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Abstract

Sample return missions are the most difficult tasks we ask robotic spacecraft to undertake in exploring our solar system, but we do so because of the high value returned samples have for the planetary science community. Thus far, we have only acquired samples from: the Moon, three asteroids, a comet's tail, and the solar wind at the Earth-Sun Lagrange Points. The National Academy's most recent decadal survey of planetary science in NASA — Origins, Worlds, Life (OWL) — emphasized the value of samples returned to Earth for analysis and called for NASA to prioritize samples returned from Mars, the Moon' South Pole, a Jupiter-family comet, and Ceres. Currently available rockets and propulsion technology impose severe, and possibly insurmountable, limits to where we can send robot explorers and return samples within a reasonable timescale. Now, the advent of large new rockets offers the potential for very high C_3 Earth escape trajectories. Parallel developments in Nuclear Propulsion yield much higher I_{SP} than chemical propulsion and can operate far away from the Sun. Our novel trajectory and mission architecture analysis shows that, combining these technologies, sample return from all across the solar system starts to become feasible within the career lifetime of a planetary scientist.

Keywords: space exploration; space propulsion; orbit transfers; space trajectory; trajectory optimization; interplanetary missions; sample return missions; mission architecture; nuclear propulsion

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1. Introduction

The best-known set of sample return missions are of course the Apollo 11 to 17 human mission to/from the Moon [1] NASA's astronauts brought back 382 kg of samples from the sites they visited, which are still surprising scientists who analyze them more than 50 years after they reached the Earth [2]. More recently, robotic missions have returned samples to Earth in much smaller quantities: NASA/JPL's Discovery mission Genesis brought back samples of the solar wind, collected at the Earth Sun L_1 and L_2 points [3]; Stardust brought back samples from the tail of comet Wild 2 [4] the Japanese Space Agency's Hayabusa and Hayabusa-2 missions returned samples from the asteroids 25143 Itokawa [5], 162173 Ryugu [6]; and in late 2023 NASA/GSFC's OSIRIS-Rex mission brought back samples from the asteroid Bennu [7]. China's Chang'E-6 mission recently returned the first samples from the South Pole-Aitken Basin area of the Moon [8].

At a roughly 10-year cadence, the National Academy of Sciences is tasked to provide NASA with advice on priorities for its scientific endeavors. In planetary science, the latest decadal survey, released in 2022, is Origins, Worlds, Life, known as OWL [9] Like its predecessor, Visions and Voyages [10], OWL stated as its highest scientific priority the flagship Mars Sample Return (MSR) mission currently in formulation for NASA at the Jet Propulsion Laboratory. Other sample return missions also featured in OWL's recommendations, particularly the Ceres and Comet Sample Return targets flagged as New Frontiers medium-class missions, and the Endurance-A long-range lunar

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rover/sample collection mission, recommended to be implemented as a strategic medium-class mission under NASA's Lunar Discovery and Exploration Program. The architecture of Endurance-A is unusual: dropped off by a commercial lander, the plan is to have a rover traverse the South Pole/Aitken Basin region of the Moon, picking up scientifically selected samples to deliver to a human landing site, for recovery by an astronaut team. OWL also discusses the value of sample return for "laboratory-based measurements of additional primitive materials that sample different reservoirs in the nebula" to "resolve the question of a supernova trigger versus later injection of isotopes" in understanding the origin of our solar system. "Different reservoirs" in this context means "Kuiper Belt objects, Centaurs, comets, and P- and D-type asteroids within the main belt, Hilda, Trojan asteroid, and irregular satellites" of the Giant Planets. A list of other targets in the solar system where OWL identifies returned samples as key to understanding their origin include Venus, Mercury, Europa, Enceladus, and Titan. OWL calls for "direct measurement of the timing of key geologic events on terrestrial planet surfaces, now only possible by sample return" and the return of "cold/cryogenic volatile samples from planetary bodies".

Clearly, scientifically selected samples returned from all across the solar system would have a very high science value. It has been suggested [11] that it is the culmination of a sequence of ever-more complex scientific investigations of any solar system body, which proceeds methodically as follows - flyby, orbit, land, rove, return samples. This argument was adopted as an organizing principle for future missions by Jim Green, the former head of planetary science at NASA Headquarters [12]. Returned samples are also a gift to future generations of scientists – as was done with the Apollo samples, a portion are set aside and preserved for analysis with laboratory tools and methods that will only be possible many decades later. Some have even put forward the idea that analysis of such samples is a "holy grail" for planetary scientists [13-14].

In section II of this paper, we present the case for sample return across the entire solar system. In section III we examine the 'tyranny of the rocket equation', and indicate how, using current propulsion technology, we may have reached a limit in our ability to bring meaningful samples back from harder to reach locations in the outer solar system, in particular. In Section IV a combination of emerging technologies—big rockets and nuclear propulsion—are proposed as a means to break through this bottleneck. We decompose into building blocks a sample return mission using a generic mission architecture with constraints on both the launch mass, and the size of the sample returned. In section V novel trajectories to/from more distant objects, including the moons of the outer planets, but also Centaurs and even Halley's Comet, are presented. Finally, we show how even the most challenging sample return missions could be made feasible by adopting the emerging combination of big rockets and nuclear propulsion.

2. Solar System Exploration: Past, Present and Future

Following Jim Green's example and taking the sequence of increasingly complex missions as an organizing principle, we can visualize the progress made in the exploration of our solar system by examining Table 1, which summarizes for each type of body what has been achieved to date. A Yes entry in Table 1 means that a mission of that particular category has been executed for that class of target. For Earth's Moon, for example, we had flyby missions, orbiters and landers in the 1960's and early 1970's, and of course the Apollo crews brought back samples from several different near-side locations, and deployed a rover that allowed them to explore farther away from their immediate landing site. The Yes, But… entries for the Moon simply mean that scientists have identified the need for greater mobility, and samples from other regions as a high priority for future lunar exploration. NASA/GSFC's Lucy Discovery mission [14] is on its way to fly by several Trojan objects in its nominal 12-year lifetime, hence its "Under Way" categorization. A blank entry in Table 1 means that no spacecraft has executed that type of mission at that type of target. For the Centaurs, for example, potentially near-pristine captured Kuiper Belt Objects that lie between Jupiter and Neptune's orbits, no spacecraft has thus far visited any of them, even fleetingly. The entire row for mission types is therefore blank for Centaur targets. As referred to above, Stardust brought back samples of the tail

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of the Jupiter family comet Wild 2, but scientists have expressed the need for a comet surface sample in both the Visions and Voyages, and OWL decadal surveys.

Table 1. Solar System exploration seen through the prism of increasing mission complexity.

Target	Flyby	Orbiter	Lander/InSitu	Mobility	Sample Return
Mercury	Yes	Yes			
Venus	Yes	Yes	Yes		
Moon	Yes	Yes	Yes	Yes, but…	Yes, but…
Mars	Yes	Yes	Yes	Yes	Not yet
Jupiter	Yes	Yes	Yes		
Saturn	Yes	Yes	Yes (Titan)		
Uranus	Yes				
Neptune	Yes				
Pluto	Yes				
Jovian Comets	Yes	Yes	Yes		Yes, but…
Trojans	Under way				
Centaurs					
NEOs	Yes	Yes	Yes	Yes	Yes
Main Belt	Yes	Yes			
Halley's Comet	Yes				

Note*: as suggested by Thompson and Coates [11] and Jim Green [12].*

Table 1 captures the state of solar system exploration to date, a set of feats that has taken over 65 years to accomplish, since the National Aeronautics and Space Administration was formed in 1958. It represents an impressive record of achievement by the planetary science community in the United States and internationally. It should be noted, however, that nearly half the cells are blank, and those are clustered towards the right-hand side of the table, where missions become increasingly complex, and harder to execute.

If we add the missions prioritized in the OWL decadal survey, we can see what this view of solar exploration might look like once all *those* missions have been successfully executed. The proposed flagship Uranus Orbiter and Probe mission [10], for example, has an architecture very like Galileo, with an orbiter component and an atmospheric entry probe for insitu measurements. This builds on the knowledge we have on the Uranus system from the Voyager II flyby in 1986 and is captured in Table 2 as two Yes entries under the Orbiter and Lander/InSitu columns. It is difficult to put an exact timeline on the new mission entries in Table 2, but it is not perhaps unreasonable to say that, at the current pace of planetary science missions, it could take at least three more decades before all of them are completed. That still leaves more than a third of mission complexity matrix unaddressed. Staying at the same pace, it could be another century before all of the mission entries are flagged with a Yes.

Here we get to the central idea that inspired this paper: what if we went straight to the right-hand column in Tables 1 and 2, to take on sample return missions from all across our solar system? We would surely achieve most of the objectives of the flyby, orbiter and lander/in situ mission entries, and in some cases the mobility objectives too. Would it be possible to shorten the timescales for such missions, so that a future planetary scientist might expect to have access to samples from all across the solar system, from Mercury all the way out to Pluto, within the time limits of their career? Herein we propose an approach that would make that possible.

Table 2. Solar System exploration with OWL decadal survey mission recommendations[1].

Target	Flyby	Orbiter	Lander/In Situ	Mobility	Sample Return

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Mercury	Yes	Yes			
Venus	Yes	Yes	Yes	**Yes**	
Moon	Yes	Yes	Yes	**Yes**	**Yes**
Mars	Yes	Yes	Yes	Yes	**Yes**
Jupiter	Yes	Yes	Yes		
Saturn	Yes	Yes	**Yes**		
Uranus	Yes	**Yes**	**Yes**		
Neptune	Yes	**Yes**			
Pluto	Yes				
Jovian Comets	Yes	Yes	Yes		**Yes**
Trojans	Yes				
Centaurs	**Yes**	**Yes**	**Yes**		
NEOs	Yes	Yes	Yes	Yes	Yes
Main Belt	Yes	Yes	**Yes**		**Yes**
Halley's Comet	Yes				

Notes*: Solar System exploration seen through the prism of missions of increasing complexity, assuming all of the OWL decadal survey mission recommendations [10], shown* ***in Bold****, are executed, and that missions such as Lucy and MSR succeed in meeting all of their objectives.*

3. How the Rocket Equation impacts Sample Return Missions

3.1. The 'Tyranny' of the Rocket Equation

The famous rocket equation, governing the mass of propellant versus useful spacecraft mass, is credited to Konstantin Tsiolkovsky [16]. It can be expressed as:

$$\Delta v = I_{SP} g_0 ln \frac{m_0}{m_f} \tag{1}$$

Where Dn is the required change in velocity of the spacecraft, I_{SP} is the specific impulse of the propulsion technology used, g_0 is standard gravity, m_0 is the mass of the spacecraft including propellant (wet mass), and m_f is the final mass of the spacecraft, after the propellant has been expended (also known as dry mass). The equation can be solved for m_0 to give:

$$m_0 = m_f e^{\Delta v / I_{SP} g_0} \tag{2}$$

$$m_0 - m_f = m_f (e^{\frac{\Delta v}{I_{SP} g_0}} - 1) \tag{3}$$

yielding the well-known result that the mass of propellant (m_0 – m_f) for a given spacecraft mass increases exponentially with the require Dn. This places a fundamental constraint on what Dn is achievable for a given launch mass. This problem is especially acute for low I_{SP} values, typical of today's chemical propulsion systems. For high Dn missions, this is what meant by the 'tyranny of the rocket equation'.

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3.2. Emerging Technologies with high mass to LEO and I_{SP} values

We live in a time when the space business is changing: big rockets (with more than 45t lift capacity), some of which are reusable, are being developed; and nuclear propulsion, after a long gestation period by nuclear engineers, is likely to be demonstrated in space by the end of the decade. The new rockets are capable of lifting enormous masses into Low Earth Orbit (LEO), as shown in Table 3.

Table 3. Launch Vehicle lift capacity to LEO.

Rocket	Lift Capability to LEO	Reusable or Expendable?
Starship	150–250 t	Fully Reusable
New Glenn	45 t	Partially Reusable
SLS Block ½	95 t/130 t	Expendable
Saturn V	140 t	Expendable

***Notes**: Lift capacity in metric tonnes to LEO for emerging launch vehicles, and the Saturn V from the Apollo era for comparison [17].*

Starship, as currently planned, will not launch directly into an Earth Escape trajectory, as the other vehicles in Table 3 can. Instead, it can be refueled in orbit, providing the potential to "transport >100 t of bulk cargo anywhere in the solar system" [18]. It's possible that other new rockets in Table 3 may be used to assemble a similar capability in orbit through multiple launches, but in this paper we will look at how we might use a fully or partially refueled Starship to achieve our goal of returning samples from all across our solar system.

At a workshop in December of 2023, held in Tempe, AZ, and organized by the Institute for Space Science and Development (henceforth The Tempe Workshop) leading planetary scientists and nuclear propulsion experts in the US gathered to discuss the prospects for future space science missions taking advantage of emerging nuclear propulsion capabilities [19]. At the time of the Tempe Workshop, NASA planned two technology demonstration missions, DRACO and JETSON, to prove out nuclear thermal and nuclear electric propulsion powered by fission reactors within a decade or so. These have since been replaced by the SR-1 Freedom mission to Mars [20] which plans to use nuclear electric propulsion in 2028. New reactor fuels, such as high-assay low-enriched uranium (HALEU) offer an alternative source of space power to address current limitations on the production of radioisotope power systems for future planetary science missions to the outer solar system [21]. The significance of these developments is that nuclear propulsion has a much higher I_{SP} than the chemical propulsion technologies in use today (see Table 4). Referring again to the rocket equation discussion above, higher I_{SP} values have a significant effect on the dry mass of the spacecraft (for a fixed launch mass) and on the amount of fuel needed to achieve a high Dn trajectory.

Table 4. Typical I_{SP} values for current and future propulsion technologies.

Propulsion Technology	Typical I_{SP} values
Mono-propellant	230 s
Solid	290 s
Bi-propellant	320 s
LOX/LH2	450 s
NTP	900 s
NTP 2.0	3000 s
NEP	4000 s
NEP 2.0	8000 s

***Notes**: NTP 2.0 and NEP 2.0 refer to the second generation of nuclear propulsion engines.*

Tables 5 and 6 summarize the characteristics of hypothetical Nuclear Thermal and Nuclear Electric Propulsion systems used in this study, based on the engineering judgment of our team. They

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are not identical to the technology demonstrations proposed for SR-1 Freedom, DRACO or JETSON, but are selected as reasonable projections of capabilities that can be derived from those demonstrations. In particular, we have used the 1st generation values for the Specific Impulse I_{SP}.

Table 5. Parameters for a spacecraft equipped with a Nuclear Thermal Propulsion system.

Assumption	Value
NTP Power & Propulsion System Mass	1000 kg
NTP Thrust Level	20 kN
NTP Specific Impulse	900 s
NTP Fuel Tank / Propellant Mass Fraction	39%
Residuals / Propellant Mass Fraction	5%
Structures / Wet Mass Fraction	15%
Thermal / Dry Mass Fraction	13%
Stirling Engine Power System	100 kg
Guidance + Avionics + Telecom	100 kg
Battery Capacity	5 kW
DTE Downlink	500 kbit/s
Relay Uplink/Downlink	200 kbit/s

Table 6. Parameters for a spacecraft equipped with a Nuclear Electric Propulsion system.

Assumption	Value
NEP Power System Mass	1700 kg
NEP Power System Power	30-65 kW
NEP Specific Impulse	3100 s
NEP Propulsion System Mass	150 kg
NEP Fuel Tank / Propellant Mass Fraction	10%
Residuals / Propellant Mass Fraction	10%
NEP Boom Mass	300 kg
Structures / Wet Mass Fraction	15%
Thermal / Dry Mass Fraction	13%
Guidance + Avionics + Telecom	100 kg
DTE Uplink/Downlink	500 kbit/s
Relay Uplink/Downlink	200 kbit/s

In our study, we assumed an NTP-equipped spacecraft with a Stirling engine for efficient conversion of thermal to electrical power [22] and that efficient thermal management strategies, e.g. using heat straps, heat pipes and radiators can be designed to avoid the necessity to add additional Radioisotope Heater Units. The fuel tanks are assumed to have a zero boil-off capability, so fuel is not lost during the long mission durations expected. Further, the tanks can be jettisoned when emptied.

Similarly, our NEP-equipped spacecraft is also assumed to have efficient thermal management strategies and the capability to jettison used fuel tanks. We also assume that mechanical booms used to offset the nuclear reactor from the spacecraft avionics can be designed to have the low mass values specified in Table 6.

For both architectures, our notional spacecraft can be assembled on Earth and launched in a single Big Rocket into Low Earth orbit or Earth Escape. There may be some advantage to on-orbit assembly, as that technology matures, which would allow the nuclear reactors to be launched separately, then integrated with the spacecraft.

3.3. Sample Return Mission Dn Values

If the high I_{SP} problem is solved as nuclear propulsion technology matures, how much *Dn* is then needed to return samples from all across the solar system? To first order, we can estimate this by

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looking at one of the many *Dn* maps of the solar system available online (see Figure 1). These maps use Hohmann transfers, and usually ignore gravity assists, atmospheric aerobraking, etc., but they do serve as a good indicator of what it takes to get out to a target in the solar system and return with a sample. Adding up the total required *Dn* to go to the surface of the Moon and back, for example (not including the *Dn* required for escaping Earth's gravity well and Earth re-entry on return), our calculations yield 4.8 km/s. Total *Dn* values for sample return from the Moon and other solar system targets are illustrated in Table 7.

Table 7. Required Dn values for sample return missions.

Target	Sample Return Dn
Mercury	25.9 km/s
Moon	4.8 km/s
Phobos	4.7 km/s
Mars	12.1 km/s
Ceres	13.0 km/s
Europa	28.5 km/s
Enceladus	24.8 km/s
Pluto	18.1 km/s

Notes*: Total required Dn values to go to the surface of targets in the solar system and back, (not including the Dn required for Earth escape and Earth re-entry). Hohmann transfer trajectories only.*

3.4. First order mass estimates for Sample Return Missions

To estimate the mass of the vehicle escaping Earth's gravity we assume an initial value for the dry mass of the spacecraft, excluding the propulsion module, of 1000 kg, which is typical for missions we send out to explore the solar system today. Exercising Equation (3) we can estimate the amount of fuel needed to send such a spacecraft out to a target in the solar system and return. Then we recalculate the mass of the spacecraft, allowing for a structure with a mass of 10% of the fuel expended as the propulsion module. As a final step we exercise Equation (2) to calculate the mass of the vehicle escaping the bonds of Earth's gravity field. This is shown in Table 8 for the target set in Table 7, sorted by *Dn*.

Table 8. Estimates of required Wet Mass for sample return missions.

Propulsion I_{SP} (s)		**Monoprop**	**Solid**	**BiProp**	**LOX/LH2**	**NTP**	**NTP 2.0**	**NEP**	**NEP 2.0**
	Dn (km/s)	230	290	320	450	900	3000	4000	8000
Phobos	4.7	1.4E+04	7.4E+03	6.0E+03	3.5E+03	1.8E+03	1.2E+03	1.1E+03	1.1E+03
Moon	4.8	1.5E+04	7.8E+03	6.3E+03	3.6E+03	1.8E+03	1.2E+03	1.1E+03	1.1E+03
Mars	12.1	4.8E+06	5.6E+05	2.7E+05	3.8E+04	5.1E+03	1.6E+03	1.4E+03	1.2E+03
Ceres	13.0	1.0E+07	1.0E+06	4.5E+05	5.3E+04	5.8E+03	1.6E+03	1.4E+03	1.2E+03
Pluto	18.1	9.4E+08	3.4E+07	1.1E+07	4.2E+05	1.3E+04	2.0E+03	1.7E+03	1.3E+03
Enceladus	24.8	3.6E+11	3.8E+09	7.3E+08	7.9E+06	4.3E+04	2.6E+03	2.0E+03	1.4E+03
Mercury	25.9	9.4E+11	8.1E+09	1.5E+09	1.3E+07	5.2E+04	2.8E+03	2.1E+03	1.4E+03
Europa	28.5	9.4E+12	5.1E+10	7.7E+09	4.1E+07	8.7E+04	3.1E+03	2.3E+03	1.5E+03

Notes*: Wet Mass (fully fueled mass) in kg for vehicles escaping Earth's gravity field to execute sample return missions to/from the solar system targets listed in Table 7, shown for representative I_{SP} values and different propulsion technologies. Shaded values indicate where the total spacecraft mass exceeds projected Earth escape capabilities of a fully refueled Starship from low Earth orbit, which is expected to be of order 1.00E+05 kg (100 t) [24].*

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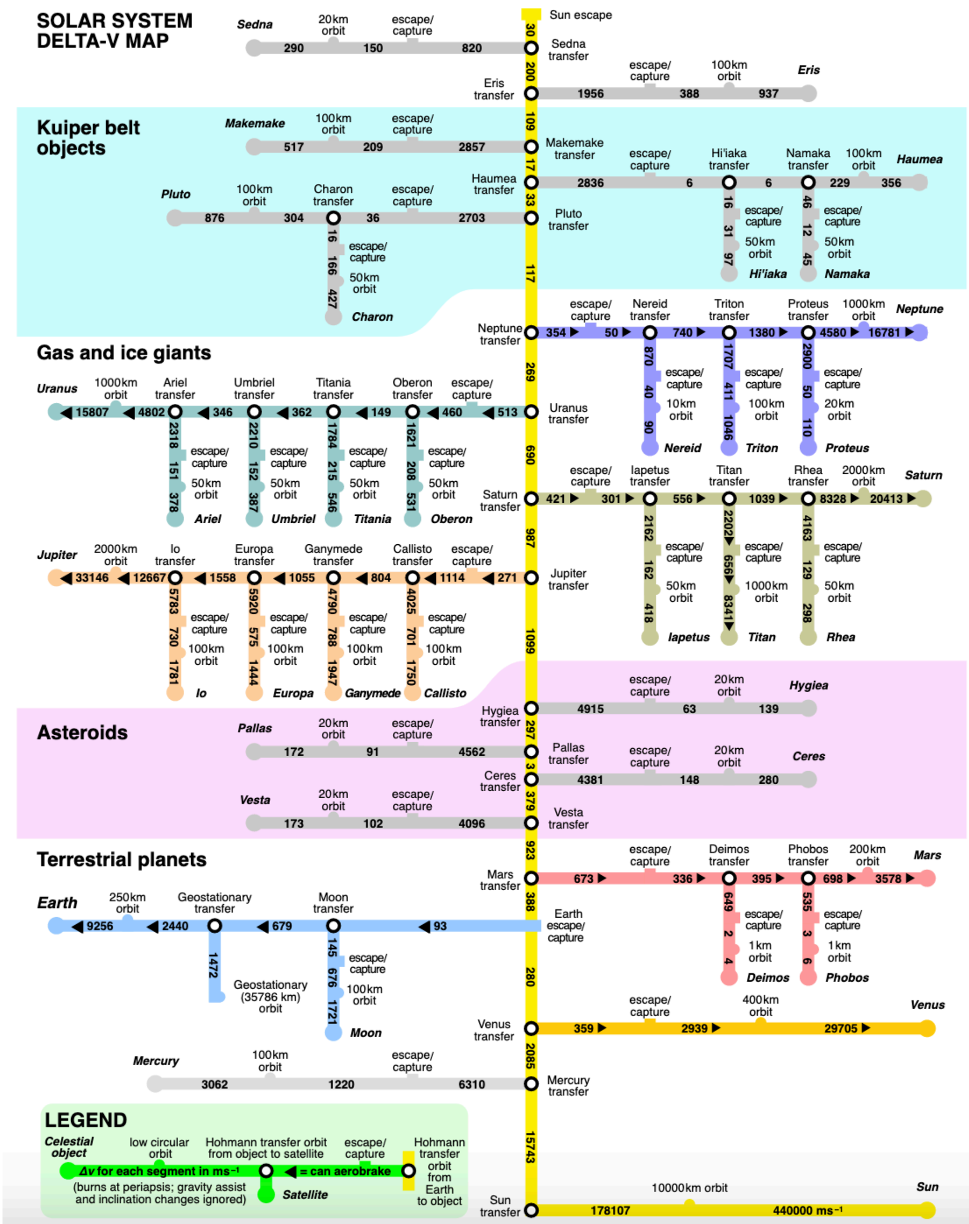

Figure 1. Dn map of the solar system, showing Dn values for Hohmann transfer trajectories [23].

To give some scale to the mass numbers in Table 8, the International Space Station, the largest space vehicle ever constructed over many Space Shuttle flights, has a mass of more than 400,000 kg, or 4.0E+05 kg in scientific notation [26]. Table 8's illustrations are perhaps overly simplistic; for example, the table does not allow for significant *Dn* provided by a high C_3 launch, or gravity assists, or using the atmosphere at the target body to slow down, as we would do at Mars. We also would not usually send out a 1000 kg spacecraft, and return all of that to Earth; typically, we would expect to leave some of the outgoing spacecraft behind, similar to what was done with Apollo [1].

Reasonable solutions for sample return have been proposed for the Moon [27] and for Phobos [28] using chemical propulsion only, which is consistent with the information presented in Table 8 for those targets. For Mars and Ceres, as seen in [10], sample return mission concept studies usually baseline Solar Electric Propulsion (SEP), which has I_{SP} values similar to Column 9 for NEP. [We can use SEP in the inner solar system where the solar radiation intensity is strong; in the outer solar system, the $1/R^2$ drop-off in solar radiation as we move further away from the Sun means it is extremely difficult to generate enough power to drive a SEP engine.]

Despite its simplification of the mission design, Table 8 does illustrate the degree of difficulty, in terms of high spacecraft mass values needed, of returning samples from more distant, harder to

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reach bodies in the solar system relying only on chemical propulsion; and in contrast the relatively low masses of vehicles that use nuclear propulsion for their primary thrust, either nuclear thermal or nuclear electric. At some point, if we are serious about returning samples from all across the solar system, especially its outer reaches, we are going to have to give up on chemical propulsion as the primary means of generating thrust, and turn to higher I_{SP} solutions, as offered by nuclear propulsion technology.

4. A Common Architecture for Sample Return Missions

For the purpose of our study, we adopted a simple, modular approach for the sample-return mission architecture, dividing it into 6 different elements: a launch vehicle, a nuclear propulsion vehicle stage, an ascent vehicle stage, a descent vehicle stage, a sample container, and an Earth-return capsule (capable of Earth atmospheric re-entry and landing). These mission architecture elements are illustrated in Figure 2 and Figure 3 provides a bat-chart diagram of how they would be used during each phase of the mission.

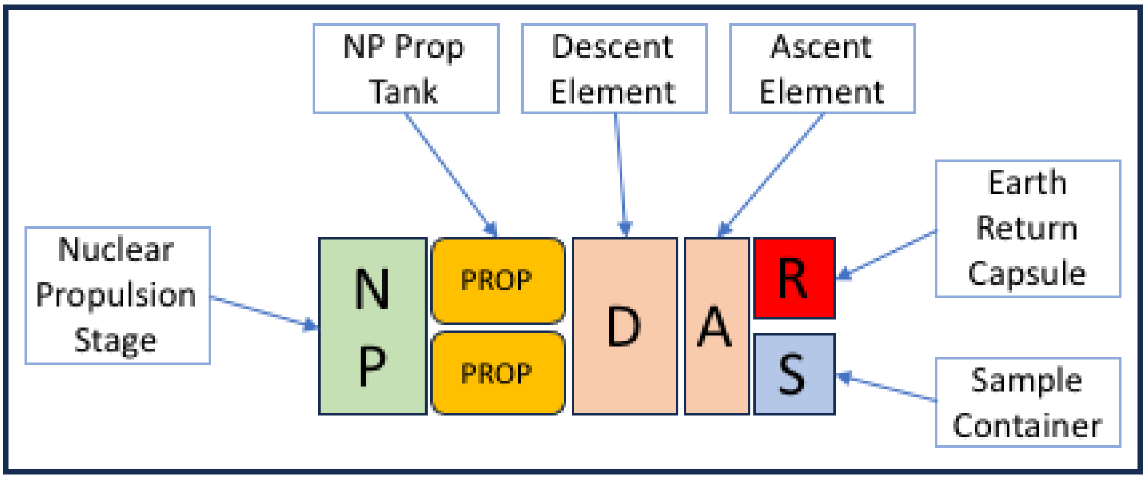

Figure 2. Generic launch stack for Sample Return Missions using Big Rockets and Nuclear Propulsion. The launch 'stack' from LEO includes: the Nuclear Propulsion stage (NP); Propulsion tanks that can be dropped when emptied; a Descent stage (D) to enable landing on the target body; an Ascent element (A) to bring the sample off the surface; a Sample container (S) ; and an Earth Return capsule (R).

This architecture has many similarities to Apollo, especially in that the Propulsion module would orbit the target body, while the descent/ascent elements go down to the surface to collect samples. The descent/ascent propulsion is assumed to be chemical, e.g. LOx-CH_4 bi-propellant, to avoid having to take a relatively heavy nuclear reactor down to the surface and back up again. The Nuclear Propulsion module, remaining in orbit, can also serve as a relay for communications between Earth and the landed elements.

The Nuclear Propulsion stage acts as the primary power/propulsion element within the vehicle throughout the mission. All primary maneuvers within the mission are conducted by the Nuclear Propulsion module. This vehicle is expected to be modular so that after significant maneuvers, the propellant tanks can be jettisoned to reduce the vehicle mass prior to the next maneuver (lowering the amount of thrust required). On top of propulsion and communications, the Nuclear Propulsion stage delivers power to meet the demands of other systems throughout the cruise phases of the mission.

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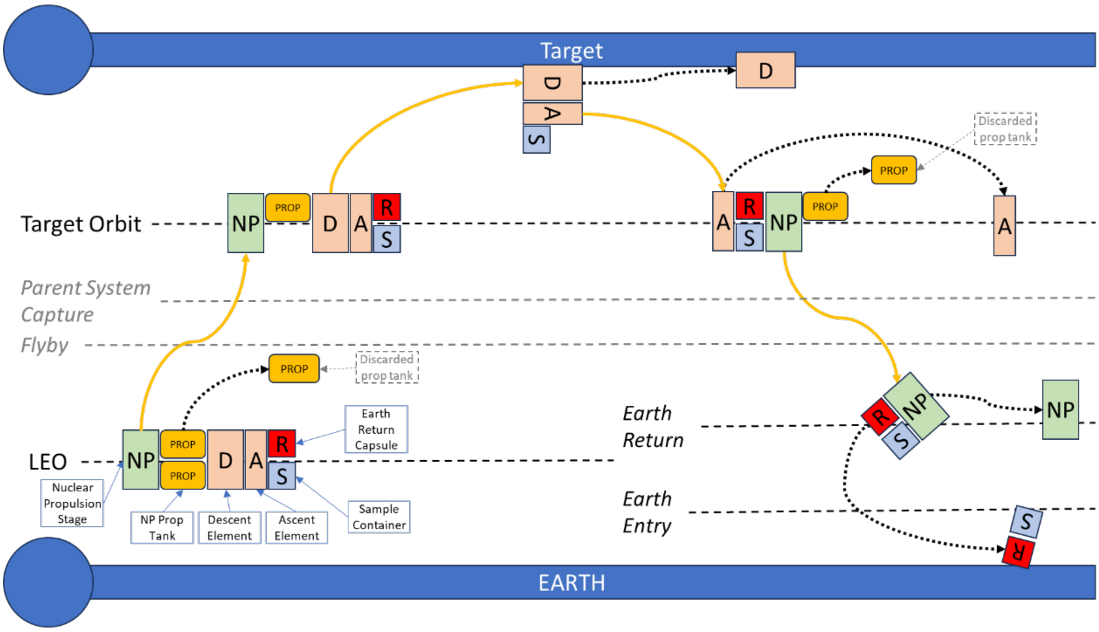

Figure 3. 'Bat Chart' showing the Generic Mission Architecture for Sample Return Mission using Big Rockets and Nuclear Propulsion. The launch 'stack' from LEO is as defined in Figure 2. The architecture of the sampling phase of the mission is similar to Apollo, in that the Propulsion module stays in orbit, while the Descent/Ascent vehicle goes down to the surface, where the Descent stage remains. On return to Earth, the Earth return capsule, carrying the sample container, separates from the propulsion module prior to atmospheric entry.

The Descent stage carries a suite of instruments to characterize the sampling site and provide appropriate context for the acquired samples. We selected a nominal mass of 20 kg for this instrument suite, which should be ample to allow instruments such as cameras, spectrometers, seismometers, compact laser ablation spectrometers to be included. The Descent stage has a power source sufficient to conduct surface operations. This could be solar if there is sufficient illumination from the Sun, batteries if the surface mission duration is short enough, or a small Radioisotope Thermal Generator like an MMRTG or a more compact RTG whose source is Americium [29] or another radioactive element. For the study, the assumed power source for the descent stage was a primary battery for the entirety of the surface mission duration.

After samples are collected, the Ascent stage launches from the surface with the Sample container, leaving the Descent stage behind on the surface, and conducts an in-orbit rendezvous with the Propulsion stage. The Ascent stage is powered by batteries, assuming the ascent and rendezvous is a short duration mission. For Mercury, which requires the largest ascent/descent *Dn* we studied, the Ascent Vehicle in our model is based on the Mars Ascent Vehicle specified by the Mars Sample Return mission, which has a solid rocket motor with and I_{SP} of 307s, a mass of 450 kg, and a *Dn* capability of approximately 4 km/s [30]. For other targets the mass of the ascent vehicle was scaled according to the required *Dn*.

Following rendezvous, the combined stack then returns Earthwards while leaving the spent ascent stage behind, in orbit at the target body. On approach to Earth, the Earth return vehicle, including the precious sample container, separates from the Propulsion module and enters the atmosphere. The Earth return vehicle mass is 57 kg in our architecture model, which is consistent with the mass of other Earth entry vehicles, e.g. OSIRIS-Rex [31]. A variant of this architecture could have the Earth return vehicle captured into an elliptical orbit, for later recovery by a crewed (or robotic) mission, if planetary protection concerns at the time prohibit direct return of samples to Earth's surface.

On return to Earth, the atmospheric entry speed is set at under 13.5 km/s, consistent with our understanding of the capabilities of current re-entry vehicle designs. The sample size is set at 250 g, twice that returned from asteroid Bennu by OSIRIS-Rex [31].

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A fundamental constraint for the architecture is that the launch 'stack' mass has to be consistent with a refueled Starship capability for the C_3 value selected, as shown in Figure 4, where the key Starship parameters are 100 mt for dry (non-payload) mass and 1200 mt of propellant with a specific impulse of 380 s [27].

Table 9 (below) summarizes the key driving assumptions of each of the elements in the mission architecture, highlighting the similarities and differences between the Nuclear Electric Propulsion and Nuclear Thermal Propulsion architectures.

Table 9. Parameters for NEP and NTP spacecraft.

Architecture Sizing Assumption	NEP Value	NTP Value
Nuclear-Propelled Vehicle		
➢ Mass (wet, max)	30,000 kg	60,000 kg
➢ Propellant Type	Xenon	Hydrogen
➢ Specific Impulse	3,100 s	900 s
➢ Mass Fraction of Propellant Tank	10%	40%
➢ Electrical Power for Propulsion	30-65 kW	*N/A*
➢ Thrust Level	*Derived*	20 kN
Descent Vehicle		
➢ Mass (wet)	230-265 kg	230-265 kg
➢ Propellant Type	Bi-Prop (LOx/CH_4)	Bi-Prop (LOx/CH_4)
➢ Specific Impulse	309 s	309 s
Ascent Vehicle		
➢ Mass (wet)	210 kg	230 kg
➢ Propellant Type	Bi-Prop (LOx/CH_4)	Bi-Prop (LOx/CH_4)
➢ Specific Impulse	309 s	309 s
Sample Container & Entry/Return Vehicle		
➢ Mass	57 kg	57 kg

Notes: *Assumed parameters for the vehicles in the architectures illustrated in Figures 2 and 3; differentiating between the Nuclear Electric and Nuclear Thermal Propulsion realizations.*

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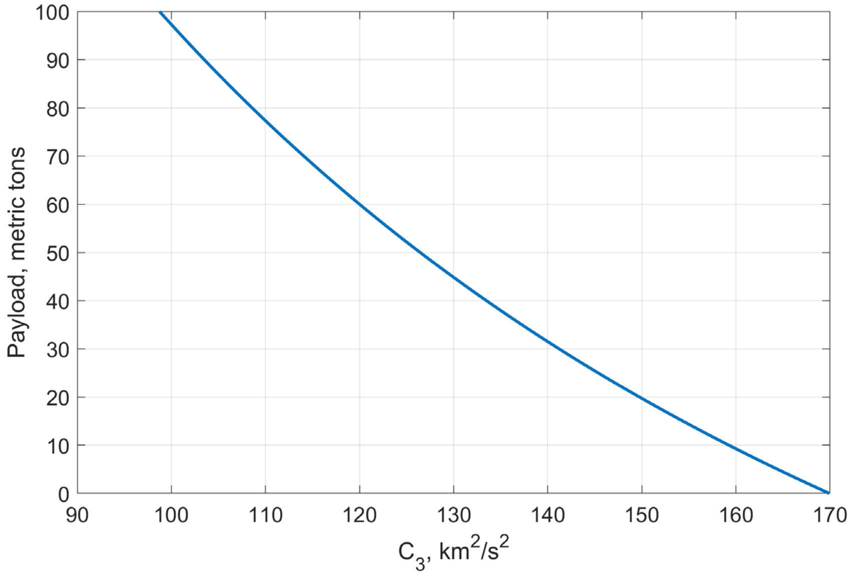

Figure 4. Example C_3 versus Payload Mass curve for Refueled StarShip from LEO.

5. Trajectory Designs for Sample Return Missions

In this section, we will examine trajectories for sample return missions. The first one we examined in more detail, using state-of-the-art mission design tools for both impulse and low thrust trajectories, was that of a sample return mission from Pluto. We selected Pluto because we thought it to be one of the hardest targets to get to in our solar system, due to its distance from the Sun; and the high degree of scientific interest in studying Pluto in much greater detail, following the surprising results obtained by the New Horizons flyby mission [32].

The results of our analysis are shown in Figure 5 for a spacecraft equipped with Nuclear Thermal Propulsion, with I_{SP} and spacecraft mass properties as shown in Table 9. The roundtrip time of flight (ToF) is just over 43 years, a number we aimed to achieve because it is close to the career span of a typical planetary scientist. But the mission profile has an arrival at Pluto 16 years after launch, and the spacecraft spends just over a year in orbit, which should yield many new scientific insights into Pluto, well before returning to Earth. We note that a Pluto Orbiter was one of the missions studied as input to [10], but ultimately seen as infeasible within current cost and technology constraints.

The landing/sampling/return-to-orbit phase of our mission can take place at any time within the year spent in orbit and takes an estimated 2.2 km/s of *Dn* to accomplish. The lander sub-vehicle for this segment of the mission uses more conventional biprop chemical propulsion, to avoid having to take the heavy nuclear reactor down to the surface. The return leg takes just over 23 years. The *Dn* required by the NTP spacecraft is 17 km/s, which is slightly lower than the number for the Hohmann transfer given above in Table 7. The mission duration may seem like a long time—but imagine if the long-lived Voyager spacecraft, which launched in 1977, instead of heading out into interstellar space, were now returning to Earth with samples from the most distant parts of our solar system.

Figure 6 shows the Pluto Sample Return Trajectory for a spacecraft equipped with Nuclear Electric Propulsion, with I_{SP} and spacecraft mass properties as shown in Table 6. The most notable difference between Figs 5 and 6 is the higher *Dn* required for the NEP mission, which is typical for low-thrust trajectories when compared with impulsive ones.

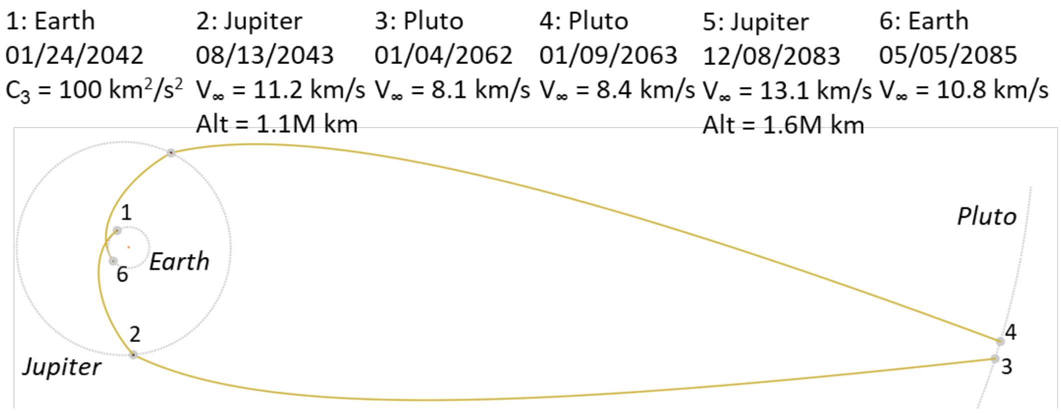

Figure 5. Trajectory design for a sample return mission from Pluto using Nuclear Thermal Propulsion. A Jupiter gravity assist maneuver provides a 4.5 km/s Dn 'kick' on the outbound trajectory, and a similar slowdown on the inbound trajectory. Total mission Dn is 19.2 km/s.

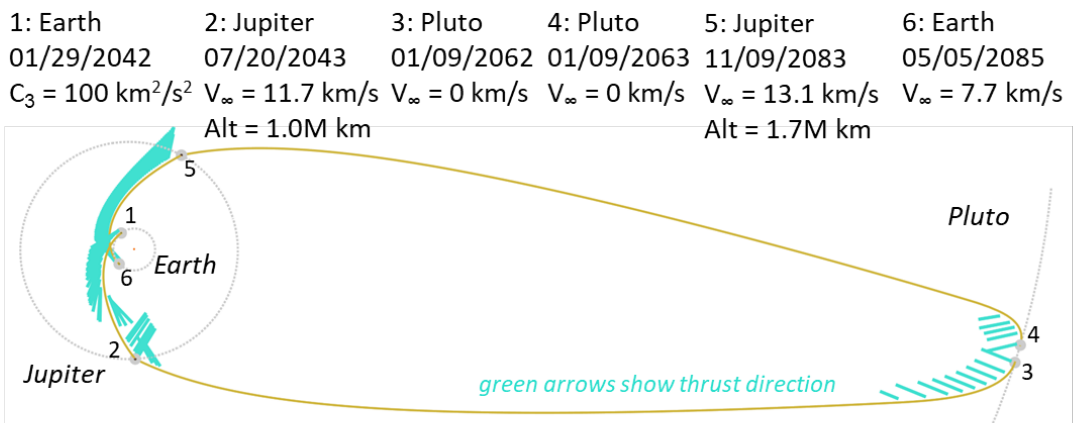

Figure 6. Trajectory design for a sample return mission from Pluto using Nuclear Electric Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 100 km^2/s^2. Total mission Dn is 27.1 km/s.

Figure 7 shows the longest duration trajectory we analyzed, which was for Arrokoth, the most remote object in our solar system encountered by one of our spacecraft so far, at more than 40 AU distant from the Sun, flown by the New Horizons spacecraft in 2019 [33]. The 43.3-year round-trip trajectory shown in the figure is for an NEP mission with a total *Dn* of 22.8 km/s. Similar duration trajectories were found for an NTP mission, with lower *Dn* requirements (by about 5 km/s). This mission stretches our goal of having the mission duration be less than 40 years, but the example is of interest because it probes the limits of what may be possible, given our assumptions of a Big Rocket and Nuclear Propulsion.

Figure 8 shows the ambitions mission designers can rise to, provided they have access to the combination of a really Big Rocket and Nuclear Propulsion. The trajectory shown has our spacecraft flying by Jupiter to pick up an enormous 9.5 km/s boost in *Dn* via gravity assist to catch up with probably the most famous long-period comet, named after the English astronomer Halley, which has a retrograde orbit with a 76-year period. After rendezvous-ing with Halley's Comet during its quiescent period as it approaches the Sun, our spacecraft flies past Jupiter again on its way back to Earth, switching back to a prograde orbit around the Sun as it does so. Samples collected at Halley's comet are returned to Earth two years after its next perihelion event.

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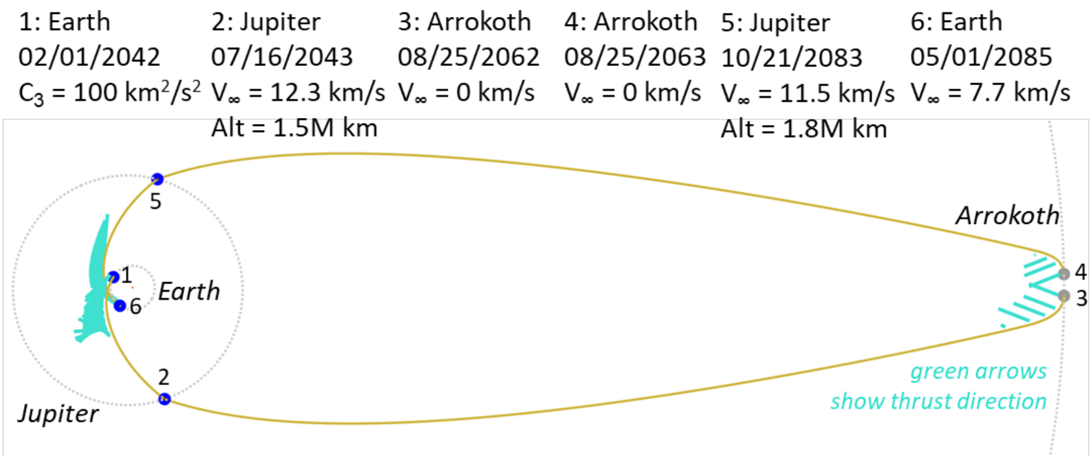

Figure 7. Trajectory design for a sample return mission from the Kuiper Belt Object Arrokoth using Nuclear Electric Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 100 km²/s²; a Jupiter gravity assist maneuver provides a 8.2 km/s Dn 'kick' on the outbound trajectory, and a similar slowdown on the inbound trajectory. Total mission Dn is 22.8 km/s.

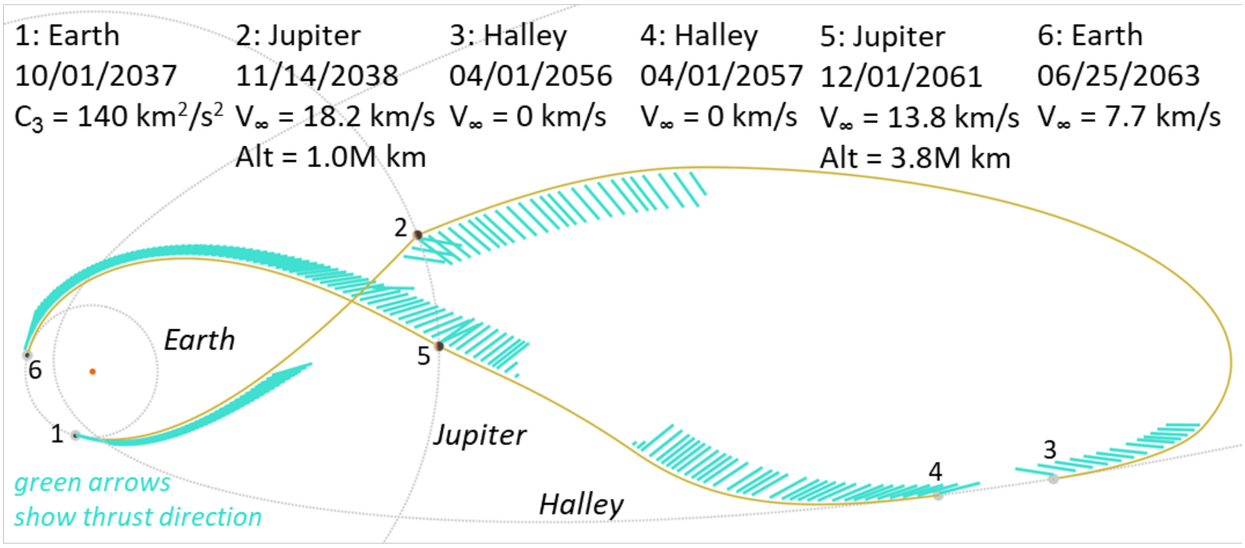

Figure 8. Trajectory design for a sample return mission from the retrograde Halley's Comet using Nuclear Electric Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 140 km²/s²; a Jupiter gravity assist maneuver provides a 9.5 km/s Dn 'kick' on the outbound trajectory, and a slowdown by 4.9 km/s on the inbound trajectory. Total mission Dn is 28.2 km/s and the mission duration is 25.8 years.

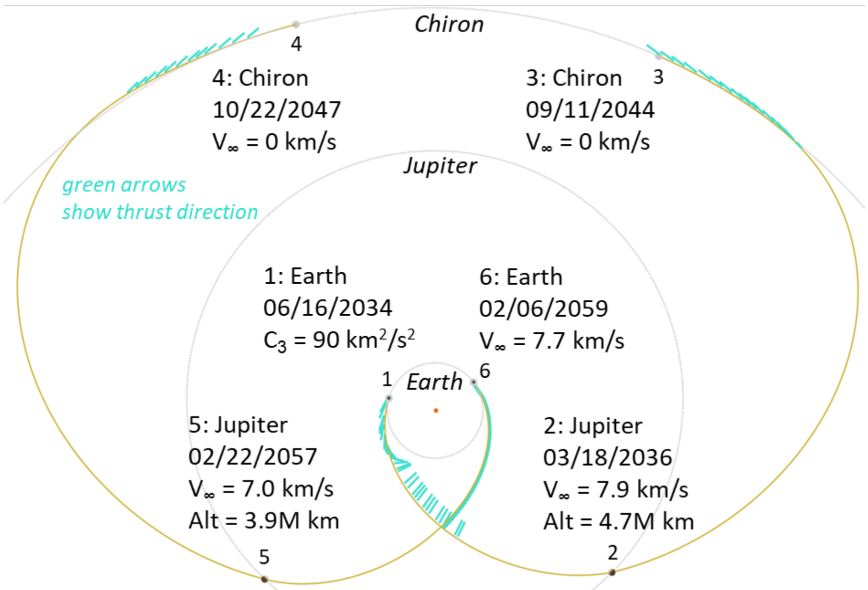

Figure 9. Trajectory design for a sample return mission from the Centaur Chiron using Nuclear Electric Propulsion. A refueled Starship launch from LEO is assumed with a C_3 of 90 km^2/s^2. Total mission Dn is 11.8 km/s and the mission duration is 24.7 years.

Moving inwards towards some more accessible targets, Figure 9 illustrates an NEP sample return mission to/from Chiron, one of the Centaurs, a population of captured Kuiper Belt Objects (KBOs) whose orbits lie between Jupiter's and Saturn's. Never having been close to the Sun, the Centaurs are expected to be relatively pristine examples of the KBO population. A Centaur rendezvous and lander is included in the New Frontiers mission target list identified in [10]. The trajectory shown again has a flyby of Jupiter less than two years after launch to pick up a boost in *D*◎ via gravity assist, then a 8.5-year cruise out to rendezvous with Chiron. Our spacecraft spends just over three years at Chiron, characterizing this exotic object from orbit and collecting a sample to return to Earth. The return journey takes 11.3 years, for a total roundtrip time of 24.7 years. The total *Dn* for this mission is 11.8 km/s which includes a 3 km/s allocation to reduce the speed of our spacecraft relative to the Earth (V-infinity) so that the atmospheric entry speed is under 13.5 km/s, consistent with what current re-entry vehicle designs can achieve. The re-entry vehicle speed for Stardust, for example, was 12.5 km/s {4]. This constraint has been applied to all our mission trajectory studies.

Having examined missions to Kuiper belt objects, comets, and Centaurs, we round out the collection of minor planets in Figure 10 with an NEP mission to/from the largest main-belt asteroid, Ceres. As a much closer target than the previous examples, a sample from Ceres could be returned less than nine years from launch. The addition of a gravity assist from Mars limits the NEP *D*◎ to 11.9 km/s; the NTP equivalent is slightly lower at 10.6 km/s due to the short burn times at departure and escape, where thrusting is most efficient. The NEP solution still provides a more efficient overall mass fraction due to its relatively high I_{SP}.

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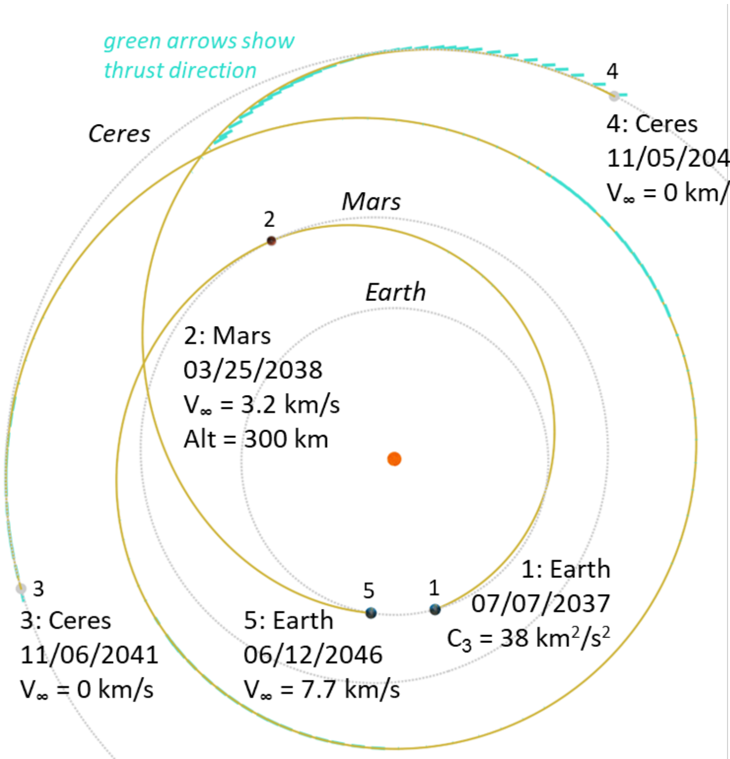

Figure 10. Trajectory design for a sample return mission from the dwarf planet Ceres using Nuclear Electric Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 38 km^2/s^2; and a Mars gravity assist maneuver provides a 3.4 km/s Dn 'kick' on the outbound trajectory. Total mission Dn is 13.0 km/s and the mission duration is 8.9 years.

Next we consider the moons of the giant planets. Figure 11 illustrates an NEP sample return mission to/from the surface of Triton, a moon of the ice giant Neptune, which is believed to be a captured KBO (because of its unusual retrograde orbit around Neptune). When Voyager 2 flew past in 1989, it saw plumes coming off Triton's surface, making it one of the few objects in the solar system known to exhibit active plumes. The trajectory shown includes a gravity assist from Jupiter en route to Neptune. Once our spacecraft becomes gravitationally captured by Neptune, it follows a spiral trajectory to achieve circular orbit around Triton. The entire spiral phase lasts 3.5 years and consumes 11 km/s of the 25 km/s total NEP *D*⊚ . After a year of reconnaissance/sample collection at Triton, the spacecraft returns to Earth, helped by another gravity assist from Jupiter.

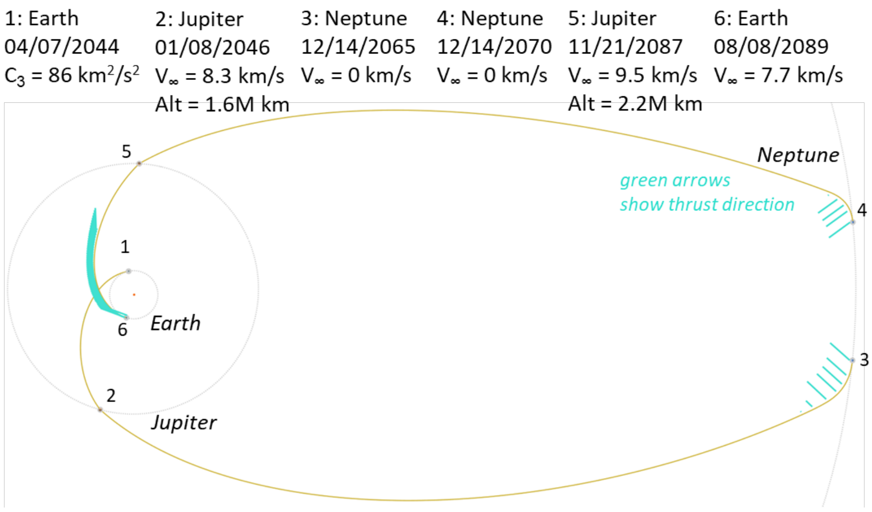

Figure 11. Trajectory design for a sample return mission from Triton using Nuclear Electric Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 86.0 km^2/s^2; which results in a flight time out to

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Neptune of 21.5 years. The spacecraft spends 1.5 years at Triton, before returning to Earth via a gravity assist from Jupiter. Total mission Dn is 27.5 km/s and the mission duration is 45.4 years.

One of the most interesting bodies in our solar system, because of its potential habitability, is one of the icy moons of Jupiter: Europa. Figure 12 illustrates the relative ease with which this ocean world can be reached if we have a Big Rocket and Nuclear Thermal Propulsion. One-way trip time out to Jupiter is three years, with a similar return flight duration. Time spent in the Jovian system is 3.7 years. Contrast this with the Europa Clipper mission, which uses conventional biprop and launches on a Falcon Heavy launch vehicle, and will take 5.5 years to get to Jupiter, then spend 4 years studying Europa [34]. For our NTP solution, insertion into a 6-month orbit with perijove at 920,000 km radius requires 1100 m/s of *Dn* following a gravity assist from Ganymede. A perijove raise maneuver of 100 m/s at apoapsis of the capture orbit commences a tour of the Galilean satellites. There are several options to get into orbit around Europa, before landing and collecting a sample [35]. We opt for a tour that has relatively low *Dn* (700 m/s) and ToF (9 mo) at the expense of higher total ionizing dose of radiation (1000 krad), which is commensurate with the "99-35" tour in [35]. These values are doubled for the return sequence to escape Jupiter. The V-infinity shown is reduced from 9.6 km/s to 7.7 km/s via a 1.9 km/s *Dn* burn before the spacecraft approaches Earth, so that the re-entry speed into Earth's atmosphere is < 13.5 km/s. Total mission *Dn* is 9.0 km/s including 3.4 km/s descent to Europa's surface plus ascent back to orbit. The *Dn* required by the NTP system is only 5.6 km/s, which is much lower than the Hohmann-transfer prediction in Table 7. This reduction in *Dn* arises from taking advantage of the Oberth effect in Jupiter's large gravity well [36] and multiple gravity assists from the Galilean satellites. An NEP trajectory solution has similar roundtrip flight time, with higher *Dn*. In this instance, the impulsive nature of NTP is probably the better solution, in comparison to low thrust NEP; impulsive propulsion burns allow solutions that spend less time in the most severe radiation environment close to Jupiter.

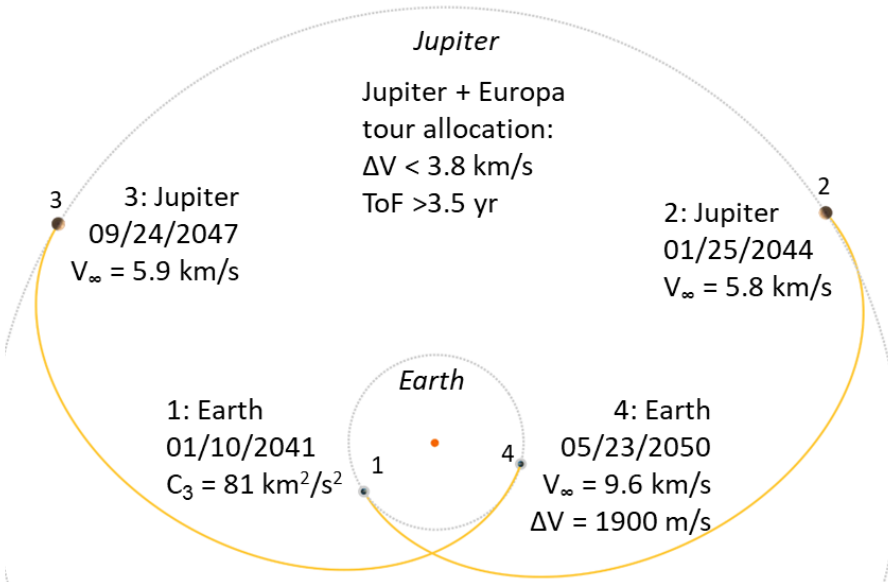

Figure 12. Trajectory design for a sample return mission from Europa using Nuclear Thermal Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 81 km^2/s^2; which results in a flight time out to Jupiter of 3.0 years. The spacecraft spends 3.7 years at Jupiter before returning to Earth. Total mission Dn is 9.0 km/s and the mission duration is 9.4 years.

Saturn's moon Enceladus also makes a tantalizing target due to its potential to harbor life. Enceladus is harder to reach than Europa due to Saturn's farther distance from the sun and smaller satellites (with the exception of Titan) to pump the spacecraft orbit down to Enceladus. However, Enceladus' relatively small mass makes it easier to descend to the surface and return to orbit. An example round trip is illustrated in Figure 13. Following a 5-year cruise to Saturn the spacecraft enters a 6-month period orbiting Saturn with periapsis at 140,000 km radius, above the F ring. A maneuver at apoapsis then raises periapsis to 1.2M km, providing a low V_∞ at Titan to commence a tour of Rhea, Dione, and Tethys before reaching Enceladus. We allocate 2.5 years and 1.8 km/s *Dn* from Saturn approach to a 100-km altitude orbit at Enceladus as a notional design point within a larger trade space of tour options [37]. These values are doubled for the return trip. On the return trip, the spacecraft first performs a flyby of Earth followed by a 600-m/s maneuver to lower the entry speed to below the 13.5 km/s limit. The NTP *Dn* is 4.2 km/s, taking advantage again of the Oberth effect and gravity assists from the moon system, and provides a launch opportunity nearly every year.

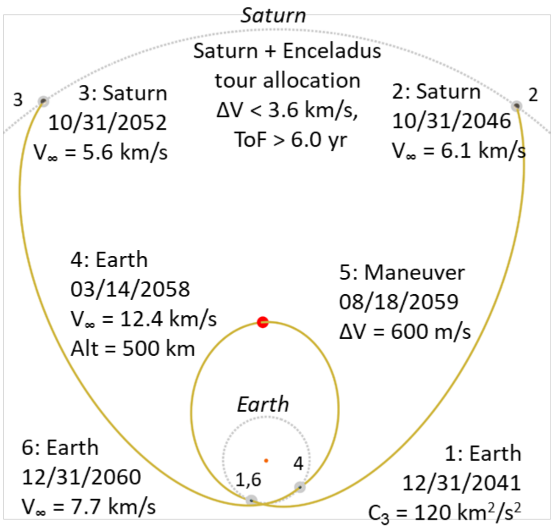

Figure 13. Trajectory design for a sample return mission from Enceladus using Nuclear Thermal Propulsion. A refueled Starship launch from LEO is assumed, with a C_3 of 120 km²/s²; which results in a flight time out to Saturn of 4.8 years. The spacecraft spends 6 years at Saturn before returning to Earth. Total mission Dn is 4.8 km/s and the mission duration is 19.0 years.

Finally, we turn to the inner solar system. Figure 14 depicts an NTP sample return mission to/from Mercury, which as seen in Table 7 is a challenging target to reach and return from. In fact, the *D*◎ required to get to Mercury, then orbit and land is so high that we could not find trajectory solutions that used purely propulsive means, instead relying on multiple gravity assists to provide the necessary additional *Dn.*

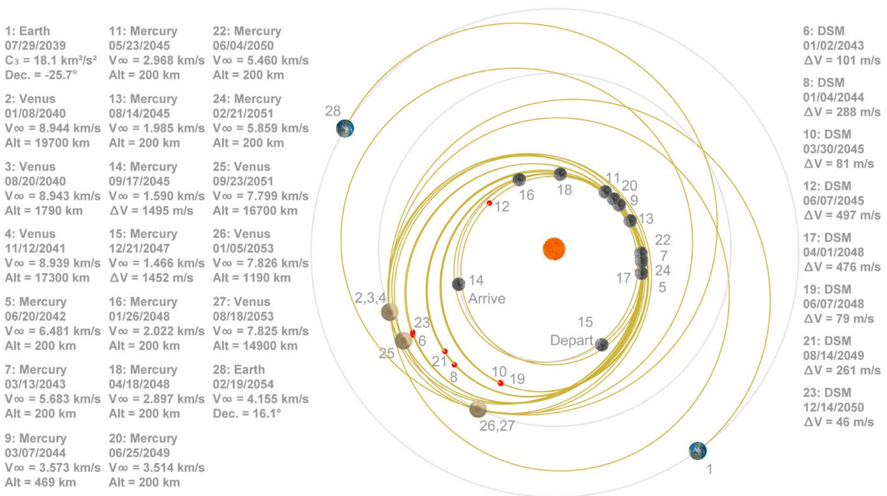

Figure 14. Trajectory design for an NTP sample return mission from Mercury. The round trip takes advantage of several gravity assists from Venus and Mercury. Total mission Dn is 11.7 km/s and the mission duration is 14.6 years with a two-year stay at Mercury.

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6. Summary and Discussion

Table 10 summarizes the results of our mission architecture analyses. We found solutions for all targets listed using either Nuclear Electric or Nuclear Thermal Propulsion, but not always using both. Most of the solutions involve fairly high C_3 values, except for Mercury and Ceres, which suggest that solutions using smaller rockets may exist for those targets. In fact, referring back to [10] we know that to be the case for Ceres.

Table 10. Key parameter estimates for sample return missions.

Destination	Nuclear Electric Propulsion				Nuclear Thermal Propulsion				Chemical
	C_3 @ Departure (km²/s²)	*Dn* (km/s)	Mission Duration (yrs)	Wet Mass (metric tons)	C_3 @ Departure (km²/s²)	*Dn* (km/s)	Mission Duration (yrs)	Wet Mass (metric tons)	Ascent/Descent *Dn*, one-way (km/s)
Pluto	100	24.9	43.3	28.7	100	17.0	43.3	Not Found	1.1
Europa	76	14.6	9.4	19.5	81	5.6	9.4	27.4	1.7
Chiron	87	11.6	24.7	17.5	87	9.2	26.6	Not Found	0.1
Mercury	16	42.0	13.8	Not Found	18	4.9	14.6	59.0	3.4
Enceladus	118	23.0	14.1	26.3	120	4.2	19.0	10.0	0.3
Halley	140	28.0	25.8	28.5	155	19.0	25.8	Not Found	0.1
Ceres	38	11.9	8.9	17.9	34	10.6	8.9	Not Found	0.55
Triton	86	25.0	45.4	27.0	120	6.7	39.3	45.7	1.25
Miranda	89	23.8	34.5	25.1	120	7.0	24.0	53.8	0.25
Arrokoth	100	22.6	43.3	29.5	100	17.1	43.2	Not Found	0.1

***Notes**: Launch energy, mission duration and allowable launch stack mass (from LEO) solutions using a fully refueled Starship and the mission architecture depicted in Figures 2 and 3. "Not Found" means that a solution could not be found that closed in terms of launch stack mass for that trajectory.*

Table 11 shows an example mass breakdown for the Enceladus NEP Sample Return mission. The nuclear propulsion system, including propellant, is by far the largest percentage mass element of the flight system, which is typical of our model results.

Table 11. Mass breakdown for an Enceladus NEP Sample Return Mission.

Block Name	Dry Mass	Propellant Mass	Wet Mass
Launch Stack	*13,306*	*12,974*	*26,280*
Nuclear Propulsion Vehicle	11,137	12,901	24,038
Nuclear Prop Tank – Outbound	514	-	-
Nuclear Prop Tank – Arrival	230	-	-
Nuclear Prop Tank – Departure	202	-	-
Nuclear Prop Tank – Inbound	746	-	-
Descent Vehicle	215	49	263
Ascent Vehicle	207	24	231
Sample Container	19	-	-
Earth Entry Vehicle	38	-	-

***Notes**: Typical mass breakdown from our mission architecture model for an example Enceladus NEP Sample Return Mission.*

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In undertaking this study, we set out to see if it might be feasible using emerging technology to fill in the right-most column in the matrix of Solar System Exploration, introduced in Tables 1 and 2. With the availability of Big Rockets and Nuclear Propulsion, either Electric or Thermal, we found that it *is* possible to imagine missions that return samples from all across our solar system, as shown in Table 12.

The results we obtained for two of the Ocean Worlds: Europa and Enceladus, are particularly noteworthy. The roundtrip times for an NTP-equipped spacecraft are short enough at 9.4 years and 19.0 years, respectively, that they compare favorably with the timescales for one-way missions to each body (see [10], [34]). A Europa mission that can dart off to Jupiter, get into orbit around that giant planet then its moon Europa, quickly acquire samples from the surface, and return them to Earth, in a timeframe similar to the mission durations for Europa Clipper [34] and ESA's JUICE mission [38] could make a very compelling follow-on to those missions.

We found that NEP mission architecture solutions closed more often than NTP solutions, but NTP mission options, when available, could have shorter duration than NEP missions. Some NTP solutions we extended by adding time for a slower return to Earth, so that the architecture did not need to carry fuel for a propulsive burn to slow down prior to Earth entry.

We did not examine here a solution to return a sample from Venus. Venus' gravity field is similar to Earth's, so sending a vehicle down to the surface and back to orbit again is incredibly challenging. That degree of difficulty is compounded by the surface conditions at Venus, with temperatures in excess of 460 deg Celsius, and an atmospheric pressure of 92 bars. Mission concepts like VATMOS-SR [39] have been proposed that skim the upper atmosphere, sucking in a sample to return to Earth. Such a sample will probably be highly fractionated, and depending on the altitude it was collected at, may not be representative of the well-mixed lower atmosphere of Venus.

We also did not look for solutions to return samples from the atmospheres of the outer planets. No compelling science case has been made for such samples at this time. In situ measurements, e.g. of the isotopic ratios of noble and trace gases, would seem to satisfy the current curiosity of the science community, at least as expressed in the most recent planetary science decadal surveys [10-11].

We did not factor in cryogenic preservation of samples into our study. Cryogenic preservation of material collected from the surface of a comet was studied as input to [10] but ultimately not included in the final recommendations, because of perceived cost and low technical maturity. We hope these problems can be addressed by the time nuclear propulsion becomes available for planetary science missions.

We did not include mobility systems explicitly in our study. Some degree of mobility can be imagined if lander vehicles can be designed to 'hop' from one surface location to another, as achieved by Surveyor 6 [40] and demonstrated again more recently by ISRO's Vikram lunar lander [41]. The value of adding a hopping capability to a wide range of missions is discussed in [40]. More advanced mobility, e.g. [42], may be needed for some targets if surface reconnaissance from orbit cannot identify the most suitable sample locations.

A degree of freedom we did not explore in this paper is reducing the mass of the primary spacecraft. One can readily see from Equations (1)-(3) that this would have a lesser effect than changing the I_{SP}, which is an exponential effect. Nonetheless, it is surprising that there has been so little progress in reducing mass (or power) for missions to the outer planets in the last few decades. The dry mass of flagship spacecraft like the Uranus Orbiter and Probe, and the Enceladus Orbilander in [10] for example are not significantly different than the dry masses of earlier spacecraft like Galileo, Cassini, and now Europa Clipper. Power needs have increased, not decreased. In this era of significant mass and power reductions for smaller spacecraft like CubeSats and NanoSats [43] we think the planetary science community could be better served by further appropriate investments in and demonstrations of deep space missions like the successful MarCO CubeSats [44] that have significantly reduced Size, Weight and Power (SWaP).

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Table 12. Updated version of the Matrix of Solar System Exploration.

Target	Flyby	Orbiter	Lander/InSitu	Mobility	Sample Return
Mercury	Yes	Yes	*Yes*		*Yes*
Venus	Yes	Yes	Yes	**Yes**	
Moon	Yes	Yes	Yes	**Yes**	**Yes**
Mars	Yes	Yes	Yes	Yes	**Yes**
Jupiter	Yes	Yes	Yes		*Yes*
Saturn	Yes	Yes	**Yes**		*Yes*
Uranus	Yes	**Yes**	**Yes**		*Yes*
Neptune	Yes	**Yes**	*Yes*		*Yes*
Pluto	Yes	*Yes*	*Yes*		*Yes*
Jovian Comets	Yes	Yes	Yes		**Yes**
Trojans	Yes	*Probably*	*Probably*		*Probably*
Centaurs	**Yes**	**Yes**	**Yes**		*Yes*
NEOs	Yes	Yes	Yes	Yes	Yes
Main Belt	Yes	Yes	**Yes**		**Yes**
Halley's Comet	Yes	*Yes*	*Yes*		*Yes*

***Notes**: An updated version of the Matrix of Solar System Exploration, with entries enabled by the use of Big Rockets plus Nuclear Propulsion shown in Italics, and shaded. Trojans missions are labeled as Probable here; although we did not study them explicitly, they should be easier to reach than Centaurs.*

We are not the first to propose the use of nuclear propulsion to expedite exploration of the outer planets – others have trod this path before us [45-48]. Nor do we expect this work to be the last word on how sample return missions are to be carried out in the future. The planetary science community is very inventive, and we anticipate that the challenge we have posed in this paper, i.e. that we are approaching the limits of what can be done to return samples using more conventional approaches involving less capable rockets, chemical propulsion, gravity assists, possibly aerocapture [49] etc. will be taken up and scrutinized heavily. Sample return missions to Io [50] and Enceladus [51] have been proposed in the literature by credible teams of planetary scientists and mission designers. Both missions aim to collect limited quantities (~100g at Io; ~2g at Enceladus) of samples by flying through plumes (volcanic at Io, liquid water at Enceladus). The planetary science community must decide on the value of such small sample quantities, relative to the cost of the mission. Sample sizes returned by the Discovery missions Genesis and Stardust were similarly small but seen as excellent value at the price tag of a Discovery mission. We welcome such debate, and hope that it will result in the prioritization of more sample return mission in future planetary science decadal surveys.

The planetary science community is by no means the only one that would benefit from the maturation of NTP and NEP nuclear propulsion technology: a 2021 report commissioned by the National Academy of Sciences pointed out their "great potential to facilitate the human exploration of Mars" [52]. If we are to evolve into a multi-planetary species, as some have argued [53], then nuclear propulsion, in combination with big rockets like Starship, may be the "double-hulled canoe" [54] that propels us towards that future. These aspirations have motivated and will continue to motivate the advancement of both Big Rockets and Nuclear Propulsion, from which planetary science can reap the benefits, as argued in this paper.

As a last comment, with the matrix more or less fully populated, as depicted in Table 12, does that state represent a time in which planetary science is "done"? It's a great question, one which we will leave for future generations of planetary scientists and engineers to answer.

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Abbreviations

The following abbreviations are used in this manuscript:

APL	Applied Physics Laboratory
DRACO	Demonstration Rocket for Agile Cislunar Operations
DTE	Direct To Earth
GSFC	Goddard Space Flight Center
ISRO	Indian Space Research Organization
JETSON	Joint Energy Technology Supplying On-Orbit Nuclear Power
JPL	Jet Propulsion Laboratory
JUICE	Jupiter ICy moons Explorer
KBO	Kuiper Belt Object
LEO	Low Earth Orbit
MarCO	Mars Cube One
MMRTG	Multi-Mission Radioisotope Thermal Generator
MSR	Mars Sample Return
NASA	National Aeronautics and Space Administration
NASEM	National Academy of Sciences, Engineering and Medicine
NEP	Nuclear Electric Propulsion
NRC	National Research Council
NTP	Nuclear Thermal Propulsion
OSIRIS-Rex	Origins, Spectral Interpretation, Resource Identification, Security-Regolith Explorer
OWL	Origins, Worlds, Life
RTG	Radioisotope Thermal Generator
SEP	Solar Electric Propulsion
SLS	Space Launch System
SR-1	Space Reactor-1
SWaP	Size, Weight and Power

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