Returning an Entire Near-Earth Asteroid in Support of Human Exploration Beyond Low-Earth Orbit

John R. Brophy
Jet Propulsion Laboratory
California Institute of Technology
John.R.Brophy@jpl.nasa.gov

Louis Friedman
The Planetary Society
Pasadena, CA 90245
tps.lf@planetary.org

Carlton Allen, David Baughman, Julie Bellerose, Bruce Betts, Mike Brown, Michael Busch, John Casani, Marcello Coradini, Fred Culick, John Dankanich, Paul Dimotakis, Martin Elvis, Ian Garrick-Bethel, Bob Gershman, Tom Jones, Damon Landau, Chris Lewicki, John Lewis, Mark Lupisella, Pedro Llanos, Dan Mazanek, Prakhar Mehrotra, Joe Nuth, Kevin Parkin, Nathan Strange, Guru Singh, Marco Tantardini, Rusty Schweickart, Brian Wilcox, Colin Williams, Willie Williams, and Don Yeomans

This paper describes the results of a study into the feasibility of identifying, robotically capturing, and returning an entire Near-Earth Asteroid (NEA) to the vicinity of the Earth by the middle of the next decade. The feasibility of such an asteroid retrieval mission hinges on finding an overlap between the smallest NEAs that could be reasonably discovered and characterized and the largest NEAs that could be captured and transported in a reasonable flight time. This overlap appears to be centered on NEAs roughly 7 m in diameter corresponding to masses in the range of 250,000 kg to 1,000,000 kg. The study concluded that it would be possible to return a ~500,000-kg NEA to high lunar orbit by around 2025. The feasibility is enabled by three key developments: the ability to discover and characterize an adequate number of sufficiently small near-Earth asteroids for capture and return; the ability to implement sufficiently powerful solar electric propulsion systems to enable transportation of the captured NEA; and the proposed human presence in cislunar space in the 2020s enabling exploration and exploitation of the returned NEA. Placing a 500-t asteroid in high lunar orbit would provide a unique, meaningful, and affordable destination for astronaut crews in the next decade. This disruptive capability would have a positive impact on a wide range of the nation's human space exploration interests. It would provide a high-value target in cislunar space that would require a human presence to take full advantage of this new resource. It would offer an affordable path to providing operational experience with astronauts working around and with a NEA that could feed forward to much longer duration human missions to larger NEAs in deep space. It represents a new synergy between robotic and human missions in which robotic spacecraft would retrieve significant quantities of valuable resources for exploitation by astronaut crews to enable human exploration farther out into the solar system. The capture, transportation, examination, and dissection of an entire NEA would provide valuable information for planetary defense activities that may someday have to deflect a much larger near-Earth object. Transportation of the NEA to lunar orbit with a total flight time of 6 to 10 years would be enabled by a ~40-kW solar electric propulsion system with a specific impulse of 3,000 s. The flight system could be launched to low-Earth orbit (LEO) on a single Atlas V-class launch vehicle, and return to lunar orbit a NEA with at least 28 times the mass launched to LEO. Longer flight times, higher power SEP systems, or a target asteroid in a particularly favorable orbit could increase the mass amplification factor from 28-to-1 to 70-to-1 or greater. The NASA GRC COMPASS team estimated the full life-cycle cost of an asteroid capture and return mission at ~$2.6B.

Carlton Allen, Carlton.C.Allen@nasa.gov
Baughman, davidcbaughman@me.com
Julie Bellerose, julie.bellerose@nasa.gov
Bruce Betts, bruce.betts@planetary.org
Mike Brown, mbrown@caltech.edu
Michael Busch, mbusch@ess.ucla.edu
John Casani, John.R.Casani@jpl.nasa.gov
Marcello Coradini, Marcello.Coradini@jpl.nasa.gov
Fred Culick, fecfly@caltech.edu
John Dankanich, John.Dankanich@nasa.gov
Paul Dimotakis, Paul.E.Dimotakis@jpl.nasa.gov
Martin Elvis, elvis@cfa.harvard.edu
Ian Garrick-Bethel, igarrick@ucsc.edu
Bob Gershman, Robert.Gershman@jpl.nasa.gov
Tom Jones, skywalking@comcast.net
Damon Landau, Damon.Landau@jpl.nasa.gov
Chris Lewicki, chris@arkyd.com
John Lewis, jsl@U.Arizona.edu
Mark Lupisella, mark.l.lupisella@nasa.gov
Pedro Llanos, llanos@usc.edu
Dan Mazanek, Daniel.D.Mazanek@nasa.gov
Prakhar Mehrotra, prakhar@caltech.edu
Joe Nuth, joseph.a.nuth@nasa.gov
Kevin Parkin, Kevin.L.Parkin@nasa.gov
Nathan Strange, Nathan.J.Strange@jpl.nasa.gov
Guru Singh, gurkirpal.singh@jpl.nasa.gov
Marco Tantardini, marco.tantardini@gmail.com David
Rusty Schweickart, rs@well.com
Brian Wilcox, Brian.H.Wilcox@jpl.nasa.gov
Colin Williams, Colin.P.Williams@jpl.nasa.gov
Willie Williams, Willie.B.Williams@nasa.gov
Don Yeomans, Donald.K.Yeomans@jpl.nasa.gov

I. INTRODUCTION

The idea to exploit the natural resources of asteroids is older than the space program. Konstantin Tsiolkovskii included in *The Exploration of Cosmic Space by Means of Reaction Motors,* published in 1903*,* the "exploitation of asteroids" as one of his fourteen points for the conquest of space [1]. More recently this idea was detailed in John Lewis' book *Mining the Sky* [2], and it has long been a major theme of science fiction stories [3]. The difference today is that the technology necessary to make this a reality is just now becoming available. To test the validity of this assertion, NASA sponsored a small study in 2010 to investigate the feasibility of identifying, robotically capturing, and returning to the International Space Station (ISS), an entire small near-Earth asteroid (NEA) – approximately 2-m diameter with a mass of order 10,000 kg – by 2025 [4]. This NASA study concluded that while challenging there were no fundamental show-stoppers that would make such a mission impossible. It was clear from this study that one of the most challenging aspects of the mission would be the identification and characterization of target NEAs suitable for capture and return.

In 2011 the Keck Institute for Space Studies (KISS) [5] sponsored a more in-depth investigation of the feasibility of returning an entire NEA to the vicinity of the Earth. The KISS study focused on returning an asteroid to a high lunar orbit instead of a low-Earth orbit. This would have several advantages. Chief among these is that it would be easier from a propulsion standpoint to return an asteroid to a high lunar orbit rather than take it down much deeper into the Earth's gravity well. Therefore, larger, heavier asteroids could be retrieved. Since larger asteroids are easier to discover and characterize this helps to mitigate one of the key feasibility issues, i.e., identifying target asteroids for return. The KISS study eventually settled on the evaluation of the feasibility of retrieving a 7-m diameter asteroid with a mass of order 500,000 kg. To put this in perspective, the Apollo program returned 382 kg of moon rocks in six missions. The OSIRIS-REx mission [6] proposes to return at least 60 grams of surface material from a NEA by 2023. The Asteroid Capture and Return (ACR) mission concept, that was the focus of this KISS study, investigates the potential to return a 500,000-kg asteroid to a high lunar orbit by the year 2025.

The KISS study enlisted the expertise of people from around the nation including representatives from most of the NASA centers (ARC, GRC, GSFC, JPL, JSC, and LaRC), several universities (Caltech, Carnegie Mellon, Harvard, Naval Postgraduate School, UCLA, UCSC, and USC), as well as several private organizations (Arkyd Astronautics, Inc., The Planetary Society, B612 Foundation, and Florida Institute for Human and Machine Cognition).

The study identified that the feasibility of a mission to retrieve an entire near-Earth asteroid rests on the successful resolution of three key issues:

1. How to discover and characterize a sufficient number of candidate asteroids to enable robust mission planning for a launch around 2020?
2. How to capture and de-spin an asteroid with a mass of order 500,000 kg in deep space?
3. How to safely transport the captured 500,000-kg asteroid back to the Earth-Moon system and place it in a high lunar orbit?

Why Now?

Given that the idea to exploit the natural resources of asteroids has been around for over a hundred years, what has changed that warrants serious investigation into the feasibility of capturing and returning entire near-Earth asteroids to the Earth-Moon system? The answer is that the technology necessary to make this possible is just now becoming available. There are three key enabling elements: 1) The ability to discover and characterize a sufficient number of small near-Earth asteroids suitable for return; 2) The ability to develop powerful solar electric propulsion systems necessary for the timely transportation of a captured NEA; and 3) NASA's proposed plans for a human exploration capability in cislunar space in a time frame that is compatible with when an asteroid could be delivered to lunar orbit. Placing a 500-t asteroid there would provide a unique, meaningful, and easy-to-reach destination for exploration by astronaut crews in the next decade.

II. RATIONALE AND BENEFITS

What are the potential benefits to NASA, the nation, and the international community of returning a 500-t asteroid, and why should the public care? The potential benefits can be grouped into five general categories: 1) Synergy with near-term human exploration; 2) Expansion of international cooperation in space; 3) Synergy with planetary defense; 4) Exploitation of asteroid resources to the benefit of human exploration beyond the Earth-moon system; and 5) Public engagement.

Synergy with Near-Term Human Exploration

An Asteroid Capture-and-Return mission (ACR) concept fits well within the current human spaceflight goals of NASA and its international partners. It would support human deep-space exploration in the following six ways.

First, the ACR mission could partially fulfill the role of a robotic precursor, yet provide far more information about asteroid structure, composition, and mechanical properties through the extensive field investigations it would enable. If conducted promptly if could feed experience and hardware forward into plans for a series of human NEA expeditions in deep space.

Second, by making available hundreds of tons of asteroidal material within the Earth-Moon system, the ACR mission concept would enable astronaut visits that would take only a few weeks, not the half a year or more required for even the most accessible NEA targets. Compared to a deep-space NEA mission, a "local" visit to the captured ACR object would enable the crew to spend a much higher fraction of their mission time actually working at the object. Such a "local asteroid" mission would clearly be a bridge between LEO operations and full-fledged deep-space NEA expeditions. The shorter duration would also reduce significantly the radiation hazard facing the crew.

Third, the ACR mission concept would put bulk asteroidal material within reach of Earth-Moon L2 (EM L2) facilities and transport systems. Visits from an L2 outpost to this small captured asteroid would be an attractive sortie option for astronaut crews, providing opportunities for sample return, in-depth scientific examination, and demonstration of resource processing methods.

Fourth, providing hundreds of tons of asteroidal material in cislunar space would open the door to large-scale use of extraterrestrial resources by NASA and its commercial partners. Extraction of propellants, bulk radiation shielding, and life support fluids from this first captured asteroid could jump-start an entire space-based industry. Our space capabilities would finally have caught up with the speculative attractions of using space resources *in situ*. One of the simplest but highly leveraged benefits from these resources might be the provision of bulk shielding material for future deep-space expeditions—a simple but effective countermeasure to galactic cosmic ray exposure.

Fifth, the public would clearly see the results from human exploration once astronauts begin the challenging task of examining and "dissecting" a ~500-ton asteroid. This ongoing robotic and astronaut operation would provide a steady stream of "real-time exploration" results to a public attracted to the scientific unknowns and the economic potential of this captured asteroid.

Sixth, the development of a high-power, 40-kW class, solar electric propulsion system would provide a high-performance transportation capability that would benefit other human missions in deep space through cargo delivery and hardware pre-deployment. It would also provide a stepping stone to even higher power SEP vehicles that could be used directly for crew transportation to NEAs and beyond.

Taken together, these attributes of an ACR mission would endow NASA and its partners with a new demonstrated capability in deep space that hasn't been seen since Apollo. Once astronaut visits to the captured object begin, NASA would be putting human explorers in contact with an ancient, scientifically intriguing, and economically valuable body beyond the Moon.

Expansion of International Cooperation in Space

The retrieval of a several-hundred-ton carbonaceous asteroid would present significant opportunities for international cooperation. The retrieval could be carried out under the same philosophy as the Apollo program, "in peace for all mankind," but with a significant advantage. An international panel could be formed to oversee both curation of the body and the review of proposals for its study. The demand for samples for engineering and scientific study of the carbonaceous chondrite material by academic, governmental, and industrial laboratories – usually severely hampered by lack of pristine material – could be met generously. Microgravity processing experiments could be carried out *in situ* in its parking orbit or at the International Space Station (ISS). Selected spacefaring nations would have access to the body under the oversight of the international curatorial panel. Nations without the ability to fly missions to the body would be encouraged to form teaming arrangements and propose jointly with those who can.

Experience gained via human expeditions to the small returned NEA would transfer directly to follow-on international expeditions beyond the Earth-Moon system: to other near-Earth asteroids, Phobos and Deimos, Mars and potentially someday to the main asteroid belt.

Synergy with Planetary Defense

The proposed ACR mission concept would lend itself also to the developing international framework for planetary defense from a NEO impact. Space agencies meeting under the auspices of the United Nations Committee on the Peaceful Uses of Outer Space are discussing the planning and operations required for an international mission demonstrating the techniques that would be required to deflect a hazardous asteroid. [7,8] In addition, the NASA Advisory Council's ad hoc Task Force on Planetary Defense recommended in 2010 that NASA pursue leadership of an international deflection mission as its long-term planetary defense objective [9]. Because the proposed ACR mission would, by definition, be a safe "deflection" of a non-hazardous asteroid, the mission concept would fit very well into this multinational effort.

Exploitation of Asteroid Resources

The capabilities demonstrated by the capture, return, and experimental processing of the first NEA would pave the way for use of asteroidal materials in human deep-space expeditions, greatly reducing required up-mass from Earth, and thus the cost, of such missions. A 500-t, carbonaceous C-type asteroid may contain up to 200 t of volatiles (~100 t water and ~100 t carbon-rich compounds), 90 t of metals (approximately 83 t of iron, 6 t of nickel, and 1 t of cobalt), and 200 t of silicate residue (similar to the average lunar surface material). As discussed below, the ACR mission concept baselines a single Atlas V 551-class launch, with an initial mass to low-Earth orbit (IMLEO) of 18,000 kg. The delivery of a 500-t asteroid to lunar orbit, therefore, represents a mass amplification factor of about 28-to-1. That is, whatever mass is launched to LEO, 28 times that mass would be delivered to high lunar orbit. Longer flight times, higher power SEP systems, or a target object in a particularly favorable orbit could increase the mass amplification factor from 28-to-1 to 70-to-1 or greater.

Galactic Cosmic Rays: Exposure to Galactic Cosmic Rays (GCRs) may represent a show-stopper for human exploration in deep space [10]. The only known solution is to provide sufficient radiation shielding mass. One of the potentially earliest uses of the returned asteroid material would be for radiation shielding against GCRs. Astronauts could cannibalize the asteroid for material to upgrade their deep space habitat with radiation shielding.

Materials Extraction: Aside from radiation shielding, initial processing work would concentrate on the extraction and purification of water. Human expeditions to the NEA placed in lunar orbit could mine and return material to the ISS where initial processing work could be conducted in a micro-gravity environment. This would take advantage of the significant infrastructure represented by the ISS. The next level of processing should be the electrolysis of water into hydrogen and oxygen and the liquefaction of both gases. Other procedures could extract nitrogen, iron, nickel and small quantities of platinum-group metals.

Prototype-scale experiments on processing the materials in the retrieved asteroid would validate concepts and refine techniques for production of radiation shielding, propellants, life-support materials, and structural metals, in support of large-scale space activities. Once developed, these processing techniques would be scaled up and located at the NEA in lunar orbit.

A rough estimate based on NASA's NLS-II agreement for launch services suggests that it costs about $100K for each kilogram of mass delivered to a high lunar orbit using conventional chemical propulsion. Therefore, delivery of 500 t of material to a high lunar orbit would cost of order $50B. As shown in below, the cost of the first ACR mission including Design, Development, Test and Engineering (DDT&E) plus the first unit, launch services, mission operations, government insight/oversight, and reserves is estimated at $2.6B. The first ACR mission would deliver asteroid material to high lunar orbit at a cost in $/kg that would be nearly a factor of 20 cheaper than launching that mass from the ground. The recurring cost for subsequent missions is estimated at approximately $1B so subsequent missions would improve that cost savings to a factor of 50.

Public Engagement

The excitement of actually capturing and moving and entire celestial object and harnessing its resources for space exploration is clear. A mission like this would engage a whole new generation of space interested persons. Apollo was based on a cold-war rationale and ever since an over-arching geo-political rationale for space ventures has been lacking. Retrieving an asteroid for human exploration would provide a new purpose for global achievement and inspiration.

III. MISSION OVERVIEW & SAFETY CONSIDERATIONS

A basic Asteroid Retrieval mission concept is illustrated in Fig. 1. The spacecraft would be launched on an Atlas 551-class launch vehicle to low-Earth orbit. A 40-kW solar electric propulsion system would then be used to reach the NEA in about 4 years. Once at the NEA, a 90-day operations period would be divided into two phases. During the first phase, the target would be studied thoroughly to understand its size, rotation, and surface topography. In the second phase the spacecraft would capture and de-spin the asteroid. To accomplish this, the spacecraft would match the target rotation, capture it using the capture mechanism described in below, secure it firmly to the spacecraft, and propulsively despin the combination. The electric propulsion system would then be used to depart the asteroid orbit, return to the vicinity of the Moon, and enter a high-lunar orbit. After reaching lunar orbit the spacecraft would stay attached to support human activity, which is anticipated to include the development of NEA proximity operational techniques for human missions, along with the development of processes and systems for the exploitation of NEA resources.

The ACR spacecraft concept would have an estimated dry mass of about 5.5 t, and could store up to 13 t of Xe propellant. The spacecraft would use a spiral trajectory to raise its apogee from LEO to the Moon where a series of Lunar Gravity Assists (LGAs) would be used in concert with SEP thrusting to depart the Earth-Moon system. This initial leg of the trajectory would take about 2 years to reach Earth escape. From escape it would take roughly another 2 years to reach the target asteroid. The return time would range from 2 to 6 years depending on the actual mass of the NEA. The concept system could return asteroids with masses in the range 250,000 kg to 1,300,000 kg.

Safety

Since even small asteroids have relatively large masses (a 7-m diameter asteroid has a mass roughly equal to that of the ISS) the final placement of the asteroid in the vicinity of the Earth must be considered carefully. Although the very low strength of a type C asteroid would minimize the likelihood that entry of such a body might inflict damage on Earth's surface, it would be more prudent to place the retrieved asteroid in an orbit from which, if all else fails, it could only impact the Moon, not Earth. Lunar orbit or possibly regions near the Earth-Moon Lagrange points would, therefore, be preferred for this criterion. The second factor regarding the choice of a "parking place" is that it is important to place the asteroid in a location that is reasonably close to and accessible from Earth (within a few days journey from LEO). A third factor is the desire to park the asteroid in a place at which there is some foreseeable future demand for water and water-derived propellants, so that early production of useful materials

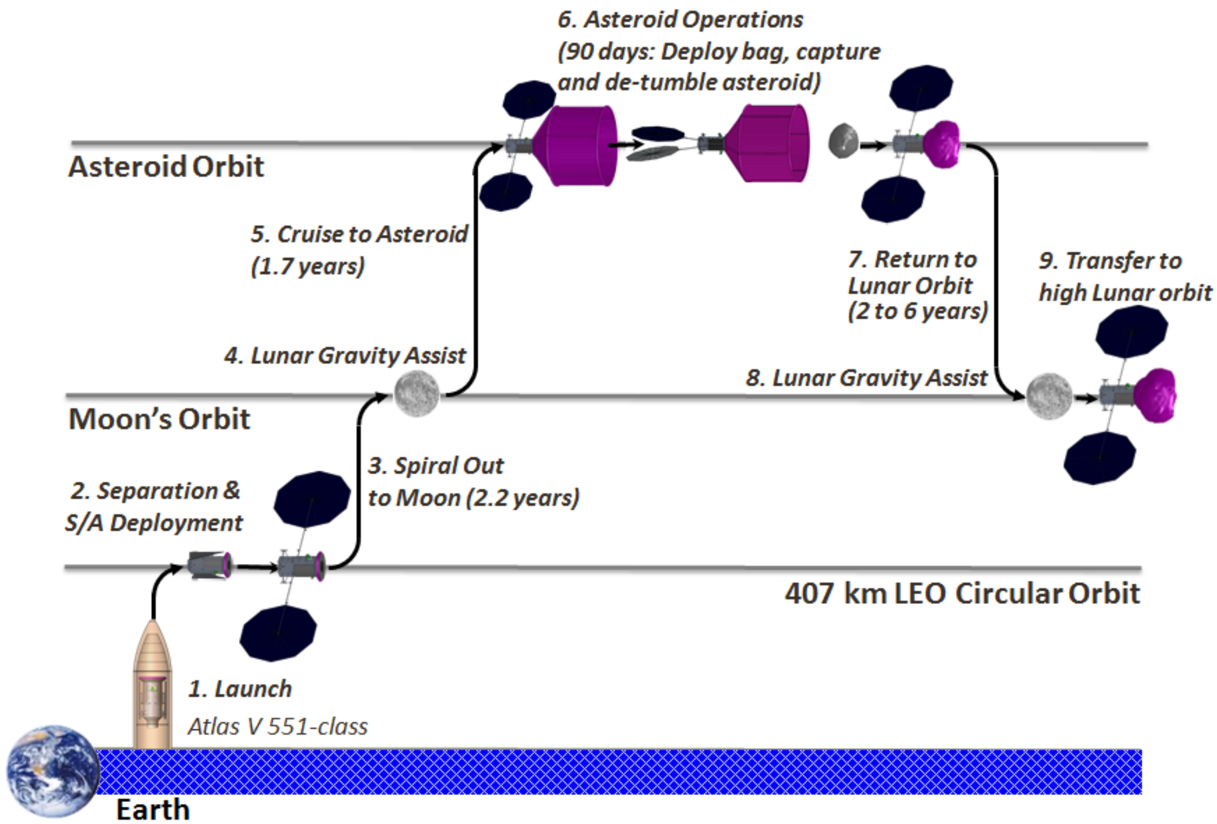

Figure 1. Asteroid return mission concept. Return flight time of 2 to 6 years depending on the asteroid mass.

could serve the needs of future space missions. This third factor suggests LEO and the lunar vicinity as the best choices. These three factors combined suggest the immediate vicinity of the Moon as a reasonable choice. Whatever the final destination the mission must clearly define the end-of-mission conditions and asteroid maintenance and disposal effort. For the purposes of the trajectory design described later, we assumed a high lunar orbit as the destination for the returned asteroid.

A key question that must be answered in the consideration of the feasibility of an ACR mission is, "could the mission be conducted safely?" In fact, moving a non-hazardous asteroid toward the Earth must not just be safe, but it must be perceived as safe to an interested, and likely concerned, public. Safety would have to be guaranteed by the mission design. The KISS study identified the following "belt & suspenders" approach to safety.

First, the size and mass of the asteroid to be returned would be like many other meteorites which routinely impact the Earth and burn up harmlessly in the atmosphere. Moving an asteroid of sufficiently small size would not add to the danger from small meteorites, which are small pieces of asteroids that approach Earth.

Second, we are selecting a carbonaceous asteroid. Asteroids of this type and size are known to be too weak to survive entry through the Earth's atmosphere, so then even if it did approach the Earth it would break up and volatilize in the atmosphere.

Thirdly the trajectory design for moving the asteroid toward the Earth would keep it on a non-impact trajectory at all times. Therefore, if the flight system fails the resulting orbit would be no more dangerous than that of thousands of natural and man-made objects in near-Earth space.

Fourth, the destination orbit would be a high lunar orbit so that even at the end of mission the natural perturbations of the trajectory would only cause an eventual impact on the Moon, not on Earth.

IV. Target Discovery and Characterization

Asteroid Type

The most desirable asteroids for return are the carbonaceous C-type asteroids that are deemed by the astronomy community to have a planetary protection categorization of unrestricted Earth return. Carbonaceous asteroids are the most compositionally diverse asteroids and contain a rich mixture of volatiles, complex organic molecules, dry rock, and metals. They make up about 20% of the known population, but since their albedo is low, they may be heavily biased against detection in optical surveys. Retrieving such asteroid material would enable the development of as many material extraction processes as possible. Carbonaceous asteroid material similar to the CI chondrites is easy to cut or crush because of its low mechanical strength, and can yield as much as 40% by mass of extractable volatiles, roughly equal parts water and carbon-bearing compounds. The residue after volatile extraction is about 30% native metal alloy similar to iron meteorites [11].

Our first priority, then, is to locate several, accessible ~7-m carbonaceous-chondrite objects that could be returned to Earth at some point in the 2020's. This requires a dramatic increase in the discovery rate of small asteroids. Such an increase is possible with relatively minor adjustments to current survey programs.

Synodic Period Constraint – The feasibility of returning an entire small, 7-m asteroid hinges mainly on the question of how to find asteroids of this size that have orbital parameters extremely close to Earth, and yet will return soon enough to be of interest. Small asteroids can only be discovered by ground-based observatories when they make a very close approach to Earth, where their intrinsic faintness is overcome by extreme closeness to the observer. In order to be able to return these objects to the vicinity of the Earth they must have orbital parameters that are very similar to Earth's. Consequently these objects will have synodic periods that are typically one or more decades long. This places a key constraint on small asteroids in order to be candidates for return. They must have synodic periods of approximately one decade. This would enable the object to be discovered and characterized followed by a mission targeted to return the NEA by the next close approach approximately 10 years later. There is an existence proof that such objects exist. The asteroid 2008 HU4 is estimated to be roughly 8-m in diameter and will make its next close approach to Earth in 2016 with a subsequent close approach in 2026. Trajectory analysis presented in Section VI assumes this asteroid and the target and demonstrates how it could be returned to the vicinity of the Earth by 2026 using a 40-kW solar electric propulsion (SEP) system.

Discovery and Characterization Techniques

Discovery and characterization of a sufficient number of candidate NEAs suitable for return is critical. Multiple good targets with launch dates covering multiple years around the nominal launch date would be required to develop a robust mission implementation plan. To support mission planning it would be necessary for each candidate target asteroid that its orbit be adequately known and have the right characteristics, that it be a volatile-rich, C-type asteroid, and that it have the right size, shape, spin state and mass, and that the values of these parameters be known with uncertainties that make the flight system design practical. The current best size frequency distributions for near-Earth asteroids suggest that there are roughly a hundred million NEAs approximately 7-m diameter, but only a few dozen of these are currently known. Fewer still have secure orbits and none of them have known spectral types. It is expected that a low-cost, ground-based observation campaign could identify approximately five good candidates per year that meet these requirements out of roughly 3,500 new discoveries per year. See [5] for details of the required observation campaign.

Size is the key to the discovery and characterization of an adequate number of candidate asteroids before the end of this decade around which a mission could be planned. Larger asteroids are easier to discover and characterize but much harder to move. Since the volume and mass scale as the cube of the diameter, but the projected area scales as the square of the diameter, smaller asteroids get less massive much faster than they get dimmer. The key feasibility issue is to determine if there is an overlap between NEAs that are bright enough (i.e, large enough) to be discovered and characterized and small enough to be moved with near-term SEP propulsion capability.

The densities of asteroids vary widely, from ~1 g/cm^3 for a high-porosity carbonaceous chondrite to ~8 g/cm^3 for solid nickel-iron meteorites. The majority of NEAs have densities between 1.9 g/cm^3 and 3.8 g/cm^3 [12,13]. The mass of an asteroid as a function of its diameter (assuming spherical asteroids) is given in Table 1 over the range of densities from 1.9 g/cm^3 to 3.8 g/cm^3. This table indicates that even very small asteroids can be quite massive from the standpoint of transporting them to the vicinity of the Earth. For example, a 7-m diameter asteroid with a density of 2.8 g/cm^3 has a mass of order 500,000 kg. Small asteroids are not spherical, but Table 1 gives a general sense of the masses of these small objects.

Table 1. Asteroid Mass Scaling (for spherical asteroids)

Diameter (m)	Asteroid Mass (kg)		
	1.9 g/cm^3	2.8 g/cm^3	3.8 g/cm^3
2.0	7,959	11,729	15,917
2.5	15,544	22,907	31,089
3.0	26,861	39,584	53,721
3.5	42,654	62,858	85,307
4.0	63,670	93,829	127,339
4.5	90,655	133,596	181,309
5.0	124,355	183,260	248,709
5.5	165,516	243,918	331,032
6.0	214,885	316,673	429,770
6.5	273,207	402,621	546,415
7.0	341,229	502,864	682,459
7.5	419,697	618,501	839,394
8.0	509,357	750,631	1,018,714
8.5	610,955	900,354	1,221,909
9.0	725,237	1,068,770	1,450,473
9.5	852,949	1,256,977	1,705,898
10.0	994,838	1,466,077	1,989,675

For NEAs with diameters larger than 100 meters, the size-frequency distribution has recently been revised downwards as a result of the WISE space-based infrared observations that were made throughout 2010 and for two months into 2011 [14]. At the small end of the NEA size-frequency distribution, there are roughly 20,500 NEAs larger than 100 meters with about 25% discovered to date, but for the smallest members of the NEA population, there are millions of NEAs larger than 10 meters and billions of NEAs larger than 2 meters.

By far the most efficient NEO search program to date is the Catalina Sky Survey (CSS) near Tucson Arizona [15]. When comparing the efficiencies of NEO search telescopes, the metric of choice, called the "entendue" is the product of the telescope's aperture and its field of view. For the CSS, its entendue is about 2. Next generation NEO search telescopes include the Panoramic Survey Telescope and Rapid Response System 1 (Pan STARRS 1) on Haleakala in Maui Hawaii, which should reach an entendue of about 13 when fully operational [16]. In addition there are plans for PanSTARRS 4, a set of four, co-located PanSTARRS 1 telescopes, which should have an entendue of about 51. The Large Synoptic Survey Telescope (LSST), is a proposed 8.4-meter aperture, wide-field telescope in Chile that has plans for first light in 2018 [17]. The entendue for LSST is about 320 so it could be about 150 times more efficient at finding NEOs than the current CSS system.

Alternative Approach

The discovery of larger objects (≥ 100 m) is, of course, much easier than those less than 10-m in diameter. These objects can be seen at >10X greater range, so much more accurate orbits can be determined with a single pass by Earth. They are visible for enough successive nights that spectroscopic and/or radar observations can be easily arranged. Almost all NEAs whose spectral types are known fall in this category.

Only a few NEAs, all >100-m diameter, have been approached sufficiently closely to get high-resolution images of their surfaces. All such objects appear to have discrete rocks ranging from gravel to house-sized boulders (and larger) on their surfaces. Analyses of spin periods indicate that larger objects have spin periods generally longer than ~2 hours, the "rubble pile limit". Objects with periods slower than this limit have self-gravity at the equator greater than the centrifugal force that would fling loose objects off into space. Objects spinning faster than this are presumed to be competent rock or otherwise coherent and cohesive objects, since the centrifugal force is larger (often much larger) than gravity at the equator. Studies of spin periods show that small objects, with few exceptions, spin faster than the rubble pile limit, while larger objects, again with few exceptions, spin slower than the limit. This suggests that larger objects are rubble piles, with a range of sizes of loose material on their surfaces.

So the alternative approach would be to target a larger NEA, knowing that the entire object would be far too massive to return intact and assume that we could take a 7-m piece off it. We'll refer to this alternative tactic as the *Pick Up a Rock* approach. The approach to capturing and returning an entire small NEA we'll refer to as *Get a Whole One*, when it is necessary to distinguish it from the *Pick Up a Rock* approach. For the *Pick Up a Rock* scenario, in the unlikely event that a single right-sized piece could not be found, then at the very least the system could be designed to collect enough regolith or many small pieces to approach the design-capacity of the system in terms of return mass (i.e., a few hundred metric tons).

V. FLIGHT SYSTEM DESIGN

A conceptual design of the flight system was developed by the COMPASS team at NASA GRC based on guidance provided by the KISS study team. The flight system in the cruise configuration is given in Figs. 2 and 3. The spacecraft configuration is dominated by two large solar array wings that would be used to generate at least 40-kW of power for the electric propulsion system (end-of-life at 1 AU) and the large inflatable structure of the capture mechanism. The solar arrays are sized to accommodate up to 20% degradation due to spiraling through the Earth's radiation belts. A margin of 9% is assumed to be added to the 40-kW power level and 1,200 W is allocated for the rest of the spacecraft. The solar array is assumed to be configured in two wings with each wing having a total area of approximately 90 m^2. There are multiple candidate solar array technologies that would have the potential to meet the needs of this proposed mission. Solar array wings based on the Ultraflex [18] design are shown in Fig. 2.

Each thruster is estimated to have a mass of 19 kg, and would operate at a specific impulse of up to 3,000 s at a PPU input power level of ~10 kW. The xenon propellant tank design is based on a cylindrical, composite overwrap pressure vessel (COPV) design with a seamless aluminum liner. Such tanks are projected to have a tankage fraction for xenon of approximately 4%. (For reference, the Dawn xenon tank had a tankage fraction of 5%.) A total of seven xenon tanks would be needed to store the 12,000 kg of xenon required for this mission. Each tank would have a diameter of 650 mm and would be approximately 3,500 mm long.

Electric Propulsion Subsystem

The EP subsystem concept includes a total of five 10-kW Hall thrusters and Power Processor Units (PPUs). A maximum of 4 thruster/PPU strings would be operated at a time. It also includes xenon propellant tanks, a propellant management assembly, and 2-axis gimbals for each Hall thruster. The electric propulsion subsystem concept incorporates one spare thruster/gimbal/PPU/XFC string to be single fault tolerant. Attitude control during SEP thrusting would be provided by gimbaling the Hall thrusters. This would provide pitch, yaw, and roll control for the spacecraft. When not thrusting with the electric propulsion subsystem, attitude control and spacecraft

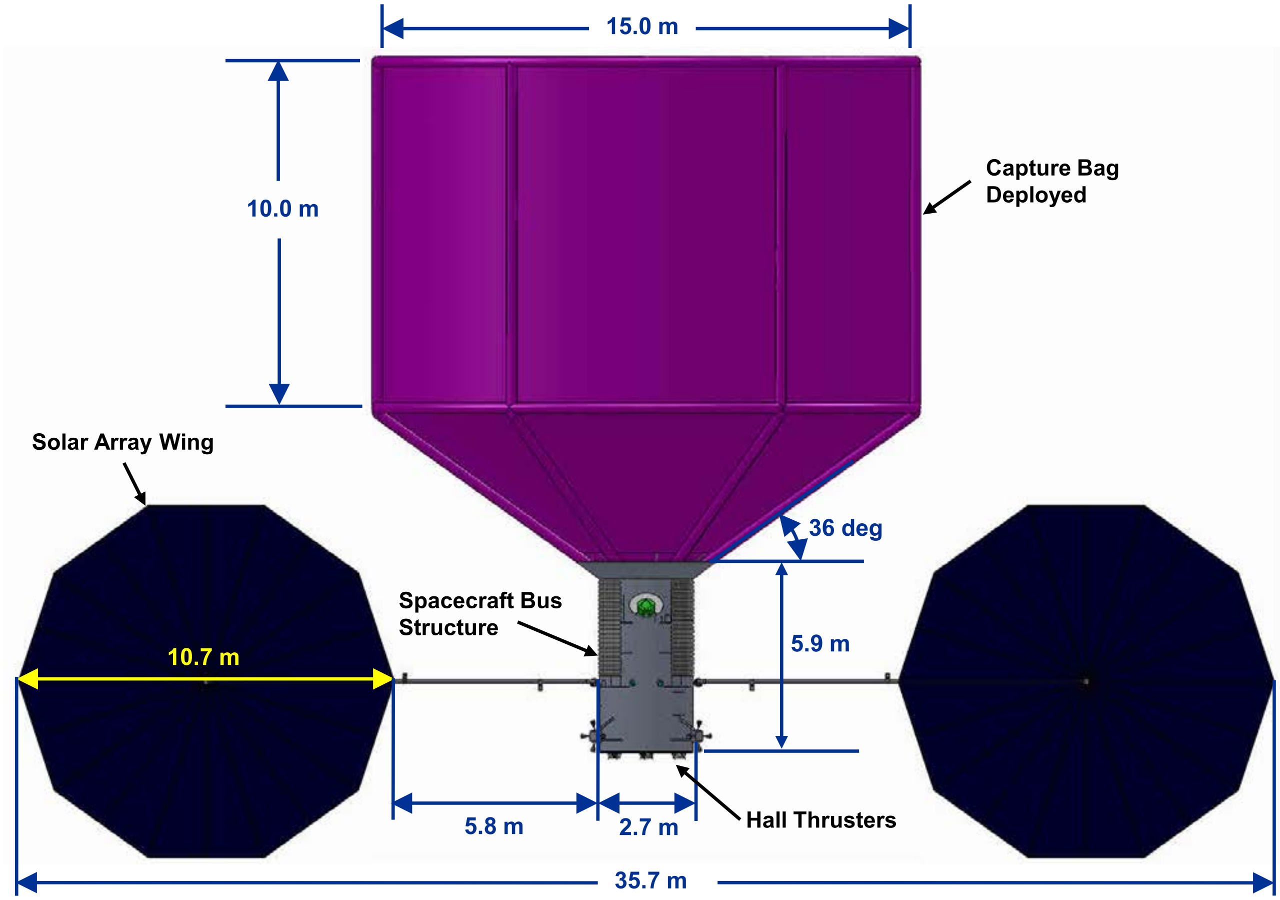

Figure 2. Conceptual spacecraft in the cruise configuration with the capture mechanism deployed.

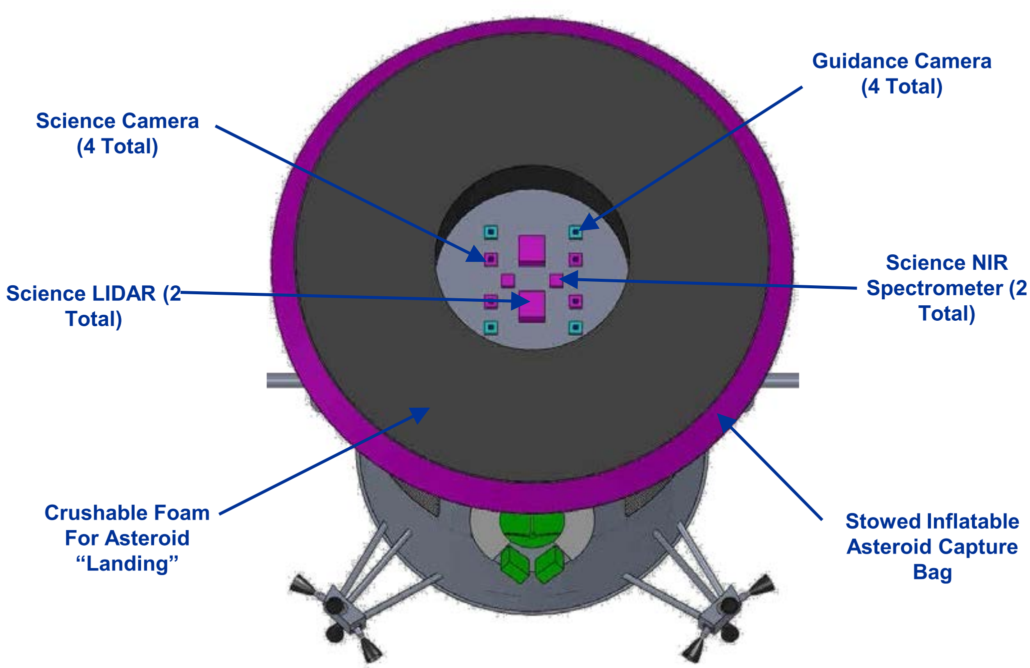

Figure 3. Top view of the conceptual ACR spacecraft showing the instrument suite and capture mechanism prior to being deployed.

translation would be provided by a monopropellant hydrazine reaction control system.

Electrical Power Subsystem (EPS)

The power system design is sized to provide 41.2 kW at 120 VDC to the user input at EOL. It would use two 10.7-m diameter Ultraflex solar arrays with 33% efficient, advanced Inverted Metamorphic (IMM) solar cells and 20-mil coverglass on front and back sides. The solar arrays could be canted toward the aft portion of the vehicle during asteroid capture and would be off-pointed at most 85° and provide at least 3.6 kW.

A secondary lithium ion battery would provide 392 W-hr at up to15% DOD. Up to 1954 W-hr available at 20°C and 80% DOD. The 120 VDC power from solar array would be down-converted to 28 VDC for use by the rest of the spacecraft (non-EP) loads.

Master Equipment List (MEL)

A preliminary MEL for the Asteroid Capture and Return flight system concept is given in Table 2. This MEL indicates a maximum expected wet mass of 15,500 kg, which is 3,300 kg less than the 18,800 kg launch vehicle capability to LEO. The low-thrust trajectory design described in the next section assumed a conservative initial vehicle wet mass of 18,800 kg and flight system dry mass of 5,500 kg. The differences between these values and those in Table 2 represent the mass margins above the maximum expected mass.

Table 2. Asteroid Capture and Return Conceptual Spacecraft MEL.

Flight System Element	Mass (kg)	Mass Growth Allowance (%)	Maximum Expected Mass (kg)
Instruments and Capture Mechanism	339	20.0%	407
Avionics	60.9	23.5%	75
Communications	61.8	24.4%	77
Guidance, Navigation, and Control	20.5	16.5%	24
Electrical Power Subsystem	929	17.3%	1089
Thermal Control Subsystem	316	18.0%	372
Structures and Mechanisms	525	18.0%	620
Electric Propulsion Subsystem	739	12.3%	830
Reaction Control Subsystem (RCS)	167	4.6%	175
Xenon Propellant	10958	0.0%	10958
RCS Propellant	877	0.0%	877
Pressurant	34.3	0.0%	34
Spacecraft Dry Mass	**3158**	**16.2%**	**3670**
Total Spacecraft Wet Mass	**15028**	**---**	**15539**

Capture Mechanism

The capture mechanism would be located at the top (the end opposite from the Hall thrusters) of the spacecraft. This end would also locate the instrumentation for asteroid characterization and capture. The capture mechanisms would include inflatable deployable arms, a high-strength bag assembly, and cinching cables. When inflated and rigidized, four or more arms connected by two or more inflated circumferential hoops would provide the compressive strength to hold open the bag, which would be roughly 10 m long x 15 m in diameter as shown in Fig. 2. This capture mechanism concept could accommodate a wide range of uncertainty in the shape and strength of the asteroid. The deployed bag assembly would be sized to accommodate an asteroid with a 2-to-1 aspect ratio with a roughly cylindrical shape of 6-m diameter x 12-m long.

The exterior finish of the capture bag assembly is designed to passively maintain the surface temperature of the captured asteroid at or below its nominal temperature before capture.

VI. MISSION DESIGN

The overall mission design, illustrated in Fig. 4, is built around the 40-kW solar electric propulsion system described above. The key mission drivers are the ΔV needed for the round trip, the upper limit on the round trip flight time, and the size and mass of the target body. The combination of flight time and upper limit on expected mass of the target determine the SEP system power and propellant quantity that would be needed, which to a first order size the spacecraft and launch vehicle. The size, spin-state, composition, and associated uncertainties of the asteroid's characteristics would also drive the designs for the capture mechanism and de-spin propellant required.

Earth Departure, Rendezvous and Pre-Capture Operations

A proof of concept trajectory analysis was performed using the known small near-Earth asteroid 2008 HU4. The pertinent design parameters are listed in Table 3. The estimated ΔVs for this particular NEA are: LEO to lunar gravity assist = 6.6 km/s; heliocentric transfer to the NEA = 2.8 km/s; NEA return to lunar gravity assist = 170 m/s. Since it is not known what type of asteroid 2008 HU4 is, its mass is highly uncertain. Table 4 summarizes the results assuming the asteroid mass is as low as 250 t and as high as 1,300 t. The trajectory details to return up to 1300 t are presented in Fig. 4. Only the heliocentric portion of the trajectory is described in Table 3 and Fig. 4.

The first five rows of Table 4 indicate that additional flight time would be required to return larger asteroid masses. However, the return date would be fixed to when the NEA naturally has a close encounter to Earth, so the additional flight time would come at the expense of earlier launch dates. Also, larger return mass would typically require additional propellant, which would increase the wet mass of the spacecraft and requires larger launch vehicles. Higher power SEP systems could reduce the flight times.

Direct transfers to Sun-Earth L2, without an intermediate lunar gravity assist, were also examined. The mission-specific parameters for a representative trajectory

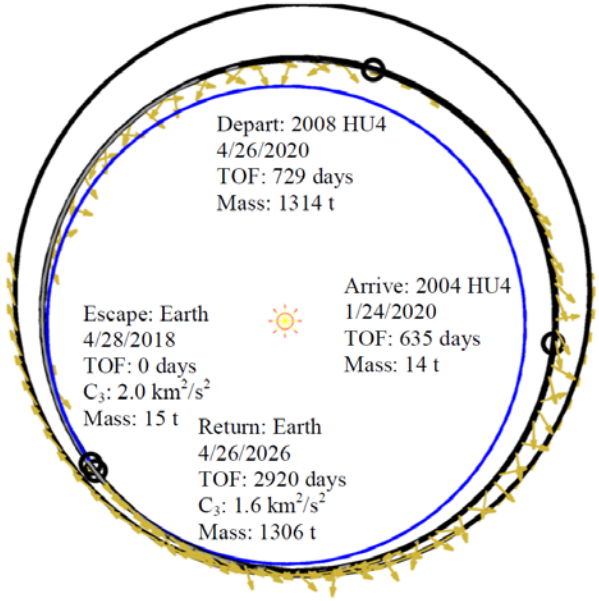

Figure 4. Example mission returning 2008 HU4, a small (~7 m), 1300 t of NEA with a radar opportunity in 2016.

are shown in row six of Table 4.

***Pick Up a Rock* Alternative Mission Approach**

In the Pick Up a Rock approach the plan would be to gather a single ~7-m diameter rock off the surface of a >100-m asteroid or, failing that, collect a similar mass of regolith or smaller rocks. Proof-of-concept trajectories using asteroid 1998 KY26 as the example were performed. 1998 KY26 is known to be a C-type carbonaceous asteroid. The relatively small number of asteroids with known types makes it more difficult to find potential targets with orbital characteristics that would permit large return masses. In this case, 1998 KY26 would require more ΔV to return a sample than was the case for asteroid 2008 HU4. For 1998 KY26 "only" 60 t could be returned. The asteroid 2008 EV5 (not examined here) is another C-type asteroid from which sizable samples could be returned.

***Get a Whole One* Pre-Capture Operations**

Since the targeted NEA is only ~7 m in diameter, the rendezvous would likely need to implement a search prior to encountering the NEA. For example, for 2008 HU4 (without radar astrometry in 2016), the ellipse uncertainty is ~200,000 km x 1,000,000 km. Assuming a navigation camera similar to the Dawn framing camera, the NEA should be visible from a distance of 100,000 km to 200,000 km.

During the 3 months prior to rendezvous, images and delta-difference one-way range (DDOR) measurements would be obtained to constrain the NEA position and obtain preliminary information for further approach and close-up characterization. The spacecraft rendezvous point could be defined at about 20-30 km out, with a residual speed of less than 1-2 m/s.

In the far-approach phase the spacecraft would approach and loiter in the vicinity of the target body by following a ground-provided SEP thrusting profile. The range to the target may be several kilometers at this point. This should permit target-relative position (target → S/C inertial position) estimation using on-board GNC sensors and functions. Once the relative state is known, the on-board station-keeping algorithms would use this data to execute desired target-relative proximity motions.

Table 3. Asteroid retrieval trajectory design parameters based on 2008HU4.

Parameter	Value	Comments
SEP Power (EOL)	40 kW	
Specific Impulse, *Isp*	3000 s	
EP System Efficiency	60%	
Spacecraft Dry Mass	5.5 t	
Launch: Atlas V 551-class		
Launch Mass to LEO	18.8 t	
Spiral Time	2.2 years	LEO to lunar gravity assist
Spiral Xe Used	3.8 t	
Spiral ΔV	6.6 km/s	
Mass at Earth Escape	15.0 t	
Transfer to the NEA		
Earth Escape *C3*	2 km^2/s^2	Lunar gravity assist
Heliocentric ΔV	2.8 km/s	
Flight Time	1.7 years	
Xe Used	1.4 t	
Arrival Mass at NEA	13.6 t	
NEA Stay Time	90 days	
Assumed Asteroid Mass	1300 t	
Transfer to Earth-Moon System		
Departure Mass: S/C + NEA	1313.6 t	
Heliocentric ΔV	0.17 m/s	
Flight Time	6.0 years	
Xe Used	7.7 t	
Mass at lunar gravity assist	1305.9 t	
Escape/Capture *C3*	2 km^2/s^2	Lunar gravity assist
Total Xenon Used	**12.9 t**	
Total Flight Time	**10.2 years**	

Table 4. Interplanetary (Earth escape to Earth capture) trajectories for example missions.

Target Asteroid Designation	Assumed Mass of Asteroid Returned (t)	Launch Vehicle	Xe (not including the Earth spiral) (t)	Earth Escape Date	Flight Time (not including the Earth spiral) (yrs)	Arrival *C3* (km^2/s^2)
2008 HU4	250	Atlas V 521-class	5.0	4/27/2022	4.0	1.8
2008 HU4	400	Atlas V 521-class	5.2	4/27/2021	5.0	1.7
2008 HU4	650	Atlas V 521-class	6.5	4/27/2020	6.0	1.6
2008 HU4	950	Atlas V 551-class	8.9	4/28/2019	7.0	1.6
2008 HU4	1300	Atlas V 551-class	9.1	4/28/2018	8.0	1.6
2008 HU4	200*	Atlas V 551-class	8.7	8/15/2017	8.0	0.0

*Returned to Sun-Earth L2.

A 7-m NEA has very little gravity, less than 10^{-6} m/s^2. Hence, the incremental approach from 20-30 km down to 1 km would be a function of the time needed to analyze images/data. A 1-km standoff distance (if hovering), or close approach distance (if slow hyperbolic flybys are adopted) would be a good distance for sub-meter imaging. Full characterization would be done at distances from 1 km to 100 m, over varying phase angles. Note that orbiting this small NEA is theoretically possible but would most likely be outside of the spacecraft proximity ΔV capabilities (too small ΔV maneuvers needed). Implementing slow hyperbolic flybys would require about 3-4 days per flyby accounting for planning maneuvers and processing tracking data.

Being most likely a fast rotator (from current statistics on < 100-m NEAs, the spin period may be as fast as 10 min), a 1-2 Hz frame rate camera would be needed for resolving the spin state. To account for a possible lack of surface features to navigate with, visible images combined with IR images would be a must-have capability. Gathering full coverage data with the candidate instrument suite given in Table 5 would total about 30-40 Gb at most within a couple of months.

In the middle-approach phase a target-relative trajectory (inertial) would be executed using relative position estimates to bring the S/C to within a few hundred meters of the target, and park it there for an extended period of time. Parking in this context implies loose station-keeping (i.e., back-and-forth coasting inside a control dead-band box defined in inertial space in the vicinity of the target body). It should be possible to use a radar altimeter during this phase.

Assuming radar observation opportunity prior to rendezvous constrain the mass uncertainty to a factor of 2, the spacecraft would need to come within 20 m of the NEA, drifting by it at less than 10 cm/s, for the radio experiment to reduce the mass uncertainty.

In addition to the candidate instrument suite in Table 5 a Gamma Ray Neutron Spectrometer (such as the GRaND instrument on Dawn) could be considered for measuring the surface composition, and a Regolith X-ray Imaging Spectrometer (such as REXIS on OSIRIS-REx) could be considered for X-ray spectroscopy.

Table 5. Candidate Instrument Suite.

Parameters	Vis Cam (OpNav)	Vis Cam (ProxOps)	NIR Spec	LIDAR
Format/ Heritage	High resolution	Ecliptic	Pushbom M3	3D flash STORRM
FOV (deg)	2 x 2	10 x 10	25 x 1	< 200 mrad
IFOV	30 mrad	200 mrad	1 mrad	0.2 mrad
Range	0.4-0.9 mm	0.45-0.9 mm	0.4-3 mm	1mm < 30 km
Resolution	<0.1 m @ 1 km	~ 0.2 m @ 1 km	2 m @ 1 km	< 200m @ 1 km
Mass (kg)	5	2	8	20
Power (W)	15	5	10	50
Telemetry rate	12 Mbits/image	12 Mbits/image	2 Mbits/sample	0.1 Mbits/sample

Capture and Post-Capture Operations

The conceptual mission design allocates 90 days for the spacecraft to characterize the NEA, capture it, and subsequently de-tumble it. These processes, which would be essential for an asteroid return mission, are outlined below.

Capture – This process must capture the NEA, which is considered to be a tumbling, non-cooperative object. The capture process must be executed largely autonomously in deep space. Sometime after the spin state has been identified, the S/C would approach the target body by following a series of closure steps consisting of several descent-stationkeeping-descent cycles. The guidance subsystem would use radar-altimeter aided relative position estimates (inertial) to plan and execute these trajectories.

The final stationkeeping location may be tens of meters from the target center. The S/C would then match the surface velocity and primary spin state of the target while maintaining station at the final station-keeping location. To make the spacecraft nimble enough to do this it may be necessary to provide the capability to fold back the large solar arrays as indicated in Fig. 5. In this configuration, the solar cells would still be facing outward, and the arrays could still generate at least 3.8 kW of power even if they're off-pointed from the sun by up to 85 deg. Final closure motion would be initiated while remaining in the synchronized motion state. Control would be disabled just before capture and re-established following a successful capture and securing of the target body.

The GNC algorithms to rendezvous with a non-cooperative space object exist for objects in Earth orbit. The algorithms, developed for rendezvous and sample capture, were exercised in a DARPA-funded study. That study demonstrated the capture of a defunct, spinning and wobbling, non-cooperative object in Earth orbit. During capture, the asteroid would be positioned inside the capture mechanism and there would only be a small residual relative velocity between the asteroid surface and the capture mechanism.

To capture the asteroid multiple "draw strings" would cinch-close the opening of the bag and also cinch-tight against the bulk material. The tightly-cinched bag containing the asteroid would be drawn up against a ring that constrains its position and attitude so that its center-of-mass is controlled and forces and torques could be applied by the S/C. Cameras positioned on the solar array yokes as indicated in Fig. 6 would be used to determine if the capture mechanism was correctly deployed, and to aid in the asteroid capture. A ring would be positioned between the bag assembly and the body of the S/C for the purpose of imparting forces on the bulk material through the bag. Although not shown in Fig. 6 it may be necessary to include a "Stewart Platform" in which six linear actuators would allow the ring to be moved in x, y, z, roll, pitch, and yaw. This would enable the center-of-mass of the final bagged asteroid to be positioned within an acceptable range of the SEP thruster gimbals so that the resultant thrust vector from all the EP thrusters could nominally be pointed through the center of mass of the whole assembly.

Due to the residual velocity between the asteroid and the spacecraft, there would be some "impact" as the asteroid is captured. Although, since the asteroid would be much more massive than the spacecraft, it is perhaps better to think of this as the asteroid capturing the spacecraft. Nevertheless, once the spacecraft and asteroid are tightly secured together, the spacecraft could then de-tumble the combination.

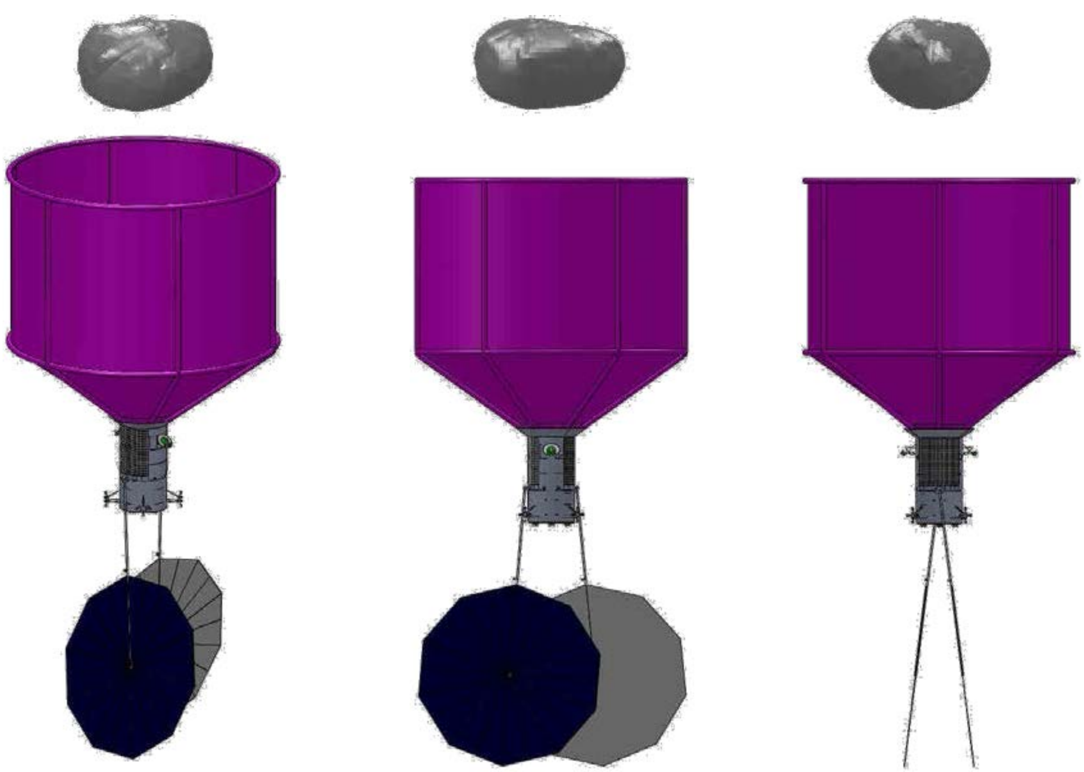

Figure 5. Conceptual spacecraft with solar arrays folded back to facilitate matching the asteroid's spin state during the capture process.

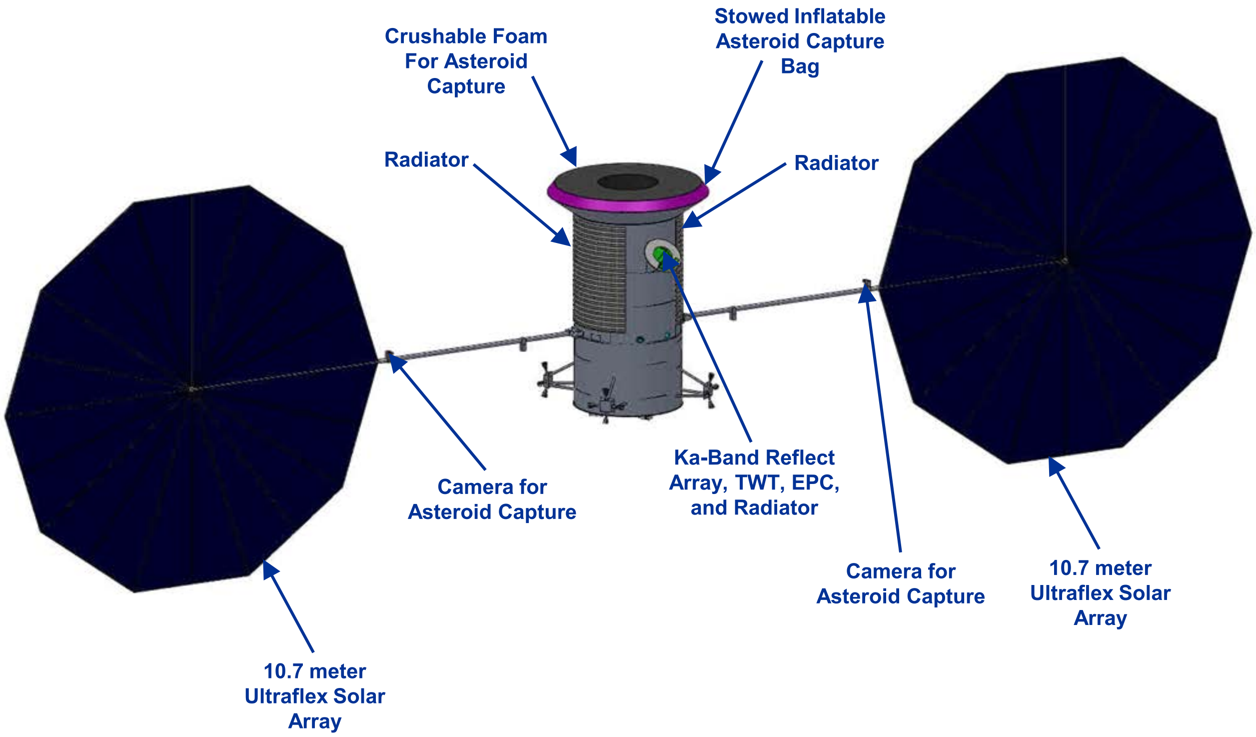

Figure 6. Conceptual flight system configuration before deployment of the capture mechanism showing the locations of the cameras on the solar array yokes used to verify proper deployment and subsequently to aid in the asteroid capture.

De-spin – To estimate the time and propellant required to de-tumble the asteroid, the object was assumed to have a mass of 1,100 t, be rotating at 1 RPM about its major axis, and have a cylindrical shape of 6-m diameter x 12-m long. RCS thrusters with a 200-N thrust capability would be used for this process assuming a moment arm of 2 m. The angular momentum of spacecraft with asteroid would be 1.7×10^6 N·m·s, and the major and minor moments of inertia (MOIs) with the spacecraft attached are estimated to be 1.65×10^7 kg·m^2 and 5.52×10^6 kg·m^2. The resulting time for despin would be ~ 33 minutes assuming continuous firing, and approximately 306 kg of propellant would be required.

Getting to Lunar Orbit

The large mass of the captured asteroid and relatively low thrust available from the electric propulsion subsystem, requires that the spacecraft + asteroid must have the ΔV necessary to target the lunar gravity assist well before the lunar encounter. This requirement, which appears feasible, is not unlike the requirement of the Dawn mission to have a forced coast period well before the Mars Gravity Assist. The asteroid would arrive into the Earth-Moon system on a hyperbolic trajectory with positive *C3*, but after the lunar gravity assist, would have a negative *C3* with respect to the Earth and would be gravitationally captured. The flyby could be targeted such that it would bring the asteroid back into a high lunar orbit, however, such an orbit would not be stable and additional ΔV from the SEP system would be required to remain captured by the Moon.

We estimate that the lunar orbit could be maintained with a station-keeping ΔV on the order of 10 m/s per year. However, the propulsion system would be limited in the rate it could apply the ΔV given the thrust limitations of the electric propulsion subsystem and the mass of the asteroid. The baseline mission concept described above does not currently include the propellant necessary for multi-year station-keeping. A xenon resupply or an additional propulsion module may be necessary for the long-term orbit maintenance of the asteroid. A proof of concept lunar orbit insertion was simulated, and a 25-N thruster was sufficient for insertion into a stable lunar orbit. The 25-N thruster lowered the asteroid *C3* with respect to the moon below -0.1 km^2/s^2.

After lowering the asteroid to a stable lunar orbit, a high-fidelity propagation was performed using Copernicus [19] and all potential perturbations for a demonstration of stability. The asteroid remained captured in lunar orbit after 20 years of simulation without any additional station-keeping as shown in Fig. 7.

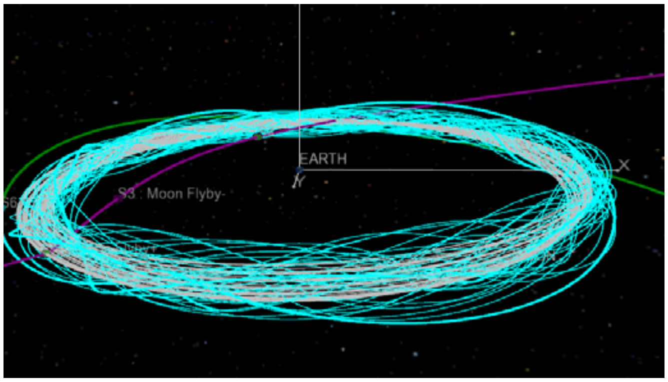

Figure 7. Long duration (20 years) stability simulation for the captured asteroid placed in lunar orbit.

Cislunar Operations

In the context of human exploration, the NEA could be used to gather engineering knowledge and assist in the development of tools and operations. In fact, having the NEA close by would provide a compelling mission objective outside of LEO for an astronaut crew to take it apart. The relative proximity of the NEA would make affordable the use of more complex payloads. Several activities could take place after the NEA is placed in lunar orbit to benefit human exploration, the development of ISRU, and science. The following measurements could be obtained by both robotic spacecraft and crewed missions.

- Remote sensing imaging obtained over various wavelengths and phase angles for composition, morphology, and high resolution mapping.
- Stereo techniques and ranging instrumentation would enable high resolution digital terrain models to be constructed to assist in further surface activity planning.
- Surface and sub-surface element and volatile composition obtained using gamma ray and neutron spectrometer such as the GRaND instrument on the Dawn spacecraft, or using X-ray spectroscopy such as the Regolith X-ray Imaging Spectrometer (REXIS) currently proposed on the OSIRIS-Rex mission.

These data would directly feed into subsequent surface and subsurface sampling operations planning, and the corresponding development of equipment and tools. Specific surface and subsurface operations could involve:

- Taking core samples at various depths for further processing tests on Earth, dust mitigation, and measuring with more accuracy mechanical and electrical properties to compare with remote sensing surveys.
- Testing of large-scale sample acquisition using various collection approaches, leading to subsequent mining activities.
- Testing of anchoring procedures and devices.
- Verification and validation of proximity operations procedures to be implemented at deep-space locations such as the moons of Mars or other near-Earth asteroid destinations.

Mining/Benefaction/Extraction/Fabrication – The technical requirements for mining asteroids would be as diverse as those used on Earth. Plausible asteroidal feedstocks cover a vast range of chemical compositions and physical properties, suggesting a careful tailoring of drilling, blasting, cutting, and crushing hardware to the chosen target—and placing a premium upon prior knowledge of the nature of the target material. Indeed, one of the central reasons for choosing a water-bearing C-type asteroid as our first target is that the chemical and physical properties of these materials are both rather well understood and benign (very low crushing strength and high content of desirable volatiles). Bench-scale prototypes of systems for processing asteroidal materials have been developed in laboratories on Earth, in some cases using real meteorite materials as the feedstock.

Further development of equipment for effecting mineral separation on asteroids, a process that would become more important in potential future missions to volatile-poor metal-bearing asteroids, could await both experience with the first retrieved asteroid and laboratory investigations on meteorite samples. Beneficiation (the selective enrichment of desired minerals) may in many cases require crushing of the target rock, followed by magnetic, electrostatic, or other means of concentration. Such concentration technologies would also be of considerable value on the Moon for the concentration of potential ores such as ilmenite.

The extraction of a desired material (water, carbon, nitrogen, iron, nickel, sulfur, platinum-group metals, etc.) may involve either chemical or physical processes. Examples include thermal decomposition of clay minerals and hydrated salts to release water vapor, Mond-process volatilization and separation of metallic iron and nickel, electrolysis of molten silicates, or any of dozens of other candidate techniques which would be chosen for their relevance to the intended target and the desired product.

Fabrication of products would likewise involve a host of different possible processes. Production of high-purity water for propulsion or life-support use may require controlled distillation of the first-cut water driven off by heating the asteroid material to separate the water from undesirable contaminants such as volatile organics and sulfur and chlorine compounds. Likewise, production of high-purity iron (99.9999% iron has the corrosion resistance of stainless steel and a very high tensile strength) could be effected by Mond-process volatilization of native metal alloys, simple distillation to separate iron and nickel carbonyls, and controlled thermal decomposition of the iron pentacarbonyl vapor in a heated mold (at about 200 Celsius and 1 atm pressure). Fabrication of refractory bricks or aerobrakes could be done by microwave sintering of appropriate metal-oxide mixtures in molds. These candidate fabrication processes could be developed sequentially as our experience with in-space processing grows, and as new classes of asteroidal feedstock become available.

Cost Estimate

The GRC COMPASS team generated an initial cost estimate for the Asteroid Capture and Return mission concept. This cost estimate, in FY'12 $, is based on the following assumptions.

- Prime contractor design, test & build based on NASA-provided specs
- Proto-flight development approach (except power and propulsion subsystems)
- Single ground spares included where applicable
- Assumes all technologies are at TRL 6 – the estimate does not include any cost for technology development up to TRL 6
- The cost estimate:
  - Represents the most likely estimate based on cost-risk simulation results
  - Includes mass growth allowance
  - Is a parametric estimate based on mostly mass-based Cost Estimating Relationships (CERs) using historical cost data
  - Includes planetary systems integration wraps
  - Includes flight software costs based on analogy to the Dawn flight system
  - Does not include the cost of propellant

With these assumptions the estimate of the Prime Contractor cost including fee given in Table 6 was generated. The total cost for the first unit including DDT&E is $1.36B. The recurring cost for the flight hardware is estimated to be $0.34B. The total cost for the first ACR mission is estimated at $2.6M as indicated in Table 6 including NASA insight/oversight, the cost of the launch services, mission operations, and reserves.

Table 6. Total cost estimate for the Asteroid Capture and Return mission concept.

Item	FY'12 $M	Comments
NASA insight/oversight	204	15% of prime contractor costs
Phase A	68	5% of Phase B/C/D
Flight System	1359	Prime Contractor B/C/D cost plus fee
Launch Services	288	Atlas V 551-class
MOS/GDS	117	10-yr mission
Reserves	611	30% reserves
Total	**2647**	

VII. SEP Technology Status and Required Development

Affordable, high-performance, deep-space propulsion technology is essential for the ACR mission concept. Solar electric propulsion is the most cost-effective technology in existence for providing substantial post-launch propulsion capability in deep space.

For the proof-of-concept low-thrust trajectories described above based on asteroid 2008 HU4, the ΔV required to move the asteroid to lunar orbit would be only approximately 170 m/s. The large asteroid mass, however, would result in a substantial required total impulse. If we assume that 2008 HU4 has a mass of 1000 t, and our spacecraft has a dry mass of 5.5 t, then from the rocket equation we get the required propellant masses shown in Fig. 8 for three different propulsion options: LOX/LH2 with an *Isp* of 450 s; a space-storable bi-propellant system with an *Isp* of 325 s; and an our Hall-thruster-based electric propulsion system with an *Isp* of 3,000 s. This figure shows only the propellant mass required for the return leg of the mission. It does not include the propellant mass required to deliver the return propellant to the NEA. The space-storable chemical propulsion system would require over 50 t of propellant to transport the NEA to lunar orbit. Even the best chemical propulsion technology, LOX/LH2, would require nearly 40 t of propellant at the NEA and assumes that long-term, zero-boil-off technology is available. Significantly more propellant, of course, would be required to deliver this propellant mass to the NEA. The SEP system, on the other hand would require just under 6 t of xenon propellant at the NEA. The use of electric propulsion would enable a single EELV-launched ARC mission.

The basic ACR mission concept would require an SEP technology characterized by an end-of-life power level of order 40 kW, a Hall thruster technology capable of operating at a specific impulse of 3,000 s, and lightweight propellant tanks capable of storing up to 12,000 kg of xenon. The current state-of-the-art for these technologies and prospects for maturing them to the levels required for the ACR mission are described below.

Solar Array Technology

The current state of the art for solar array technology is probably best represented by the solar arrays in use on the largest commercial communication satellites. These satellites use rigid-panel arrays with triple-junction cells and beginning-of-life (BOL) power levels up to 24 kW. At least one commercial satellite manufacturer is now offering a 30-kW BOL capability. A typical rigid-panel solar array has a specific power of approximately 80 W/kg.

The alternative to rigid-panel solar arrays are flexible-blanket arrays. Flexible-blanket arrays have been flown on the International Space Station (ISS) in a rectangular configuration with 12% efficient single-junction solar cells giving a specific power of about 40 W/kg, and on the

Phoenix mission in the circular Ultraflex [18] configuration with 27% efficient solar cells resulting in a specific power of about 110 W/kg.

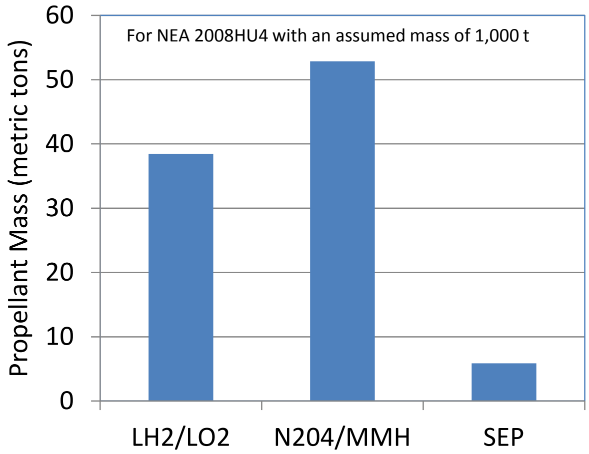

Figure 8. The estimated propellant mass required to return a 1000-t NEA to lunar orbit would be prohibitive without solar electric propulsion (SEP).

The ACR flight system concept described above assumes the use of a flexible blanket solar array in the Ultraflex configuration with 33% efficient inverted metamorphic (IMM) cells. The BOL specific power, however, would be a conservative 73 W/kg, because this includes 500-micron thick cover glass on the front and back of the cells to reduce the radiation damage during the spiral out through the Earth's radiation belts.

Ultraflex solar arrays were scaled up by nearly an order of magnitude from 0.75 kW per wing for the Phoenix spacecraft to about 7 kW per wing for the Orion vehicle [20]. The ACR mission concept would need an additional factor of four increase in the Ultraflex solar array power to about 29 kW per wing. The circular configuration of the Ultraflex solar array means that a factor of four increase in power per wing could be achieved by increasing the wing radius by only a factor of two. IMM solar cells with an efficiency of 33% are expected to be flight qualified well in advance of the 2020 launch date assumed for the ACR mission concept.

Electric Propulsion Technology

The electric propulsion technology required for the ACR mission concept has three key components: Hall thrusters capable of processing an input power of 10 kW each while producing a specific impulse of 3,000 s; Power Processing Units (PPUs) capable of providing the power necessary to operate the Hall thrusters at this specific impulse; and propellant tanks capable of storing the required xenon load with a tankage fraction of approximately 4%.

The state-of-the-art in Hall thruster technology is represented by the BPT-4000 thrusters that are currently flying on the Air Force Advanced Extremely High Frequency (AEHF) satellite [21]. These thrusters operate at up to 4.5 kW and a specific impulse of up to 2,000 s. Hall thrusters under development have been operated at specific impulses over 3,000 s at around 6 kW [22]. Other Hall thrusters have been designed and tested for operation at power levels of 20 kW and higher [23,24]. The thrusters are assumed to incorporate recently developed technologies which mitigate channel wall erosion so that no additional thrusters need to be added because of propellant throughput limitations [25,26]. The ACR mission concept requirements for a 10-kW, 3000-s Hall thruster represent a capability that could easily be developed.

The specific impulse of 3000 s needed for the ACR mission design would require an input voltage to the Hall thruster of approximately 800 V. Voltages of this level are currently considered to be too risky for solar array operation and so direct-drive was not considered for the ACR flight system concept. Consequently, the ACR spacecraft concept assumes the use of a conventional PPU with an output voltage capability of 800 V and 10 kW. Hall thruster PPUs are under development that could produce the required voltage level and others that could produce the required power level. Therefore, development of a PPU with the required capability should be straight forward.

The ACR mission design would require the storage of about 12,000 kg of xenon. This is nearly a factor 30 greater than the 425 kg launched on the Dawn mission – the largest xenon propellant load launched to date. The Dawn xenon tank has a tankage fraction of 5% [27]. Lightweight tank technology currently under development is projected to enable a xenon tankage fraction of 3%. For the ACR mission concept we have assumed a tankage fraction of 4% as a low-risk extension of the current state-of-the-art.

Near-Term Application of SEP Technology for Human Missions to NEAs

The development of a 40 kW-class SEP system would provide the valuable capability of being able to pre-deploy several tons of destination elements, logistics, and payloads. Initial estimates identify that approximately 3,100 kg of elements and logistics, along with approximately 500 kg of destination payload, could be pre-deployed in support of a human NEA mission, rather than carried with the crew. This approach would reduce the requirements for the launch vehicles and in-space propulsive elements required to conduct a human mission. The amount of mass that could be pre-deployed along with the SEP system is primarily a function of the launch vehicle utilized, the orbital energy requirements of the NEA target, the efficiency of the SEP system, and the desired amount of returned mass. Although a SEP system and associated cargo could be delivered to low-Earth orbit (LEO) by the launch vehicle and spiraled out to escape the Earth's gravity, the time required to perform this operation along with the radiation and micrometeoroid and orbital debris (MMOD) exposure resulting from the spiral from LEO would make it desirable for the launch vehicle to be able to propel the SEP system and payload to an escape C3. Additionally, since the departure windows for accessible NEAs could be short and since it is likely that pre-deployed assets would be required to be at NEA prior to crew departure from Earth, the duration of the pre-deploy mission would be a critical factor.

Another important capability that could be leveraged is the ability to return several metric tons of asteroid samples to cislunar space and/or the possible return and reuse of mission elements. Currently, the Orion Multi-purpose Crew Vehicle (MPCV) is limited in the amount of mass it could return to the Earth's surface. The current estimate for the MPCV return capability is 100 kg of samples and associated containers. These samples would be returned to cislunar space and they could either be cached or analyzed and high-graded before the final samples were returned to Earth over some period of time. Being able to return several tons of samples would greatly increase the value of a human NEA mission, and returning critical, high-value mission elements could reduce the cost of subsequent human missions.

A mission concept utilizing pre-deployment and providing multi-ton sample return capability is depicted in Fig. 9.

If the SEP system could deliver ~4,000 kg of payload to the target NEA for a human mission, this would likely be sufficient to provide the necessary elements and equipment to be able to utilize the SEP as an excursion vehicle (e.g., airlock, robotic arms, anchoring system, etc.) for exploring the surface of the NEA. A preliminary analysis indicates that using SEP for excursions from the mission deep space habitat to the NEA appears feasible from a daily travel time/distance standpoint, but the ability to perform local proximity operations needs further detailed analyses. A conceptual excursion spacecraft is depicted in Fig. 10. Developing confidence in the SEP system (i.e., the power and propulsive systems) could also lead to the development of higher powered SEP systems (200-300 kW-class) with greater pre-deploy and return capability which could also be used for the direct transfer of crew to and from the NEA target.

Additionally, the anchoring/capture hardware developed for the asteroid retrieval mission would provide valuable testing of the systems and the operational approaches. The SEP system could also provide resource

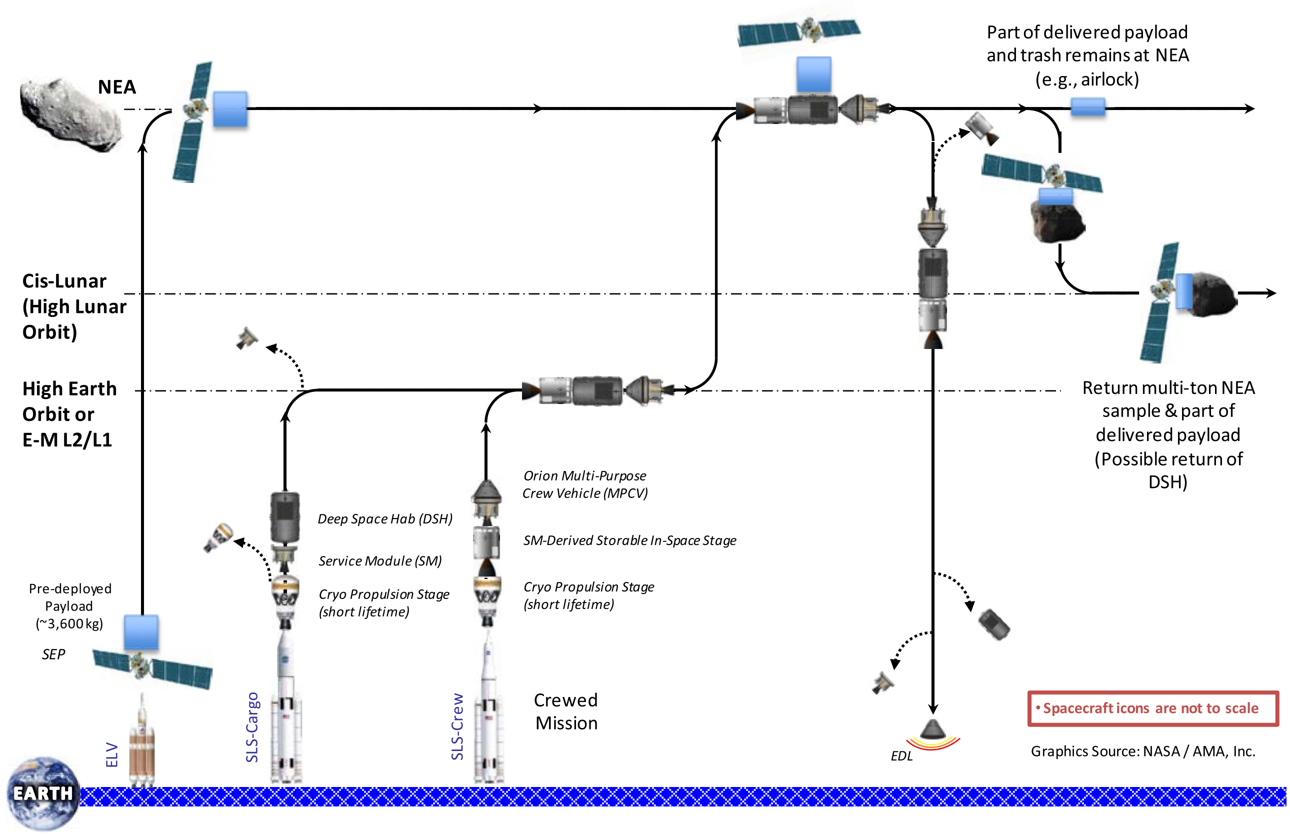

Figure 9. Notional NEA Human Mission Concept of Operations with Pre-deploy

redundancy at the destination (e.g., power and communications) during the crew mission, which could help reduce mission risk and provide additional capability at the destination.

Another important synergistic application of the SEP system would be to facilitate a multi-target robotic precursor to select the human mission NEA target(s). The SEP system could be utilized to deploy multiple independent NEA probes (rendezvous/surface) to provide reconnaissance of human targets and return a large boulder and regolith from a human target prior to conducting the human missions.

The asteroidal material delivered to cislunar space could be used to provide radiation shielding for future deep space missions and also validate in-situ resource utilization (ISRU) processes (water extraction, propellant production, etc.) that could significantly reduce the mass and propulsion requirements for a human mission. The introduction of ISRU into human mission designs could be extremely beneficial, but until the processing and storage techniques have been sufficiently tested in a relevant environment it is difficult to baseline the use of ISRU into the human mission architecture. Bringing back large quantities of asteroid materials to an advantageous location would make validation of an ISRU system significantly easier. Small asteroids could benefit the planetary defense initiatives by providing a better understanding of the nature and properties of potential Earth impactors and by facilitating the maturation of mission hardware and operational approaches. One day, in the more distant future, it is possible that a small NEA (10-m, 1500-t) returned to E-M L2/L1 could act as an orbiting platform/counter weight for a lunar space elevator to allow routine access to and from the lunar surface and also function as a space resource processing facility for mining significant quantities of materials for future human space exploration and settlement and possible return and inclusion in terrestrial markets.

Figure 10. Conceptual Human NEA Mission Excursion Vehicle Using SEP.
(Image Credit: Source: NASA / AMA, Inc.)

VIII. CONCLUSIONS

The two major conclusions from the KISS study are: 1) that it appears feasible to identify, capture and return an entire ~7-m diameter, ~500,000-kg near-Earth asteroid to a high lunar orbit using technology that is or could be available in this decade, and 2) that such an endeavor may be essential technically and programmatically for the success of both near-term and long-term human exploration beyond low-Earth orbit. One of the key challenges – the discovery and characterization of a sufficiently large number of small asteroids of the right type, size, spin state and orbital characteristics – could be addressed by a low-cost, ground-based observation campaign identified in the study. To be an attractive target for return the asteroid must be a C-type approximately 7 m in diameter, have a synodic period of approximately 10 years, and require a ΔV for return of less than ~200 m/s. Implementation of the observation campaign could enable the discovery of a few thousand small asteroids per year and the characterization of a fraction of these resulting in a likelihood of finding about five good targets per year that meet the criteria for return.

Proof-of-concept trajectory analysis based on asteroid 2008 HU4 (which is approximately the right size, but of an unknown spectral type) suggest that a robotic spacecraft with a 40-kW solar electric propulsion system could return this asteroid to a high-lunar orbit in a total flight time of 6 to 10 years assuming the asteroid has a mass in the range of 250,000 to 1,000,000 kg (with the shorter flight times corresponding to the lower asteroid mass). Significantly, these proof-of-concept trajectories baseline a single Atlas V-class launch to low-Earth orbit.

The study also considered an alternative concept in which the spacecraft would pick up a ~7-m diameter rock from the surface of a much larger asteroid (> 100-m diameter). The advantage of this approach is that asteroids 100-m in diameter or greater are much easier to discover and characterize. This advantage is somewhat offset by the added complexity of trying to pick up a large 7-m diameter rock from the surface, and the fact that there are far fewer 100-m class NEAs than smaller ones making it more difficult to find ones with the desired orbital characteristics. This mission approach would seek to return approximately the same mass of asteroid material – of order 500,000 kg – as the approach that would return an entire small NEA.

The proposed Asteroid Capture and Return mission would impact an impressive range of NASA interests including: the establishment of an accessible, high-value target in cislunar space; near-term operational experience with astronaut crews in the vicinity of an asteroid; a new synergy between robotic and human missions in which robotic spacecraft return resources for human exploitation and use in space; the potential to jump-start an entire industry based on *in situ* resource utilization; expansion of international cooperation in space; and planetary defense. It has the potential for cost effectively providing sufficient radiation shielding to protect astronauts from galactic cosmic rays and to provide the propellant necessary to transport the resulting shielded habitats. It would endow NASA and its partners with a new capability in deep space that hasn't been seen since Apollo. Ever since the completion of the cold-war-based Apollo program there has been no over-arching geo-political rationale for the nation's space ventures. Retrieving an asteroid for human exploration and exploitation would provide a new rationale for global achievement and inspiration. For the first time humanity would begin modification of the heavens for its benefit.

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