Strategic Implications of Human Exploration of Near-Earth Asteroids

Bret G. Drake
NASA Lyndon B. Johnson Space Center
Houston, Texas 77058

Abstract - The current United States Space Policy [1] as articulated by the White House and later confirmed by the Congress [2] calls for "*[t]he extension of the human presence from low-Earth orbit to other regions of space beyond low-Earth orbit will enable missions to the surface of the Moon and missions to deep space destinations such as near-Earth asteroids and Mars.*" Human exploration of the Moon and Mars has been the focus of numerous exhaustive studies and planning, but missions to Near-Earth Asteroids (NEAs) has, by comparison, garnered relatively little attention in terms of mission and systems planning. This paper examines the strategic implications of human exploration of NEAs and how they can fit into the overall exploration strategy. This paper specifically addresses how accessible NEAs are in terms of mission duration, technologies required, and overall architecture construct. Example mission architectures utilizing different propulsion technologies such as chemical, nuclear thermal, and solar electric propulsion were formulated to determine resulting figures of merit including number of NEAs accessible, time of flight, mission mass, number of departure windows, and length of the launch windows. These data, in conjunction with what we currently know about these potential exploration targets (or need to know in the future), provide key insights necessary for future mission and strategic planning.[1,2]

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[2] IEEEAC paper #1069, Version 5-Draft, Updated 8/18/2011

1. CAPABILITY DRIVEN FRAMEWORK

During the past few years the direction for future human exploration beyond low-Earth orbit [3] has undergone revision and a broader, less destination specific framework has emerged. This strategy, referred to as a Capability Driven Framework (CDF) [4], is based on the idea of an ever expanding human presence beyond low-Earth orbit in terms of duration and distance from the Earth. It is based on evolving capabilities which are utilized after operational experience has been established from less demanding missions. In theory, the Capability Driven Framework enables multiple destinations and provides increased flexibility, greater cost effectiveness, and sustainability. But the utility of a Capability Driven Framework can only be measured and fully understood when put into context of actual missions. Thus, to help formulate the strategies, technologies, and systems needed to support the framework, example destinations are being examined including low-Earth orbit, Geostationary missions, cis-lunar space (including lunar fly-by, lunar orbit, and lunar surface), Near-Earth Asteroids, as well as missions to the Mars and the moons of Mars. Before examining how human missions to Near-Earth Asteroids fit into the overall Capability Driven Framework a brief review of the missions associated with the CDF is necessary.

Geostationary Orbits (GEO)

This mission class includes missions to GEO or other high-Earth orbit destinations generally for the purpose of deploying or repairing ailing spacecraft. Due to the high delta-v associated with these destinations, a split-mission approach is typically used where the crew is sent to the destination separate from the cargo assets to be used at the destination. The cargo assets can include habitats, mobility systems, robotic systems, and repair equipment.

Earth-Moon Libration

This mission class includes missions to the Earth-Moon L1 or L2 points or high lunar orbit. As with the GEO mission, cargo for these missions is sent separately from the crew. L1 can also serve as a staging node for other destinations such as to the lunar surface, NEAs, or perhaps even Mars. Thus, crew missions to L1 may serve as the initial crew transport leg at the beginning or end of a different mission class.

Lunar Surface

Missions to the lunar surface will encompass a range of mission durations, beginning with short stays to prove the performance of the systems, to longer duration test beds for more challenging missions such as the surface of Mars. As with both the GEO and L1 missions, a split mission approach is typically used separating the crew from cargo.

Near-Earth Asteroids

This mission class represents human missions to and from asteroids which are in close proximity to Earth, orbit perihelion typically less than 1.3 AU. Near-Earth Asteroids are of interest because they represent a class of missions which truly leave Earth vicinity[3]. Since these missions are conducted in heliocentric space and the orbits of NEAs have long synodic periods, perhaps decades long (discussed later), it becomes very difficult to pre-deploy mission assets prior to the crew mission. Thus, these missions are typically constructed as all-up missions, whereby all of the required mission assets are transported with the crew (deep space habitat, destination exploration systems, and Earth entry vehicle). Characteristics of a typical NEA mission design will be discussed later in subsequent sections.

Mars Orbit

This mission class includes missions to the moons of Mars (Phobos and Deimos) as well as Mars orbit. Missions to Mars occur approximately every 26 months. Since these missions avoid planetary surfaces, the crew is exposed to the deep space environment for the entire mission duration. Thus, these missions are generally constructed to reduce this crew exposure by flying the trajectories as fast as possible within the constraints of the propulsion technologies and number of heavy lift launches. Since missions to Mars occur on a frequent basis (every 26 months), pre-deployment of mission exploration vehicles is usually employed.

Mars Surface

This mission class represents missions to the surface of Mars. Strategies for exploring the surface of Mars typically utilize pre-deployed cargo vehicles and flying lower energy conjunction class missions. Details of this type are consistent with the NASA Mars Design Reference Architecture 5.0 [5].

2. NEA TRAJECTORY SCANS

When planning human missions to Near-Earth Asteroids it is important to establish an approach which can not only achieve the overarching mission objectives, but also, and more importantly, is safe and affordable. Mission concepts which can provide short round-trip missions are desirable in order to reduce the exposure of the crew to the deep-space radiation and micro-gravity environment. But it must also be recognized that as humans extend its reach outward towards Mars, concepts for supporting humans in deep space for extended durations, two to three years, will be necessary.

The process for establishing mission architectures begins with a thorough scan of potential round-trip trajectories for all known NEAs. Both high-thrust ballistic [6] and low-thrust [7] trajectories were constructed for each of the over 7,600 known NEAs as of February 3, 2011 within the Small Body Database of the International Astronomical Union Minor Planet Data Center[4]. Aggressive trajectory parameters were established in terms of total energy and mission duration such that all viable NEAs would be considered and none were eliminated a priori. This scan methodology resulted in over 79 million potential trajectories.

Each of these 79 million trajectories represent a continuum of mission duration and total energy required (as measured by total change in velocity, or delta-v). But in order to determine if a specific trajectory is viable, it must be put into a physical architecture, namely a first order approximation of mission mass.

3. EXAMPLE NEA ARCHITECTURES

A streamlined process was established to translate each of the 79 million trajectories into an estimate of total mission mass, and resulting number of launches, for four different mission architectures which are depicted in Figure 1.

All Chemical Propulsion Option

The all chemical propulsion option (Figure 1a) is a high thrust ballistic architecture utilizing cryogenic liquid oxygen and liquid hydrogen engines for each propulsive stage. Each stage is ideally sized for each of the major maneuvers and each stage is expended at the conclusion of the maneuver. The primary payloads for this architecture include a deep space habitat which is taken round trip, but expended at Earth return, as well as a Space Exploration Vehicle (SEV) which is left at the asteroid.

[3] Unlike GEO, HEO, L1, and lunar missions all of which remain in the vicinity of Earth, NEA missions must break from Earth's orbit and fly in trajectories bound by the Sun (Heliocentric trajectories). This results in missions which are longer in duration with limited Earth return opportunities for aborts.

[4] The trajectory scans produced by both the GSFC and JPL teams turned out to be a monumental effort requiring great expertise and computing time. The author greatly appreciates the hard work and dedication provided by the trajectory teams in this effort.

All Nuclear Thermal Propulsion Option

The all Nuclear Thermal Propulsion option (Figure 1b) is also a high thrust ballistic architecture similar in construct to the all chemical option. The only major difference is the use of a single higher performing nuclear thermal propulsion core stage with expendable drop tanks.

Solar Electric Propulsion for Deep Space Option

The Solar Electric Propulsion (SEP) for deep space option (Figure 1c) is characterized by the incorporation of a high power solar electric stage for a majority of the propulsive maneuvers. In this mission architecture, the SEP is used to spiral itself along with the mission payloads including the deep space habitat and space exploration vehicle to a high energy staging orbit. The crew is then transported from the Earth to the staging point separately in a high thrust chemical stage. The SEP is then used to transport the crew, deep space habitat, and SEV to and from the NEA. As with the previous two architectures, all elements in this architecture are expended in order to keep the total mission mass lower.

Solar Electric Propulsion / Chemical Propulsion Hybrid Option

The SEP / chemical hybrid option (Figure 1d) is similar to the all SEP for deep space option with the exception of the insertion of a high thrust chemical stage at the beginning of the deep-space portion of the mission (L1 escape). This additional chemical stage is transported to L1[5] via a separate SEP stage.

4. ELEMENT MODELING

The first step in constructing an appropriate physical architecture is to decompose each operational concept into functional elements of the architecture. The elements necessary for the in-space transportation function for missions to Near-Earth Asteroids are shown in Figure 2. Each of these elements performs a specific function within the operational concept for each reference mission as discussed below.

Orion

The Orion Multi-Purpose Crew vehicle was assumed to be the primary vehicle used for transportation of the crew from the surface of the Earth to Earth orbit. Within the Capability Driven Framework, the Orion utilizes the heavy-lift Space Launch System (SLS) as the launch vehicle. Orion's design legacy from the Constellation Program has shown to be a good fit with the mission needs of human exploration of Near-Earth Objects (such as 4 crew, a minimum of 21 days active operation, rendezvous and docking, contingency EVA capability, high speed direct entry, to name a few). When combined with a chemical propulsion stage the Orion can be used to transport crew to HEO, GEO, and lunar vicinity (L1, L2, or lunar orbit) in a single SLS launch. In addition to providing transportation function to LEO, Orion also is used by the mission crew for return from the hyperbolic return trajectory for a direct entry and landing on the Earth. For the NEA mission modeling, it was assumed that the inert mass and an independent delta-v capability of the Orion system were approximately 15 t and 1,450 m/s respectively.

Deep Space Habitat

The Orion system is a very capable vehicle for transporting crews in near-Earth space for mission durations of 21 days or less. But missions to Near-Earth Asteroids are much longer in duration. These longer duration missions require additional crew support systems including life support, food, accommodations, exercise, etc. all of which are dependent on the total duration of the mission. The estimated mass of the appropriate Deep Space Habitat as a function of mission duration is shown in Table 1. Additional food and water must also be included based on the mission duration. It was assumed that the life support system in the Deep Space Habitat would provide an adequate level of closure for air and water, and thus 2 kg/person/day were included for these items. This value is consistent with ISS level of closure of food and water.[8]

Space Exploration Vehicle

There are still many unknowns with respect to how exploration of Near Earth Asteroids will be conducted. How humans interact and explore NEAs will be dependent on the physical characteristics of the NEA itself, how fast it will be spinning or tumbling, the physical relief of the NEA, and the amount of dust present to name a few. Since it is unclear what that exploration strategy will be like, some equipment and capabilities will need to be taken with the crew in order to perform the necessary exploration. For modeling purposes, it was assumed that an independent vehicle would be taken with the crew for this purpose. The Space Exploration Vehicle concept envisions a small livable volume for a crew of two up to two weeks in duration.[9] The SEV would provide limited translational delta-v, on the order of 200-300 m/s, suit locks for quick EVA capability, and remote manipulation systems. Preliminary designs of the SEV have been conducted and a total vehicle mass of the SEV was assumed to be 6.7 t.

[5] For the SEP mission architectures, the Earth-Moon libration point L-1 was assumed as the Earth vicinity staging point. Subsequent analysis indicates that departures from a high Earth orbit may be a superior approach, but is subject to further assessment.

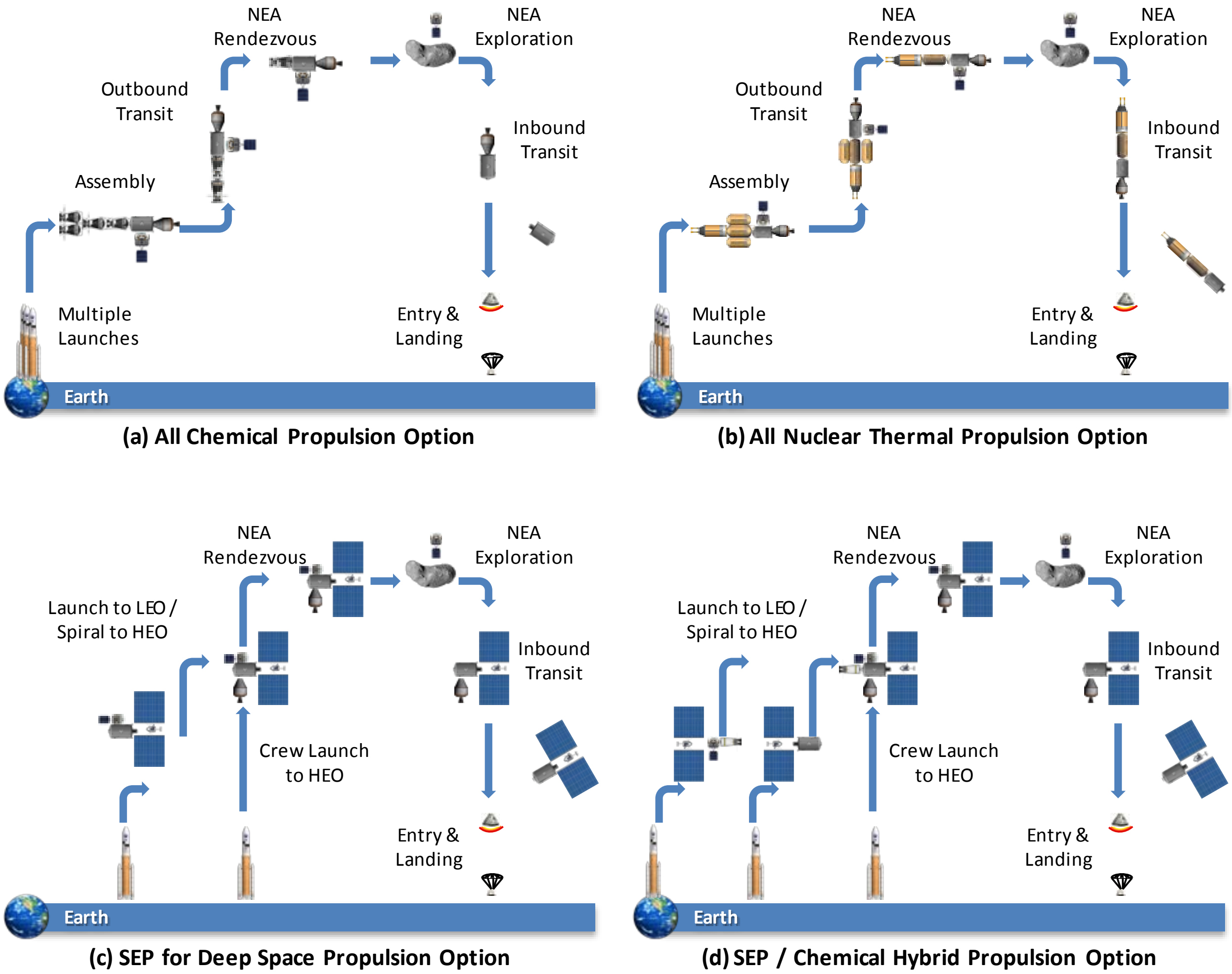


Figure 1 NEA Architectures Considered.

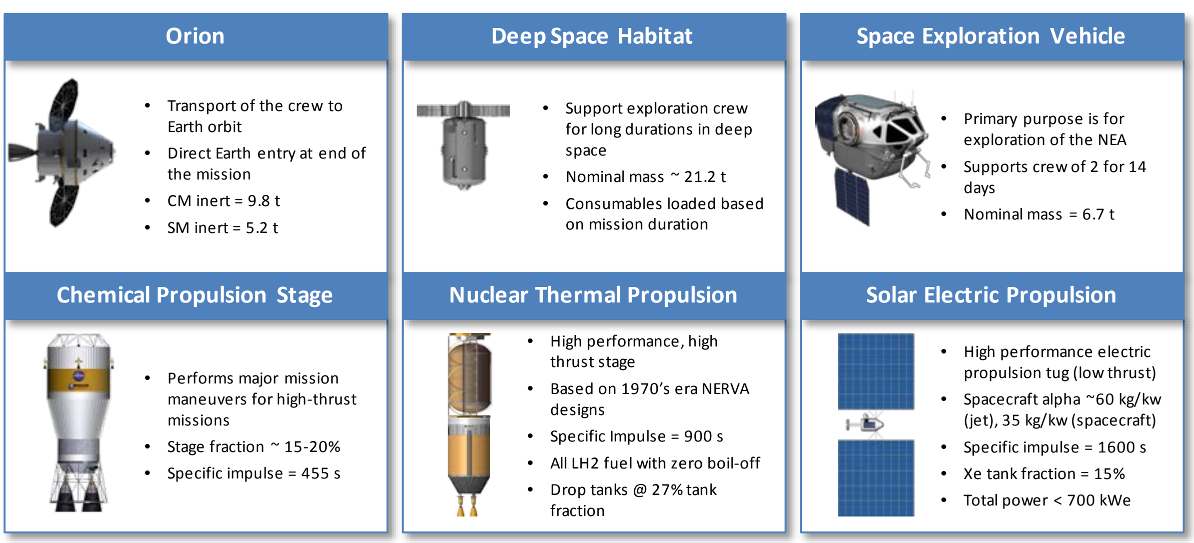


Figure 2 Exploration Elements

Table 1 Deep Space Habitat Sizing Schedule

Habitat Size	Min Duration (days)	Max Duration (days)	Low Mass (kg)	Expected Mass (kg)	High Mass (kg)
CM Alone	-	21	-	-	-
CM + Inflatable	21	50	3,600	4,000	5,200
Small Habitat	51	180	15,410	17,123	22,260
Medium Habitat	181	360	18,970	21,078	27,401
Large Habitat	361	540	22,529	25,033	32,543

Cryogenic Propulsion

The mass of each chemical propulsive stage was determined using the required change in velocity, delta-V (ΔV), for each trajectory burn, the mass of the total payload at the beginning of the phase (the payload mass, M_{begin}, typically mission systems plus remaining propulsive stages), the stage specific impulse (I_{sp}), and stage inert mass fraction (f_{inert})[6].

The "wet" mass for each propulsion stage of a particular mission phase is calculated as follows:

$$R = e^{\frac{\Delta V}{g * I_{sp}}} \qquad (1)$$

$$M_{propellant} = M_{begin} * (R - 1) * \frac{1 - f_{inert}}{1 - R * f_{inert}} \qquad (2)$$

$$M_{inert} = M_{propellant} * \frac{f_{inert}}{1 - f_{inert}} \qquad (3)$$

$$M_{stage} = M_{propellant} + M_{inert} \qquad (4)$$

where g is the Earth's acceleration level (9.806 m/s^2). Each major propulsive maneuver was modeled as an expendable stage. Thus, the total mission vehicle stack mass (M_{Total}) can be determined by summing the payload mass ($M_{payload}$) and each individual wet propulsion stage:

$$M_{Total} = M_{Payload} + \sum_{n=1}^{N} M_{stage_n} \qquad (5)$$

where N is the number of stages required for the mission. It should also be noted that some of the trajectory maneuvers become quite large for many of the difficult NEAs or as the trip time is decreased. As the required delta-v increases, the maneuver should be divided into separate similar size stages in order to improve the efficiency of the overall system. A fist order approximation of the appropriate staging strategy was determined by comparing the total delta-v for the maneuver to the characteristic exit velocity (V_{exit}) of the stage, where

$$V_{exit} = g * I_{sp} \qquad (6)$$

Determination of the appropriate number of stages and the percentage of the total delta-v for that specific maneuver were determined from the burn schedule shown in Table 2.

Table 2 High-Thrust Stage Burn Schedule

ΔV/Vexit	# Stages	% 1st	% 2nd	% 3rd	% 4th
0.0-0.6	1	100%			
0.6-1.0	2	39%	61%		
1.0-1.4	3	21%	29%	50%	
1.4+	4	9%	18%	27%	46%

Nuclear Thermal

Nuclear Thermal Propulsion systems typically consist of a core NTP engine system and associated hydrogen tanks. But unlike the cryogenic propulsion system, the NTP core engines are taken round-trip and only the hydrogen tanks are "dropped" to increase the overall system efficiency through staging. The mass of the Nuclear Thermal Propulsion system was determined in a similar fashion as the chemical propulsion stage, where stage fractions are actually tank fractions. Thus, the total NTP mission mass can be computed by summing the payload mass ($M_{payload}$), NTP core mass (M_{core}), and all required wet hydrogen drop tanks (M_{tank}).

$$M_{Total} = M_{Payload} + M_{core} + \sum_{n=1}^{N} M_{tank_n} \qquad (7)$$

[6] For our analysis purposes we defined stage fraction to be the ratio of the inert stage mass divided by the total wet mass of the stage (inert plus propellant). This stage fraction was then used to size "rubber" stages for each propulsive maneuver.

where N is the number of tanks required for the mission.

Solar Electric Propulsion

The strategy for scaling SEP systems is more complex than the high-thrust ballistic counterparts discussed previously. The trajectory data provides a round-trip net mass delivery capability based on the trajectory and spacecraft constraint conditions such as launch date, departure C3, trip times, vehicle specific impulse, etc. Direct scaling can be accomplished based on the fact that the trajectory describes the necessary acceleration history of the integrated vehicle. From the initial and final mass ratios along with vehicle characteristics such as specific impulse and spacecraft specific mass, the required jet power ($P_{Jet_{Helio}}$) necessary to match the required acceleration level for the heliocentric portion of the mission can be computed from:

$$P_{Jet_{Helio}} = \frac{M_{payload}\left[e^{\frac{\Delta V}{g * I_{sp}}} - 1\right]}{\frac{M_i - M_f}{P_{jet_i}} - \left[e^{\frac{\Delta V}{g * I_{sp}}} - 1\right]\left[f_{tank}\left(\frac{M_i - M_f}{P_{jet_i}}\right) + \alpha_{sep}\right]} \quad (8)$$

where,

$M_{payload}$	= Round trip payload mass
ΔV	= heliocentric delta-v
g	= Earth gravitational constant
I_{sp}	= Electric propulsion specific impulse
M_i	= Trajectory initial mass
M_f	= Trajectory final mass
P_{jet_i}	= Trajectory initial jet power
f_{tank}	= SEP propellant tank fraction
α_{sep}	= SEP total spacecraft specific mass

Once the required jet power necessary for the heliocentric portion of the trajectory is known, the total spacecraft power mass (M_{power}), propellant mass ($M_{propellant}$), and tank mass (M_{tank}) can be computed from the specific mass (α_{sep}).

$$M_{power} = P_{Jet_{Helio}} * \alpha_{sep} \quad (9)$$

$$M_{propellant} = \frac{M_i - M_f}{P_{jet_i}} * P_{Jet_{Helio}} \quad (10)$$

$$M_{tank} = M_{propellant} * f_{tank} \quad (11)$$

Thus, the total SEP departing mass (M_{SEP}) is:

$$M_{SEP} = M_{payload} + M_{power} + M_{propellant} + M_{tank} \quad (12)$$

Space Launch System

The strategy for the Space Launch System, NASA's future heavy lift launch vehicle program, continues to evolve. During the period of time in which the NEA accessibility assessments were conducted it was unclear what the actual lift performance of the SLS would be. Thus the strategy utilized for the system modeling was to assume a generic launch capacity of 100 t net vehicle performance. From this net performance appropriate performance margins, payload adapters, and packing inefficiencies must be accounted for. Packing inefficiencies occur due to the fact that the size of the integrated vehicles does not always utilize the entire payload capability. For instance, volume constraints of the launch vehicle payload shroud can limit the amount of payload delivered. For this assessment, 10% was assumed to cover performance margin as well as packing inefficiencies, and 2.5% launch vehicle adapter knockdowns were assumed, thus resulting in a net launch vehicle performance of 87.75 t to Low-Earth Orbit.

5. NEA ACCESSIBILITY CRITERIA

As the Capability Driven Framework is applied to human exploration of Near-Earth Asteroids, some important strategic needs must be considered:

- How short can the trip times be reduced in order to reduce crew exposure to the deep-space radiation and micro-gravity environment?
- Are there options to conduct easy, early missions?
- What is the affect of infusion of advanced propulsion technologies on NEA target availability?
- When do the departure opportunities open up, how frequent and how long are they?
- How many launches are required to conduct a round trip human mission to a NEA?
- And, based on the above, how many Near-Earth Asteroids are available for a given time of interest?

True accessibility can only be determined in the context of specific constraints in which the question of "How many are available" can be answered. Key considerations of NEA accessibility include:

Date of Interest

It is important to understand the temporal frequency of asteroid availability. The year 2025 is being used by many communities as the notional date for planning the first human missions to an Asteroid.[10] The timing of a human mission to a NEA has an important impact on the availability of key technologies as well as required budget. Due to the number of new spacecraft systems required for a human NEA mission, departure dates prior to 2025 will experience significant budgetary pressures as well as potentially eliminate certain technological options. To help understand the sensitivity of the number of available asteroids to time frames of interest, the period between 2025 and 2035 was used as the prime period for searches.

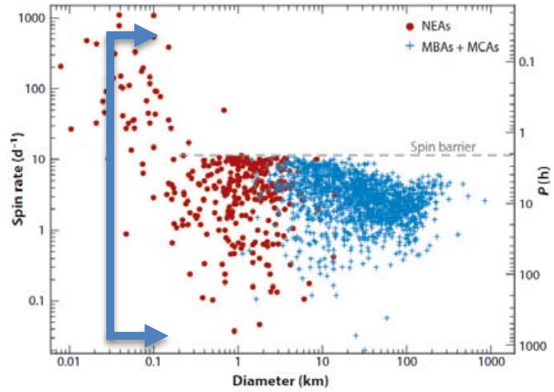
Asteroid Size

Perhaps one of the most important criteria on the suitability, and thus accessibility, of an asteroid is the estimated size. Remote sensing data of both main-belt and near-Earth asteroids show that there is a strong correlation of asteroid size and estimated spin rate as shown in Figure 3. [11]. Small asteroids, on the order of 50-100 meters, have a tendency to be fast rotating and are more likely to be monolithic with less surface regolith. Large asteroids, 100 meters or larger, tend to rotate more slowly and have a high probability of being rubble piles comprised of a variety of particle sizes. Internal NASA studies have shown that of the seventeen top driving capabilities for human exploration of NEAs, over half require direct interaction of the surface [12]. Surface exploration via direct contact, will require the ability to maintain a fixed relative position with the NEA. A slow NEA spin rate and the ability to anchor to the surface are highly desirable. Attaching to a micro-gravity bound rubble pile may also pose significant operational complexity. Larger NEAs are desired to maximize the diversity of surface terrains and composition for scientific study (although smaller NEAs will still provide significant scientific return). Finding a NEA that is "just right" (for example substantially monolithic and with a slow spin rate) could be enabling for human exploration. Thus, asteroids with an estimated diameter greater than 30 meters were used as the lower threshold for the search criteria.[7]

Total Mission Mass

The number of available NEAs is also highly dependent on the total energy required to reach the NEA and return to Earth and how much mass must be pushed through each maneuver (typically driven by mission duration) as described previously. Since a round-trip NEA mission is operationally very similar to a human mission to Mars orbit and back, an upper mass limit approaching those that approximate the crew portion of Mars mission was used. For this assessment a total mission mass limit was set to no greater than that of the current NASA Mars architecture.[5]. This equates to a launch limit of three, four, and five for the Nuclear Thermal, Solar Electric, and Chemical Propulsion architectures respectively.

[7] It must be noted that there is very little direct data on asteroid size and rotation rate. Most NEA size estimates, and consequently rotation rate, are based on observed visual brightness (absolute magnitude) which can lead to large uncertainties of up to a factor of approximately 5.



Asteroid rotation rate ω_{rot} *(rotations per day, left axis) and rotation period* P_{rot} *(h per rotation, right axis)* plotted versus asteroid diameter *D (km), obtained by brightness (absolute magnitude) and brightness* variation (periodicity) studies of hundreds of asteroids. The "spin barrier" at P_{rot} ** 2.2 h appears to be an* abrupt threshold transgressed by only one known asteroid larger than ~300 m diameter. Near-Earth asteroids (NEAs) are shown as circles; main-belt asteroids (MBAs) and Mars-crossing asteroids (MCAs) are shown as small crosses. Figure courtesy of Petr Pravec. Ref. Asphaug, 2009.

Figure 3 NEA Size/Rotation Rate

Mission Duration

Total mission duration is another key criterion in determining the potential accessibility of Near-Earth Asteroids. Crew exposure to the deep-space environment, including zero-gravity and radiation, particularly exposure to Galactic Cosmic Radiation (GCR), is an important criterion to consider in designing human missions beyond low-Earth orbit. Minimizing crew exposure is generally accomplished by flying faster trajectories, which in turn increases the amount of propulsive capability (delta-v) required. Mission duration is also a significant contributor to the overall reliability of the systems. The longer the mission the higher the probability those systems will fail. System and component failure can be mitigated through additional redundancy or proper sparing and in-flight maintenance techniques. This is especially exaggerated since round-trip human NEA missions do not exhibit the opportunities for just-in-time resupply or logistics missions that are possible for current ISS missions and future mission in cis-lunar space (L1, L2, lunar orbit, and lunar surface). For NEA missions, the crew must carry with them all equipment, supplies, and spare parts necessary for the entire mission duration. But it must also be recognized that as humans extend our reach outward towards Mars, concepts for supporting humans in deep space for extended durations, two to three years, will be necessary.

Orbit Condition Code

Another key aspect of determining the accessibility of a potential NEA is determination of the expected orbit of the target. Most NEAs are "discovered" during relatively short passes near Earth when observational data is collected. This data is then utilized to determine the orbit elements which are fed into the Small Bodies Database managed by JPL. But the orbital parameters are only as good as the data gathered, and there is often significant uncertainty, especially when propagating the estimated orbit into the future. The Orbit Condition Code (OCC) is a measure of how much anticipated uncertainty there is in the asteroid's orbit (longitudinal runoff) after 10 years (without more data). This condition code can be improved with additional observations or via robotic precursor missions prior to the human mission. But assessments have yet to be made on determining the OCC threshold required for human missions. In general, lower OCCs are better. Since it not possible to predict how OCC will improve over time due to additional ground or space-based observations, orbit condition code was not used as a filter criteria. But it is suggested that OCC should be used as a post-assessment criteria in determining "acceptability" of a potential target.

6. ACCESSIBILITY TRENDS AND FINDINGS

Determining the set of "accessible" targets required translating the 79 million trajectories into the four different physical architectures as described previously in order to determine the total mission mass and first order approximation of number of launches. It should be noted that due to the sheer magnitude of number of simulations run, parametric mass sizing was utilized. Although the parametric results have been validated with results from more detailed assessments, the results contained herein should be used for comparative purposes only. That is, understanding the relative trends of one approach versus another is appropriate. Caution should be observed when making definitive statements, and additional detailed assessments are required before such statements can be made. Future assessments should focus on the more promising targets and include mission, operational, and system design including launch packaging and volume.

Figure 4 provides a summary of the sensitivity analysis and accessibility trends for representative high-thrust ballistic and low-thrust electric propulsion architectures. As of February 3, 2011 there were 7,655 total known near-Earth Asteroids in the Small Bodies Database. For mission departure dates in the period 2015-2040, only 2-4% (180-294) could be considered "accessible" with transportation capabilities approaching those needed for human Mars missions. When the date of interest is limited to the 2025-2035 timeframe, the number of available targets drops to less than 3% of the total population (128-233 total NEAs). Many of these targets are small, and when the minimum size is limited to 30 meters, only 1-2% of the total known population remains viable (75-159). Further examination of Figure 1 provides some additional interesting trends.

For the high-thrust architectures, Nuclear Thermal and Chemical propulsion, there is a great payoff at one year duration. Note that there is a rather modest increase in the number of available targets as the mission duration is increased from 365 to 450 days. But there is also a sharp drop off of available targets as mission duration is reduced to 270 days and shorter. This trend is not as remarkable in the low-thrust architecture predominately due to the fact that the low-thrust systems provide continuous thrust profiles, thus smoothing out this sensitivity of trip time.

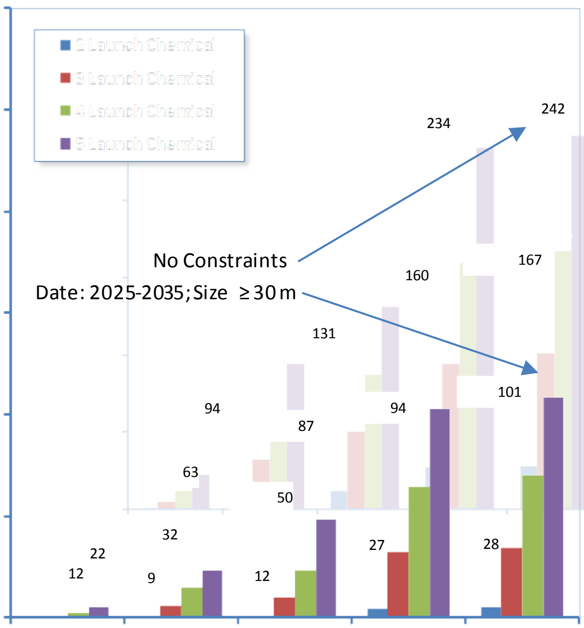
As budgetary constraints are applied, emphasis is placed on trying to find solutions which minimize system and technology development costs, require fewer launches, and are shorter in duration. Figure 4 shows that there is a paucity of targets when less technology and capability is considered available – the budget constrained approach. For instance, there are very few targets for the all chemical propulsion architecture allowing only two heavy lift launches. This is especially exaggerated when trip times are reduced to those within our current experience base, that is, less than 180 days as the current standard rotation period on the International Space Station. The trends depicted in Figure 4 demonstrate that human missions to Near-Earth Asteroids require more performance than what is needed in cis-lunar space in terms of total number of launches and technologies, namely propulsion and long-duration human support.

7. THE SYNODIC PARADOX

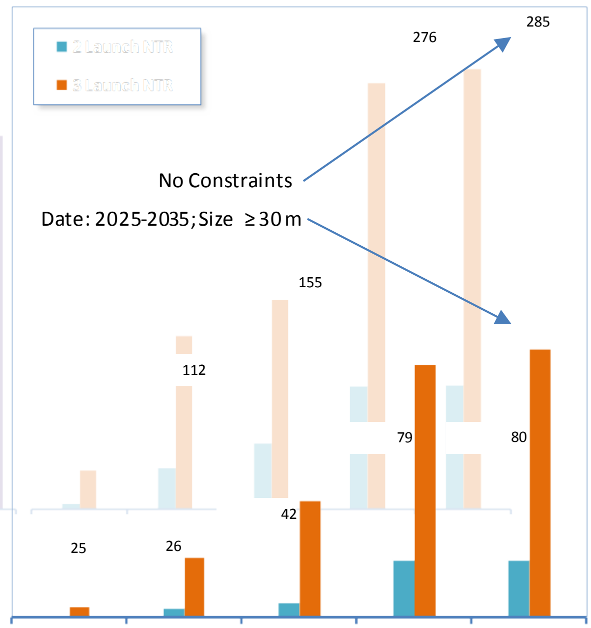
It should be noted that in the previous discussion, a target was assumed to be accessible if it had at least one trajectory solution, but this does not describe how robust the NEA is in terms of mission opportunities. In the early stages of advanced mission design, mission architects often utilize trajectory search algorithms and optimization techniques to find the minimum delta-v for a specified round trip time. This minimum delta-v point often represents the minimum mass point.[8] In actual flight operations, the minimum delta-v point is never flown due to a myriad of considerations and constraints including uncertainties in propulsion system performance, targeting guidance, and operational considerations including providing an adequate departure window to provide operational flexibility for unknown or unplanned situations. Thus, a viable target is one which provides an acceptable departure window of sufficient width or consisting of multiple successive opportunities.

When constructing human missions to Near-Earth Asteroids it is important to establish an approach which can not only achieve the overarching mission objectives, but is also safe, affordable, and sustainable. Architecture complexity and mission duration are key drivers of safety, and affordability is driven predominately by the cost of the transportation systems including launch. Throughout the design cycle, emphasis is placed on reducing total mission mass because mass provides a very good first order approximation for

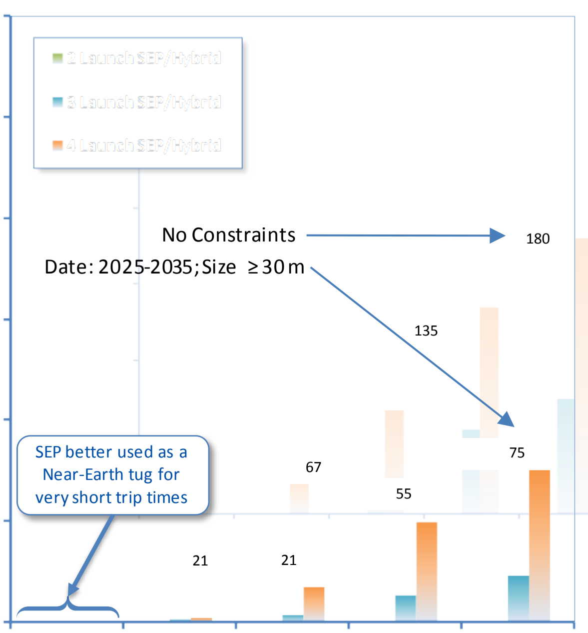
8 Minimum delta-v does not always represent minimum mass, but must be determined through more detailed mission design associated with the specific operational concept.



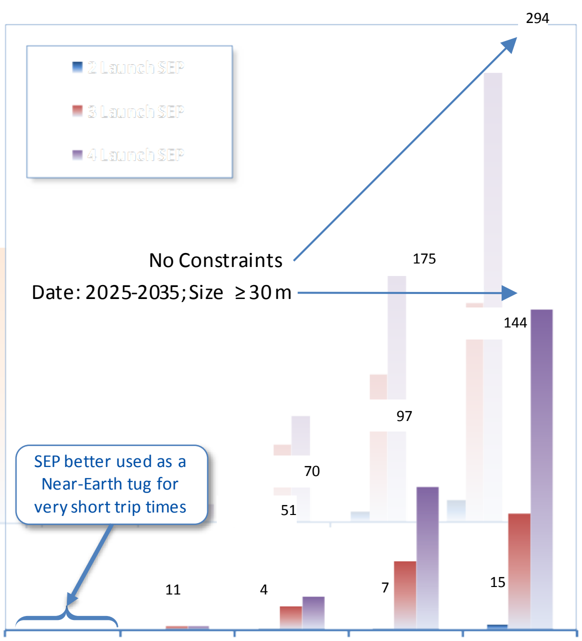
(a) All Chemical Propulsion



(b) All NTR Propulsion



(c) All Solar Electric Propulsion



(d) Solar Electric / Chemical Hybrid Propulsion

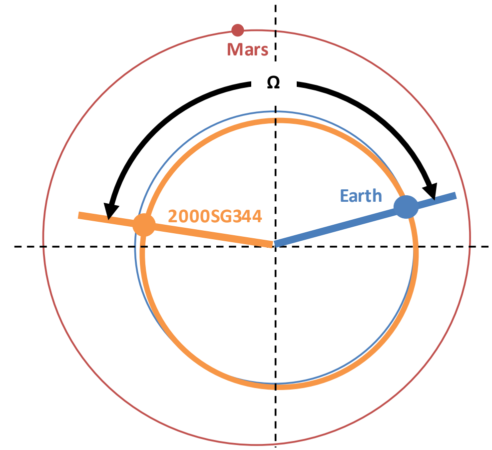
The background (lighter color) data set represent the total number of NEAs accessible with no constraints applied. The darker foreground data represent the total number of targets available within the 2025-2035 time period and NEAs estimated size greater than or equal to 30 m. Accessibility here means that at least one trajectory solution exists, but these data do not represent how robust the trajectories are.

Figure 4 NEA Accessibility Sensitivity Results.

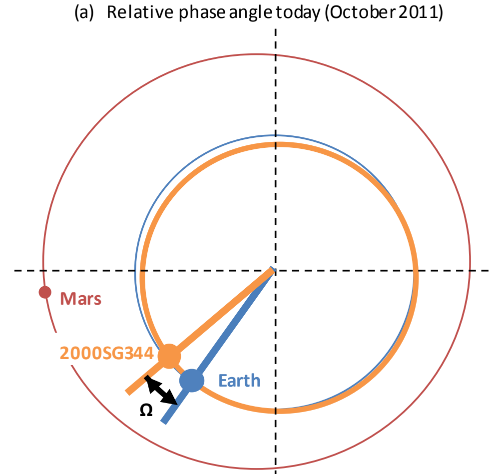
total cost and complexity (safety). Infusing advanced technologies is one way to reduce the overall mission mass, but technology development takes additional lead time and early funding which often competes with the mainstream system development efforts. Mission designers typically seek balanced strategies which can incorporate the proper mission design with technology content consistent with available budget and schedule.

One of the most powerful techniques to reduce mission mass is to design the overall mission by finding ways to reduce mission delta-v by proper selection of the Earth departure date. Optimum Earth departure windows occur when the relative phase angle between the Earth and the NEA target are such that a round-trip minimum energy transfer trajectory exists. A depiction of this geometry for NEA 2000SG344 is shown in Figure 5. Note that as of today, the relative phase angle for this NEA is quite large (Figure 5a). In fact, the specific target, 2000SG344, is nearly on the opposite side of the Sun, and the relative phase angle for this target does not become reasonable until 2027 (Figure 5b). This is due to the fact that orbital period of 2000SG344 is very near that of the Earth. Thus, this optimum relative phase angle repeats over long periods of time (synodic period). The synodic period is the time required for any phase angle to repeat itself. The synodic period has a strong influence on the overall exploration architecture. It influences when and how often Earth departure opportunities occur as well as when additional physical characteristics can be obtained from Earth-based assets. But synodic period alone does not determine accessibility. Take for instance NEA 2003LN6 as depicted in Figure 6. This figure shows that departure opportunities to 2003LN6 occur with its synodic period every four years. Further examination of Figure 6 shows that there is a significant variation in the total delta-v across the departure opportunities. Thus, further assessment of the specific characteristics of each departure window is warranted before it can be determined if the specific target is accessible.

Earth departure window characteristics were generated for the high-thrust architectures (chemical and nuclear thermal) for each trajectory in the scan set.[9] In order to determine the departure window length for each NEA, the total mission mass estimate for the architectures was searched to determine when each window "opens" and "closes" within a given number of launches, as described by the total mission mass, and total mission duration. An example departure window for 2000SG3244 is shown in Figure 7. It must be noted that this example shown is for a rather good target, in fact the best found for the time period of interest. Most NEAs do not exhibit this deep or wide of a window, but this one provides a good example of how complex the behavior



(a) Relative phase angle today (October 2011)



(b) Relative phase angle at opening of departure window (May 2027)

Figure 5 Phase angle at departure.

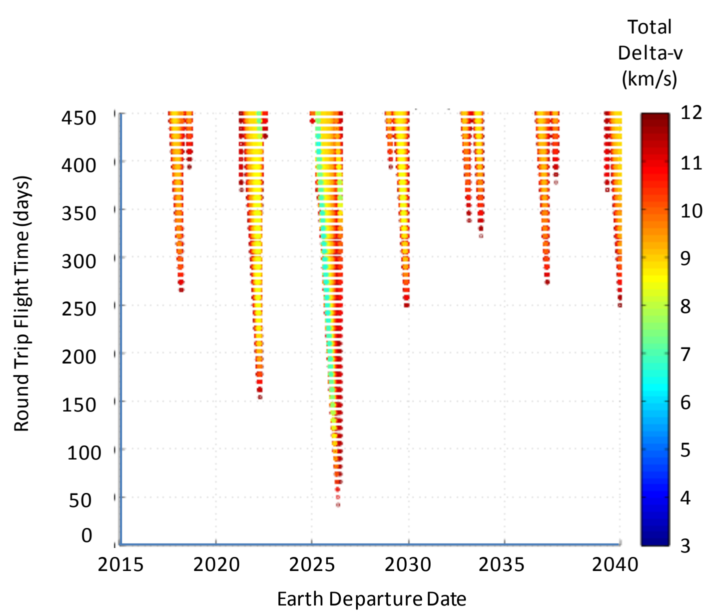


Figure 6 Synodic Period for Asteroid 2003LN6

(Courtesy Barbee, et al. [6])

[9] Departure windows for electric propulsion were not conducted at this early stage due to the complexity of low thrust trajectory analysis and limited study time. In addition, other strategies, such as three-burn departures, have not yet been analyzed.

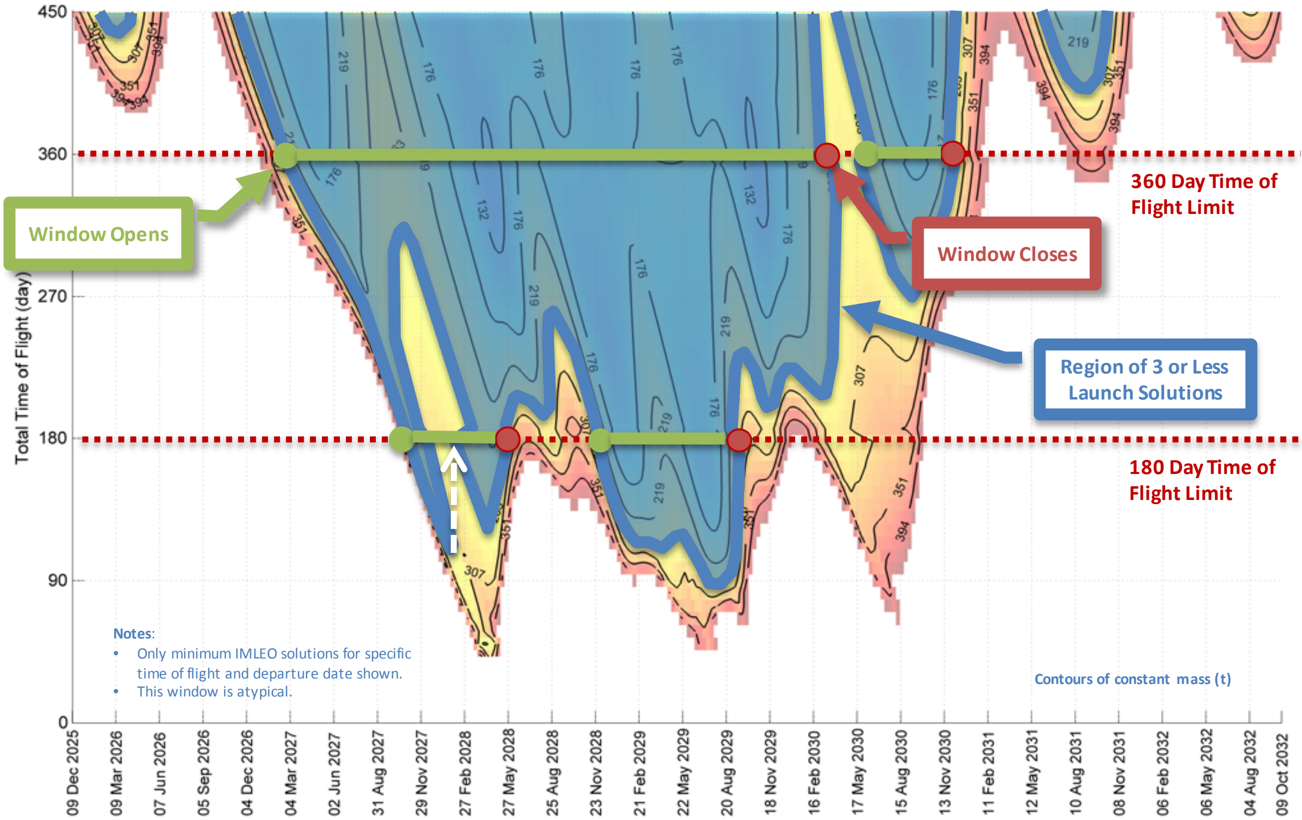
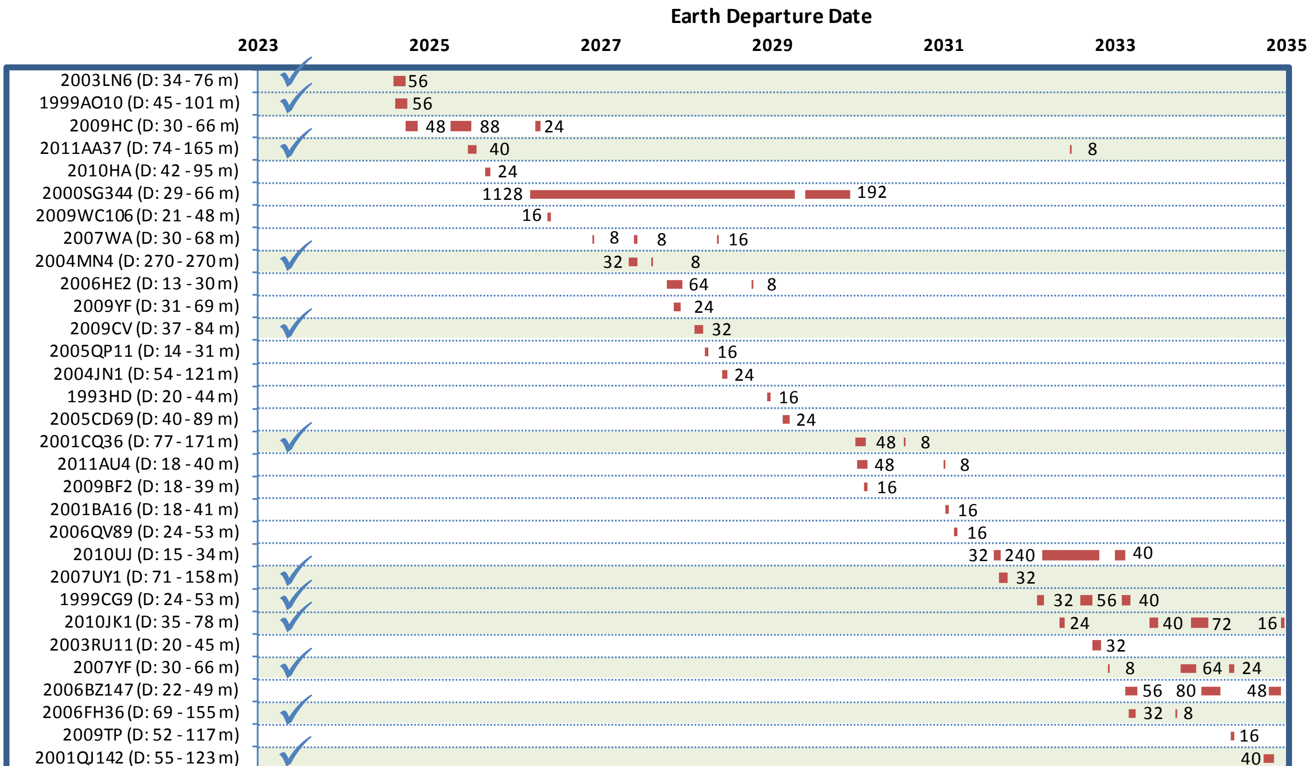


Figure 7 Anatomy of a NEA Earth Departure Window

can be. From this type of analysis and given a set of specific criteria (number of launches and trip time) departure window characteristics such as the number of departure windows and representative width of each window can be determined. These data can then be used to define how robust a NEA is as well as sequencing between available targets can be determined.

Figure 8 provides the departure window results for missions consisting of three heavy lift launches, chemical propulsion, total mission duration of 365 days, departure dates between 2025 and 2035, and a NEA size of at least 30 meters. Of the 31 NEAs which meet these constraints less than half (denoted with green shading) could be considered viable targets when other criteria such as ability to gather additional Earth-based observational data prior to the departure date, probability of high spin rate as represented by small estimated size (diameter less than 30 m), and length of departure window (< 30 days). It should be noted that very little rotation rate data exists for the current NEA population, especially for the small NEAs, and thus most rotation rates are inferred from statistical trends based on size as discussed previously. Figure 8 also shows that for some of the NEA targets there is a departure "season" where multiple departure opportunities can occur over a span of a little over a year. This indicates that as the departure conditions approach optimum alignment, the length of the departure window can be extended with additional transportation performance (via number of launches or advanced propulsion), but only to a point. Departure opportunities are eventually lost even with advanced propulsion technologies. This demonstrates that repeat opportunities to a specific NEA are very limited, and thus a human visit to a particular NEA will be a one time event.

The Synodic period can have a significant strategic impact on human missions to Near-Earth Asteroids. Figure 9 provides a graphical view of the top 127 NEAs for the three launch chemical architecture with no constraint on minimum size. It can be seen from this figure that the better targets, denoted as green with total delta-v between 4.5 and 5.5 km/s, tend to be small and have long synodic periods. Increasing the total available delta-v, and corresponding mission mass, allows missions to be flown to NEAs which are larger, but also those that have shorter synodic periods. NEAs with medium length synodic periods, on the order of tens of years, have a distinct disadvantage of infrequent opportunities to conduct missions and gather Earth-based observational data. NEAs with relatively short synodic periods are desirable because they provide frequent repeat departure and observational opportunities. But NEAs with extremely long synodic periods could provide a very wide and long relative phase angle providing nearly continuous departure and observational opportunities [14]. This



Asteroid Name	Number of Windows	Total Window Days	Size Range (m)	Rotation Period (Hr)	OCC	Last Observed	Next Radar Observation	Next Mission Opportunity	Synodic Period (years)	Notes
2003LN6	1	56	34 - 76	-	5	Jun-03	May-22	Jul-25	4	
1999AO10	1	56	45 - 101	-	6	Feb-99	Dec-18	Aug-25	7	
2009HC	3	160	30 - 66	-	4	Jun-09	Aug-25	Sep-25	18	1
2011AA37	2	48	74 - 165	-	5	Mar-11	Mar-11	Jun-26	8	
2010HA	1	24	42 - 95	-	3	May-10	May-11	Aug-26	16	1,2
2000SG344	2	1320	29 - 66	-	2	Oct-00	May-28	Mar-27	29	1
2009WC106	1	16	21 - 48	-	8	Nov-09	Apr-28	May-27	9	1,2
2007WA	3	32	30 - 68	-	6	Nov-07	Sep-27	Nov-27	20	1,2
2004MN4	2	40	270 - 270	30.40	0	Jan-08	Jan-12	Apr-28	8	
2006HE2	2	72	13 - 30	-	5	Apr-06	Apr-17	Oct-28	11	3
2009YF	1	24	31 - 69	-	7	Jan-10	Jan-20	Nov-28	10	1
2009CV	1	32	37 - 84	-	3	Sep-09	Nov-15	Feb-29	7	
2005QP11	1	16	14 - 31	-	3	Sep-05	May-29	Mar-29	27	1,2,3
2004JN1	1	24	54 - 121	-	6	Jun-04	Oct-11	May-29	9	2
1993HD	1	16	20 - 44	-	9	Apr-93	Oct-11	Dec-29	6	1,2
2005CD69	1	24	40 - 89	-	7	Feb-05	Aug-14	Feb-30	8	1,2
2001CQ36	2	56	77 - 171	-	1	Mar-11	Mar-11	Dec-30	10	
2011AU4	2	56	18 - 40	-	6	Jan-11	Jun-31	Dec-30	10	1,3
2009BF2	1	16	18 - 39	0.0 2	6	Feb-09	Jul-19	Jan-31	11	2,3
2001BA16	1	16	18 - 41	-	5	Feb-01	Feb-22	Jan-32	10	2,3
2006QV89	1	16	24 - 53	-	6	Sep-06	Jul-19	Feb-32	4	2
2010UJ	3	312	15 - 34	-	9	Oct-10	Sep-21	Jul-32	11	1,3
2007UY1	1	32	71 - 158	-	2	Jan-09	Oct-19	Aug-32	13	
1999CG9	3	128	24 - 53	-	6	Mar-99	Apr-11	Jan-33	12	
2010JK1	4	152	35 - 78	-	5	Mar-11	May-11	May-33	26	
2003RU11	1	32	20 - 45	-	3	Sep-03	Aug-34	Sep-33	5	1
2007YF	3	96	30 - 66	-	5	Jan-08	Jan-22	Nov-33	13	
2006BZ147	3	184	22 - 49	-	3	Sep-07	Jan-35	Feb-34	29	1
2006FH36	2	40	69 - 155	-	3	Jun-07	Apr-19	Feb-34	14	
2009TP	1	16	52 - 117	-	6	Oct-09	May-11	May-35	23	1,2
2001QJ142	1	40	55 - 123	-	6	Sep-01	Jan-12	Sep-35	11	

Notes: These data are for 3 launch, all chemical, departure dates 2025-2035, at least 30 m diameter, and 365 day round trip missions only.

(1) Location uncertain and/or limited Earth-based observation opportunity to improve prior to the human mission.

(2) Very limited departure opportunities.

(3) Likely too small based on estimated albedo.

Figure 8 Departure Windows for the Three-Launch Chemical Architecture

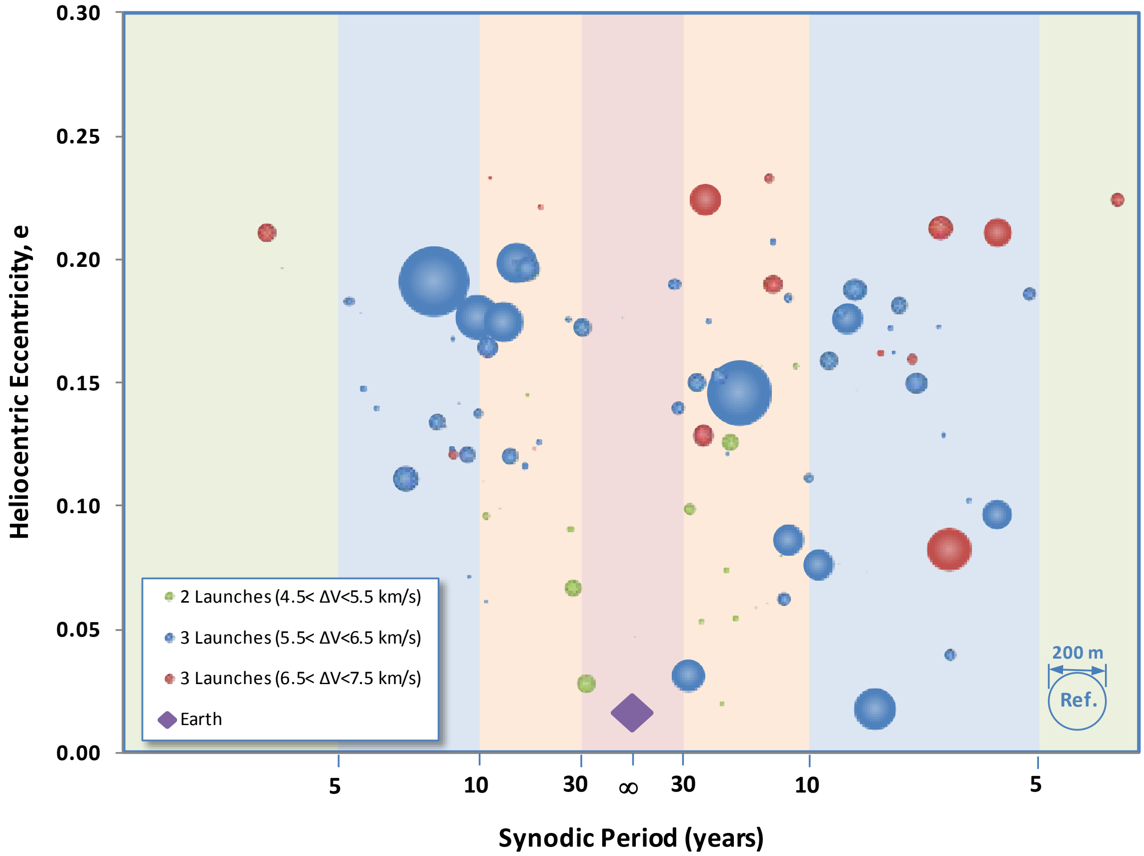


Figure 9 Example NEA Accessibility for the Chemical Architecture.

strategy would only be viable if it were extremely long. This "yet to be discovered" NEA to be strategically useful, observational opportunities must occur well in advance of conducting the actual human mission. Unfortunately for the time period of interest, 2025-2035, this notional NEA does not currently exist, based on what we know today.

Synodic period has a strong influence on the overall exploration architecture. It influences when and how often Earth departure opportunities occur as well as when additional physical characteristics can be obtained from Earth-based assets including both optical and radar observations. This is important because many of the NEAs are poorly understood especially in terms of rotation rate, physical characteristics, and orbital elements. Take for instance a notional mission to the very good target 2000SG344. Although the WISE [13] spacecraft was able to obtain additional orbit data[10] on this target improving the orbit condition code, much still remains unknown. There are no direct observations of the size of 2000SG344, and consequently the size and rotation rate must still be estimated from the visual brightness. Other key data such as spin mode, activity, internal structure, geotechnical surface properties and gravitational field remain unknown. Lastly, because of the synodic period of this target, the next opportunity to gather these data from Earth-based assets will not occur until May 2028, just about the time that the Earth departure window opens. The only way to gather the necessary data would be via robotic mission flown on a non-optimum trajectory to obtain direct measurements years before the human mission.

There is still much debate about what NEA characteristics are required prior to sending humans, and much of that debate centers around what overall level of risk is acceptable. Conducting a mission with limited a priori knowledge, which is critical to developing the systems and operational strategies, could lead to unacceptable risks. An internal NASA team, the NEO User Team (NUT) established both required and recommended critical knowledge needs to enable human exploration of NEAs [12]. Many of the needed NEA characterization data identified by the NUT are best gathered from robotic missions sent to orbit and study the NEA prior to the eventual human mission. But any robotic precursor data gathered must be obtained in advance of the human mission in order for the data to be useful – coupling robotic precursor and human missions. A strategically robust program is one which will provide multiple departure

[10] The WISE spacecraft was able to improve the OCC of 2000SG344 from a 3 to a 2 as contained in the Small Bodies Database.

opportunities to one or multiple targets in order to mitigate unknown and unplanned events including programmatic delays, technical difficulties or funding shortfalls. Thus, for NEA missions, coupling the robotic and human missions necessitates precursors be flown to multiple targets in order to retain overall acceptable programmatic risk.

8. CONCLUSIONS

Debate over what the next destination for human exploration beyond-low Earth orbit continues. Some argue that missions in cis-lunar space (such as the lunar surface) represent missions of the past and missions to NEAs represent a bold new future. Others argue that our next venture beyond LEO should be to the Moon leading to eventual commercialization and settlement. But these arguments are generally made from a policy perspective with little or no physics and engineering behind them. Establishment of a Capability Driven Framework provides an excellent construct to determine the fundamental needs and technologies required to expand human presence beyond low-Earth orbit, but it still lacks a temporal focus – namely which destination comes first?. Exploration destinations such as high-Earth orbit, cis-lunar space, lunar surface, near-Earth asteroids, and eventually to Mars can best be accomplished by proper assessment of the performance, cost, risk and technology needs along the various pathways. As was mentioned earlier, the utility of a Capability Driven Framework can only be measured and fully understood when put into context of actual missions. Table 3 provides a summary of some of the key characteristics of the major destinations considered as part of the Capability Driven Framework. As can be seen from both Figure 8 and Table 3 some general trends appear.

Cis-Lunar Space

Missions in cis-Lunar space (destinations to the left of the green bar in Table 3 including GEO, HEO, Libration Points, lunar orbit and lunar surface) provide frequent mission opportunities and essentially anytime return options. For these missions Earth remains the principal body and the missions remain close to Earth at all times. Subsequently, the mission duration can be easily tailored to meet the exploration objectives from a few days to weeks or more in duration. Anytime return can serve as a key element in risk reduction, especially for early missions when systems are less mature and their performance less demonstrated. These missions also provide the opportunities to easily pre-deploy assets, as well as provide the capability to send mission logistics and spares on a frequent basis. Thus, mission duration, and consequently mission risk and crew exposure to the harsh deep-space environment, can be gradually increased as experience is gained. With the exception of destination cargo, such as deep space habitats or lunar landers, crew missions in cis-lunar space can be conducted in a single heavy lift launch. Recent analyses are showing that sustainability of cis-lunar missions is greater (more economically feasible) when international partnerships and contributions are included in the mission formulation [15]. Lastly, missions in cis-lunar space do not require enabling technologies, although advances are always sought in order to reduce cost, increase mission performance and improve safety.

Mars Missions

Though not discussed in detail here, human missions to Mars are on the far spectrum in terms of challenging missions. Due to orbital mechanics and reasonable projection of transportation capabilities, they will be two – three years in duration demanding advances in almost every technological area.

So Where to NEAs Fit in the Capability Driven Framework?

Human missions to Near-Earth Asteroids represent neither the hardest, nor easiest in the CFD mission spectrum. Just because a mission to a NEA avoids the difficulties of planetary landing [16], these missions will remain very challenging. Human NEA missions truly break the orbit of Earth and are flown in heliocentric space, thus crossing a threshold boundary of capability needs in terms of delta-v, mission duration and resulting capabilities and technologies (the right side of Table 3). These heliocentric missions will have very limited fast return abort capabilities, and once the abort is initiated the return time will be commensurate with the mission elapsed time when the abort is declared measuring weeks to months in duration. Without advances in advanced propulsion technologies there are very few available NEA targets and the mission durations will be long (on the order of a year). Those NEAs which are easier to reach in terms of propulsive capability are generally small, with a high probability of spinning, and have long synodic periods. If a departure season is missed there is a long time before the next mission can be conducted, if it can be flown at all, and thus a backup NEA target must be chosen. Due to the long synodic period of most NEAs, repeat opportunities do not exist, and thus a mission a particular NEA will be a one- time event.

Of those NEAs which are currently considered “accessible” there is relatively very little known about them and they are believed to be small and potentially fast rotating and thus may be difficult if not impossible to explore via direct contact. The current lack of knowledge and uncertainty in NEA characteristics may lead to the need to obtain physical data for the specific targets to be explored by humans. Obtaining additional data in terms of their anticipated location, dynamics (spin/tumble characteristics), their size and composition are keys to determining how to safely explore them with humans. Conducting a mission with limited a priori knowledge, which is critical to developing the systems and operational strategies, could lead to unacceptable risks. Acquiring these data from Earth is not always possible due to the infrequency in their close passage. If required to reduce future human exploration risk, obtaining these in-situ data necessitates a robust robotic precursor program, gathering the needed data on multiple targets years prior to the human mission.

Due to their challenging nature, NEA missions are very similar to Mars orbital missions in terms of delta-v, systems, and technologies. Near-Earth Asteroid missions require further advances in terms of life support reliability, deep-space radiation protection, zero-g countermeasures and advanced propulsion. Even with the inclusion of advanced technologies and capabilities, human NEA missions require multiple, usually three or more, heavy lift launches each. These very characteristics suggest that a human mission to a NEA are more distinctive of a human mission to Mars, and thus would more suitably serve as a final demonstration test of the Mars systems. Human missions to a Near-Earth Asteroids should not be viewed as the first step for humans beyond low-Earth orbit, but rather one potential future step on the path to Mars and beyond.

Table 3 Characteristics of the Capability Driven Framework

	Geostationary Orbit	Earth-Moon Libration	Lunar Surface	Near-Earth Asteroids	Mars Orbit (Phobos)	Mars Surface
Typical Mission Design						
In-Space Delta-v (km/s)	5.9	4.8	~5.6	4.0-9.0+ (1)	~9.0-15.0	5.5-7.3
Descent/Ascent or Vicinity Delta-v (km/s)	-	-	4.2	-	2.4	6.3
Total Mission Duration (days)	10 (2)	16 (2)	16-180 (2)	365 (2)	660	900
Outbound Time (days)	0.5	4	5	170 (2)	250 (4)	180
Time at Destination (days)	9 (2)	8 (2)	7-180 (2)	25 (2)	60 (4)	540
Return Time (days)	0.5	4	4	170 (2)	350 (4)	180
Crew Mission Mode	Zero-g	Zero-g	Zero-g	Zero-g (?)	Artificial-g	Zero-g
Cargo Mode	Split	Split	Split	All-up	Split	Split
Typical Mission Opportunities	Daily	Weekly-Monthly	Weekly-Monthly	10-50+ Years	Every 26 Months	Every 26 Months
Quick Abort to Earth Availability	Anytime	Anytime	Nearly Anytime	None	None	None
Typical Systems Required						
Earth Entry Vehicle	✔	✔	✔	✔	✔	✔
Heavy Lift Launch	✔	✔	✔	✔	✔	✔
In-Space Propulsion	✔	✔	✔	✔	✔	✔
Destination Exploration Systems	✔	✔	✔	✔	✔	✔
Deep-Space Habitat		✔		✔	✔	✔
Planetary Lander			✔			✔
Key Technologies						
Cryogenic Propulsion	✔	✔	✔	✔	✔	✔
Radiation Protection			✔	✔	✔	✔
Advanced Propulsion (SEP, NEP, NTR)				✔	✔	✔
Near-Zero Boiloff Cryogenic Fluid Storage				✔	✔	✔
High-speed Earth Entry				✔	✔	✔
Life Support System Enhancements				✔	✔	✔
Zero-g Countermeasures				✔	✔	✔
In-Situ Resource Utilization			✔			✔
Entry, Descent and Landing			✔			✔
Nuclear Surface Power						✔
Typical Mission Parameters						
Number of crew launches	1	1	1	1	1	1
Number of cargo launches (5)	1	1	1+	2+ (3)	5-9+	7-10
Total mass injected from LEO (t) (5)	30 (6)	30 (6)	75	100-150+ (3)	200-400	500

Notes:
(1) Total in-space delta-v depends on the specific target chosen.
(2) Times are typical, but can be shorter or longer depending on the target chosen, time spent at the destination, and total delta-v
(3) NEA mission mass is highly dependent on the specific target chosen
(4) These durations are typical for opposition class (short stay) missions and will vary by mission opportunity and delta-v
(5) Number of launches highly dependent on the launch vehicle, technologies inserted, and choice of in-space propulsion concept
(6) Mass exclusive of destination support systems (habitat, work platform, remote manipulators, etc.)

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Biography

Mr. Drake currently serves as the Deputy Chief Architect for the NASA Human Spaceflight Architecture Team. For the past several years Mr. Drake helped lead the Agency in the design and analysis of human exploration mission approaches beyond low-Earth Orbit including missions to the Moon, Near-Earth Objects, and Mars. Mr. Drake has been involved in various agency strategic planning activities for NASA's exploration efforts for over twenty years including the NASA 90-day study and the White House Synthesis group, Integrated Space Plan, Exploration Systems Architecture Study, and the Review of Human Space Flight Plans Committee (aka Augustine Committee). Previously, Mr. Drake served as Chief of the Advanced Missions & System Design Office for the Constellation Program and Interim Program Manager for the Lunar Prospector mission at NASA Headquarters and has served as the project lead for many design efforts at the Johnson Space Center. Mr. Drake graduated from the University of Texas at Austin with a Bachelor of Science degree in 1984 with a degree in Aerospace Engineering.

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Study Guidance / Overview
Paul Abel, NASA/JSC
Victoria Friedensen, NASA/HQ

Lindley Johnson, NASA/HQ
Dan Mazanek, NASA/LaRC
GSFC NHATS Trajectory
Ron Mink, NASA/GSFC
Dave Folta, NASA/GSFC
Paul Abell, NASA/JSC
Brent Barbee, GSFC/ESTI
Timothy Esposito, GSFC/ESTI
Elfego Piňon III, GSFC/ESTI
Dan Adamo, JSC

Mass Estimation
Rob Falk, NASA/GRC
Kurt Hack, NASA/GRC
Steve Hoffman, JSC/SAIC
D.R. Komar, NASA/LaRC
Dan Mazanek, NASA/LaRC
Steve Voels, JSC/SAIC
Juan Senent, JSC/OSR
Paul Kessler, LaRC/AMA
Shaun Deacon, LaRC/AMA

NEO Users Team
Doug Craig, HQ ESMD
Victoria Friedensen, HQ ESMD
Brent Sherwood, JPL
Chris Culbert, JSC
Mike Hess, JSC
Peggy Guirgis, JSC
Andy Thomas, JSC
Mike Gernhardt, JSC
Frank Bauer, HQ ESMD
Dee Dee Healey, HQ OSMA
Kevin Larman, LARC/AMA
Dave Liskowsky, HQ OCHMO
Dave Reeves, LaRC
Dan Mazanek, LaRC
Rob Landis, JSC
Paul Abell, JSC
Scott Vangen , KSC
Randy McDaniel, JSC
Greg Oliver, JSC
Brian Johnson, JSC
Dave Westheimer, JSC
Julie Williams-Byrd, LARC

JPL NHATS Trajectory
John D. Baker, JPL, Lead
Lance Benner, JPL
Paul Chodas, JPL
Brent Sherwood, JPL
Brian Muirhead, JPL
Don Yeomans, JPL
Damon Landau, JPL
Nathan Strange, JPL
Tim McElrath, JPL
Jon Sims, JPL
Joe Guinn, JPL
Dennis Byrnes, JPL
Jerry Condon, NASA/JSC/EG5
Jacob Williams, JSC/ESCG
Elizabeth Davis, JSC/ESCG
David Lee, NASA/JSC/EG5
Kurt Hack, NASA/GRC
Rob Falck, NASA/GRC
Steve Chesley, JPL
Jon Giorgini, JPL
Alan Chamberlain, JPL

ACRONYMS AND VARIABLES

α_{sep}	Alpha, SEP spacecraft specific mass
AU	Astronomical Unit
C_3	Characteristic Energy
CDF	Capability Driven Framework
CPS	Cryogenic Propulsion Stage
CTV	Crew Transfer Vehicle
Delta-v	Change in velocity
DRA	Design Reference Architecture
DSH	Deep Space Habitat
EDL	Entry Descent and Landing
E-M	Earth-Moon
EP	Electric Propulsion
f_{inert}	Stage inert mass fraction
f_{tank}	SEP propellant tank fraction
g	Earth gravitational constant
GEO	Geostationary Orbits
GCR	Galactic Cosmic Radiation
GSFC	Goddard Space Flight Center
HEFT	Human Exploration Framework Team
HEO	High-Earth Orbit
Hmag	Absolute visual magnitude
HSF	Human Space Flight
IMLEO	Initial Mass in Low-Earth Orbit
I_{sp}	Electric propulsion specific impulse
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
km/s	Kilometers per second
L1	Earth-Moon Libration Point 1
LaRC	Langley Research Center
LEO	Low Earth Orbit
M_{begin}	Mass, beginning
M_i	Mass, initial
M_f	Mass, final
$M_{payload}$	Mass, round-trip payload
M_{power}	Mass, power system
$M_{propellant}$	Mass, propellant
M_{SEP}	Mass, SEP system
M_{Stage}	Mass, stage
M_{tank}	Mass, tank
M_{Total}	Mass, total
MPCV	Multi-Purpose Crew Vehicle
NEA	Near Earth Asteroid
NTP	Nuclear Thermal Propulsion
NUT	NEA User Team
OCC	Orbit Condition Code

PCC	Pork Chop Contour
$P_{Jet_{Helio}}$	Power, heliocentric jet power
P_{jet_i}	Power, initial jet power
SEP	Solar Electric Propulsion
SEV	Space Exploration Vehicle
SLS	Space Launch System (Heavy Lift)
SM	Service Module
T	Tons, metric
V_{exit}	Velocity, exit
ΔV	Velocity, change (Delta-v)