6; however, the HP ASRGs provide 550 W of power, well above the 134 W provided by a standard ASRG. The ASRGs also produce a constraint on the available transit time to Chiron. The total ASRG life is defined as 17 years, with 3 years of that being consumed by ground operations prior to launch. The study parameters require a 3-year science mission. Therefore, the available transit time is defined as 11 years (17 year life minus 3 years ground operations and 3 years science mission).

2.2 Concept Maturity Level

Upon receiving the Chiron Orbiter study questionnaire, the study team rated the concept at a Concept Maturity Level 2, Initial Feasibility. Previous studies and flight missions had shown it was technically feasible to orbit an object in the outer solar system. Studies to orbit Centaur objects have been performed (Oleson, et. al.); however, specific studies to orbit Chiron using various propulsion types did not exist. For this study, the team examined various propulsion methods for placing a spacecraft in orbit around Chiron and evaluated the performance and cost of these methods. Upon completing this study, the architecture and objectives trade space for a Chiron orbiter have been evaluated for cost, risk and performance. The concept is now at Concept Maturity Level 3, Trade Space Evaluated.

2.3 Preliminary Concepts

The various propulsion types for a mission to Chiron are classified into five preliminary options. These options are:

- A - Star 48 solid rocket/Chemical propulsion
- B - All Chemical propulsion
- C - Chemical/Solar Electric propulsion
- D - Solar Electric/2 ASRG Radioisotope electric propulsion
- E - 6 ASRG Radioisotope electric propulsion

An engineering assessment was performed on all preliminary options. Seventy-seven launch date/transit time cases for preliminary Options A and B were run. The figure of merit for these cases was the total mass delivered to Chiron orbit. These runs reveal the amount of delta-velocity (Delta-V) arriving at Chiron is large, ranging from 4

to 7.7 km/sec. These large delta-Vs make single stage braking into Chiron orbit non-optimal for preliminary Option A. Additionally, the 11-year transit time further exacerbated the arrival delta-V to the point that the available mass delivered to Chiron is not feasible for the mission. The runs also reveal preliminary Option B design does not close because no mass can be delivered to Chiron with this option. Preliminary Option B is therefore dropped from further consideration.

For preliminary Option C, the NASA team defined a notional solar electric propulsion (SEP) upper stage for the Atlas V 551 that utilizes an Earth gravity assist (to stay closer to the sun and hence, thrust longer) and a major planet gravity assist. In this scenario, the spacecraft would then cease thrusting and be ejected in the vicinity of the Main Asteroid Belt. Seven trajectories with an 11-year transit time were defined and a chemical propulsion upper stage and satellite were sized for each. This option produces higher useable masses for the Chiron Orbiter satellite than all-chemical options (preliminary Options A and B). However, this option also shares the high arrival delta-Vs, hence one-stage braking to Chiron orbit is not optimal and the 11-year transit times also exacerbate the delta-Vs, which make the trip time unrealistic.

For preliminary Option D, analysis reveals that the engines of a 200 W electric propulsion system produce very little thrust. In-depth analysis of this option in the study time frame is impractical. Therefore, preliminary Option D is dropped from further consideration.

Initial sizing runs of preliminary Option E (all radioisotope electric propulsion and six ASRGs), leveraging NASA GRC's prior concept of a Centaur Orbiter, show the 11-year transit time is unfeasible due to the low mass delivered to Chiron.

Table 4 summarizes the findings and disposition for each initial option studied. An option that does not close means the delivered mass is so low that a viable satellite cannot be designed.

2.4 Final Concept Selection

Because none of the preliminary solutions deliver an acceptable mass to Chiron with an 11 year transit time, the Science Champions advised the study team to define configurations that deliver acceptable masses into Chiron Orbit. The team selected a 13-year transit time as the baseline, with the goal of finding some configuration(s) capable of an 11-year transit time. The Chemical, Chemical/SEP and Radioisotope electric propulsion types were re-evaluated using these new guidelines.

2.4.1 Chemical Propulsion Options

In re-evaluating the chemical propulsion systems, a second round of cases was analyzed for preliminary Option A using:

- Two stage liquid chemical braking into Chiron orbit.
- 40 kg current-best-estimate (56 kg with growth) science package.
- 13-year transit time to Chiron.
- Available satellite subsystem mass as the figure-of-merit for selecting an optimal design (total mass delivered to Chiron orbit minus propulsion mass, ASRG mass and science payload mass)

The study team baselined candidate instruments that match the strawman instrument package and are lighter than the 80 kg allocated to the science package in the original mission questionnaire.

The propellant mass allocation between the first and second braking stages is varied to find a configuration that produces the maximum satellite subsystem mass. Results of the propellant mass parametric study are provided in Appendix A. The mission description is discussed in Section 3.1.

2.4.2 Chemical/Solar Electric Propulsion Options

Preliminary Option C was re-evaluated using a 13-year transit with a launch date in 2019. Additionally, a Chemical/SEP option was examined that launched in 2023. A key element of this option is the use of SEP in the inner solar system to

Table 4: Initial Propulsion Options and Disposition

Option	Propulsion Option	Launch Date	Trajectory	Disposition
A	Star 48/Chemical	2019	Jupiter Assist	Does not close, but best all-chemical option
B	All Chemical	2019	Jupiter Assist	Dropped - Does not close
C	SEP/Chemical	2019	Earth-Jupiter Assist	Does not close but better performance than Option A
D	SEP/2 ASRG REP		N/A	Dropped – Small radioisotope electric engine not feasible
E	Star 48/REP	2024	Direct	Design closes

increase spacecraft mass delivered to Chiron. The SEP system is implemented in an SEP module. After launch and initial checkout, the SEP system is operated for up to 3 years, and in combination with an Earth gravity assist, puts the spacecraft on a transfer trajectory to Chiron. After completing the electric propulsion phase, the SEP module is separated from the science spacecraft. This architecture has shown to be beneficial for a variety of outer solar system destinations.

The SEP module for this study is based on the design proposed for the Titan/Saturn System Mission (TSSM), shown in Figure 2. TSSM is an outer planet flagship mission proposal developed by JPL with NASA GRC support. The module contains a NEXT-based ion propulsion system and two Ultraflex solar arrays providing approximately 15 kW of power at 1 AU. The design has three NEXT thrusters (two primary thrusters to perform the mission and one back-up). The module is controlled by the spacecraft processor, thus has limited other subsystems onboard. The TSSM SEP module mass was estimated to be 778 kg, including mass growth. The module has a xenon propellant capacity of 660 kg, which is more than sufficient for the Chiron mission. The TSSM design provides a sufficient baseline for this study; however, further detailed study could result in a Chiron mission optimized module sizing. Results are discussed in Section 3.2.

2.4.3 Radioisotope Electric Propulsion Options

The radioisotope electric propulsion (REP) systems were analyzed in more detail. The concept leverages previous studies to Centaur and Kuiper Belt object missions. This approach results in multiple conceptual point design spacecraft concepts. The REP concept delivers a science orbiter to Chi-

ron with arrival 11 to 13 years after launch, ending in a 3-year science mission. Appendix B lists the Chiron orbiter power system and total trip time trades run for this mission design. The study team looked at various RPS approaches including use of 6 or 8 standard ASRGs, as well as use of high power (HP) variants of the ASRG. The baseline 6 ASRG case is Case 1 and the 2 HP ASRG baseline case is Case 5 in the Appendix.

Both of the options, 6 ASRG and 2 HP ASRG, use the same set of major subsystem components. They differ only in propellant necessary to perform their different trajectories based on total delta-V requirements of the trajectory. The main propulsion system consists of two High Throughput BHT-600 Hall Long Life Hall engines with one extra for life and one cold spare, three (plus one spare) PPUs and an off-the-shelf hydrazine system with New Horizons heritage for reaction control (RCS). A Star 48B is used to perform the Earth Departure Burn. The casing is jettisoned after the burn. The sizing of the Star 48 assumed a 94% Propellant Mass Fraction. Results are discussed in Sections 3.3 and 3.4.

3.0 Architecture Options

The final concepts are renamed final Options 1–5 (Table 5) and analyzed to determine the total delivered mass to Chiron. A parametric mass allocation tool is used to allocate the total delivered mass to satellite subsystems resulting in subsystem masses for each option. The resulting subsystem masses are shown in Table 6. These parametric subsystem masses for each final option are then compared to strawman subsystem masses developed by the study team and described in Section 4.0. The feasibility and merit of each final option is determined by this comparison.

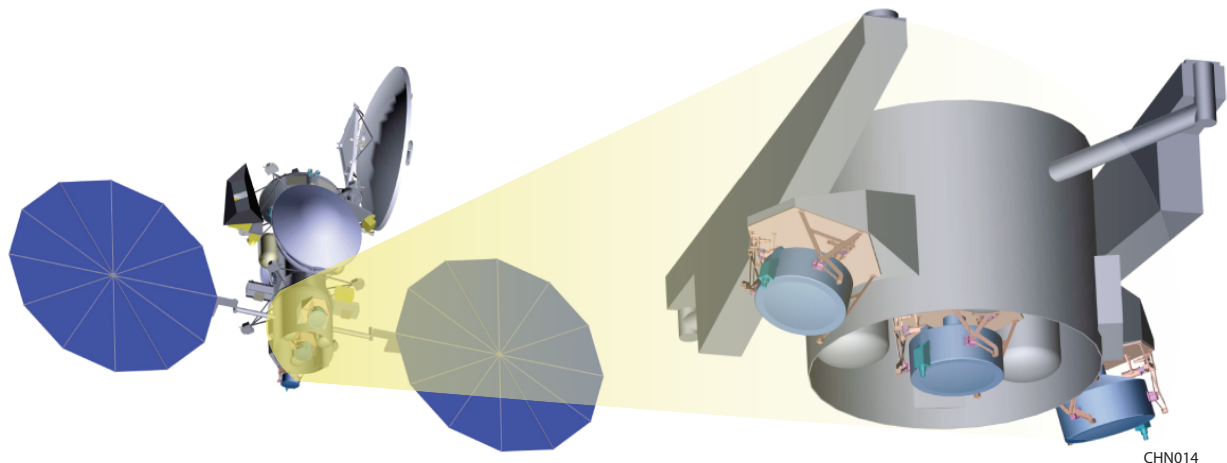


Figure 2: Conceptual SEP Module (Courtesy of JPL)

Table 5: Final Options for Chiron Orbiter

Option	Propulsion Type	Launch date	Transit time (years)	Total Delivered Mass (kg)	Instrument Mass (kg)	Cost Bin
1	All Chemical	2/2019	13	342	56	New Frontiers
2	Solar Electric/Chemical	3/2019	13	528	56	New Frontiers
3	Solar Electric/Chemical	4/2023	13	576	56	New Frontiers
4	Radioisotope (6 ASRGs)	5/2024	13	831	72	Flagship
5	Radioisotope (2 High Power ASRGs)	5/2025	11	737	76	Flagship

Table 6: Resultant Masses for Chiron Spacecraft Subsystems

Spacecraft Subsystem	Grassroots Mass Estimate (kg)	Final Option 1 Mass (kg)*	Final Option 2 Mass (kg)*	Final Option 3 Mass (kg)*	Final Option 4 Mass (kg)*	Final Option 5 Mass (kg)*
C&DH	45	16	29	33	46	46
GNC/ACS	20	16	29	33	22	22
Thermal	46 - 60	16	29	33	40	40
Comm	54	22	40	47	46	43
Power Dist'n	15	9	17	20	67**	75**
Total Subsystem	243	79	144	166	221	226
Instrument Suite	56	56	56	56	76	72

*-Mass with margin

** = Additional mass needed above grassroots to accommodate extra ASRG's

3.1 Final Option 1 – All Chemical Propulsion

Final Option 1 uses an Atlas V 551 with a Star 48 upper stage and launches in 2019. A liquid propulsion system decelerates the spacecraft into orbit around Chiron. Two ASRGs are used on the satellite for power production (no electric propulsion onboard). Initial sizing runs indicate that single stage braking into Chiron orbit is not optimal and two braking stages are needed. The total mass delivered to Chiron with this option is 342 kg.

As Table 6 shows, the final Option 1 masses are much lower than the grassroots masses. Some further optimization may be possible with development of a point design; however, this option is not feasible with the current instrument suite and launch vehicle.

3.2 Final Options 2 and 3 (2019 and 2023) – Solar Electric and Chemical Propulsion

Final Options 2 and 3 use a solar-electric upper stage to achieve the high C3 required for this mission, with a liquid propulsion system to decelerate the spacecraft into orbit around Chiron. Two ASRGs are used for power production (no electric propulsion onboard the satellite).

The solar-electric propulsion stage ends in the vicinity of the Main Asteroid Belt, when the solar intensity is too low to maintain efficient thruster firing. The trajectory is optimized to keep the solar-electric stage in the inner solar system as long as possible to maximize the time available for

solar-electric thrusting. This also allows an Earth gravity assist to further enhancing performance.

Initial sizing runs indicated that single stage braking into Chiron orbit is not optimal and that two braking stages are needed. The configurations of final Options 2 and 3 utilize this two-stage braking concept.

Two different launch date/gravity assist configurations were examined for solar electric and chemical propulsion:

- Final Option 2 - A 2019 launch with Earth and Jupiter gravity assists.
- Final Option 3 - A 2023 launch with Earth and Saturn gravity assists.

528 kg of total mass is delivered to Chiron with final Option 2. The mass breakout for this option is shown in Table 6. The resultant mass available for satellite subsystem nearly doubles that for final Option 1; however, most of the resulting spacecraft subsystem masses are lower than the grassroots generated subsystem masses. Some further optimization is possible with development of a point design.

Final Option 3 delivers 576 kg of total mass to Chiron. The mass breakout for final Option 3 is also shown in Table 6. This configuration is better performing than final Option 2 due to the higher delivered mass. The resulting subsystem masses come closer to the grassroots estimates, but are still low.

3.3 Final Option 4: 6 ASRG Radioisotope Electric Propulsion

Final Option 4 uses six ASRGs (0.578 kW to thrusters) to provide power to the spacecraft and propulsion system. Launched in 2024 on an Atlas V 551 with Star 48, the spacecraft begins a 13-year direct cruise and arrives at Chiron in 2037. The trajectory plot is shown in Figure 3. The trajectory parameter details are captured in Table 7. This option delivers 831 kg to Chiron.

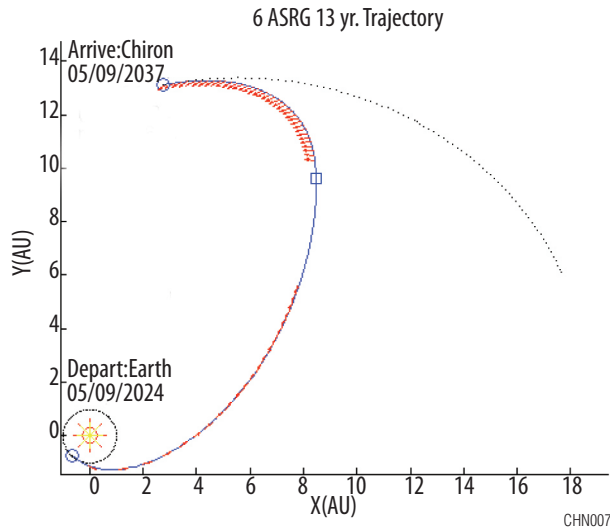


Figure 3: The red arrows indicate periods of thrusting along the trajectory.

Table 7: 6 HP ASRG Trajectory Parameter Details

Parameter	Value
Mass, Xenon total	440.6445 kg
Mass, Xenon Useable	405.75 kg
Mass, Xenon Nav. and Traj. Margin	20.2875 kg
Mass, Xenon Residuals	14.607 kg
Mass, Xenon Nav. and Traj. Margin	5.00 %
Mass, Xenon Residuals	3.60 %
Thruster Name	BHT-600
Thruster Efficiency	55.00 %
Specific Impulse	2229.52 s
Quantity, Number of Thrusters Operating	1
Average Power into the Thruster	0.578 kW
Time, Transfer to Chiron	4748.25 days
Time, Transfer in Years	13 years
Date, Launch	5/9/2024
Date, Jupiter Flyby	
Date, Chiron Arrival	5/9/2037
Launch Vehicle	Atlas 551 w/STAR 48
Energy, C3	102.6169 km ² /s ²
ELV Performance (pre-margin)	1302.42

3.4 Final Option 5: 2 HP ASRG Radioisotope Electric Propulsion

This spacecraft option used two hypothetical, high-power ASRGs (0.744 kW to thrusters) to provide power to the spacecraft, including the propulsion system. This approach reduces the electrical power system dry mass while using the same amount of plutonium as final Option 4. It is launched in 2025 on an Atlas V 551 and uses a Star 48 motor. Total cruise time is 11 years and arrives at Chiron in 2036 on a direct trajectory. The trajectory plot is shown in Figure 4. The trajectory parameter details are shown in Table 8. This option delivers 737 kg to Chiron.

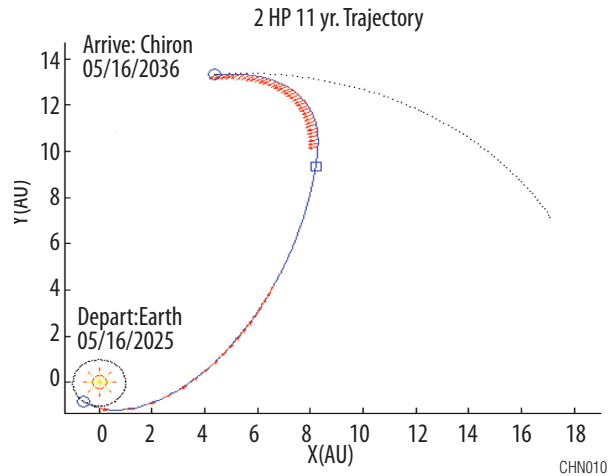


Figure 4: The red arrows indicate periods of thrusting along the trajectory.

Table 8: 2 HP ASRG Trajectory Parameter Details

Parameter	Value
Mass, Xenon total	543.3801 kg
Mass, Xenon Useable	500.3 kg
Mass, Xenon Nav. and Traj. Margin	25.0175 kg
Mass, Xenon Residuals	18.0126 kg
Mass, Xenon Nav. and Traj. Margin	5.00 %
Mass, Xenon Residuals	3.60 %
Thruster Name	BHT-600
Thruster Efficiency	55.00 %
Specific Impulse	1997.15 s
Quantity, Number of Thrusters Operating	1
Average Power into the Thruster	0.744 kW
Time, Transfer to Chiron	4017.75 days
Time, Transfer in Years	11 years
Date, Launch	5/16/25
Date, Jupiter Flyby	
Date, Chiron Arrival	5/16/36
Launch Vehicle	Atlas 551 w/STAR 48
Energy, C3	102.2121 km ² /s ²
ELV Performance (pre-margin)	1311.15 kg

4.0 Chiron Orbiter Mission Elements

The study team developed a strawman satellite that provides a framework to show that after a satellite has been placed in orbit around Chiron, the science objectives can be met. The strawman satellite subsystems are developed from previous missions and provide a baseline for mass that can be compared against the delivered mass for each option. Table 6 shows the strawman satellite subsystem masses compared against final Options 1–5 subsystem masses. Figure 5 shows a block diagram of the strawman satellite.

4.1 Instrument Payload

Table 9 details candidate instruments selected to fulfill the requested payload suite. All instruments have relatively recent heritage and are ca-

pable of meeting the measurement requirements for this mission. Changes (see the Notes column) are accounted for in the mass, power, and cost numbers. These numbers are based on CBE and do not include margin over heritage values. The total amount of data returned to Earth for this instrument suite is 102 Gbits. Table 10 details the data type and volume.

4.2 Trajectories

Trajectories were calculated for flights between Earth and Chiron starting in 2018. The team also investigated using Jupiter and/or Saturn flybys. There is generally one launch opportunity per year as the Earth moves into position for an efficient transfer orbit. Because Chiron is very far from the Sun (approximately 17 Astronomical

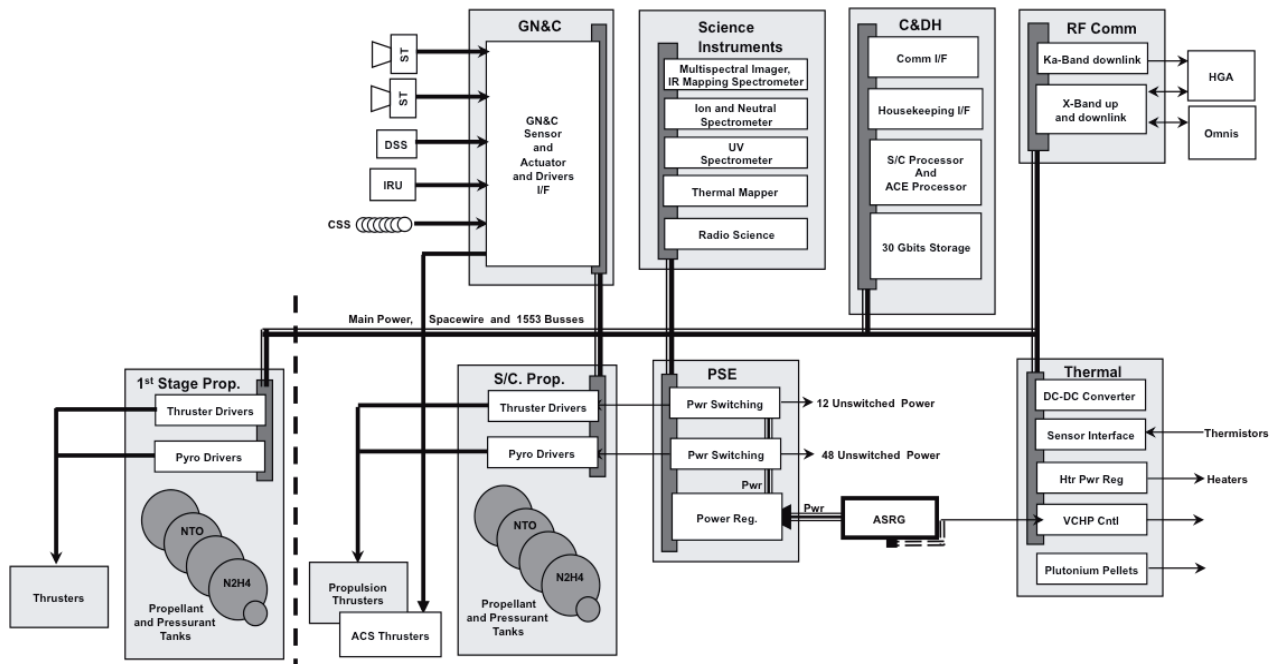


Figure 5: Strawman satellite block diagram

Table 9: Chiron Orbiter Mission Candidate Instrumentation

Required Instrument Type	Candidate Instrument	Specifications	Notes
Multispectral Imager, IR Mapping Spectrometer (0.3 – 1.1 μ m, 1 – 5 μ m)	Stern/SWRI	12 kg, 7 W, 10 Mbps 0.4 – 0.98 μ m, 1.2 – 2.5 μ m	Radiator area needs to be increased to run detectors colder and extend response to 5 μ m
Ion and Neutral Spectrometer (1 – 120 dalton)	Mahaffey/GSFC	10 kg, 30 W, 2.1 Kbps 1 – 300 dalton	
UV Spectrometer (Lya to 4000 Å)	Stern/SWRI	8 kg, 7.5 W, 10 Kbps 520 – 1870 Å	Requires additional detector to cover range from 1870 – 4000 Å
Thermal Mapper (10 – 20 μ m)	Christensen/ASU	2.4 kg, 5.4 W, 0.24 Kbps	
Radio Science	Transponder + USO		
Total	Mass 32.4 kg + transponder, USO	Power 49.9 W + transponder, USO	Cost \$81 M + transponder, USO

Table 10: Data Volume

Data Type	Data Volume (Gbits)
Multispectral Imager - Initial Map	27
Multispectral Imager - Change Monitoring	13
Other Instruments (5% duty cycle)	49
Housekeeping	13
Total	102

Units (AU)), it moves very slowly, and there is flexibility in arrival time over a period of years. It is thus possible to create trajectories that take a specific amount of time for any launch window.

For the chemical option, the team first looked at direct trajectories. Each trajectory has a launch energy (C3) and arrival velocity (VHP) that must be eliminated by using a chemical propulsion system. The team used these figures to calculate the maximum mass delivered to Chiron. Due to the launch vehicle selected (Atlas V 551/Star 48), the C3 is limited to 190. Direct trajectories have lower C3 in 2018 and rise in subsequent years, becoming higher than 190 in 2023. While longer flight times have lower VHP than shorter times, they have slightly higher C3. When these effects are combined, the most desirable trajectories have a launch in 2018 and a flight time of 11.5 to 12 years. Even with the most desirable trajectory, the delivered mass is less than 150 kg, which is small for a useful spacecraft.

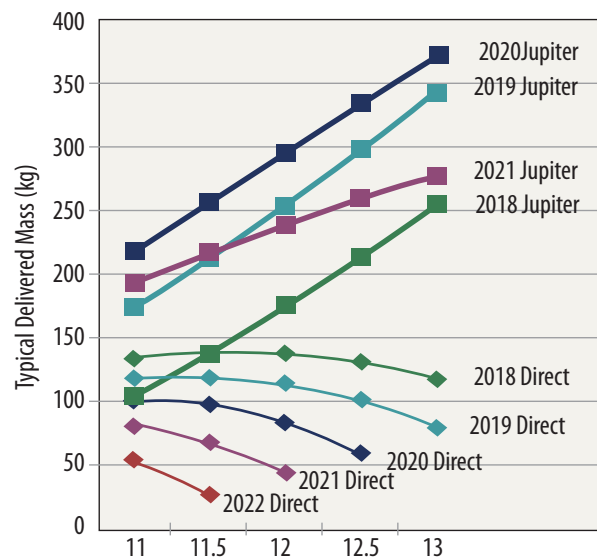
If Jupiter flybys are used, the situation improves. The most desirable launch date is 2020, and by 2022, Jupiter is out of flyby position. The arrival VHP continues to improve with longer flight duration up to and beyond the maximum considered value of 13 years. Using the best opportunity of 2020 launch and 13-year flight time, the delivered mass exceeds 350 kg. This mass might allow for a mission with minimal instruments. Figure 6 summarizes the findings.

4.3 Spacecraft Subsystems

Propulsion: For final Options 1, 2 and 3, the propulsion subsystem for both stages of the Chiron mission is a bi-propellant system using nitrogen tetroxide (NTO) and hydrazine (N_2H_4). The advantage of this type of system is that it allows the use of high specific impulse thrusters. For this mission, an R-4D-15DM, 100-lbf (444.8 N) high performance apogee engine (HiPAT™) is selected to be integrated on each stage for the breaking maneuvers because it provides a specific impulse of 329 seconds using a 375:1 nozzle expansion ratio. For breaking maneuver control and orbital

maneuvers (i.e., orbit capture, orbit resize, etc.) four AMPAC-ISP DST-11H engines are used on each stage. The bi-propellant NTO/ N_2H_4 engine was initially qualified to produce 5-lbf (22.2 N) of thrust at 235 psia and a specific impulse of 309 seconds at a mixture ratio of 0.85. The advantage of this engine is that it has been qualified to perform long burns; therefore it can be used as a back-up to the HiPAT™ engine. For fine spacecraft pointing and other attitude control maneuvers six 0.2-lbf (0.9 N) monopropellant N_2H_4 engines are used. The other major component for both stages is the propellant tank. The tanks are selected based on the calculated mass and volume for each type of propellant.

Power: The Chiron Power System Electronics (PSE) provides power to all observatory loads throughout all phases of the mission. It accepts the raw power from the ASRGs, conditions it, and distributes it to the spacecraft and payload systems in a voltage range of 24–30 VDC. The PSE weighs approximately 15 kg, is 279 mm wide, 292 mm long, and 244 mm high, and is designed with a throughput capability of 900 W. It consumes 43 W of power. The PSE can provide up to 12 un-switched power services to critical subsystems such as C&DH, ACS, propulsion, communications, and thermal control survival heaters that must be active throughout the mission. An additional 48 switched power services are available for other spacecraft systems and instruments as required during the orbit phase of the mission. All switched services have re-settable over current protection and current monitoring


Figure 6: Typical Delivered Mass vs. Flight Time for Chemical Propulsion Systems

capability. The PSE accepts commands from and provides telemetry to the C&DH through a 1553 bus interface. The unit described in this study is based on the LRO version and could be optimized for mass, power, and capability based on specific Chiron requirements. The PSE consists of components and circuits that have flown on many missions, is TRL 9, and is very low risk.

Communications System: The communications system for the baseline two-ASRG configuration is an X/Ka band system. X-band is used for commanding and low rate health and safety telemetry. The Ka band telemetry is used to transmit the science data at a rate of 4 kbps with 3 dB of margin, worst case. This allows the spacecraft to return the 100 Gbits of baseline science data within 2.5 years, using one 8-hour pass per day to a 34-meter DSN ground station. Science data transmission uses the dual polarization capability identified in the guidelines for this study. This requires two Small Deep Space Transponders (SDST) and Traveling-Wave Tube Amplifiers (TWTAs) for normal operations. This system includes a spare SDST and TWTA and has no single points of failure with the exception of the 2.2 meter fixed High Gain Antenna (HGA). The RF power is 80 Watts. The mass estimate for the communications system is 54 kg and the peak power requirement is 180 Watts. If additional ASRGs are available for the mission, additional power will be available for communications while in orbit around Chiron. These alternate configurations may have higher downlink rates and potentially a smaller, lower mass HGA.

Thermal: The Chiron Orbiter employs standard thermal control components with proven flight heritage. The thermal design relies on waste heat from the avionics and proper radiator sizing using approved coatings and Multi Layer Insulation to maintain the orbiter in an acceptable temperature regime. Radioisotope Heating Units (RHU), aluminum spreaders, and doublers are utilized to maintain components above limits that do not have their own power dissipations (e.g., prop tanks, lines, valves, etc.) if needed to preserve ASRG power for avionics. A total radiator area of $\sim 0.55\text{m}^2$ per ASRG (including 30% margin) is needed to reject the 134 W/ASRG power to space and maintain temperatures at 30°C . Assuming the Orbiter has a uniform cold sink in all directions, it is likely that a single monolithic radiator is not needed, and hence existing mechanical structure could be used as radiator for internally-mounted avionics. Additional ASRGs will require additional radiator area.

During the cruise phase, all ASRG power will nominally be used for the avionics and the cruise phase thermal design is identical to the mission phase once beyond Earth and solar environmental effects. A minimum avionics power configuration may be needed for early cruise when solar and planetary environmental heating is larger. If supplemental ASRG power is to be diverted to an electric propulsion system, a more complex active thermal control system could utilize some of the waste heat from the ASRG to maintain avionics temperatures above limits.

The thermal mass is primarily from Multi Layer Insulation, which is needed to keep the avionics and propulsion components warm. For final Option 1, an estimated blanket mass of 34 kg is needed; for final Options 2 and 3, approximately 48 kg of blankets is required. To maintain propulsion components within limits, 2.4 kg of RHUs provide 60 W of heating. Further mass is needed for mounting RHUs to the propulsion lines (saddles, ~ 5 kg aluminum) and for heat spreading (doubler, ~ 5 kg aluminum). This yields a thermal subsystem of 46.4 kg for final Option 1 and 60.4 kg for final Options 2 and 3.

Attitude Control System: The Chiron Orbiter's Attitude Control Subsystem (ACS) controls spacecraft orientation during thrust and attitude maneuvers, establishes and maintains critical high gain antenna lock on Earth during communications, and points the science instruments during science operations. It is comparable to Voyager's ACS in that it is a reaction wheel-free, three-axis stabilized, celestial or gyro referenced system. The ACS components are eight Course and two Digital Sun Sensors (~ 1 kg total), dual arc-second quality Star Trackers (~ 2 -5 kg) and an IRU (~ 1 -5 kg). Attitude control electrical functions are provided by the Command and Data Handling System. The total subsystem mass is 20 kg. The mission life total monopropellant expenditure is on the order of 10 kg, which is accounted for in the propellant budget.

Command and Data Handling System: The command and data handling (C&DH) system includes the processor, data storage (~ 30 Gbits), interface cards, and other avionics components. This subsystem is fully redundant. The C&DH provides a safe mode that places the spacecraft and instruments into a safe configuration in the event of an unexpected event. A safe mode is defined for each phase of the mission. The C&DH mass is estimated at 45 kg and the power is 55 W in cruise mode and 70 W in science observing mode.

Environments: The total ionizing radiation dose for the Chiron Mission is under 50 krad. This mission will use 100 krad parts for the flight system. In addition, this mission shall employ micrometeorite protection techniques comparable to CASSINI. Planetary protection is not addressed in this study.

5.0 Concept of Operations and Mission Design

Figure 7 shows the Mission Phases. The launch phase, which occurs between 2019 and 2025, includes the launch as well as the initial checkout of the spacecraft, instruments, and propulsion components. The cruise phase has duration of between 11 and 13 years and includes the use of planetary flybys that differ based on the various propulsion options used for each scenario. The cruise phase makes use of hibernation mode, similar to that pioneered by New Horizons in its cruise to Pluto; this option saves money on operations costs. The hibernating spacecraft is powered up at regular intervals for status and tracking checks. During hibernation mode, a key feature for maintaining health and safety is Safe Hold Mode (SHM), which permits automated acquisition and long-term attitude control of the spacecraft in a power-positive and thermally-safe condition while maintaining ground communication. Entry into SHM may be triggered by an event or condition detected autonomously onboard, such as a problem with one of the electric propulsion units. SHM may also be commanded manually based on the mission operations engineering assessment of a failure or degraded condition, however, recovery to normal operating mode attitude from SHM involves significant planning and engineering preparation.

A gravity assist of Earth, Jupiter, or Saturn is an opportunity to demonstrate the instrument operations and perform serendipitous science. After the flyby, the spacecraft points the HGA and dumps data to Earth. The potential high data acquisition rate of the imaging instrument and the relatively low data rates to Earth could result in

it taking several months to return all flyby data.

The orbit insertion is relatively uneventful for options using Radioisotope Electric Propulsion, with the spacecraft entering Chiron's vicinity at low relative velocity. For the all-Chemical and Solar-Electric/Chemical Propulsion options, orbit insertion requires a large and multistage chemical insertion burn. The spacecraft orbits the Centaur Chiron body for 3 years. The initial orbit at Chiron uses a high survey orbit, allowing the navigation and science team to determine the mass of Chiron, the orientation of its spin axis, and the dust environment in the vicinity of the desired final orbit of nominally 100 km. Once these have been determined, the ground teams define and execute the maneuvers required to get into the science orbit. This reconnoitering phase could last several months.

The Science Data Collection Phase is constrained by the limited bandwidth to return science data to Earth. The Imaging Spectrometer maps the accessible surface of Chiron early in this phase. This generates approximately 27 Gbits, which will take about 1 year to fully return to Earth. The other instruments operate on a duty cycle that averages 5%. When a change is detected, either indicated by the other instruments, or possibly by taking new mapping data and sending highly compressed thumbnails to Earth, the areas of change are re-imaged by the Imaging Spectrometer and the full resolution data is stored for transmission. This process occurs throughout the mission. The nominal science mission ends three years after arrival at Chiron and the two-ASRG version of the spacecraft returns 102 Gbits of data during this time. The six-ASRG configuration and the two-high-power-ASRG options have additional power available for communications and are able to transmit at higher data rates.

5.1 Data Downlink/Uplink/Tracking

The Deep Space Network (DSN) is used for all communications with the Chiron mission. X and Ka band are used, X-band for commands and

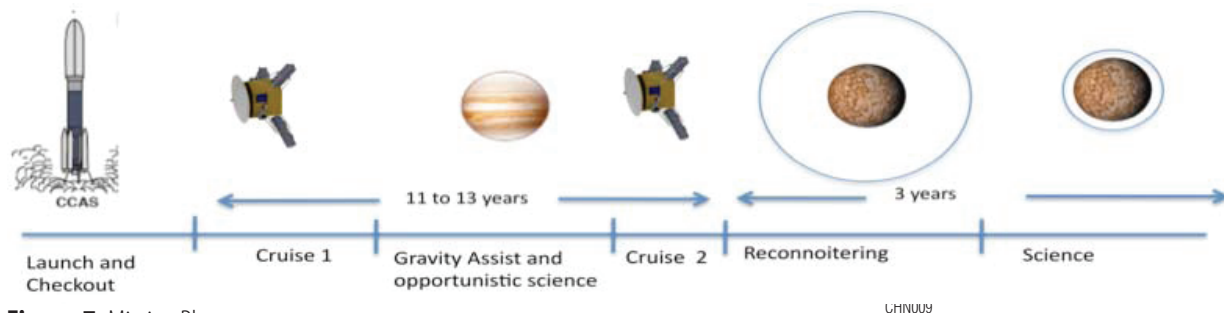


Figure 7: Mission Phases

low rate telemetry and Ka-band for science data. The 34-meter stations are used for normal operations. During the reconnoitering and science data collection phases, the DSN stations are used for 8 hours per day. Communications are suspended once per year during solar conjunctions. The Ka-band communication uses the dual polarization suggested in the study guidelines, which results in a lower mass communications system.

5.2 Ground System

A block diagram for the Chiron Orbiter ground system is shown in Figure 8. The 34-meter DSN ground station system is the single point for the following functions: command generation, telemetry tracking and data processing, navigation/orbit determination, maneuver planning and evaluation, mission planning and scheduling, flight software management, raw science data temporary storage, data archive, and mission operations (including real-time console operations, offline engineering and trending, bus and instrument health and safety and performance monitoring, anomaly detection and resolution, procedure development, spacecraft resource accounting, special operations planning and execution), science operations, and science data processing.

During the cruise phase, the spacecraft generates tones to indicate its condition. If the spacecraft configuration is nominal, there is no telemetry data downlinked or commands uplinked. This reduces the cost of the log cruise phase, since the operations team can be small and DSN costs are lower.

The science data system development and staffing can evolve after launch. The planetary flyby is an opportunity to use the instrumentation and to prototype the systems and procedures that will be used for science operations at Chiron. The systems used for science operations and data processing do not have to be fully operational until shortly before the spacecraft arrives at Chiron.

5.3 Risk

The primary risks that must be addressed for future studies of a Chiron Orbiter are:

- **Mission lifetime:** This study identified an issue with the qualification lifetime of the planned ASRG system (17 years, 3 of which are assumed to be ground assembly and testing) driving unmanageable transit times to Chiron (11 years to insure a 3 year science mission at Chiron). Longer ASRG qualification times are necessary, or higher-power 17 year lifetime ASRG's need to be developed in order for this design to close. Additionally, all other spacecraft components

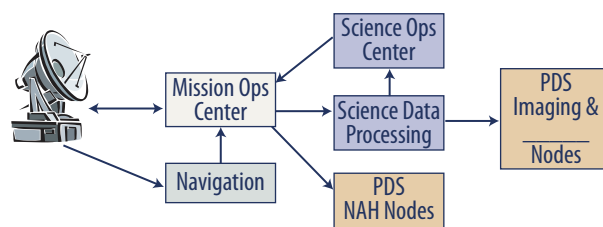


Figure 8: Ground System Block Diagram

will have to survive an 11–13 year transit to Chiron along with the 3 year science mission once the orbiter has arrived at Chiron. Development of these long life components could incur performance, schedule, and cost risks to the development of the Chiron Orbiter.

- **Launch window:** This study has shown that the only propulsion systems capable of a direct trajectory to Chiron are the radioisotope-electric propulsion systems that have their own set of risks (see above). All other propulsion concepts needed differing sets of gravity assists to optimize their performance (which still was insufficient). All of the launch windows identified in this study have backup opportunities one year later, but often with reduced performance. The selection of an actual target launch date (and set of gravity assists) and backup opportunities must go hand-in-hand with the propulsion technology development. Selection of these launch window opportunities and their backups could incur performance, cost, and schedule risks.

6.0 Schedule

6.1 Chiron Orbiter Mission Schedule

A standard template is used to generate generic Chiron mission final option schedules. The template provides 2 years for Phase A and 5 years for Phase B–D. The generic schedule provides project lifetime information. A detailed schedule would be developed in a point design study. Final Options 1 and 2 violate the 2013 project new start date in the decadal survey ground rules. The 2013 project start date will be satisfied if either of these options is studied in detail. Table 11 shows key start dates for each mission option.

6.2 Technology Development Plan

6.2.1 ASRG

The Advanced Stirling Radioisotope Generator, (Figure 9) is designed for multi-mission use in environments with and without atmospheres

Table 11: Key dates for mission options, start dates by option launch dates layout the generic schedule for the mission and the arrival date at Chiron.

Final Option	Phase A start	Phase B start	Phase C start	Phase D start	Launch	Arrive at Chiron	Mission complete
1	2/2012	2/2014	8/2015	2/2017	2/2019	2/2032	2/2035
2	3/2012	3/2014	9/2015	3/2017	3/2019	3/2032	3/2035
3	4/2016	4/2018	10/2019	4/2021	4/2023	4/2036	4/2036
4	5/2017	5/2019	11/2020	5/2022	5/2024	5/2037	5/2040
5	5/2018	5/2020	11/2021	5/2023	5/2025	5/2036	5/2039

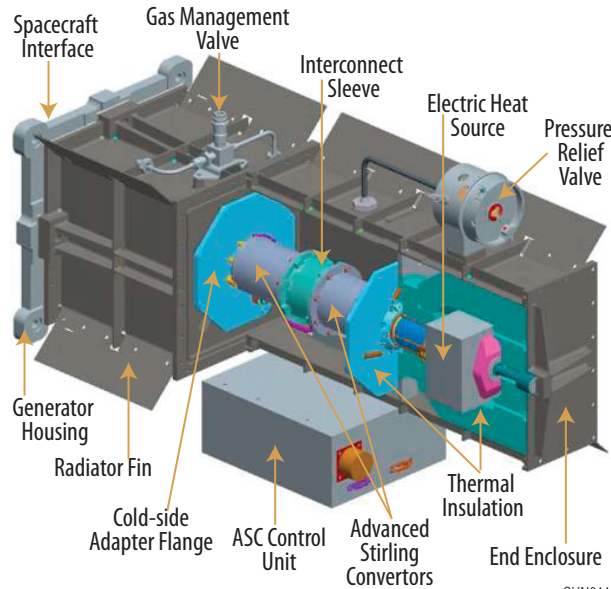


Figure 9: Advanced Stirling Radioisotope Generator (ASRG) ^{CHN011}

and provides continuous electrical power with a specific power of at least 7 W/kg (143 W) at startup. The power degrades by 0.8% per year due to natural Plutonium decay. An ASRG low temperature engineering unit is currently being successfully ground tested (Figure 10), and a final design review is scheduled for a higher temperature ASRG (>800° C) later in 2010. The higher temperature unit will provide 160 W (8.2 W/kg). The ASRG also has an option for active cooling (via a plumbing connection) that could potentially be used to help cool the unit.

Power generation with a radioisotope heated Stirling converter (such as ASRG) during cruise/coast and orbit phases are largely identical processes. It is possible to turn off the ASRG with the use of a variable conductance heat pipe that can shunt the heat from a General Purpose Heat Source (GPHS) module away from the converter—though this would require a modification of the flight-qualified ASRG. During cruise, the Plutonium's heat release decreases by 0.8% a year—this must be factored into the trajectory thrust



Figure 10: ASRG Low Temperature Engineering Unit ^{CHN012}

calculation for electric propulsion final Option 4 and 5. During orbit at Chiron, the ASRG operates normally, as long as the orbit-acquisition deceleration forces are below 30 g, which is within the fatigue limits of the ASRG's Beryllium housing. Since it is likely the ASRG will continue operating beyond 17 years, provision should be made to allow for extended mission operation capability.

The all-ion propulsion final options require more power than a single ASRG can provide. In that case, multiple ASRG units can be employed, as they are designed to share the same DC bus. In addition, a higher power converter is another

option. However, one advantage of using multiple ASRG units is a natural fault tolerance in the event of a single unit failure. The converters are designed for a lifetime of 17 years, and their primary wear mechanism is creep of the pressure vessel. The converters are not expected to fail at 17 years, but the performance could degrade due to the effects of creep that would cause a potential bypass of Helium around the regenerator.

6.2.2 HP ASRG

The technology readiness level (TRL) for the proposed 550 W high power ASRG is at the concept level and the low temperature ASRG is at TRL 6. The high temperature ASRG will be tested in 2011. The Sunpower 1000 W Stirling converter is not space qualified, but has been in operation for years. The higher power Stirling converters are currently being developed under the fission surface power and advanced Stirling technology projects. A thermo-acoustic Stirling power converter is also being tested at the 4000 W level in April 2010. The cost of each of these efforts is several million per year. The cost to purchase an ASRG for future missions is estimated to be \$20 M.

6.2.3 Ion Propulsion

The REP mission options assume use of Hall thrusters that combine existing high TRL designs with a life-extension technology currently under development. A derivative of the Busek BHT-600 thruster, shown in Figure 11, is assumed for this study. The current lifetime capability of this thruster is approximately 30 kg of xenon throughput (xenon expended for thrust). GRC has developed a technology concept for extending Hall thruster life in a separate ISPT project, the High Voltage Hall Accelerator (HiVHA) thruster project. Under a NASA innovative partnership program, NASA GRC and Busek are transferring this technology to the BHT-600 thruster to ex-

tend the thruster life by up to a factor of ten. This development is being pursued specifically to support REP missions like the final Options 4 and 5 approaches to the Chiron Orbiter mission. Two short duration thruster wear tests are planned in 2010 to demonstrate successful technology integration.

To support the Chiron Orbiter mission, additional development is required:

- 2011 to 2012: Design, fabricate and test an engineering model BHT-600LE
- 2013 to 2015: Perform thruster long duration test

The Chiron Orbiter study assumes three operating Hall thrusters and one spare, with 400–500 kg xenon throughput required to perform variations of the final Option 4 and 5 missions. Life extension to 170 kg is thus sufficient to accomplish this mission.

With respect to the critical elements of the NEXT ion propulsion are in development under the In-Space Propulsion Technology (ISPT) project in the NASA Planetary Science Division. The thruster, power processing unit, and xenon feed system assemblies will be at TRL 6 and the thruster gimbal at TRL 4/5 at the completion of this project in 2010. The NEXT project has been authorized additional budget from 2011 to 2013 to extend thruster life testing and address additional risk reduction activities to support transition to the first use on a NASA mission. Assuming successful completion of the project, no further technology development is required prior to initiation of flight system development.

7.0 Mission Life-Cycle Costs

7.1 Costing Methodology and Basis of Estimate

The model used to develop the cost of each of the five options is the NASA/Air Force Cost Model (NAFCOM). NAFCOM is an automated parametric estimating tool for space hardware that uses Cost Estimating Relationships (CERs) that correlate historical costs to mission characteristic to predict new project costs. As with any cost model, the fidelity of the input determines the precision of the output. The subsystem inputs to the model were developed from previous missions. Chiron Orbiter requirements may drive the subsystems to alternate design solutions which would affect the total cost.

7.2 Cost Estimate

Each of the five options was costed and binned in either a New Frontiers cost range or a Flagship cost range. The New Frontiers maximum cost is

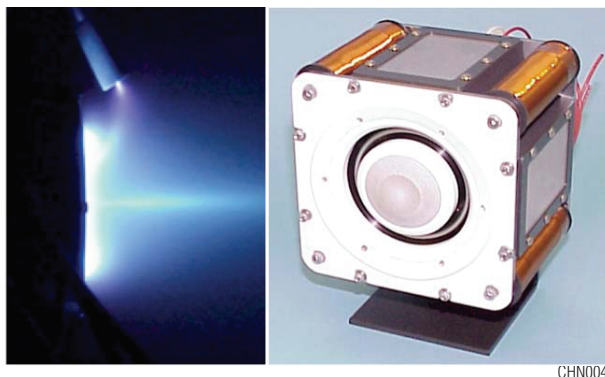


Figure 11: Busek BHT-600 Hall thruster

assumed to be less than \$800 M (FY15) without launch vehicle while a Flagship cost is greater than \$800 M without launch vehicle. Final Options 1–3 (all chemical, SEP/chemical propulsion) fit within the New Frontiers cost range. Final Options 4 and 5 (radioisotope electric propulsion) fall outside the New Frontiers cost range and are binned as Flagship missions.

8.0 Recommendation for Future Concepts

8.1 Optical Communication

NASA's long term plans for its communication infrastructure include development of optical communications ground terminals by the 2020s. If there is a firm commitment for this infrastructure prior to implementation of the Chiron Mission, then use of optical communications could significantly enhance the volume of data returned.

For about the same mass and power, the communications system could have a small RF system for commanding, tracking, and low rate telemetry plus an optical communications system that could provide a data rate of up to 200 kbps to Earth under favorable conditions. The optical communications components on the spacecraft consist of a 10 W laser operating at a wavelength of 1 micron and a 30-centimeter telescope. It communicates with a network of 10-meter ground terminals and 200 kbps can be transmitted under optimal conditions.

Further study is required to optimize the spacecraft laser communication components for the Chiron mission and to characterize the communication under all conditions. When the ground terminal is required to point close to the sun, its field of view must be reduced, or a narrower filter must be used, or both. The Sun-Chiron-Earth angle will always be quite small, complicating pointing the spacecraft's laser at Earth. An Earth-based high-power laser beacon may be required.

8.2 Liquid Oxygen Propulsion for the Inter-Planetary Spacecraft

LO₂/LH₂ propulsion systems are attractive because their high Isp (~450 sec) provides large payload mass fractions. Historically, the use of cryogenic propulsion systems for small spacecraft has been viewed as complicated, due to propellant storage challenges. However, recent advances in cryogenic technologies make cryogenic propellant storage significantly more manageable. Two noteworthy facts make the development of LO₂/LH₂ propulsion specifically for interplanetary spacecraft worth reconsidering:

1. The propulsion requirements of interplanetary missions are momentous, often in the 3–7 km/s range. Propulsion typically dominates mission design (case in point: Chiron Orbiter). As a specific example, imparting a delta-V of 5,000 m/s on a spacecraft with a dry mass of 1,000 kg requires a prohibitive 3,800 kg of MMH/NTO propellant (Isp = 325 sec). With an LO₂/LH₂ propulsion system (Isp = 450 sec), the same delta-V can be achieved with only 2,100 kg propellant, a savings of 1,700 kg. Note that the dry mass of cryogenic propulsion systems is higher because cryogenic propellants are less dense and therefore require larger tanks.
2. In a deep space environment, suitably built hardware can naturally cool below 40 K with just the judicious choice of materials, paint, blanketing, and orientation. Such temperatures will passively maintain the liquid oxygen cryogenic with zero loss to venting during an entire interplanetary cruise. Thus, only the liquid hydrogen may need cryogenic control during the cruise phase.

Two promising technologies help passively reduce hydrogen loss without using cryocoolers. One is subcooling, which extends prevent hold time. Preliminary analyses at GSFC Cryogenics indicate that in a 40 K environment subcooled hydrogen can be held for up to 4 years before venting starts. After that, advanced Cryogenic Fluid Management (CFM) techniques can keep hydrogen vent rates low for long duration cruise phases (15 to 20 year). Specifically, for the above example of 2,100 kg LO₂/LH₂, the hydrogen vent rate can be kept as low as 0.3 mg/s. The total mass of hydrogen lost to venting during a 15 year mission would only be 115 kg. The LO₂/LH₂ propellant load would have to be increased from 2,100 kg to 2,215 kg—still a gain of 1,585 kg over the MMH+NTO required for the same delta-V.

Subcooling can also maintain cryogens at their desired thermodynamic condition on the launch pad. An interesting possibility of circumventing launch phase cryogenics issues altogether is the transfer of liquid oxygen and hydrogen to the spacecraft after launch, either from the reservoirs of a Centaur upper stage, or from on-orbit cryogenic depots. This technology is presently being investigated by NASA and United Launch Alliance for exploration purposes.

Future studies of a Chiron Orbiter may consider examining the technology development required to implement LO₂/LH₂ propulsion as part of a trade against electric propulsion.

9.0 Conclusion

This study examined several options of delivering a useful mass into orbit around Chiron. While this was the driving challenge, the results showed that it is feasible to deliver a science package into orbit around Chiron. The five options discussed focused mainly on the propulsion and trajectories needed to place a satellite, with a given science package, into orbit around Chiron. The results of this study showed final Option 1, all-chemical (storable propellant) propulsion system, delivered a mass which cannot support a viable Chiron science mission. Conversely, final Options 2 and 3, a combined solar electric and storable chemical propulsion system, delivered a useful mass but it will require a reduction in the science instrument package. Final Options 4 and 5, the radioisotope electric propulsion systems, delivered a viable mass to meet all the science requirements. However, the radioisotope electric propulsion systems exceeded the limited supply of the two ASRG's provided for this study.

The task for this study was successfully accomplished in providing various options for delivering mass to Chiron. Future studies may be used to explore quantifying an optimal payload that could be carried to Chiron and investigating other Centaur objects like Okyrhoe and Echeclus that are closer to Earth. Further development of critical technologies, such as ASRGs and ion propulsion systems, will enable larger science suites to be delivered to Chiron. Also, larger launch vehicles may allow storable Chemical and storable-Chemical/SEP propulsion systems to deliver viable spacecraft into Chiron orbit.

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Appendix A - Propellant Mass Parametric Study for Preliminary Concept Options A, B & C

		Launch Properties				Mission Description				Propulsion Stages										Total Dry to Chiron			2nd Stg Propulsion Dry			ASRGs Total (Manual Entry)			Science Payload (Manual Entry)			S/C S/S's Dry (Mass excl. Sci, Prop, ASRG)		Structure/ Mech.		ACS		Thermal		Power Distr. System		C&DH		RF Comm		Harness		Launch Adapter		Spacecraft As Allocated TOTAL																																																																																																																																																																																																																																																																																																																																																																																																																																					
A	Option	Option Description / Stages				Mission Description				Propulsion Stages										Total Dry to Chiron			2nd Stg Propulsion Dry			ASRGs Total (Manual Entry)			Science Payload (Manual Entry)			S/C S/S's Dry (Mass excl. Sci, Prop, ASRG)		Structure/ Mech.		ACS		Thermal		Power Distr. System		C&DH		RF Comm		Harness		Launch Adapter		Spacecraft As Allocated TOTAL																																																																																																																																																																																																																																																																																																																																																																																																																																					
	Star 48 / Chem / Chem	Launch Date				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]																																																																																																																																																																																																																																																																																																																																																																																																																																			
	Jupiter	Total Cruise Duration [years]				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]																																																																																																																																																																																																																																																																																																																																																																																																																																	
	13	Atlas 551				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]																																																																																																																																																																																																																																																																																																																																																																																																																															
	Star 48	Launch Vehicle				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]																																																																																																																																																																																																																																																																																																																																																																																																																													
	81	Star Motor				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]																																																																																																																																																																																																																																																																																																																																																																																																																									
	1718.5	LV C3 [km2/s2]				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]																																																																																																																																																																																																																																																																																																																																																																																																																					
	4648	TromMass or (or SEP Initial Mass)				1st Stage Braking Burns / Ops Concept				1st Stage Prop System Dry [kg]										CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]			CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]		CBE [kg]</	

Chiron Orbiter Mission

[illegible]

Appendix B - Radioisotope Electric Propulsion Trade Summary

Mass Rackup Options 4 (6 ASRG) and 5 (2HP ASRG)

		Launch Properties								Mission Description				Propulsion Stages												Total S/C Dry		2nd Stg Propulsion Dry		ASRGs Total (Manual Entry)		Science Payload (Manual Entry)		S/C S/S's Dry (Mass excl. Sci, Prop, ASRG)		Structure/ Mech.		ACS		Thermal		Power Distr. System		C&DH		RF Comm		Harness		Launch Adapter																																																																																																																																																																																																																																																																																																																																																																																																																																																											
Option	Option Description / Stages	Launch Date	Trajectory [Direct / Planetary Assist]	Total Cruise Duration [years]	Launch Vehicle	Star Motor	LV C3 [km2/s2]	Trow Mass or (or SEP Initial Mass)	Arrival Delta-V [km/s]	Number of ASRGs (if any)	Required Pu238 [kg]	Solar Array Size (f any) [m2 or kW @ 1AU]	Power into Thruster (if any) [W]	Post Boost Stage (or SEP) Initial Mass	Total Delta-v [m/s]	1st Stage Isp [s]	1st Stage Propellant [kg]	1st Stage Propelled ("Dry") Mass [kg]	1st Stage Delta-v [m/s]	1st Stage Prop System [%]	1st Stage Prop System Dry [kg]	1st Stage Jettisoned Mass [kg]	2nd Stage Isp [s]	2nd Stage Delta-v [m/s]*	2nd Stage Wet Mass [kg]	2nd Stage Propellant [kg]	2nd Stage Prop System [%]	2nd Stage Prop System Dry [kg]	2nd Stage Dry [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max Exp'd Value [kg]	CBE [kg]	Max 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COMPASS Chiron Orbiter GLIDE Trade Space Summary

Case #	Summary	Power to thrusters (kW)	Initial mass (ELV Performance)	Basic Dry Mass (kg)	Science Payload (Basic Mass) kg	Trip Time (yr)
1	6 ASRGs	0.578	1302	581	58	13
2	6 ASRGs	N/A	N/A	N/A	Not closed	N/A
3	8 ASRGs	0.833	1354	581	Not closed	11.5
4	8 ASRGs	0.833	1367	599	27	11.5
5	2 HP ASRGs	0.744	1311	516	55	11
6	2 HP ASRGs	0.744	1348	585	124	12
7	2 HP ASRGs	0.744	1352	625	164	13
8	3 HP ASRGs	1.209	1461	625	98	11
9	3 HP ASRGs	1.209	1470	727	200	13
10	6 ASRGs	0.578	1254	546	18	13

Appendix C - References

S. R. Oleson, M. L. McGuire, et. al. (2007) COMPASS Final Report: Radioisotope Electric Propulsion (REP) Centaur Orbiter New Frontiers Mission

Appendix D - Acronym List

ACS	Attitude Control Subsystem
ADL.....	Architecture Design Laboratory
ASRG.....	Advanced Stirling Radioisotope Generators
AU	Astronomical Unit
C&DH	Command and Data Handling
CBE.....	Current Best Estimate
CERs.....	Cost Estimating Relationships
CFM.....	Cryogenic Fluid Management
COMPASS	Collaborative Modeling for Parametric Assessment of Space Systems
Delta-V	Delta-Velocity
DSN	Deep Space Network
FY	Fiscal Year
Gbits.....	Gigabits
GNC.....	Guidance, Navigation, and Control
GRC	Glenn Research Center
GSFC.....	Goddard Space Flight Center
HGA.....	High Gain Antenna
HiPAT™	High Performance Apogee Engine
HiVHAc	High Voltage Hall Accelerator
HP	High Power
N ₂ H ₄	Hydrazine
INMS	Ion Neutral Mass Spectrometer
IR.....	Infrared
IRU.....	Inertial Reference Unit
ISPT	In-Space Propulsion Technology
K.....	Kelvin
Kbps.....	Kilobits Per Second
kg.....	Kilogram
kW.....	KiloWatt
lbf	Pounds (force)
LRO.....	Lunar Reconnaissance Orbiter
MMH/NTO	Monomethylhydrazine / Nitrogen Tetroxide
NAFCOM	NASA/Air Force Cost Model
NEXT	NASA Evolutionary Xenon Thruster
NTO.....	Nitrogen Tetroxide
PPU	Power Processing Unit
PSE.....	Power System Electronics
RCS	Reaction Control
RF.....	Radio Frequency
RHU.....	Radioisotope Heating Units
SEP	Solar Electric Propulsion
REP	Radioisotope Electric Propulsion
RPS.....	Radioisotope Power Systems
SDST	Small Deep Space Transponders
SHM.....	Safe Hold Mode
TRL.....	Technology Readiness Level
TSSM	Titan/Saturn System Mission

TWTA Traveling-Wave Tube Amplifiers
USO Ultra Stable Oscillator
UV Ultraviolet
VDC Voltage Direct Current
VHP Hyperbolic Velocity
W Watts
Xe Xenon