AAS 13-409

TRAJECTORIES TO NAB A NEA (NEAR-EARTH ASTEROID)

Damon Landau,[*] John Dankanich,[†] Nathan Strange,[‡] Julie Bellerose,[§] Pedro Llanos,[**] and Marco Tantardini[††]

In 2010 and 2011 NASA and Keck Institute for Space Studies sponsored investigations into the feasibility of identifying, capturing, and returning an entire (albeit small) NEA to the vicinity of Earth, and concluded that a 40-kW solar electric propulsion system launched on an Atlas 551 provided sufficient propulsion to control an asteroid's trajectory. Once secured by the spacecraft, a NEA with a naturally close encounter with Earth is nudged over a few years to target a lunar gravity assist, capturing the object into Earth orbit. With further use of solar perturbations, up to 3,600,000 kg of NEA could be placed in high-lunar orbit.

INTRODUCTION

In 2010 NASA sponsored a study to investigate the feasibility of identifying, capturing, and returning an entire Near-Earth Asteroid (NEA) to the International Space Station.[1] This study identified no technological showstoppers, and found that a solar electric propulsion system with reasonable power (40 kW) could return up to 10,000 kg from a near-Earth orbit to the space station. With these promising results the Keck Institute for Space Studies sponsored a workshop to flesh out the concept, and found that the returned mass could be improved a hundredfold by keeping the returned NEA in high-Earth orbit instead of spiraling it down to space station orbit.[2,3] Once captured, the object could be pulled by the Sun and Moon into Earth-Moon L_2, where it could provide a target for astronauts to test space operations in preparation for missions to larger NEAs, providing a crucial step to enable human exploration beyond low-Earth orbit.[4]

The goal of the Keck study was to assess the feasibility of the asteroid retrieval concept over the course of a week-long workshop. The resulting trajectory design connected the disparate phases of Earth escape spiral, low-thrust interplanetary transfer, capture via lunar gravity assist, and solar-perturbed trans-lunar approach to a storage orbit. The preliminary design techniques used to estimate trajectory performance in "real time" during the study are discussed, with particular emphasis on how to maximize the return mass with constrained SEP power and launch mass.

[*] Outer Planet Mission Analyst, Mission Design and Navigation Section, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Dr., M/S 301-121, Pasadena, CA 91109.
[†] Mission Analyst, NASA Programs, AeroDank, Inc., 21000 Brookpark Road, M/S 142-5, Cleveland, OH, 44135.
[‡] Systems Engineer, Mission Concepts Section, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Dr., M/S 301-170S, Pasadena, CA 91109.
[§] Research Faculty, Carnegie Mellon University / NASA ARC, MS202-3, Building 202, Room 101, Moffett Field 94035.
[**] Marie Curie Postdoctoral Research Fellow, Flight Mechanics Group, GMV, Isaac Newton 11, Tres Cantos, 28760 Madrid, Spain.
[††] Visiting Fellow, The Planetary Society, 85 South Grand, Pasadena, CA 91105.

MISSION OVERVIEW

The asteroid retrieval mission is designed to be consistent with currently available technologies. For example, a 40 kW SEP system with 10 kW thrusters is roughly double the level currently flown on satellites. Further, the launch mass of the spacecraft is limited to the capability of Atlas launch vehicles. The initial study[1] indicated that the propellant requirements would exceed this capability for direct launches, thus the mass to Earth escape is increased by first launching to Earth orbit then spiraling up to a lunar flyby for escape. Similarly, lunar-assisted capture is particularly effective at increasing the asteroid mass that can be returned to Earth orbit. Instead of decreasing the energy with respect to Earth with the SEP system, the Moon can be used to transfer the asteroid from a hyperbolic flyby trajectory to a loosely captured orbit at Earth. Because the SEP system does less work, the required trajectory acceleration decreases, permitting larger asteroid return masses for a given thrust level. It was found that masses on the order of several hundred tons could be moved to lunar intercept. As shown in Table 1 this mass corresponds to asteroids that are about 7 m across, which is near the limit of objects currently being detected. An overview of the asteroid retrieval mission sequence is depicted in Figure 1 and the mission design parameters are listed in Table 2.

Table 1 Expected asteroid mass (assuming spherical objects) is affected by size and density uncertainty (Reference 2).

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

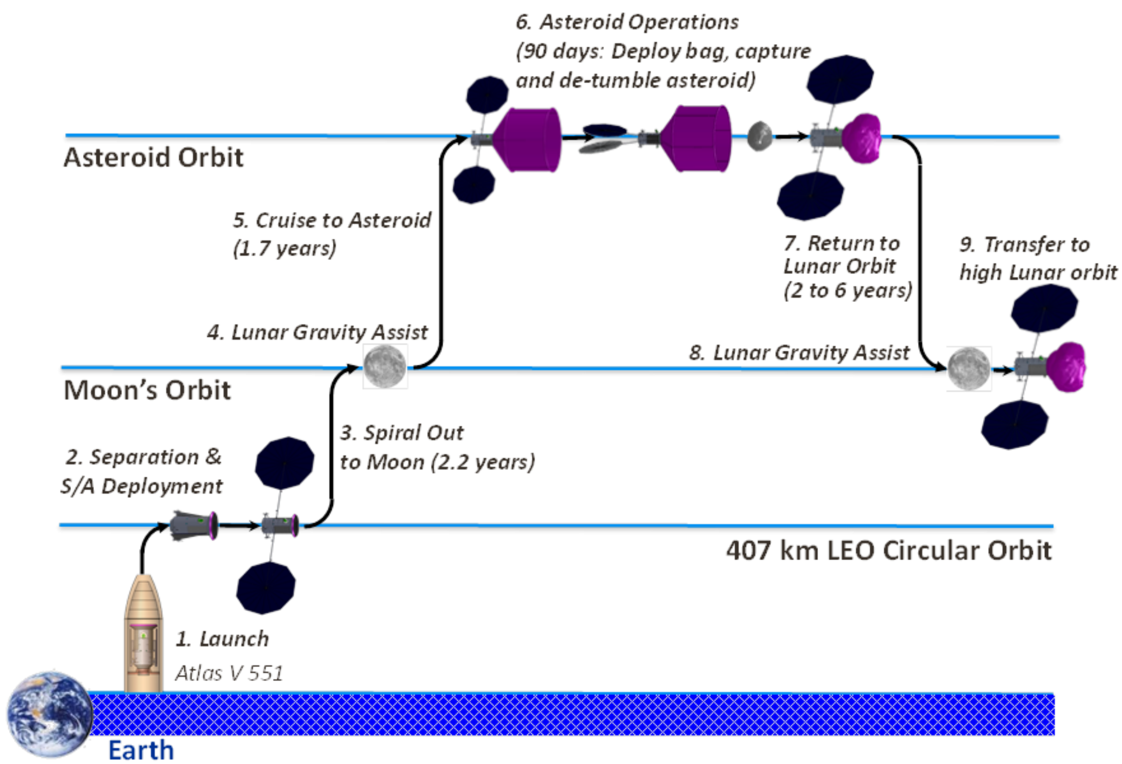

Figure 1 Asteroid retrieval sequence involves Earth spiral, interplanetary transfer, lunar gravity assist and trans-lunar transfers (Reference 2).

Table 2 Asteroid retrieval design parameters.

Parameter	Value	Notes
SEP Power	40 kW	
I_{sp}, efficiency	3000 s, 60 %	Hall thrusters, 1.6 N
s/c dry mass	~5.5 t	Reference 1
Atlas V (521) escape mass	10.7 t	13.5 t to LEO
Atlas V (521) spiral time, Xe	1.6 yr, 2.8 t	without shadowing
Atlas V (551) escape mass	15.0 t	18.8 t to LEO
Atlas V (551) spiral time, Xe	2.2 yr, 3.8 t	without shadowing
Spiral ΔV	6.6 km/s	LEO-Intersect Moon
Escape/Capture C_3	2 $(km/s)^2$	Lunar assisted
NEA stay time	90 days	

METHODOLOGY

Interplanetary Trajectories

The design process begins by identifying asteroids that have naturally Earth-like orbits. If a given object has a close approach to Earth (of say < 0.2 AU) at a relatively low relative velocity (of say < 2 km/s), then the close approach date is used as an initial guess for the Earth return date. The NEA-to-Earth return leg is then optimized for maximum return mass for the given SEP system parameters and an unbounded NEA departure mass. Figure 2 provides an example of the maximum return mass for the asteroid 2008 HU4 (which has a close approach of 0.15 AU at 1.3 km/s in 2026) over a range of flight times at 40-kW power and 3,000 s specific impulse. It is notable that the return ΔV can be approximated assuming a linear B-plane by dividing the close approach distance by the time to intercept. This approximation can be used to parameterize the maximum return mass (assuming constant thrust) by close approach distance (Figure 3) or energy (Figure 4), assuming the Moon can capture objects with C_3 up to 2 km^2/s^2. It is noteworthy that very large return masses of 1,000 t can be returned from objects that approach up to 80 lunar distances (0.2 AU) away, but the approach energy cannot increase much above the lunar capture limit. Objects with distant approaches become more retrievable at longer flight times because the ΔV decreases with flight time, while the impulse generated by the SEP system continues to increase.

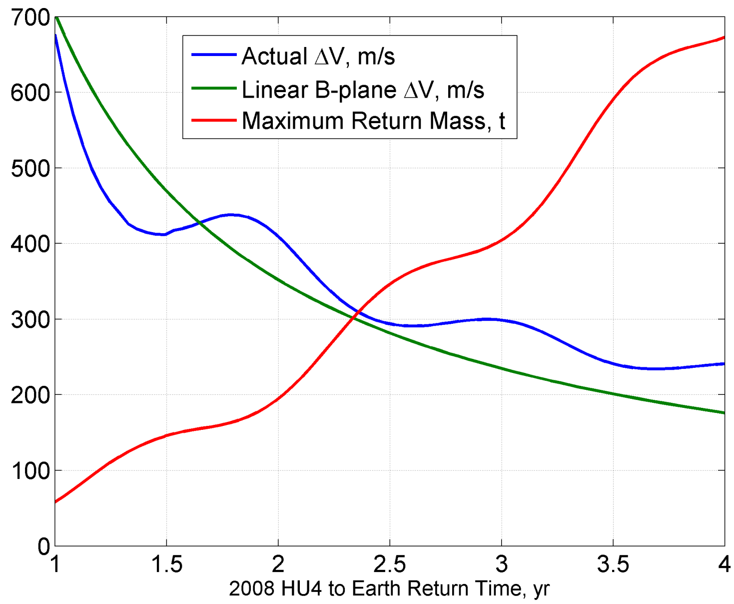

Figure 2 Several hundred tons can be returned with flight times of a few years.

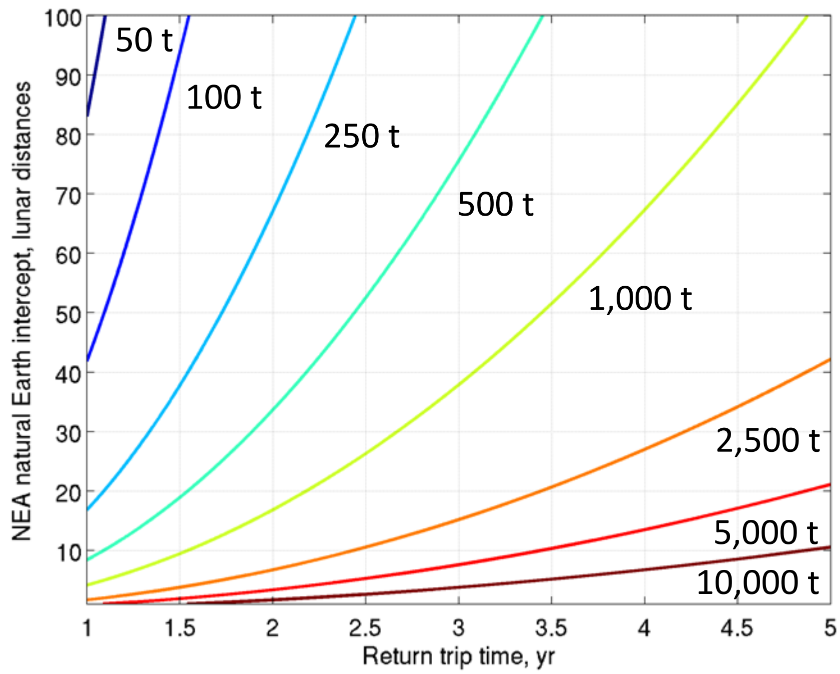

Figure 3 Possible NEA retrieval mass for objects with a natural Earth close approach $C_3 < 2$ km^2/s^2. (Assuming 40 kW power, 3,000 s I_{sp}, and linear B-plane approximations.)

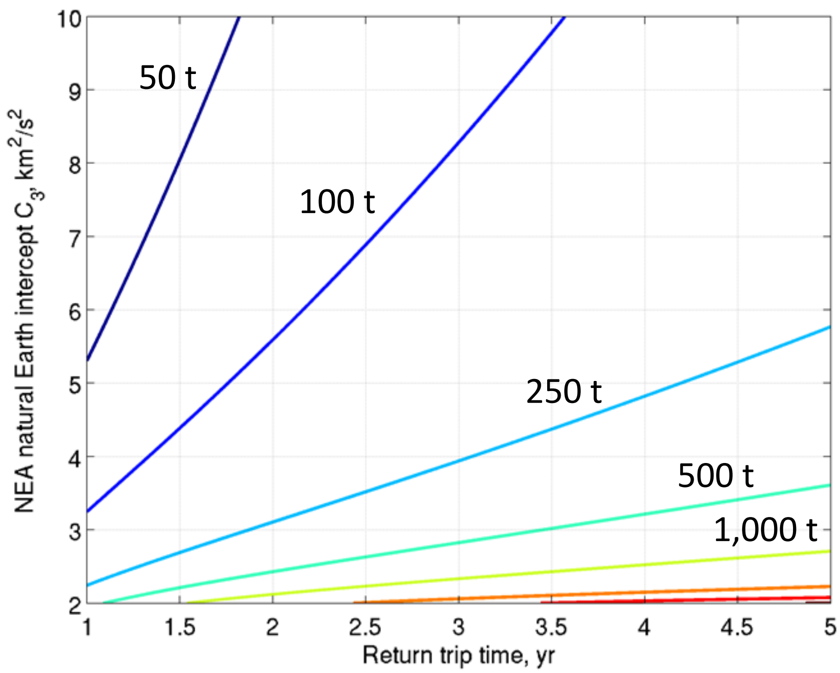

Figure 4 Possible NEA retrieval mass for objects with natural Earth close approach < 10 lunar distances. (Assuming 40 kW power, 3,000 s I_{sp}, and linear B-plane.)

Cislunar and low-energy transfers

The notional return scenario is to use a lunar gravity assist to decrease the Earth-relative energy of the return mass from a positive $C_3 < 2\ km^2/s^2$ to a C_3 of around $-0.5\ km^2/s^2$. [5] As shown in Figure 5 a C_3 of $2\ km^2/s^2$ permits asteroid capture from declinations ranging up to 30 degrees from the ecliptic. Capture from higher energies are achievable in the special case of in-plane arrivals and favorable lunar phase.[6] Following lunar flyby the spacecraft transfers to the vicinity of a Sun-Earth L_1 or L_2 point to reduce the energy with respect to the Moon, so that it can then be transferred to a Moon-Earth L_1 or L_2 orbit. The capture sequence thus uses the Moon to reduce energy with respect to Earth, then uses the Sun to reduce energy with respect to the Moon. A representative trajectory that captures an asteroid from a C_3 of around $2\ km^2/s^2$ to high lunar orbit is shown in Figure 6.

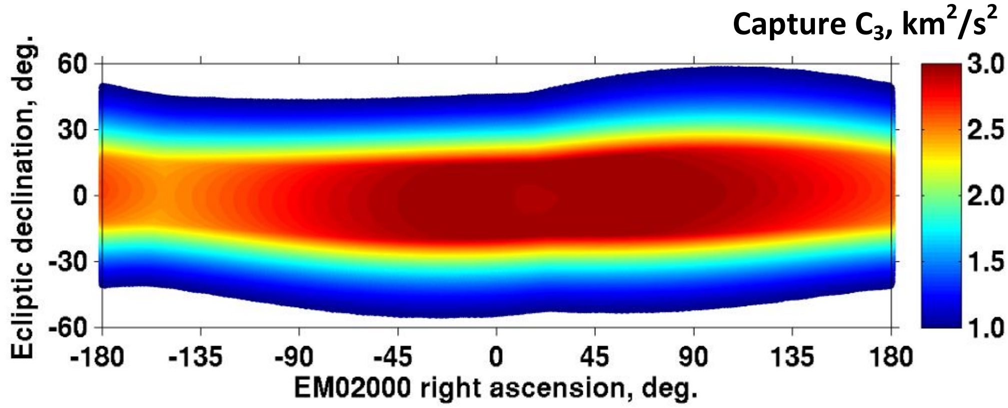

Figure 5 A lunar flyby can capture an asteroid with a C_3 of $2\ km^2/s^2$ into Earth orbit from a range of directions.

Once in the vicinity of the Moon there are different options for where to place the asteroid depending on the desired lifetime of the spacecraft. The asteroid could be transferred to Earth-Moon L_1 or L_2 with minimal ΔV, but the unstable dynamics in those regions require frequent station keeping maneuvers. While the yearly ΔV for station keeping is low, the frequency of maneuvers places additional requirements on the design lifetime of the spacecraft. Alternatively, the spacecraft can spiral down to a high lunar orbit, where the end-of mission scenario would allow the returned mass to ultimately crash into the moon. The transfer to loose lunar orbit requires around 50 m/s ΔV which translates to an additional 1.5 t of propellant and a year of thrusting following capture of a 1,000 t object. The lifetime of this orbit is on the order of a year, which significantly decreases the demand for station keeping. If the asteroid is required to remain in orbit passively, then it could be placed in a distant retrograde orbit (DRO) around the moon for approximately 100 m/s additional ΔV. This option requires two extra years of thrusting before reaching the storage orbit, but allows the asteroid to remain uncontrolled in the vicinity of the Moon for several decades.

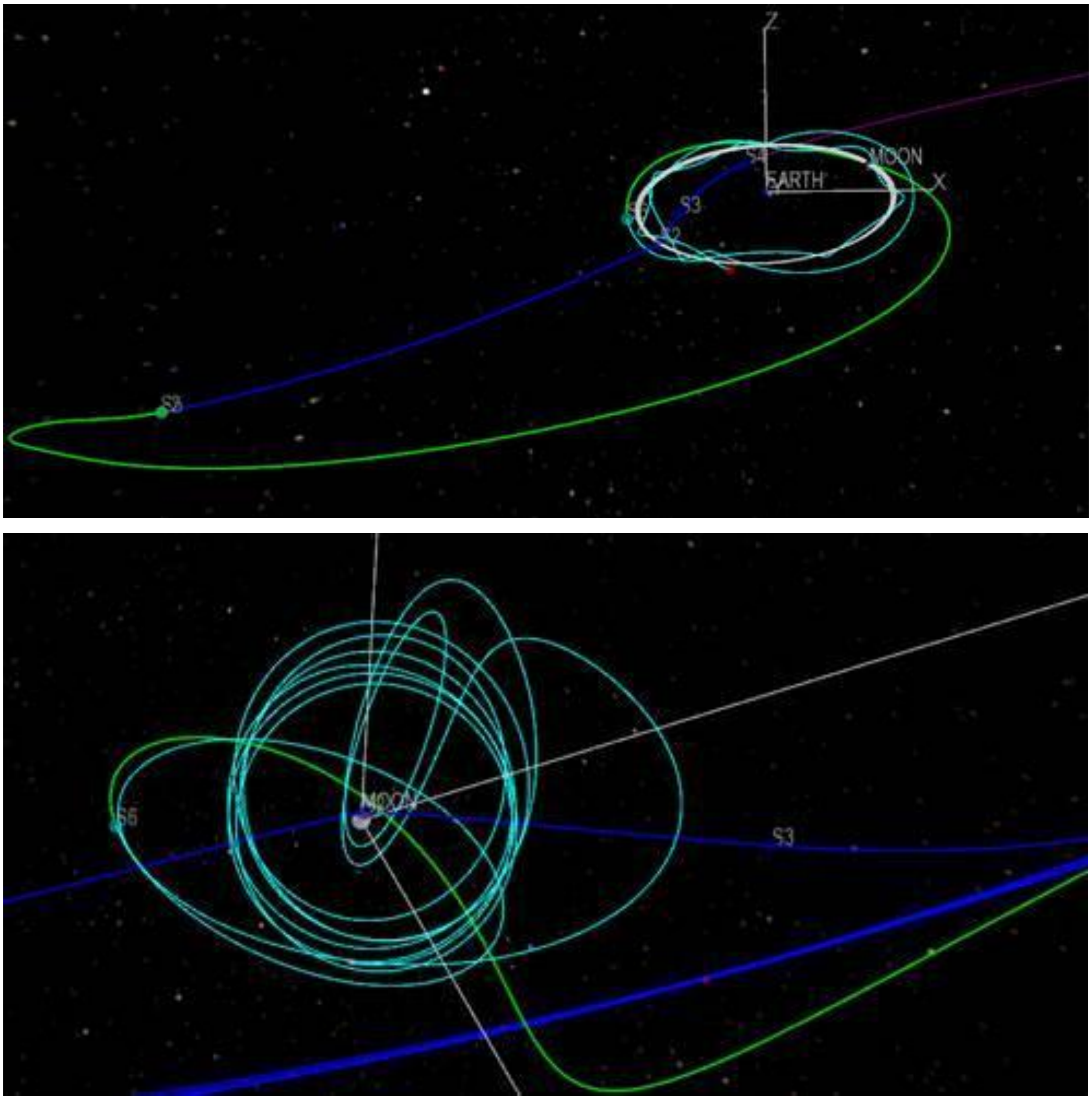

Figure 6 Solar perturbations can cause the asteroid to capture into orbit at the Moon.

Direct transfers to Sun-Earth L_2, without an intermediate lunar gravity assist, have also been examined. This option connects the low-thrust interplanetary trajectories to a stable manifold that asymptotically approaches L_2. The first step is to generate a table of state vectors that define the manifold. Then the state (position and velocity) of the target over the time span of interest is called from an ephemeris and rotated into the same frame as the manifold data. A particularly useful frame is Earth-centered radial-tangential-normal (RTN), where the radial component is Earth's position with respect to the Sun and the normal component is Earth's orbital angular momentum, because the manifolds are independent of the reference epoch in this frame (i.e. they don't significantly vary over Earth's orbit around the Sun). A heuristic cost function may now be calculated by taking the difference in position between the NEA and the manifold and dividing it by an assumed transfer time to estimate the ΔV for intercept, then an additional ΔV is required to match velocity and place the NEA on the manifold. This process provides a quick comparison to the lunar-targeting estimate of Figure 2. The resulting three-dimensional cost function can be parameterized by 1) the absolute time along the NEAs orbit, 2) the relative time from L_2 on the manifold, and 3) the arrival position along the L_2 orbit. This heuristic function is plotted for NEAs 2008 HU4 and 2000 SG344 in Figure 7 and Figure 8 where each line arrives at a different position in the L_2 orbit, and the left figure is a minimization across all arrival times on the manifold (providing a minimum for each date along the orbit) and the right figure is a minimization across all encounter dates (providing a minimum for each time along the manifold). We note that the minimum ΔV to place a NEA on the manifold is similar to the ΔV to target the Moon. However, for a fixed flight time and power level, the returned mass is generally less to Sun-Earth L_2 because a higher acceleration is necessary to match velocities near the manifold.[7,8,9] In contrast, the lunar-capture approach allows the velocity to be free (within a magnitude bound that is naturally low) at lunar intercept, thus lowering the average acceleration that enables capture.

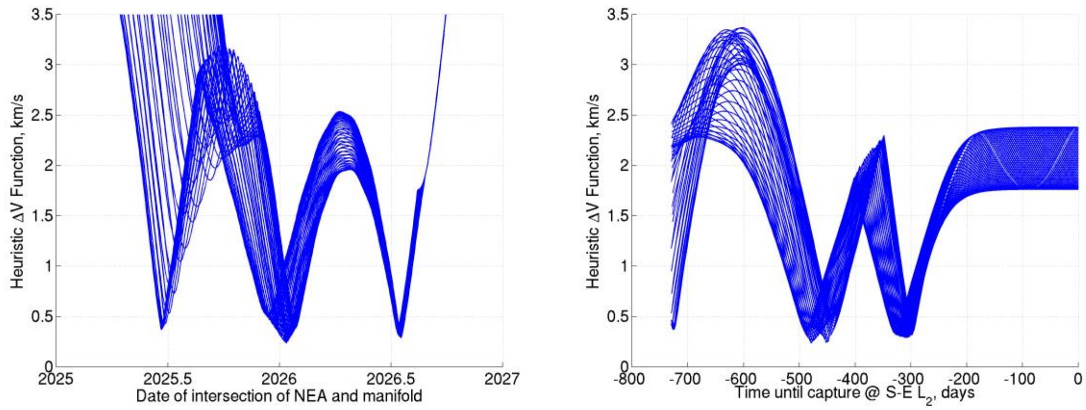

Figure 7. Approximate ΔV to place 2008 HU4 on a stable Sun-Earth L_2 manifold.

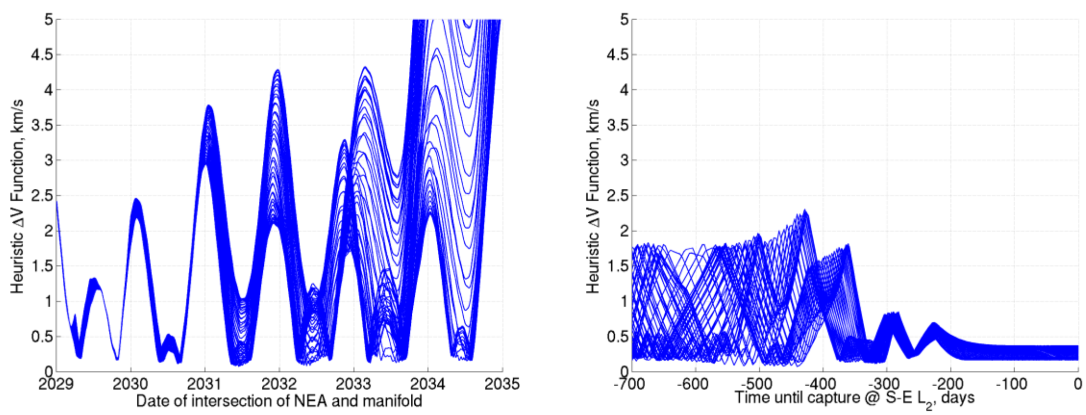

Figure 8. Approximate ΔV to place 2000 SG344 on a stable Sun-Earth L_2 manifold.

RESULTS

Two mission scenarios are considered: 1) return an entire asteroid of possibly unknown type, and 2) return a sizable portion of a well-characterized target. Because there are many uncharacterized NEAs, it is possible to find a few small objects with orbits similar enough to Earth's to return large (>1,000 t) payloads. From this set 2008 HU4 was chosen as a baseline target because it also has an opportunity for observation in 2016. It has an absolute magnitude of 28.2, placing its size in the 5–12 m range. (For most NEAs only the absolute magnitude is known, thus the diameter is estimated from an assumed albedo range.) An example trajectory returning 1,300 t to Earth is presented in Figure 9. Generally the round trip interplanetary missions take 1–2 years to reach the target, then several years to guide the asteroid to lunar intercept.

Alternatively, if a target of a particular type is desired, then 1998 KY26 is known to be carbonaceous, but requires more ΔV to return a sample. This object has a natural close approach of only twelve lunar distances, but approaches with a C_3 of around 13 km^2/s^2. Because typed asteroids are rare, it is more difficult to find a potential target that permits large return masses. For 1998 KY26 "only" 60 t could be returned (as in Figure 10). In this case an Earth-Earth leveraging transfer was added to the end to more efficiently reduce the C_3 for capture.

There is also the option of returning a big chunk from a larger uncharacterized target. The asteroid 2000 SG344 has an orbit very similar to Earth's and can provide extreme return masses as shown in Figure 11. In this case, the spacecraft does not have to travel far from Earth to rendezvous with the asteroid, which permits a moderately short flight time. In fact, the optimal departure from the asteroid occurs when it makes a close approach in 2028, and departing earlier (and farther from Earth) reduces the return mass.

As demonstrated in the first five rows of Table 3, additional flight time can permit increasingly larger return masses. However, the return date is fixed to when the NEA naturally has a close encounter to Earth, so the additional flight time comes at the expense of earlier launch dates. Also, larger return mass typically entails additional propellant, which increases the wet mass of the spacecraft and requires larger launch vehicles. The difference between rows five and six (1998 KY26) is the addition of an Earth gravity assist in row six to leverage down the naturally high encounter velocity of 1998 KY26. The return mass remains moderate even at a higher C_3 of 4 km^2/s^2 (row seven). When the encounter C_3 is naturally high, the return mass is relatively insensitive to the lunar capture C_3 constraint.

On the other hand, as expected from Figure 4 the return mass becomes very sensitive when the approach C_3 is near the limit (2 km^2/s^2) and the return mass is high. In this case, the retrieval ΔV is extremely low, and decreasing the C_3 limit can easily double the return ΔV. Comparing rows eight and nine of Table 3, the addition of 0.1 km^2/s^2 doubles the return mass. The propellant requirement for the 2000 SG344 missions are well below the launch vehicle capability of the Atlas 521, thus the spiral time for these missions could be reduced from 1.6 yr to 0.4 yr by launching to a higher orbit.

There is also a possible trade between launch vehicle capability and specific impulse. As the specific impulse decreases the available thrust increases (at fixed power) and larger asteroids can be returned. However, the propellant mass also increases and the maximum return mass becomes bounded by the amount of propellant that can be launched with the spacecraft. Thus there is an optimal balance between high thrust to move large objects and high specific impulse to maintain a reasonable propellant mass. For example in the first row of Table 4, the maximum return mass decreases at the lower Isp of 2000 s compared to 3000 s when the propellant mass is constrained by the Atlas 521 launch vehicle. The return mass increases dramatically with the larger Atlas 551 vehicle because sufficient propellant can be launched to take advantage of the higher thrust to move more mass. In this case, the 2000 s specific impulse design (row 4) returns 50 t more than the 3000 s solution (row 5). As the specific impulse increases, the return mass is no longer constrained by launch vehicle capability, but instead becomes bounded by the available thrust on the return leg. Thus the return mass with 4000 s specific impulse is the same whether the Atlas 521 (row 3) or Atlas 551 (row 6) vehicle is used.

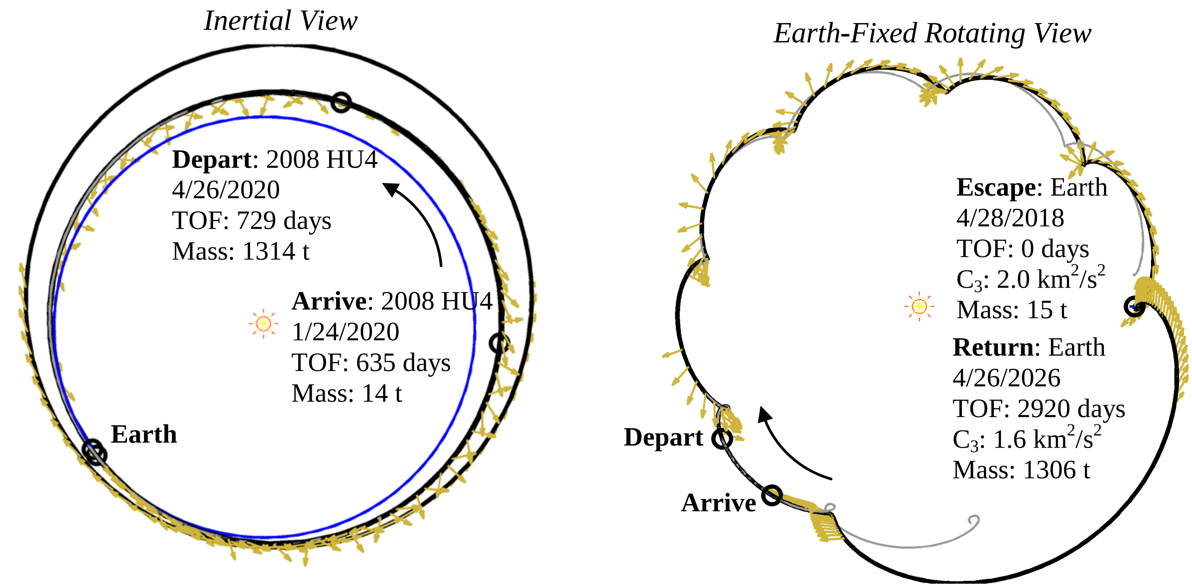

Figure 9. Example mission returning 1300 t of a small (~7 m) NEA with a radar opportunity in 2016.

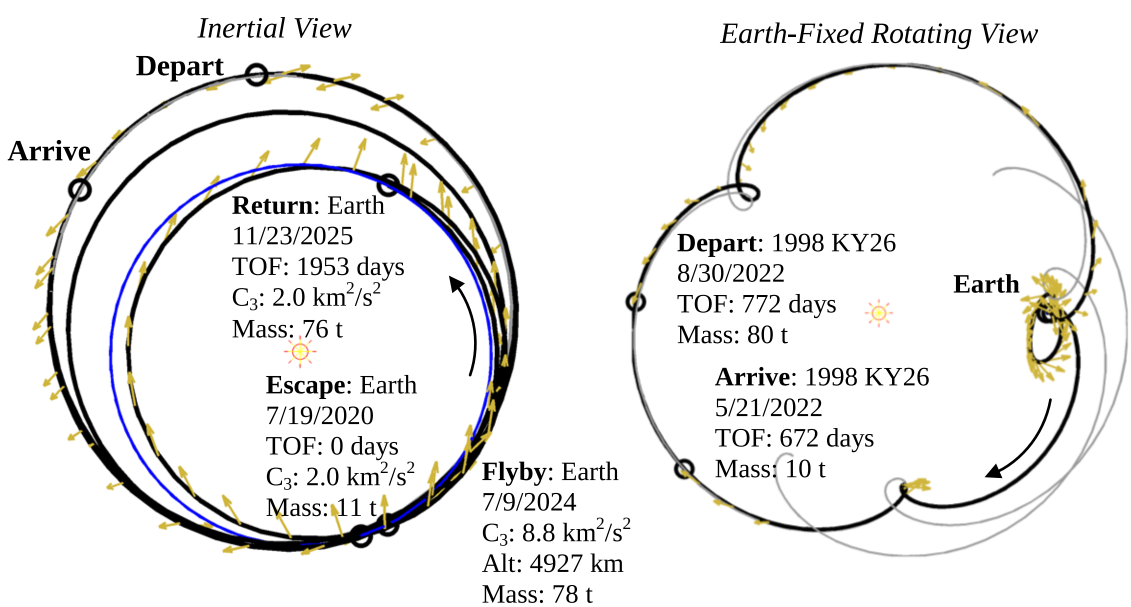

Figure 10. Example mission returning 60 t of a well-characterized carbonaceous NEA.

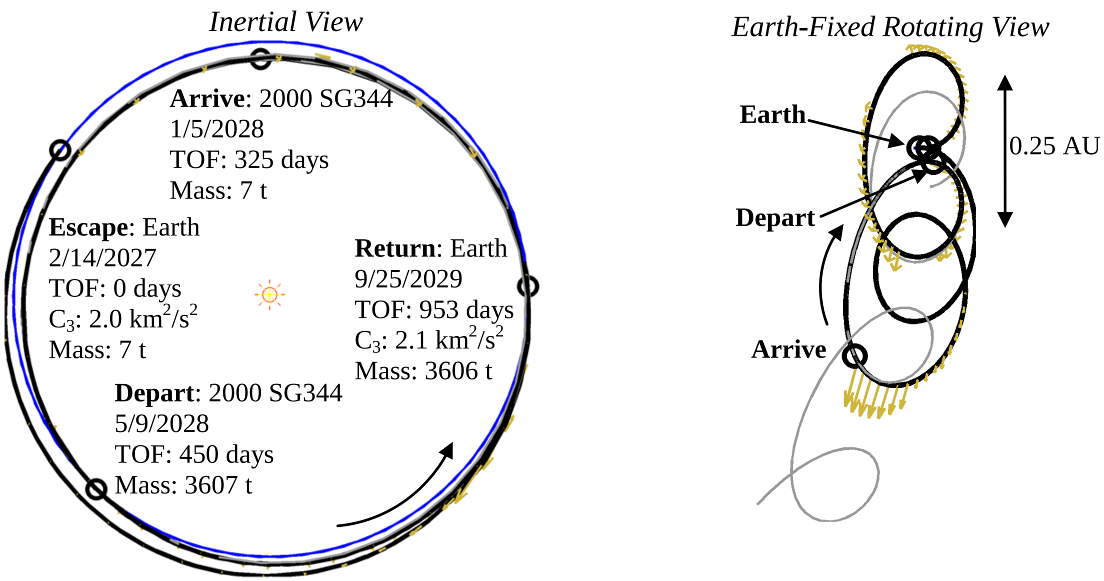

Figure 11 Example mission returning 3,600 t of a NEA with an Earth-like orbit.

Table 3 Example interplanetary (Earth escape to Earth capture) trajectories.

Designation	Returned Mass, t	Propellant, t (no spiral)	Earth Escape	Flight time, yr (no spiral)	Arrival C_3, km^2/s^2
2008 HU4	400	5.2	4/27/2021	5.0	1.7
2008 HU4	600	5.6	4/27/2020	6.0	1.6
2008 HU4	950	8.9[a]	4/28/2019	7.0	1.6
2008 HU4	1300	9.1[a]	4/28/2018	8.0	1.6
1998 KY26	30	4.9	11/11/2019	4.7	2.0
1998 KY26	70	4.8	7/19/2020	5.3	2.0
1998 KY26	110	5.2	6/25/2020	5.4	4.0
2000 SG344	1800	1.8	3/8/2027	2.6	2.0
2000 SG344	3600	1.7	2/14/2027	2.6	2.1

[a]Requires Atlas V (551) launch vehicle. All others assume an Atlas V (521) launch.

Table 4 The maximum return mass varies with specific impulse and launch vehicle.

Designation	Returned Mass, t	Xe, t (no spiral)	Specific Impulse, s	Launch Vehicle
2008 HU4[a]	450	5.7	2000	521
2008 HU4	600	5.6	3000	521
2008 HU4	550	4.8	4000	521
2008 HU4	700	9.4	2000	551
2008 HU4	650	5.9	3000	551
2008 HU4	550	4.8	4000	551

[a]April 2020 launch, 6 year interplanetary transfer.

CONCLUSIONS

The idea to capture and return an entire celestial 0bject has become possible with the advent of the technology to develop powerful solar electric propulsion systems and the ability to discover and characterize small near-Earth asteroids. A 40-kW SEP system operating at a few thousand seconds specific impulse provides sufficient thrust to alter the course of a one-million-kg object to intercept the Moon. Flight times of several years are required to sufficiently alter the course of objects that naturally approach within a few dozen lunar distances and have an encounter C_3 less than about 2 km^2/s^2. These transfers expend several tons of propellant, which can exceed the launch vehicle capability for direct launches to escape. However, provided sufficient lead time, ample propellant can be launched to Earth orbit, then spiraled out to lunar-assisted escape to meet the propulsion demand to move an asteroid. Once captured and slowly nudged onto a lunar intercept trajectory, the object can eventually be placed in orbit near the Moon following a series of lunar gravity assists and low-energy transfers. There are several options for the ultimate storage location of a small asteroid where longer orbit lifetimes are generally achieved with additional propellant. Placing this new asset in a storage location that is also accessible to astronauts in the next decade would provide a tantalizing stepping stone on the path to deep space exploration.

ACKNOWLEDGEMENTS

The research described in this paper was sponsored by the Keck Institute for Space Studies (KISS) and was carried out in part at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The authors are grateful to John Brophy, Lou Friedman, and Fred Culick for leading the Asteroid Retrieval Feasibility Study and to Dan Mazanek for his contributions to the mission design effort. We also thank Laura Burke for her design of the transfer trajectory to high lunar orbit.

REFERENCES

[1] Brophy, J. R., Gershman, R., Landau, D., Polk, J., Porter, C., Yeomans, D. et al., "Asteroid Return Mission Feasibility Study," AIAA Paper 2011-5565, July 2011.

[2] Keck Institute for Space Studies, Asteroid Retrieval Feasibility Study, April 2, 2012, *see also* http://kiss.caltech.edu/study/asteroid/asteroid_final_report.pdf

[3] Brophy, J.R., Friedman, L., and Culick, F., "Asteroid Retrieval Feasibility," IEEE Conference Proceedings, 2012.

[4] Brophy, J.R., et al., "Returning an Entire Near-Earth Asteroid in Support of Human Exploration Beyond Low-Earth Orbit," GLEX-2012.11.1.7x12334, May 2012.

[5] Landau, D., McElrath, T., Grebow, D., and Strange, N., "Efficient Lunar Gravity Assists for Solar Electric Propulsion Missions," Paper AAS 12-165, AAS/AIAA Spaceflight Mechanics Meeting, Charleston, SC, January 29–February 2, 2012.

[6] McElrath, T.P., et al., "Using Gravity Assists in the Earth-Moon System as a Gateway to the Solar System," GLEX-2012.05.5.2x12358, May 2012.

[7] Massonnet, D. and Meyssignac, B., "A Captured Asteroid: Our David's Stone for Shielding Earth and Providing the Cheapest Extraterrestrial Material," *Acta Astronautica*, 2006, pp. 77–83.

[8] Sanchez, J.P., Yarnoz, D.G., and McInnes, C.R., "Near-Earth Asteroid Resource Accessibility and Future Capture Mission Opportunities," GLEX-2012.11.1.5x12412, May 2012.

[9] Llado, N., Ren, Y., Masdemont, J.J., and Gomez, G., "Capturing Small Asteroids into Sun-Earth Lagrangian Points for Mining Purposes," IAC-12.C1.6.7, October 2012.