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Spacecraft Conceptual Design for Returning Entire Near-Earth Asteroids

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In situ resource utilization (ISRU) in general, and asteroid mining in particular are ideas that have been around for a long time, and for good reason. It is clear that ultimately human exploration beyond low-Earth orbit will have to utilize the material resources available in space. Historically, the lack of sufficiently capable in-space transportation has been one of the key impediments to the harvesting of near-Earth asteroid resources. With the advent of high-power (or order 40 kW) solar electric propulsion systems, that impediment is being removed. High-power solar electric propulsion (SEP) would be enabling for the exploitation of asteroid resources. The design of a 40-kW end-of-life SEP system is presented that could rendezvous with, capture, and subsequently transport a 1,000-metric-ton near-Earth asteroid back to cislunar space. The conceptual spacecraft design was developed by the Collaborative Modeling for Parametric Assessment of Space Systems (COMPASS) team at the Glenn Research Center in collaboration with the Keck Institute for Space Studies (KISS) team assembled to investigate the feasibility of an asteroid retrieval mission. Returning such an object to cislunar space would enable astronaut crews to inspect, sample, dissect, and ultimately determine how to extract the desired materials from the asteroid. This process could jump-start the entire ISRU industry.

I. Introduction

The idea to exploit the natural resources of asteroids has been around for over a hundred years.[1] In the 1970's the use of near-Earth asteroids (NEAs) as a source of resources figured prominently in O'Niells' concepts for the colonization of space[2] and detailed concepts for the mining and retrieval of material from near-Earth asteroids, as envisioned in the 1970's, are given by O'Leary.[3-6] This body of work concluded that "*the asteroid-retrieval option is competitive with the retrieval of lunar materials for space manufacturing.*" It also concluded that "*…a carbonaceous object would provide a distinctive advantage over the Earth as a source of consumables and raw materials for biomass in space settlements…*" Consequently, O'Leary, et al.,[5] recommended an increased search program to identify and characterize attractive targets for mining, robotic precursor missions, and the development of supporting technologies including in-space transportation. These studies, however, were fantastically ambitious and optimistic. They targeted the return of an asteroid fragment with a mass of a million tons or more and suggested that it would be possible "*using existing technology.*"[3] Nevertheless, these studies identified the key attractive features for asteroid mining including that:

1. NEAs are more desirable targets than main-belt asteroids because they are energetically easier to get to and from.
2. There is five to ten times the solar flux at near-Earth asteroids than in the main asteroid belt enabling higher power solar electric propulsion (SEP)-based transportation systems.
3. NEAs are potentially a much richer source of materials than the lunar surface.
4. Energetically many near-Earth asteroids are easier to return material from than the lunar surface.
5. In-space transportation is one of the key enabling technologies for asteroid mining.
6. Lunar gravity assist trajectories could be used to "kill" up to ~2 km/s of excess hyperbolic velocity on return to cislunar space to facilitate capture into a highly elliptical Earth orbits.[5]

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7. The value of the material returned to cislunar space is derived largely from its delivery to this location.
8. A key objective of asteroid mining is to minimize the cost per unit mass of the material returned to the point of use.[3]

In the 1970's however, the required transportation technology was not up to the task and several different propulsion technologies were considered including linear[6] and rotary[7] mass drivers, solar sails, and even using material from one asteroid to collisionally decelerate a second asteroid.[8] All of the concepts considered in the 1970's assumed the use of reaction mass obtained from the asteroid itself. This conclusion was driven by the assumed scale of the asteroid mining operations. O'Leary considered the capture and transportation back to cislunar space of a 200-m diameter near-Earth asteroid with a mass of 10^7 tons, and estimated that the cost of such a mission would be around $1 billion (in 1976 dollars, equivalent to about $4 billion in 2012 dollars). This price tag didn't include the cost for the development of the required mass-driver based transportation technology. Later authors recognized that it is essential to minimize the initial costs for asteroid mining[9] or such endeavors will never get started. So while the use asteroid material for reaction mass to transport the rest of the asteroid may ultimately be the most attractive approach, the complexity, and therefore the cost, required to do this makes it unlikely to be the first method used.

A review of near-Earth asteroid mining concepts at the start of the new millennium is given by Ross.[10] This review again highlights that near-Earth asteroids (NEAs) are potentially a much richer source of desirable raw materials than the Moon, and notes that one of the early applications could be the use of "unprocessed" asteroidal material for shielding against galactic cosmic rays. However, the transportation problem was still unsolved and Ross discusses various options to augment lunar gravity assist trajectories including "*propulsive breaking using some of the Asteroid-derived propellant*," and aerobraking using an "*Earth-fabricated, LEO-fabricated, or asteroid-fabricated aerobrake made of metallic or refractory silicate*." Ross recognized that the fabrication of an aerobrake on an asteroid would add considerable complexity to the endeavor and may require a human presence which would "*increase the cost substantially*."

It is clear that ultimately the expansion of human exploration beyond low-Earth orbit will require the use of in situ resources. It is also clear that where you are going will dictate the most attractive source of those resources. Near-Earth asteroids, the Moon, and Mars have all received serious consideration as sources for in situ resource utilization (ISRU).[11] For operations in cislunar space, near-Earth asteroids appear to be the most attractive from both an energy standpoint and from the richness of the available resources. This paper outlines a solution to the transportation problem for the exploitation of near-Earth asteroid resources. It provides a way to minimize the initial cost and risk for asteroid mining and could provide the way to jump-start an entire ISRU industry ultimately resulting in the exploitation of resources on the Moon and Mars in addition to NEAs.

II. Approach

There are three generic approaches for mining asteroids: 1) Mine and process the material at the asteroid and return only the processed material; 2) Mine the asteroid and return the raw material for processing; 3) Return an entire small asteroid for processing. The first approach has the advantage of minimizing the return mass since only the high-value material is brought back. This approach has been the subject of numerous studies (see Erickson[12] for example), but it suffers from the need to deliver all of the mining, material-extraction, and material-storage hardware to the asteroid. This requires a sufficiently detailed knowledge of how to mine and then extract the desired substances from the asteroid raw material in a microgravity environment such that highly reliable machines for these functions could be developed. It is hard to see how such developments could take place without significant in-flight testing or subjecting the first mission to unacceptable risk. To further complicate matters, the long synodic periods of attractive target asteroids suggest that mining operations will only get one shot at each asteroid and orbital mechanics may dictate that the available mining "season" could be relatively short.[10] The very high degree of automation required for this approach and/or the possible requirement to have astronauts present to solve problems as they arise suggests that this is unlikely to be the first viable approach. The second and third approaches would require less sophisticated automation and were the topic of recent papers,[13-15] as well as a detailed study by the Keck Institute for Space Studies (KISS).[16]

A. Returning Entire NEAs

One potential approach for jump-starting the ISRU industry would be to minimize the initial cost for exploiting in-space resources. This is the approach described in this paper which is intended to minimize the cost of the flight system development and mission operations, and maximize the use of other in-space resources, either existing or planned, to demonstrate the feasibility of harvesting near-Earth asteroids. This approach, detailed in the KISS study,[16] involves the discovery and characterization, rendezvous and capture, and subsequent transportation of an entire, small near-Earth asteroid to cislunar space. It relies on the following three key features:

1) The development in this decade of the capability to discover and characterize an adequate number of sufficiently small NEAs per year around which a robust mission could be planned.
2) The development of a sufficiently powerful solar electric propulsion system to rendezvous with and transport the NEA in a reasonable total flight time from a single launch.
3) The existence of a human exploration capability in cislunar space which could examine, sample, dissect, and ultimately learn how to extract the desired materials from the retrieved asteroid in a microgravity environment. In this final step it could be highly beneficial to take advantage of the \$100B worth of infrastructure represented by the International Space Station (ISS) to test extraction technologies in space. Significant quantities of material mined from the returned asteroid could be brought to the ISS to test processing approaches and hardware.

Affordable in-space transportation is the key to the asteroid retrieval mission concept. For such a mission to be feasible there must be an overlap between near-Earth asteroids that are sufficiently large that they can be discovered and characterized and those that are sufficiently small that they can be transported in a reasonable flight time. More capable propulsion technologies push this overlap toward larger asteroids. The KISS study[16] suggested that this overlap currently occurs around asteroids approximately 7 m in diameter with a masses of order 500,000 kg (which is approximately equal to the mass of the International Space Station). The approximate asteroid mass versus diameter is given in Table 1 for asteroid densities in the range 1.9 g/cm^3 to 3.8 g/cm^3.

The best technique for determining the asteroid size is radar imaging. The Goldstone radar can currently image asteroids with 3.75-m resolution[17] and future upgrades may improve this to 2-m resolution. The gray shaded region in this table indicates that even if the uncertainty in the diameter of the asteroid can be reduced to ± 1 m, the estimated mass could range from 200,000 kg to 1,000,000 kg. The flight system design must accommodate the wide range in mass uncertainty.

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

The size-frequency distribution from Harris[18] is reproduced in Fig.1 and suggests that there may be as many as a hundred million 7-m diameter NEAs. If this is true it suggests that there are roughly a thousand times more NEAs of this size than asteroids larger than 100 m diameter. This in turn suggests that there may be a thousand times better chance of finding 7-m diameter asteroids with the right combination of characteristics to make them attractive targets for retrieval for than for larger NEAs. Generally there hasn't been much interest in very small asteroids since they are not considered to be potentially hazardous objects. Approximately 280 NEAs ≤ 10-m diameter have been discovered to date. Few of these have secure orbits and none have known spectral types.

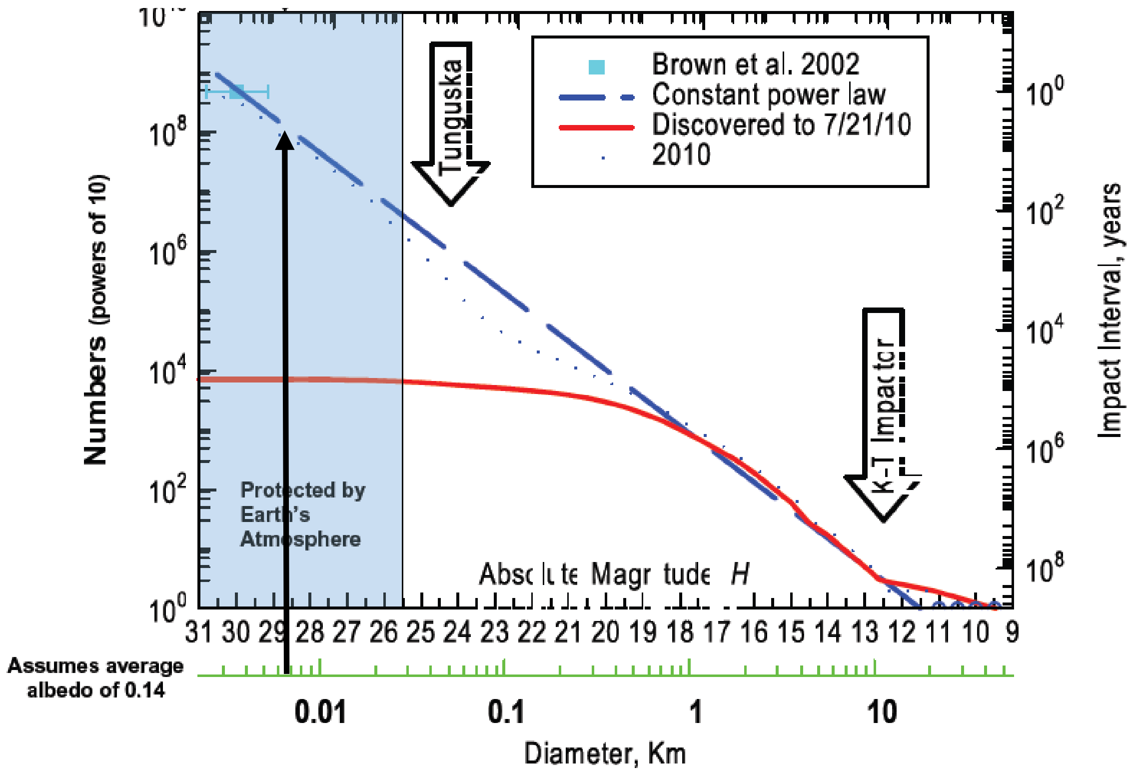

Fig. 1 NEA size frequency distribution[18] suggests that there are roughly a hundred million 7-m diameter near-Earth asteroids.

There is one known NEA, 2008 HU4, that is about the right size (~8-m diameter) and has orbital characteristics that make it an attractive candidate for return. The spectral type for 2008 HU4 is unknown, so it couldn't be used for actual mission planning, but instead it is used for proof-of-concept trajectory analyses to determine what propulsion capabilities are required to return such an object. This NEA has a synoptic period of about 10 years and will make its next close approach to Earth in 2016. At this opportunity the asteroid could be characterized to determine its spectral type and spin state, and reduce the uncertainty in its size, mass, and orbital elements. If it were of the right spectral type it could become a candidate for retrieval in 2026 at its subsequent close approach to Earth.

B. Candidate Propulsion Technologies

Three different propulsion technologies were evaluated for the asteroid retrieval mission based on asteroid 2008 HU4, and assuming that this asteroid has a mass of 1,000 metric tons. These three technologies were zero-boil-off LOX/LH2 with an assumed specific impulse of 465 s, a space storable bi-prop. system (N_2O_4/MMH) with an *Isp* of 325 s, and a 40-kW end-of-life Hall thruster system with a specific impulse of 3,000 s. Each flight system is assumed to start from low-Earth (LEO) orbit and uses its on-board propulsion system to do the transfer to the asteroid and then return the asteroid to cislunar space. The delta-Vs for the high-thrust and low-thrust trajectories to/from 2008 HU4 are given in Table 2. Note, the delta-V for the return leg of the mission is assumed to be the same for the chemical and electric propulsion options under the assumption that the asteroid is sufficiently massive that the chemical propulsion technologies will be effectively low-thrust for the return trajectories. Also shown in Table 2 are delta-Vs for going from LEO to the lunar surface and from the lunar surface to Earth-Moon L2. The total delta-V for such a mission is 8.4 km/s. It is noteworthy that this is significantly greater than the 4.6 km/s for the high-thrust mission to capture and return 2008 HU4. The total low-thrust delta-V to capture and return 2008 HU4 is 9.6

km/s, which is larger than the lunar surface mission, however, because of the negligible gravity of the NEA this mission can be accomplished entirely with the much higher Isp Hall thruster system.

Table 2 Mission Delta-V Requirements

From	To	Delta-V (km/s)
LEO	Asteroid 2008 HU4	4.4 (high-thrust)
LEO	Asteroid 2008 HU4	9.4 (low-thrust) + Lunar Gravity Assist
Asteroid 2008 HU4	High Lunar Orbit	0.17 + Lunar Gravity Assist
LEO	Lunar Surface	5.9
Lunar Surface	Earth-Moon L2	2.5

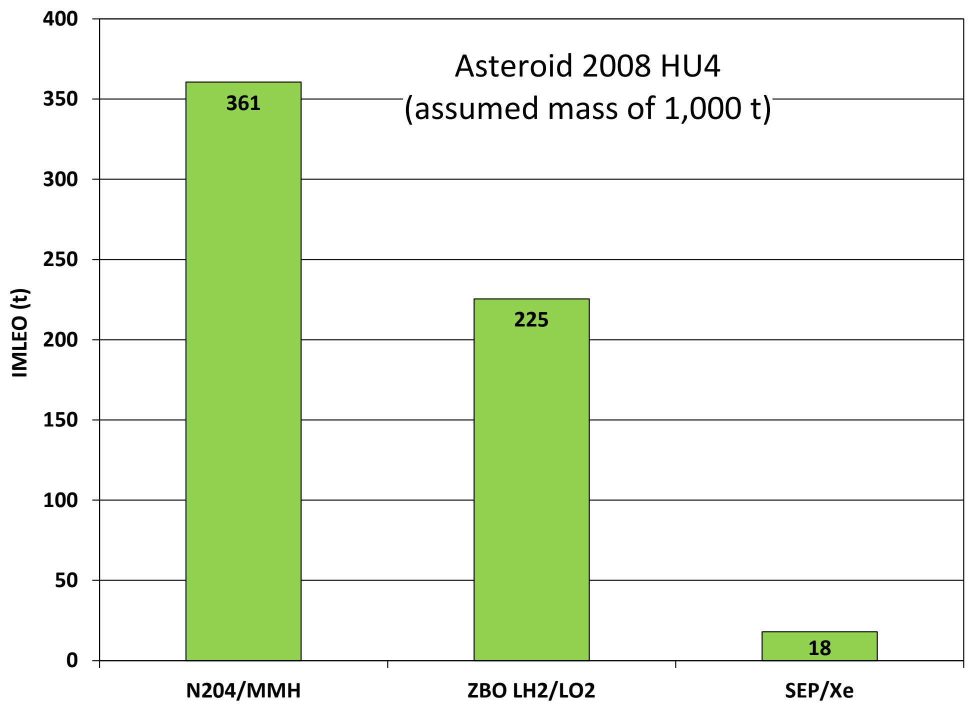

Fig. 2. Initial mass in low-Earth orbit (IMLEO) for three propulsion options for the rendezvous, capture and return of asteroid 2008 HU4 assuming the asteroid has a mass of 1,000,000 kg (1,000 t).

The results for the three propulsion technologies are compared in Fig. 2 on the basis of the initial mass in low-Earth orbit (IMLEO) required to perform the asteroid capture and return mission. The space-storable bi-prop. system has by far the greatest IMLEO. At 361 tons, this is 36% of the mass of the asteroid itself, and would require approximately five 70-t, heavy lift launches, or four 105-t launches to LEO, plus on-orbit assembly. The LO_2/LH_2 option requires the development of zero-boil-off (ZBO) technology, three 70-t launches or two 105-t launches, plus on-orbit assembly. The SEP system, on the other hand, requires the delivery of only 18 t to LEO which could be accomplished with a single Evolved Expendable Launch Vehicle (EELV) such as the Atlas V 551. This enormous order-of-magnitude reduction in IMLEO is enabling for the asteroid retrieval concept.

III. Orbit Transfer

Two general approaches have been identified for the asteroid retrieval mission concept. The first is to identify, characterize and subsequently rendezvous with, capture and return and entire near-Earth asteroid that is approximately 7-m diameter. This approach is referred to as the "get-a-whole-one" mission concept. The other approach is to pick a bolder that is approximately 7-m diameter off a much larger near-Earth asteroid. This approach is referred to as the "pick-up-a-rock" scenario.

A. Get-a-Whole-One

The overall mission design for the get-a-whole-one concept is illustrated in Fig. 3. This concept is built around a 40-kW end-of-life (EOL) solar electric propulsion system described in Section IV. The spacecraft would be launched to low-Earth orbit (LEO) using a single Atlas V 551-class launch vehicle. The SEP system would then spiral the spacecraft to a high-Earth orbit where a lunar gravity assist (LGA) would put the vehicle on an escape trajectory with a positive C3 of about 2 km^2/s^2. The SEP system would then complete the heliocentric transfer to the target NEA. Once at the asteroid, the mission design concept would allocate 90 days for characterization of the NEA, determination of its spin state, creation of a detailed shape model, and the subsequent capture and de-tumbling of the asteroid. The SEP system would then transport the NEA back to the vicinity of the Earth-moon system where another lunar gravity assist would be used to capture the vehicle plus NEA to a slightly negative C3. Approximately 4.5 months after the LGA, the asteroid and spacecraft would complete the transfer to a stable high lunar orbit with essentially zero additional ΔV.[17]

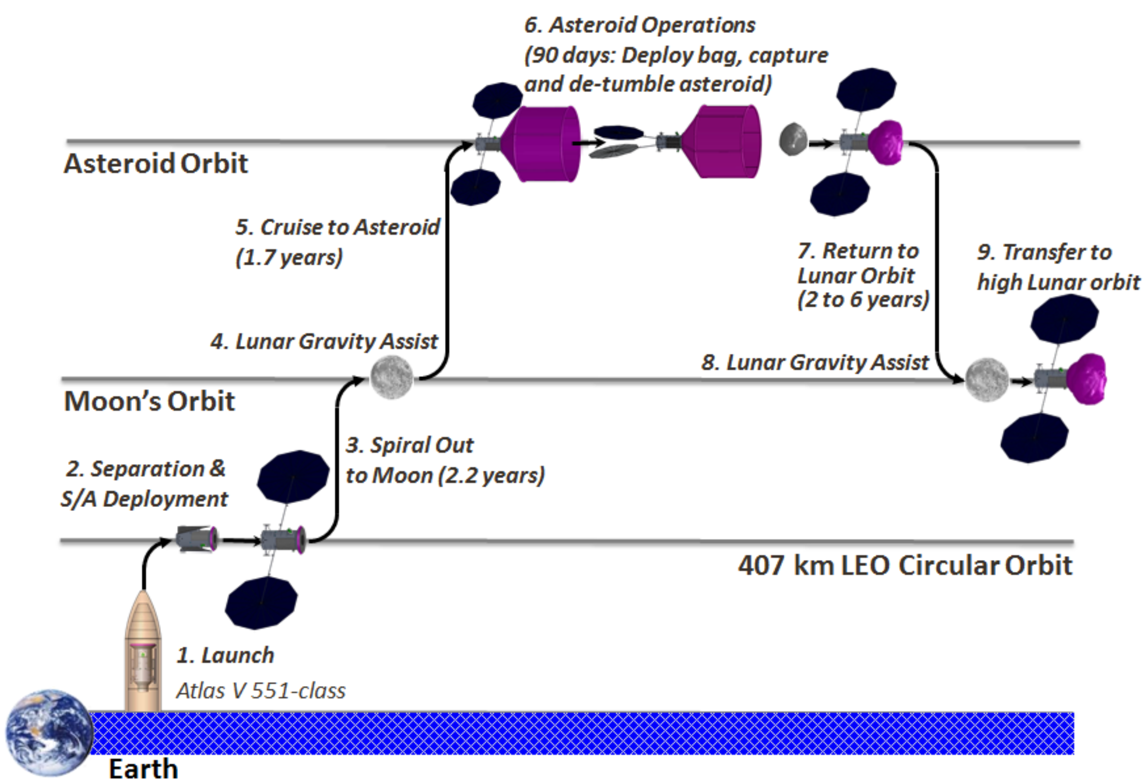

Fig. 3. Asteroid return mission concept. Return flight time of 2 to 6 years depending on the asteroid mass.

The heliocentric trajectory results for 2008 HU4 are summarized in Table 3 and Fig.4. Since this asteroid is of an unknown type, its mass is highly uncertain. Therefore, the data in Table 3 cover a range of assumed asteroid masses from as low as 250 t to as high as 1,300 t. As indicated in this table, and not unexpectedly, larger assumed masses for the asteroid would require longer return flight times. However, the return date would be fixed to when the NEA naturally has a close encounter to Earth (2026 for 2088 HU4), so the additional flight time would come at the expense of earlier launch and arrival dates at the asteroid. The 4th row in Table 3 indicates that if 2008 HU4 has a mass of 950 t, the 40-kW EOL SEP vehicle could be capture and return it to a high lunar orbit in a total flight time of about 9.2 years from a single Atlas V 551-class launch vehicle. Higher power SEP systems would reduce the flight times in Table 3.

Table 3. Trajectory performance summary for the asteroid retrieval option for different assumed asteroid masses to be returned by 2026.

Target Asteroid	Assumed Asteroid Mass (t)	Launch Vehicle	Xe Mass (not including the Earth spiral) (t)	Launch Date	Heliocentric Flight Time (not including Earth spiral) (years)	Total Flight Time (years)	Arrival *C3* (km^2/s^2)
2008 HU4	250	Atlas V 521-class	5.0	Feb. 2020	4.0	6.2	1.8
2008 HU4	400	Atlas V 521-class	5.2	Feb. 2019	5.0	7.2	1.7
2008 HU4	650	Atlas V 521-class	6.5	Feb. 2018	6.0	8.2	1.6
2008 HU4	950	Atlas V 551-class	8.9	Feb. 2017	7.0	9.2	1.6
2008 HU4	1300	Atlas V 551-class	9.1	Feb. 2016	8.0	10.2	1.6

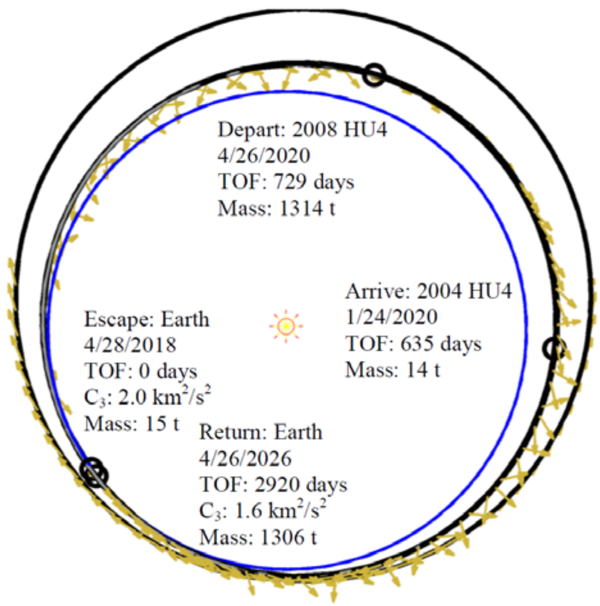

Fig. 4. Heliocentric trajectory for the rendezvous, capture and return of asteroid 2008 HU4.

B. Pick-Up-a-Rock

The key to the get-a-whole-one mission option is the observation campaign necessary to discover and characterize a sufficient number of attractive small NEAs for capture and return. This is a difficult undertaking whose success is not assured. Since larger NEAs of 10's to 100s of meters in diameter are much easier to characterize, the pick-up-a-rock approach was identified as an alternative that could return a large quantities of material from a well-characterized asteroid. Proof-of-concept trajectories were performed by Landau for asteroid 1998 KY26.[17] This asteroid is believed to be only about 30-m diameter, and is known to be a water-rich object.[19] However, this particular asteroid is also known to be spinning too fast to be a rubble pile, and so may not be a good candidate for an actual pick-up-a-rock mission, but it was used here simply for proof-of-concept trajectory analysis just as 2008 HU4 was used for the get-a-whole-one analysis.

The trajectory results for 1998 KY26 are summarized in Table 4 assuming the use of the same 40-kW EOL SEP vehicle assumed for the get-a-whole-one option. These results suggest that this mission could be launched from an Atlas V 521-class launch vehicle and potentially return 30 to 60 t of asteroid material in a total flight time of 6.9 to 7.5 years. An example of the trajectory for returning 60 t of asteroid material by November 2025 is given in Fig. 5. This trajectory assumes a launch to low-Earth orbit and a spiral out to Earth escape using the SEP system. The vehicle at Earth escape would have a mass of about 11 t. This is about the same as the projected performance of the Space X Falcon Heavy to a C3 of zero.[20] Suggesting that the use of a launch vehicle with Falcon Heavy-like performance could eliminate the Earth spiral out with the SEP system and shave about two years off the total mission duration.

Table 4. Trajectory performance summary for the pick-up-a-rock mission option.

Target Asteroid	Mass of Returned Material (t)	Launch Vehicle	Xe (not including the Earth spiral) (t)	Launch Date	Heliocentric Flight Time (not including the Earth spiral) (years)	Total Flight Time (years)	Arrival *C3*, (km^2/s^2)
1998 KY26	30	Atlas V 521-class	4.9	Sep. 2019	4.7	6.9	2.0
1998 KY26	60	Atlas V 521-class	4.2	May 2018	5.3	7.5	2.0

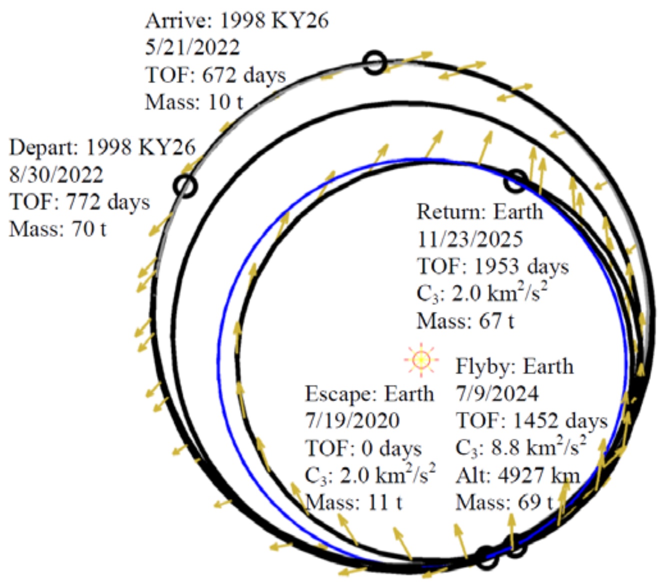

Fig. 5. Heliocentric trajectory for the pick-up-a-rock approach from asteroid 1998 KY26.

IV. Conceptual Flight System

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 and stowed in a 5-m launch vehicle fairing are given in Figs. 6 and 7. The spacecraft cruise 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. For example, solar array wings based on the Ultraflex[21] design are shown in Fig. 6.

The overall spacecraft configuration was determined by the results of the following first order trades:

1. Should the spacecraft push or tow the asteroid back?
2. Should the asteroid SEP transportation vehicle and the asteroid capture vehicle be one spacecraft or two?
3. What range of asteroid dimensions, for a nominal 7-m diameter asteroid, should the flight system be capable of handling?
4. What is the basic capture mechanism concept?

The tradeoffs for pushing or pulling the asteroid, as identified by the COMPASS team are summarized in Table 5. While pulling the asteroid back had several attractive features and the team recommended this option be investigated further, in the end it was believed that pushing the asteroid represented a lower-risk, nearer-term solution.

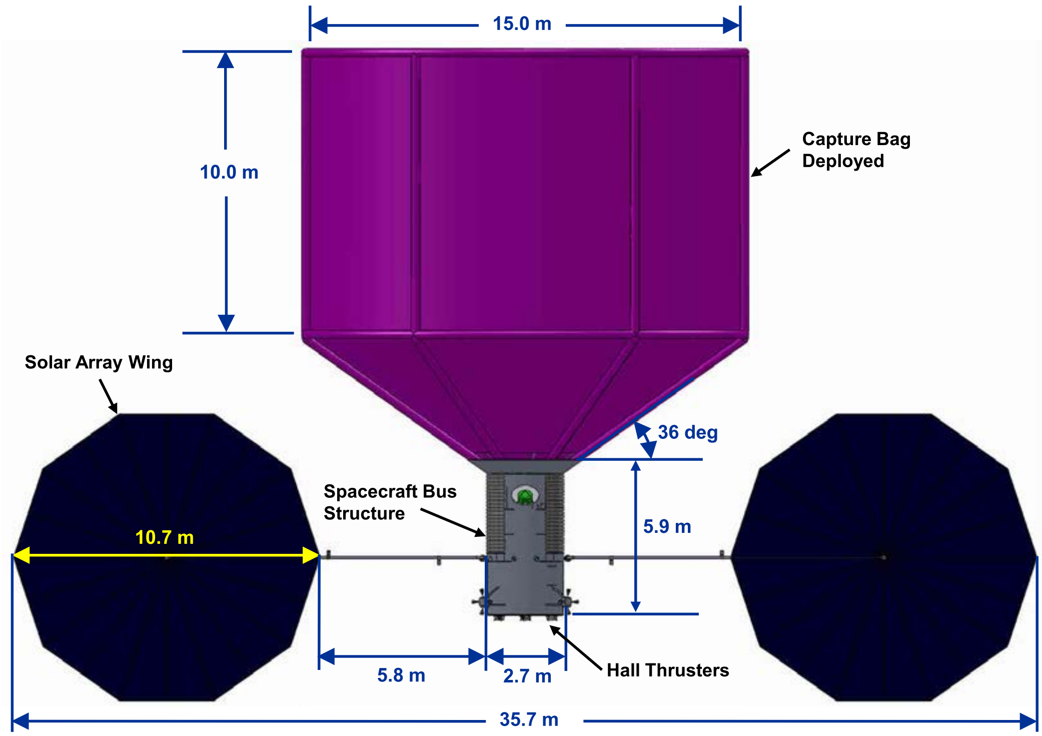

Fig. 6. Conceptual spacecraft in the cruise configuration with the capture mechanism deployed.

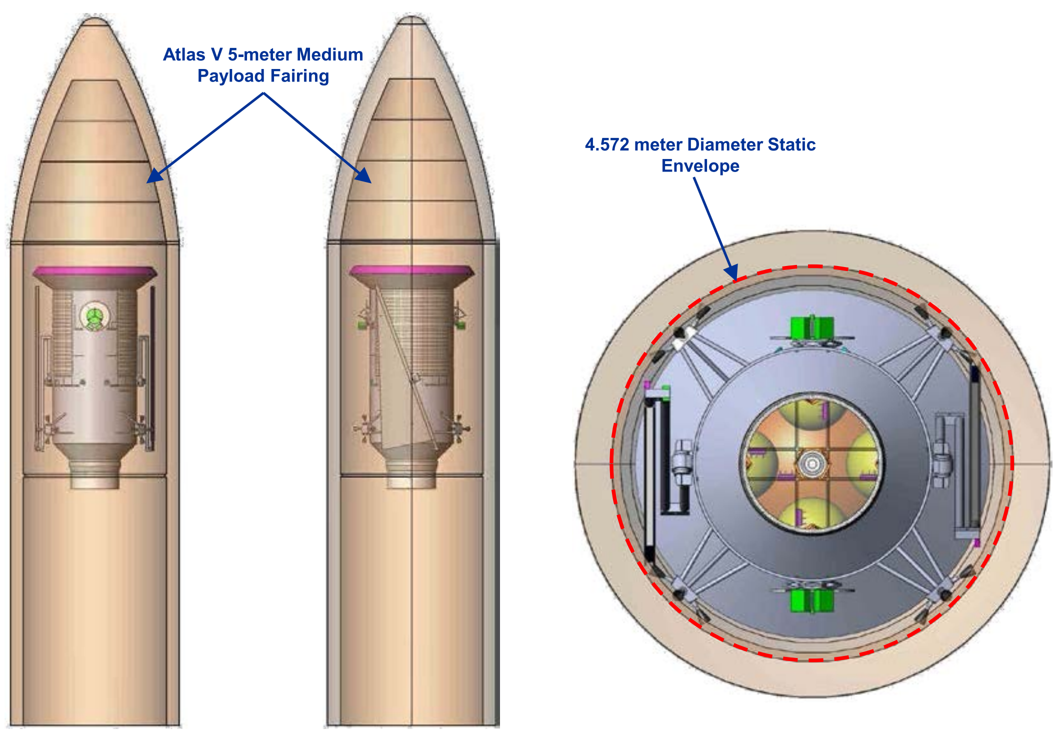

Fig. 7. Conceptual ACR spacecraft in the stowed configuration.

Table 5. Tradeoffs for pushing or pulling the asteroid.

Area	Push		Pull	
	Pros	Cons	Pros	Cons
Mission/System	Pushing things in space well understood	Proximity to Asteroid risk of damaging power/propulsion/other systems	Could avoid close approach of asteroid by S/C (attachment device)	Canting thrusters to avoid tether will cost thrust/isp (~30%)
GN&C	Hard dock –no flexibility to deal with	Need to thrust through center of mass (COM)	No worry of center of gravity	Need to deal with dynamics of the asteroid/tether for entire return, harder to transfer motions
C&DH/ Comm	---	Antenna blockage by asteroid	No blockage of comm. signal	---
Thermal	---	Shadowing by asteroid (cycling)	Less shadowing by asteroid	---
Power	---	Long solar array yokes required to avoid shadowing by the asteroid	No shadowing of the solar arrays by the asteroid	---
Propulsion	---	Need to cant the thrusters through COM, big gimbals	---	Loss of Isp/thrust to avoid sputtering tether, change to ion thrusters?
Mechanical	No tether needed	Stable attachment needed to asteroid	Shock dampened by tether, S/C can be decoupled from asteroid	Destruction of tether by thruster plume, tangle of tether during safing, tether development

The COMPASS team briefly considered the two-spacecraft option where the SEP vehicle provides the transportation and once at the asteroid the capture spacecraft separates from the SEP vehicle captures and detumbles the asteroid and then re-docks the capture spacecraft with the asteroid to the SEP vehicle for the return trip. This approach was rejected under the expectation that it would be significantly more expensive than the single spacecraft configuration.

The capture mechanism was sized to handle non-spherical asteroids in the following way. The first requirement was that it be large enough to handle asteroids with aspect ratios up to 2-to-1. A 7-m diameter asteroid with a 2-m uncertainty could have a projected area of about 63 m^2. The same projected area is given by a 2-to-1 aspect ratio asteroid with dimensions 5.6 m x 11.2 m, which were rounded up to 6 m x 12 m. To provide margin, opening of the capture mechanism shown in Fig. 6 is 15-m in diameter.

The same basic capture mechanism would be used regardless of the mission architecture (get-a-whole-one or pick-up-a-rock). It 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. This capture mechanism concept could accommodate a wide range of uncertainty in the shape and strength of the asteroid. The exterior finish of the capture bag assembly would be designed to passively maintain the surface temperature of the captured asteroid at or below its nominal temperature before capture.

Spacecraft subsystems key to the asteroid retrieval concept are described below.

A. Solar Array

The key considerations for the solar array design would include: the required end-of-life power level; the impact of the slow initial spiral through the Earth's radiation belts; and the impact of the asteroid capture process on the solar array. The power system design would be required to provide 41.2 kW at 120 VDC at EOL. To meet this requirement the COMPASS team selected two 10.7-m diameter Ultraflex solar arrays with 33% efficient, advanced Inverted Metamorphic (IMM) solar cells and 20-mil (0.508 mm) coverglass on front and back sides and the power system architecture shown in Fig. 8. A secondary lithium ion battery would provide 392 W-hr at up to15% DOD, and 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.

The solar array wings were over-sized by 20% as an allocation for radiation degradation by the Earth's radiation belts and by an additional 9% as an allocation for all other degradation effects. This would result in a beginning-of-life (BOL) solar array power level of 53.1 kW. While such a solar array is large, its development could be a straightforward extension of the technology to be developed under NASA's new Solar Array System technology development activity.[22]

To capture the asteroid in the single spacecraft configuration the spacecraft must be able to match the rotation and nutation of the tumbling asteroid. Several options were considered for how to deal with the large solar arrays in order to make the spacecraft nimble enough to enable this capability including: restowing the arrays prior to asteroid capture and redeploying them afterward; folding up just the solar array blanket for capture but leaving the structure fully deployed; and canting the solar arrays in the direction opposite of the capture mechanism. Restowing the arrays entirely would minimize the potential for inadvertent contact with the asteroid, protect them from dust from the asteroid, and maximize the nimbleness of the spacecraft. It would, however provide no power to the spacecraft and the mission-critical asteroid capture would have to be performed entirely on battery power. Mission success would also depend on the ability to redeploy the arrays after asteroid capture. To successfully stow the arrays and then redeploy them after they had been in use in space for about four years was considered to be too risky. This coupled with the resulting lack of solar power during the capture process prompted the search for alternatives.

Folding up just the solar array blankets may be less risky that stowing the blankets and structure and it has many of the same advantages. It also has the same disadvantage of providing no power during the capture process. If multiple attempts at the capture process would be necessary, it would be impractical to stow and deploy the blankets between attempts.

For these reasons it was decided that the least risky approach would be to fold back the wings as indicated in Fig. 9. This approach gets the wings out of the way for the asteroid capture, is expected to improve the nimbleness of the spacecraft (although this was not quantified), and still provides significant power to the flight system. It was estimated that even with a solar array wing off-pointed by 85° from the sun it would still provide at least 1.8 kW, more than enough power to operate the spacecraft. This also highlights why the DC/DC Converter shown in Fig. 8 is necessary. The power system architecture in this figure would enable the DC/DC Converter to have "access" to all of the power produced by each solar array wing.

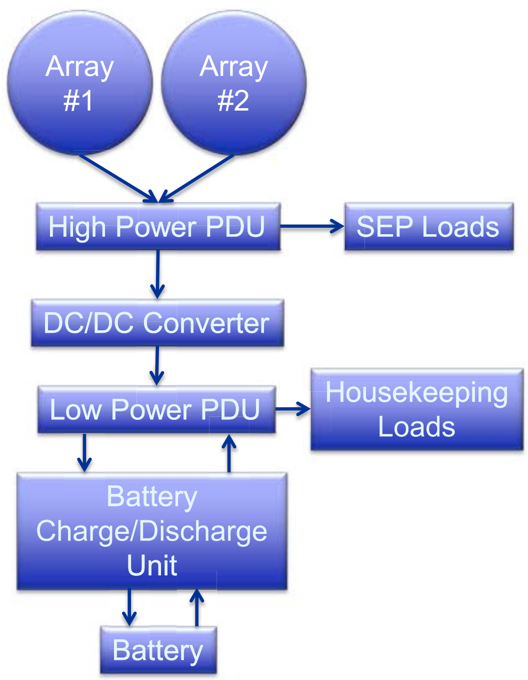

Fig. 8. Conceptual power system architecture for the asteroid retrieval spacecraft.

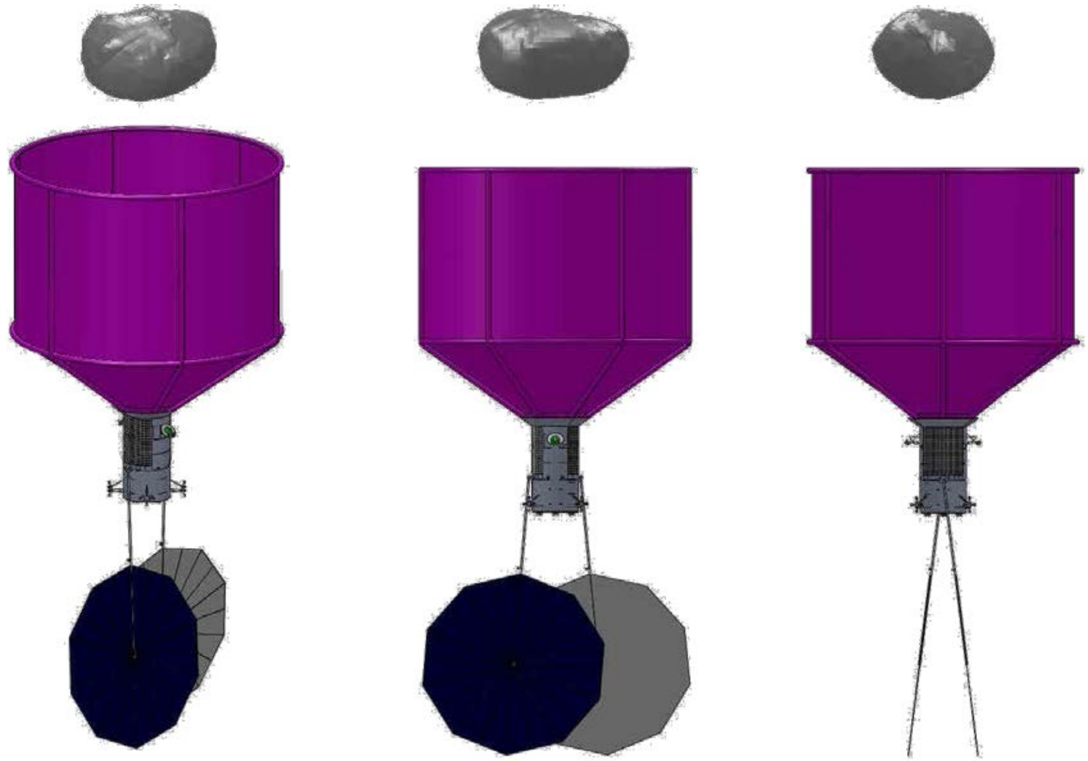

Figure 9. Conceptual spacecraft with solar arrays folded back to facilitate matching the asteroid's spin state during the capture process.

The final driver for the conceptual solar array design was the requirement to strengthened the array to endure an impact loading of up to 2-g during asteroid capture. The maximum g-loading during asteroid capture was very conservatively estimated to be 0.45 g and this was "rounded-up" to 2-g to cover unknowns in the analysis. This requirement had a significant effect on the solar array mass resulting in a BOL specific power of 133 W/kg (current-best-estimate plus mass growth allowance). Follow-on studies should revisit the required g-loading for the solar array.

B. Electric Propulsion (EP) Subsystem

The EP subsystem would include five 10-kW Hall thrusters and Power Processor Units (PPUs), with a maximum of four thruster/PPU strings operated at a time. It would also include a xenon propellant tank, a propellant management assembly, xenon flow controllers, and 2-axis gimbals for each Hall thruster. The EP subsystem would incorporate one spare thruster/gimbal/PPU/xenon flow control string to be single fault tolerant as indicated in Fig. 10. Each thruster would have to operate at a specific impulse of 3,000 s at a PPU input power level of 10 kW. 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.[23] 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.[24] Other Hall thrusters have been designed and tested for operation at power levels of 20 kW and higher.[25,26] 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.[27,28] The asteroid retrieval mission concept requirements for a 10-kW, 3000-s Hall thruster represent a capability that could readily be developed.

The high specific impulse of 3000 s needed for the asteroid retrieval 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 near-term solar array development and so direct-drive was not considered for the asteroid retrieval flight system concept. Consequently, the asteroid retrieval spacecraft 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 can produce the required power level. Therefore, development of a PPU with the required capability should be straight forward. No tradeoffs were performed to look at the effects of direct-drive and lower specific impulses on the initial spacecraft wet mass or trip time. Such tradeoffs should be performed in the future.

The asteroid retrieval mission concept 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%.[29] The xenon propellant tank assumed by the COMPASS team was a single 2.2-m diameter spherical composite overwrapped pressure vessel COPV with a tankage fraction of 5%. An alternative approach would be to consider the use of multiple cylindrical COPVs to potentially reduce the cost of the xenon tank development.

Attitude control during SEP thrusting would be provided by gimbaling the Hall thrusters. This would provide pitch, yaw, and roll control for the spacecraft. Thrusting with the electric propulsion system would be the normal operating mode for the spacecraft, i.e., this is the mode in which the spacecraft would spend the vast majority of its time during the mission. At other times attitude control and spacecraft translation would be provided by a reaction control system (RCS). The Hall thruster-gimbals must have sufficient gimbal-angle capability such that the nominal thrust vector can be pointed through the c.g. of the spacecraft+asteroid system after asteroid capture.

For the return trip to cislunar space it may be necessary to slew the vehicle to obtain a clear line of sight to Earth for communications. This could be accomplished by the RCS, but the required propellant may be excessive due to the mass of the captured asteroid. For this reason the COMPASS team examined performing these slews with the EP subsystem. This analysis indicated that gimballing all four Hall thrusters by 10 degrees could slew the vehicle+asteroid 45 degrees about the minor axis in 6.2 hours, assuming a 7 meter moment arm. It would take about 0.22 kg of propellant to accomplish such a maneuver and return to the original attitude. If the vehicle performs one of these maneuvers per week over the ~313 weeks for the return leg of the mission, then ~138 kg of propellant would be required. This, however, is very conservative and it is not expected that the spacecraft will have an obstructed line of sight to Earth each week. If this maneuver is needed only 10% of the time, the resulting propellant requirement would be reduced a negligible 14 kg.

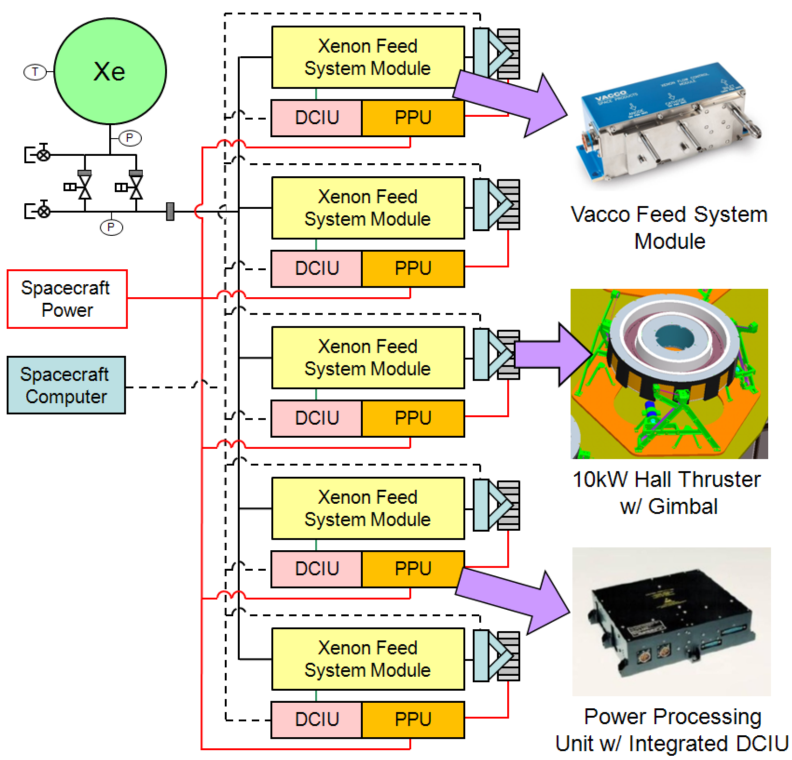

Fig. 10. The conceptual EP subsystem architecture include five xenon feed module/PPU/thruster-gimbal/thruster strings. Only four strings are used at a time. The fifth string makes the subsystem single fault tolerant.

C. RCS

The sizing function for the RCS would be the requirement to de-tumble the asteroid after capture. To estimate the propellant required for this function the following assumptions were made. The asteroid was assumed to have a mass of 1,100 t, a cylindrical shape of 6-m x 12-m, and to be rotating about its major axis at 1 revolution per minute (RPM). This rotation rate is expected to be very conservative, and most asteroids would likely have rotation rates significantly slower than this. The RCS thrusters were assumed to have an *Isp* of 287 s, a thrust of 222.4 N, and a moment arm of 2 m.

With these assumptions it was calculate that it would take about 33 minutes of continuous firing to de-tumble the asteroid requiring about 306 kg of propellant. This propellant load was increased by 50% to 459 kg to account for uncertainties in the analysis. Adding to this other RCS functions, margin, and residuals would bring the total RCS propellant load to 877 kg.

The RCS concept would be a single fault tolerant, hypergolic bipropellant subsystem using monomethylhydrazine (MMH) and nitrogen tetroxide (NTO) with a gaseous nitrogen pressurization system. It includes four pods of four thrusters. A preliminary schematic of the RCS concept design is shown in Fig. 11. The RSC could store up to 900 kg of propellant.

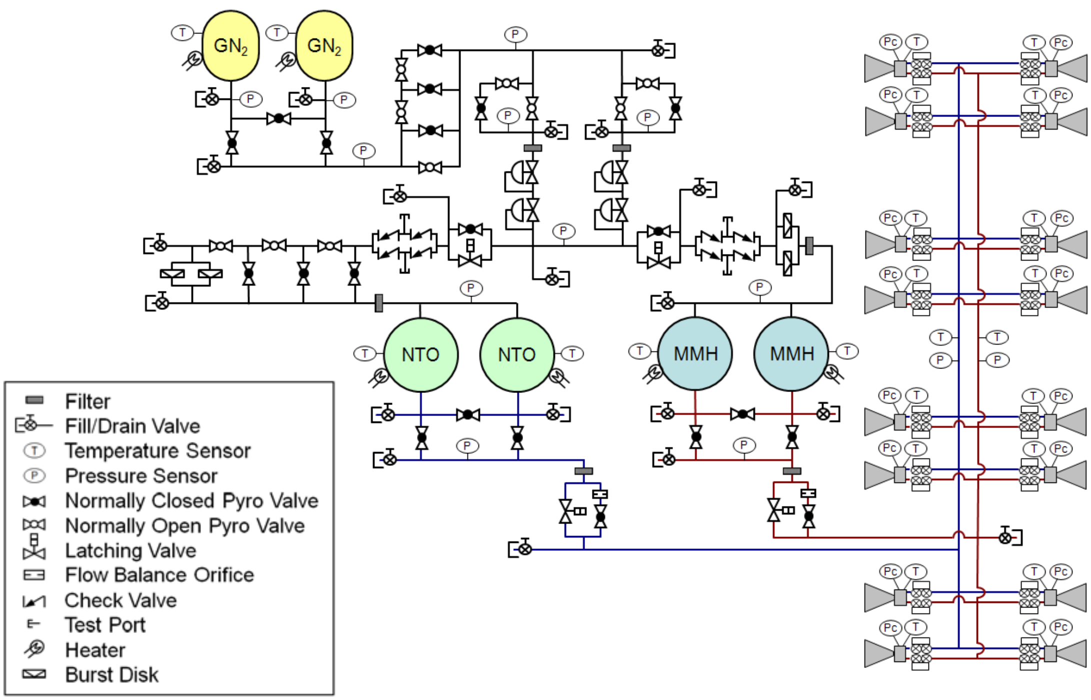

Fig. 11. Conceptual design of the Reaction Control Subsystem (RCS).

D. Capture Mechanism

The COMPASS team briefly discussed different capture mechanism concepts before adopting the concept selected by the KISS team. This capture mechanism 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 hold open the high-strength bag. This capture mechanism concept was selected because it would accommodate a wide range of uncertainty in the shape and strength of the asteroid and would completely contain any loose debris from the asteroid. Initially the high-strength bag would line the inflated, rigidized structure. After the bag is cinched closed around the asteroid and the spacecraft is pulled up tight against the asteroid, the rigidized structure would retain its original cylindrical shape. Once the asteroid is returned to cislunar space the rigidized structure could form the basis of an enclosure for working on the asteroid. For example, astronaut crews could bring up a circular end cap that would fit over the open end of the rigidized structure completely enclosing the captured asteroid that is still contained within the high-strength capture bag. Ports in the end cap could allow astronauts access to the asteroid for inspection, sampling, and dissection while containing any loose debris from the asteroid as the high-strength bag is partially removed.

E. Master Equipment List (MEL)

The overall flight system MEL developed by the COMPASS team is given in Table 6. This mass estimate indicates that the flight system is consistent with the launch capability of an Atlas V 551-class vehicle to low-Earth orbit. The COMPASS team estimated the cost for the first asteroid retrieval mission at approximately $2.6B.[16]

Table 6. MEL for conceptual asteroid retrieval flight system.

Subsystem	CBE Mass (kg)
Payload -- Instruments and Capture Mechanism	339
Avionics	60.9
Communications	61.8
Attitude Control	20.5
Power	928.8
Thermal	315.6
Structures and Mechanisms	525.1
Electric Propulsion	739.3
RCS	167.4
Total Dry Mass CBE	**3158**
Dry Mass with 30% Margin	**4106**
Total Xenon with Margin	10958
Total Biprop. with Margin	876.6
RCS Pressurant	34.3
Initial Wet Mass	**15975**

V. Conclusion

Rendezvousing with, capturing, and subsequently transporting an entire near-Earth asteroid, with mass is of order 1,000 metric tons, and placing it in cislunar space is not a trivial in-space propulsion task. To do this with a conventional space-storable bi-propellant system would require launching to low-Earth orbit an initial mass of 361 metric tons. Even a LO2/LH2 system with zero-boil-off technology would require an initial mass in low-Earth orbit of 225 t. Both of these approaches would require multiple heavy lift launches and on-orbit assembly. In contract, a 40-kW end-of-life, SEP system that could be launched on a single Atlas V 551-class launch vehicle would be capable of returning a 1,000 t asteroid in a total flight time of less than 10 years (based on the orbital characteristics of asteroid 2008 HU4 as a proof-of-concept example). The enormous reduction in initial mass in low-Earth orbit enabled by SEP would make an asteroid retrieval mission affordable for the first time in history. This has the potential to jump-start the in situ resource utilization industry.

The conceptual flight system developed by the GRC COMPASS team addressed the key issues associated with an asteroid capture and return mission, and demonstrated the basic feasibility of the concept. Due to the immaturity of the concept, conservative approximations were made to size many of the subsystems. Future work should be performed to refine the subsystem designs and improve the concept maturity.

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