Overview of Mission Design for NASA Asteroid Redirect Robotic Mission Concept

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Abstract: Part of NASA's new asteroid initiative would be a robotic mission to capture a roughly four to ten meter asteroid and redirect its orbit to place it in translunar space. Once in a stable storage orbit at the Moon, astronauts would then visit the asteroid for science investigations, to test in space resource extraction, and to develop experience with human deep space missions. This paper discusses the mission design techniques that would enable the redirection of a 100-1000 metric ton asteroid into lunar orbit with a 40-50 kW Solar Electric Propulsion (SEP) system.

Nomenclature

ΔV	=	change in velocity
C3	=	v_∞ squared
I_{sp}	=	specific impulse
v_∞	=	hyperbolic excess velocity

I. Introduction

THE Asteroid Redirect Robotic Mission (ARRM) concept is to capture an entire 4-10 m Near Earth Asteroid (NEA) and redirect it into an orbit around the Moon where astronauts in the Orion spacecraft could explore it. This mission concept has been developed by NASA, following a study by the Keck Institute of Space Studies[1], with two primary objectives: (1) to demonstrate high-power SEP technology that could be used in future human deep-space missions and (2) to increase the deep-space exploration experience and knowledge that can be learned from the initial Orion translunar missions planned for the early 2020s. Although the concept was conceived as a technology demonstration mission as opposed to a science mission, there clearly will be valuable science investigations enabled by the bulk retrieval of an entire asteroid. In addition, the retrieved asteroid could provide many opportunities to experiment with asteroid resource extraction and commercialization of asteroid resources.

A high-power (~50 kW) Solar Electric Propulsion (SEP) spacecraft would be used to rendezvous, capture, an asteroid that is naturally flying by the Earth to a Lunar Gravity Assist (LGA) that would capture it into the Earth-Moon system. From there, the ARRM spacecraft (i.e., the Asteroid Redirect Robotic Vehicle or ARRV) would use the SEP system in concert with Solar and Lunar perturbations to nudge the asteroid into a long lifetime storage orbit around the Moon. Once the ARRV has placed the asteroid into the storage orbit, astronauts would launch in an Orion spacecraft on an SLS (Space Launch System) rocket to rendezvous with the ARRV and the asteroid.

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This ambitious mission concept is a radically different scale from past "sample return" concepts and requires innovative new mission design techniques to accomplish. To establish feasibility, the ARRM Mission Design team has developed a design that combines many astrodynamics techniques developed in the past few years: low thrust trajectory optimization, low-energy transfers, asteroid proximity operations, gravity assists, and n-body system orbit design. The ARRM is an entirely new class of mission that could benefit greatly more research into new astrodynamics techniques. This paper details the first iteration of the mission design for the ARRM concept in hopes of motivating such research.

II. The ARRM Mission Design

The ARRV spacecraft is designed to return an asteroid with a mass of up to 1000 t (10^6 kg) to the Earth-Moon system. At the time of asteroid capture, the wet mass of the ARRV would be in the range 7 t to 10 t depending on how much propellant would be expended getting to the asteroid. This means that the amount of propellant that would be needed for a given ΔV increases by a factor of 100 after the asteroid is captured. This scaling drives the mission design to minimize the post-capture ΔV. It is also why SEP is enabling for the ARM concept.

A nominal ARRM would consist of the following seven mission phases as indicated in Figure 1: 1) Launch & Earth Escape Trajectory; 2) Go To Asteroid Leg; 3) Asteroid Rendezvous & Capture; 4) Asteroid Fetch Leg; 5) Lunar Endgame; 6) Storage Orbit; 7) Crew Access.

The design drivers that govern the trajectory performance include the asteroid parameters (mass, v_∞, and Earth encounter geometry), the SEP performance parameters (I_{sp}, Power, and efficiency), and launch vehicle performance capabilities. Designing the SEP module with higher I_{sp} would require less propellant for the same maneuver than a lower I_{sp} system at the cost of increased time. Increasing the power to the thrusters increases thrust and would lower the time to perform these maneuvers. A larger capability launch vehicle allows either more propellant mass to be sent (good for lower I_{sp} systems) or launch to a higher energy trajectory that reduces flight time (good for higher I_{sp} and/or lower power systems).

It is very difficult to estimate the mass of a non-binary asteroid from ground-based observations. Currently, the strategy of the ARM mission is to design the flight system and trajectory to accommodate the 99% probability upper bound estimate of the asteroid mass. Because mass estimates for even a well-characterized asteroid could have a residual uncertainty of an order of magnitude, a design for returning a 1000 t asteroid could end up returning a significantly smaller object.

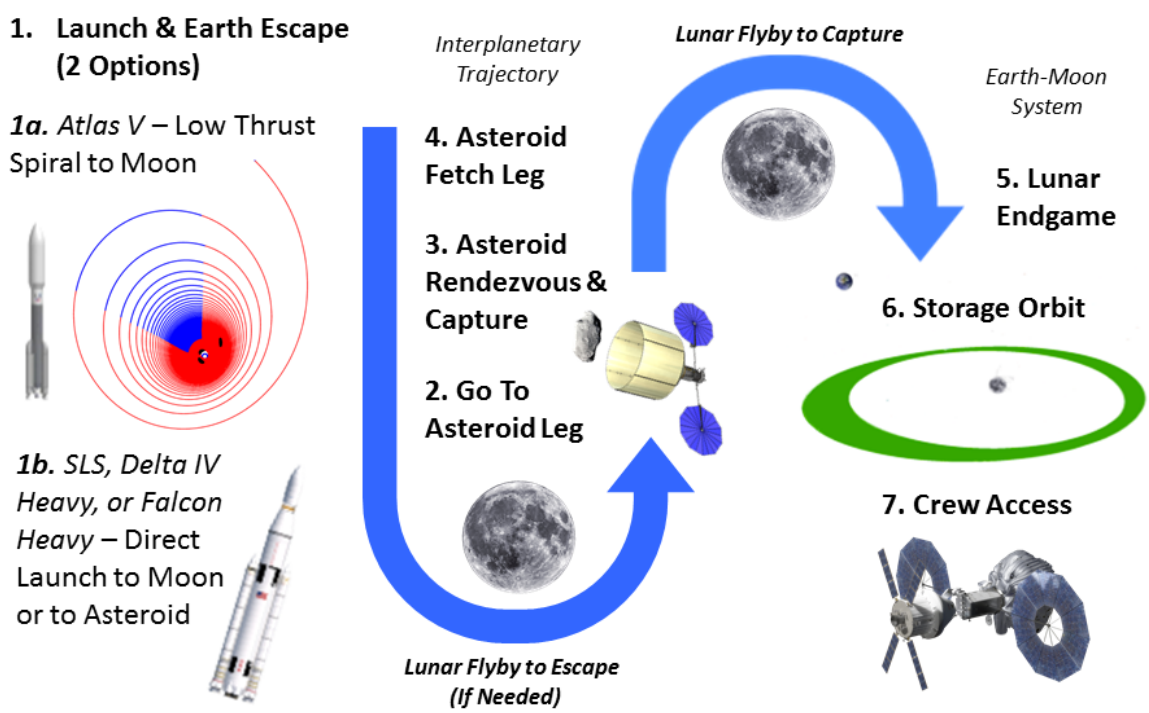


Figure 1. Mission Timeline

Table 1: 2009 BD example mission timeline.

Activity	Date
Atlas V Launch Period	Dec 26 2016 to Jan 16, 2017
Begin Low Thrust Spiral	Jan 23, 2027
First Departure LGA	May 14, 2018
Second Departure LGA (This would be the close of the Falcon Heavy or SLS Launch Period)	Jun 11, 1018
2009 BD Arrival	Apr 12, 2020
Departure with Captured Asteroid	Jun 11, 2020
Capture LGA	Jun 9, 2023
DRO Insertion LGA	Feb 15, 2024
Long Term Storage DRO	Oct 1, 2024

Table 2: 2009 BD example mission ΔV and phase durations.

Mission Leg	ΔV (m/s)	Duration (yr)
Spiral	4662	1.4
To Asteroid (Go to leg)	3868	1.8
Asteroid Ops	(hydrazine)	
Earth Return (Fetch leg)	152	3.0
To Storage Orbit	60	1.4
Total SEP Module DV	8742	

The nominal ARRM retrieval strategy is to find an asteroid with a natural Earth close approach so that it could be redirected to a Lunar flyby.[2] The resulting Lunar gravity assist would loosely capture it into the Earth-Moon system. From this point, solar perturbations and thrust from the spacecraft's SEP system would be used to target a second Lunar flyby that would then be used to enter a long-term storage orbit accessible by crewed missions. It is highly desirable to place the asteroid in an orbit with a very long lifetime. Although the current mission concept would return an asteroid that is too small to survive Earth entry, it would be still important to preserve the asteroid in a stable orbit as a resource for future deep space human missions.

The following sections detail the mission phases identified in Figure 1, using asteroid 2009 BD as an example mission target. Table 1 provides a sequence of events for this example case, and Table 2 indicates that the corresponding mission ΔV of 8.7 km/s is required. Note that the SEP system on the Dawn mission has provided over 7.9 km/s to the much small Dawn spacecraft and will provide approximately 11 km/s by the end of the mission. So, large post-launch ΔVs are possible with solar electric propulsion.

Currently there are medium fidelity trajectories for 10 potential mission targets listed in the top half of Table 3 and high fidelity, end-to-end trajectories for the asteroids 2009 BD and 2011 MD. The detailed analysis for these targets was used to verify the analysis methodology for the other targets and to search for potential problems that may arise when the analysis fidelity is increased. In addition, the detailed analysis provides high confidence that 2009 BD and 2011 MD could be retrievable if pending characterization by the Spitzer Space Telescope (2009 BD in Oct. 2013 and 2011 MD in Jan. 2014) can adequately constrain the uncertainty in its mass.

The medium fidelity trajectories for broad searches were developed using the low-thrust tool Malto.[3] High fidelity interplanetary trajectories were developed using two independent tools (Mystic[4] and Copernicus[5]), which produced ΔVs that agreed to within 3 m/s. Mystic is the trajectory tool currently used on the Dawn mission. Earth-spiral trajectories were built independently at NASA/JPL and NASA/GRC using custom integrators and agreed within 1%. The endgame and DRO trajectories were built in Mystic. Integrations were done using gravitational perturbations from the Earth, Moon, Sun, Jupiter, and Venus. Earth oblateness effects were included for the spiral trajectories. An I_{sp} of 3000 s with a 60% efficiency for the electric propulsion (EP) system was used with a maximum power input to the EP system of 40 kW (end of life at 1 AU) and a conservative $1/r^2$ power variation in available solar array power with solar range. A typical duty cycle of 90% for thrusting with the SEP system was assumed. This duty cycle is appropriate for early formulation studies. The Dawn mission has an actual duty cycle of approximately 95%. On the transfer to Vesta, Dawn would typically thrust for 160 hours per week then shut down the electric propulsion system and spend 8 hours communicating with Earth. The ARM flight system would include

a gimbaled high-gain antenna and would not need to turn off the SEP system to communicate with Earth, making the 90% duty cycle even more conservative.

During the two months prior to the asteroid arrival for ARM the planned duty cycle would be dropped to 50% to allow additional time for acquiring optical navigation images. This is very similar to the methodology used by Dawn in its approach to Vesta. Additional xenon propellant is assumed to be carried to provide margin for unintentional missed thrust periods, and 6% Xe margin is applied on top of the xenon required for the trajectory (including missed thrust margin) to cover other typical uncertainties including flow rate errors, fill errors, and leakage. Furthermore, the propellant mass is calculated for the maximum launch capability (i.e., the fully margined spacecraft mass plus the launch vehicle margin), so that the launch vehicle margin is a true margin useable by the mission.

A NEOWise non-detection of 2009 BD places a 3σ upper bound on 2009 BD diameter of 10 m. Micheli et al. (2012)[6] estimate the area to mass ratio of 2009 BD as $2.97\pm0.33\times10^{-4}$ m^2/kg, which provides a 3σ upper bound of 397 t on 2009 BD's mass. Farnocchia et al. (2013)[7] estimate the area to mass ratio as $2.72\pm0.39\times10^{-4}$ m^2/kg, which would place a 3σ upper bound of 507 t on the mass. Spitzer observations in Oct. 2013 are expected to provide a better upper bound on the asteroid size and therefore better a estimate the mass of 2009 BD. For the analysis in this paper, a mass of 400 t is assumed for 2009 BD. Once the Spitzer derived mass estimate is available, we will update the trajectory analysis.

Phase I: Launch & Earth Escape Trajectory

The ARRV is conceptually designed to launch on either an existing EELV (Atlas V 551 or Delta IV Heavy) or on one of two heavy lift vehicles currently in development (the Falcon Heavy or the SLS). International launch vehicles have not been included in the design space, but any rocket that can meet or exceed to capability of the Atlas V 551 could be an option.

In order to maximize the possible asteroid return mass from an EELV, the ARRV could be launched into Low Earth Orbit (LEO) and fly a low-thrust spiral trajectory to one or more Lunar Gravity Assists (LGAs) that would be used to escape Earth and inject onto an interplanetary trajectory to the asteroid. Such a spiral trajectory would take ~1.4 years with a 40-kW SEP system.

To avoid the long flight time of a spiral trajectory, the next best performance is achieved by launching directly to a trans-lunar orbit and using an LGA to escape. This approach requires about a month of phasing orbits to reach the needed LGA encounter time from any day in the launch period. Finally, if we have sufficient launch capability and a well enough characterized asteroid, it could be possible to launch directly to the asteroid eliminating the LGA entirely, reducing the associated operational cost and complexity.

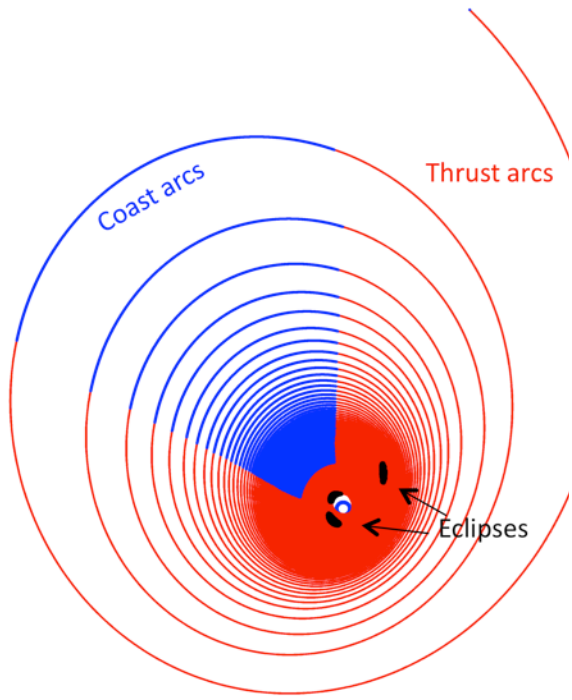


Figure 2: Spiral trajectory showing eclipses

Figure 2 illustrates a spiral trajectory corresponding to a launch on an Atlas V 551-class launch vehicle in January 2017 and an Earth departure in June 2018, using the asteroid 2009 BD as an example target. In this figure, red denotes a thrust arc, blue denotes a coast arc, and black denotes a period in which the spacecraft is in eclipse. The eclipses occur during the first 10 months on the spiral and range from 35 to 95 min. In order to reduce the spiral time, this particular trajectory sequence begins the spiral in a 200-km by 17878-km elliptical orbit and a launch mass of 11590 t.

Figure 3 depicts an example trajectory with two LGAs used to transition from the spiral trajectory to an Earth escape with a C3 of 1.1 km^2/s^2 on June 11, 2018. For an Atlas V-class launch, the escape mass is 9900 kg and the spiral trajectory consumes 1690 kg of xenon. Both the Falcon Heavy and the SLS are expected to have the capability of launching at least 10,000 kg directly to this C3, and therefore do not need a departure LGA for the 2009 BD example trajectory.

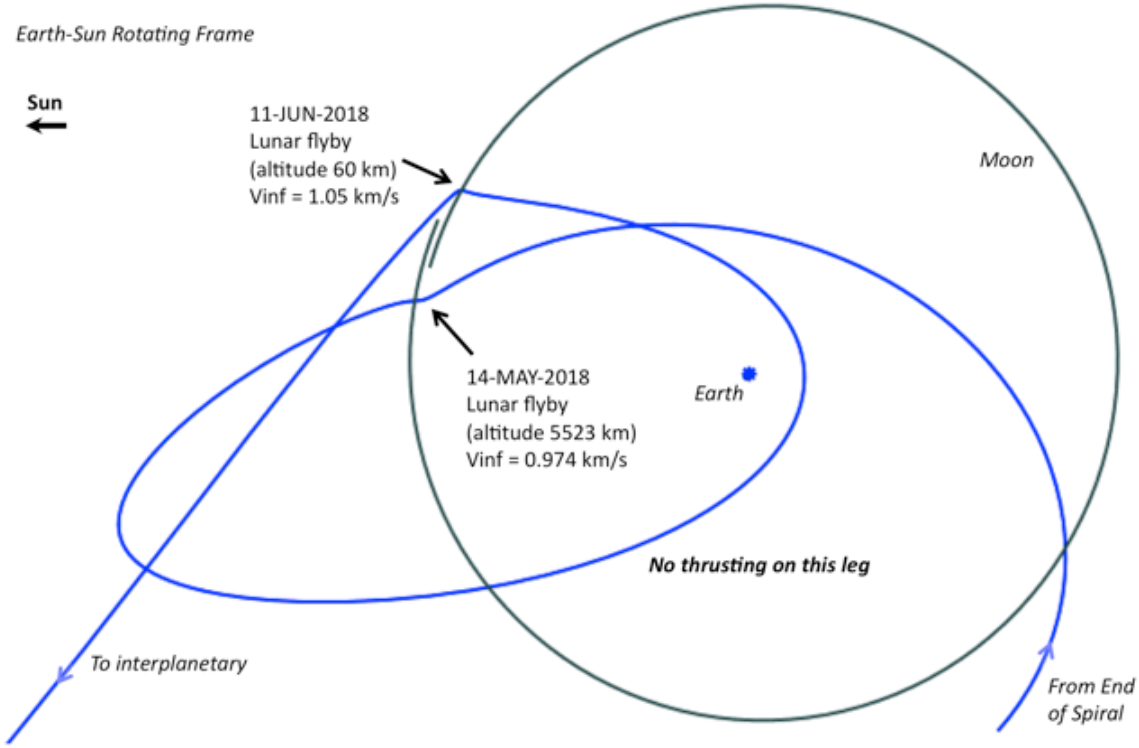


Figure 3: Two Lunar Gravity Assists (LGAs) would be used to escape at the end of the low thrust spiral.

A launch period that opens on Dec 26 2016, would allow for a 21-day launch period and a 7-day operational margin on the start of initial thrusting. A low thrust spiral that starts by Jan 23 2017 would then have margin for an additional 14 days of missed thrust and still be able to reach the first of the two LGA escape sequence on May 14, 2018.

Phase II: Go To Asteroid Leg

The trajectory for the Go To phase is generally a high ΔV leg in order to minimize the flight time to the asteroid. Minimizing the flight time to the asteroid would maximize the time available for the fetch leg, which would minimize the return ΔV needed after asteroid capture. Because the asteroid mass increases the propellant required for a given ΔV on the fetch leg by roughly two orders of magnitude compared to the Go To leg, this approach minimizes the total mission propellant. The current ARRV flight system concept carries solar shields designed for a perihelion as low at 0.7 AU. For the 2009 BD example the minimum range is 0.83 AU. For the example case the Go To leg is 1.7 years, requiring 3940 m/s and 1242 kg of xenon propellant. This leg is shown in both the inertial and Earth-Sun rotating reference fames in Figure 4.

The SEP duty cycle is reduced to 50% for two months before arrival to allow time for approach activities including optical navigation. Although we expect to be able to characterize any target asteroid's orbit with ground observations to know its position better than 50,000 km at the time of encounter, the ARRV optical navigation cameras would be able to find an asteroid with a 500,000 km uncertainty on approach without significant changes to the approach trajectory. (If we permit additional ΔV or time for an extended search phase, the ARRV could find an asteroid with a position error of several million km.)

The last 100–150 days of the Go To leg would be the most sensitive part of the interplanetary trajectory to missed thrust. If the spacecraft stops thrusting due to a fault condition during this phase either additional propellant would be needed prior to rendezvous or the rendezvous would be delayed and additional propellant would be needed on the Fetch Leg. An additional 365 kg of xenon would be carried to remain robust to a 7-day thrust outage during this period. Based on analysis of thrust outages throughout the interplanetary trajectory, this is sufficient to cover more than 7 days of thrust outage throughout the other portions of the trajectory. Note, the Dawn mission has been in operation for about five and a half years, and has never experienced a thrust outage of 7 days. As the ARRM mission design matures, we will continue to analyze the mission tolerance to missed thrust periods and will develop new methods for optimizing the nominal trajectory to reduce the amount of extra propellant that must be carried.

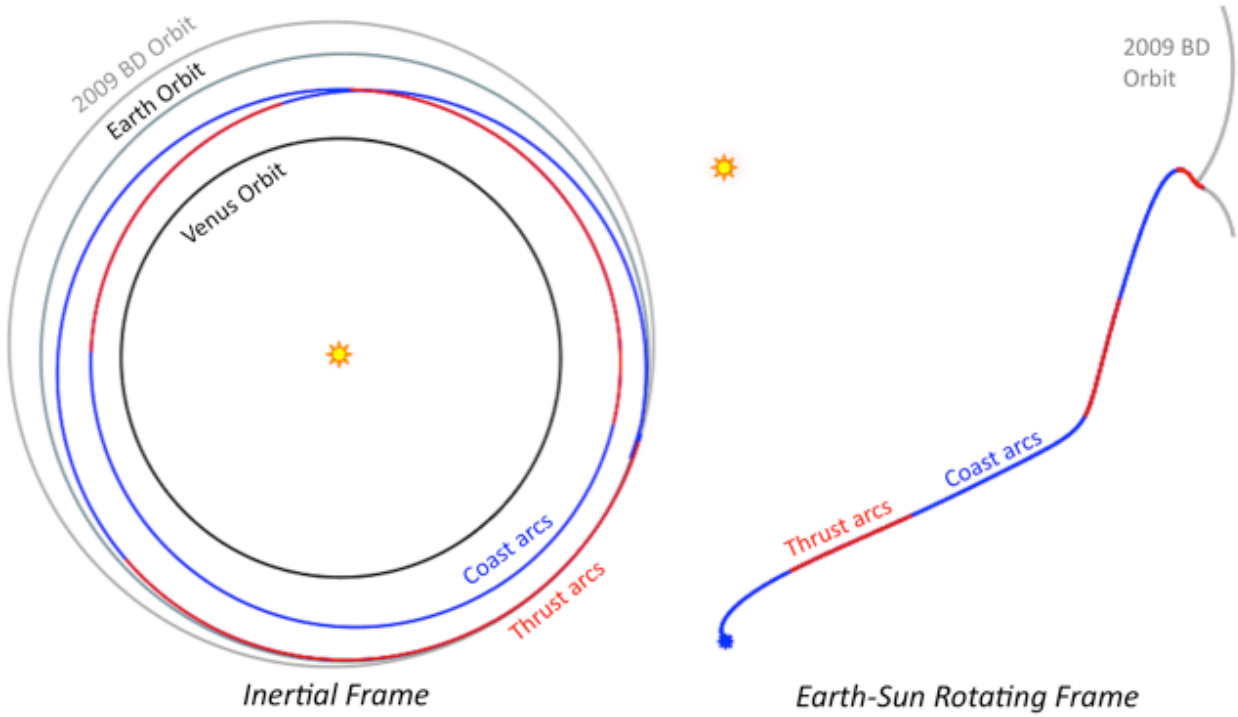


Figure 4: Go To asteroid leg example trajectory for asteroid 2009 BD.

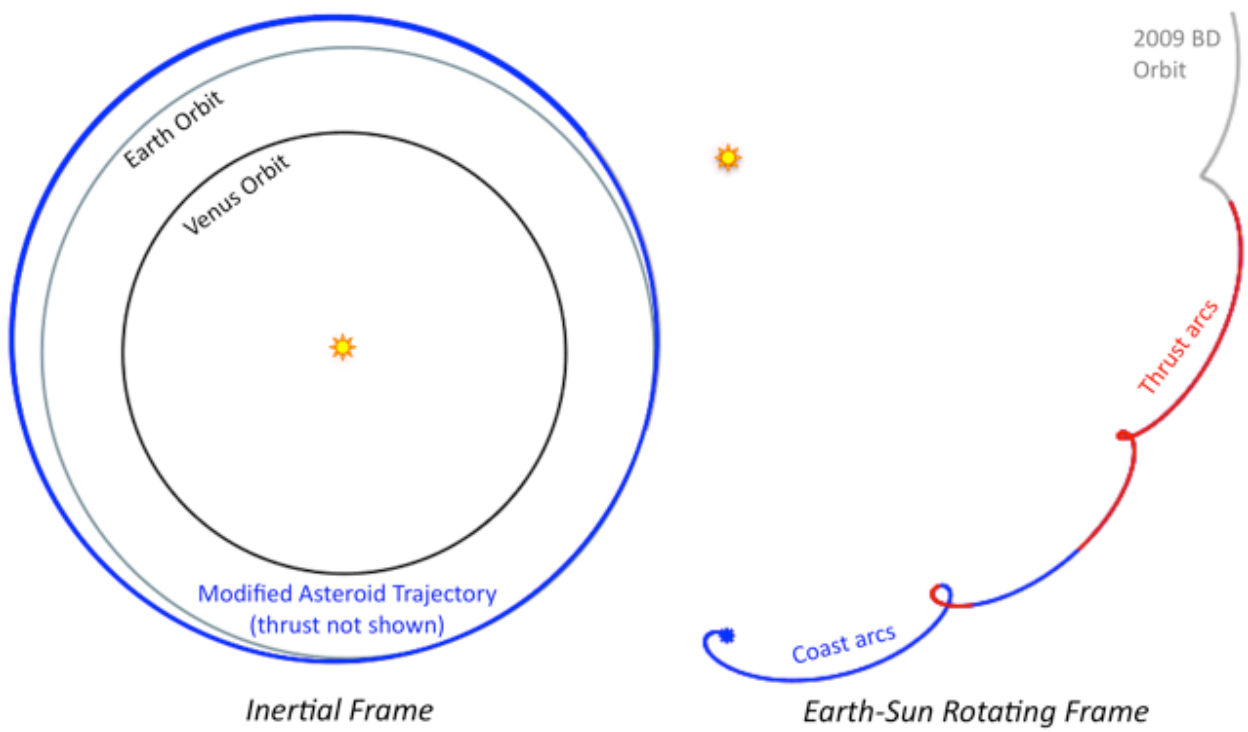


Figure 5: Asteroid Fetch Leg example for asteroid 2009 BD.

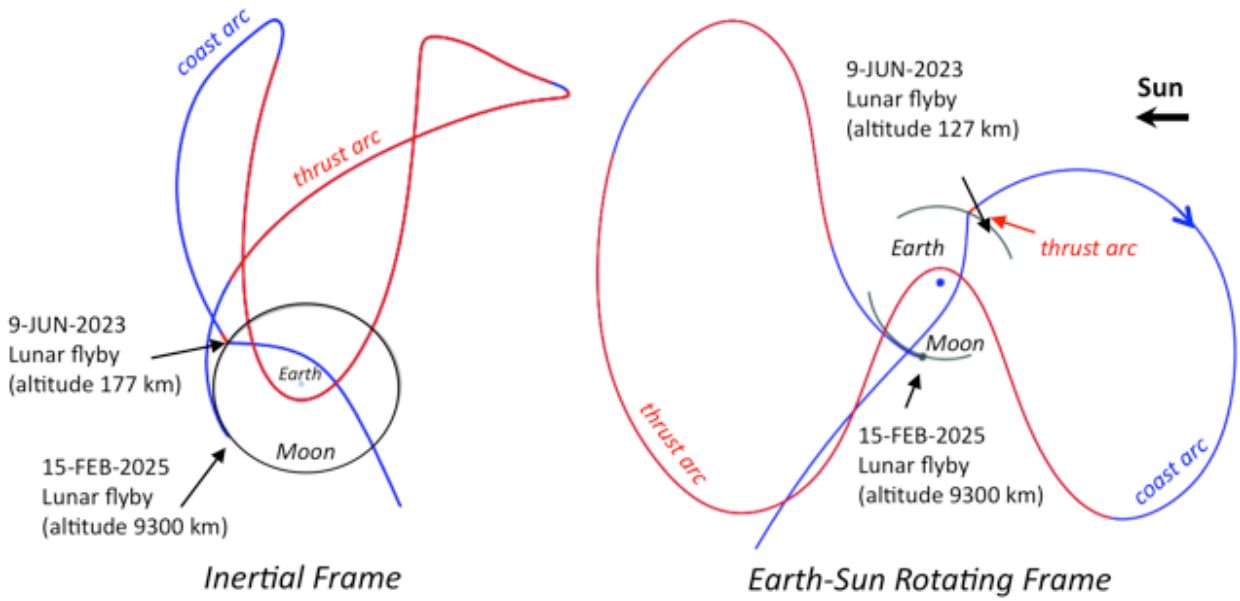


Figure 6: The Endgame transitions from the initial capture orbit to a Lunar Distant Retrograde Orbit (DRO)

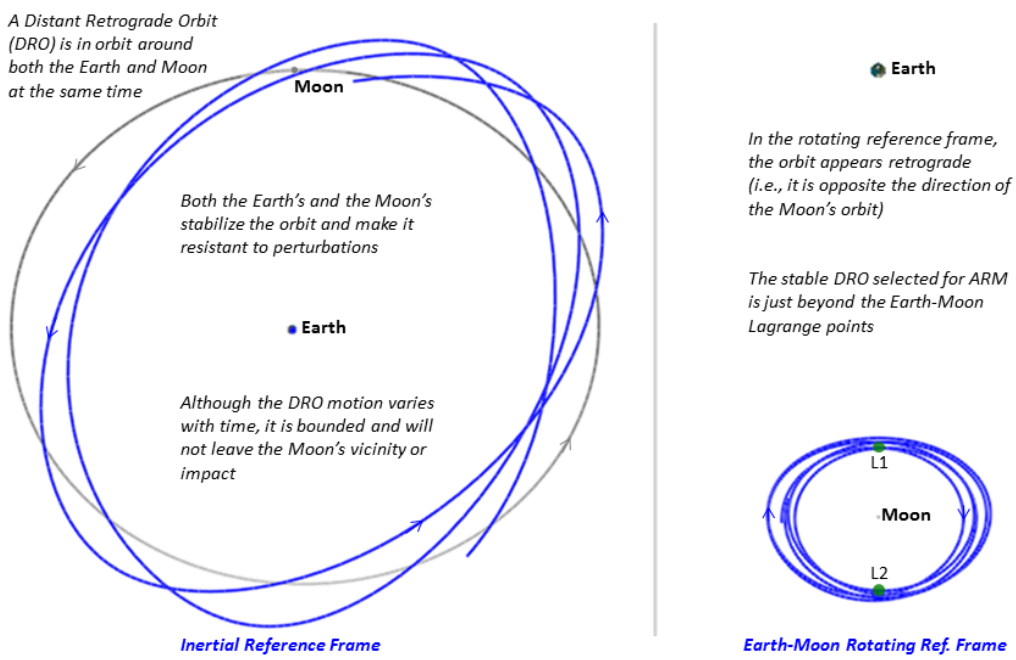
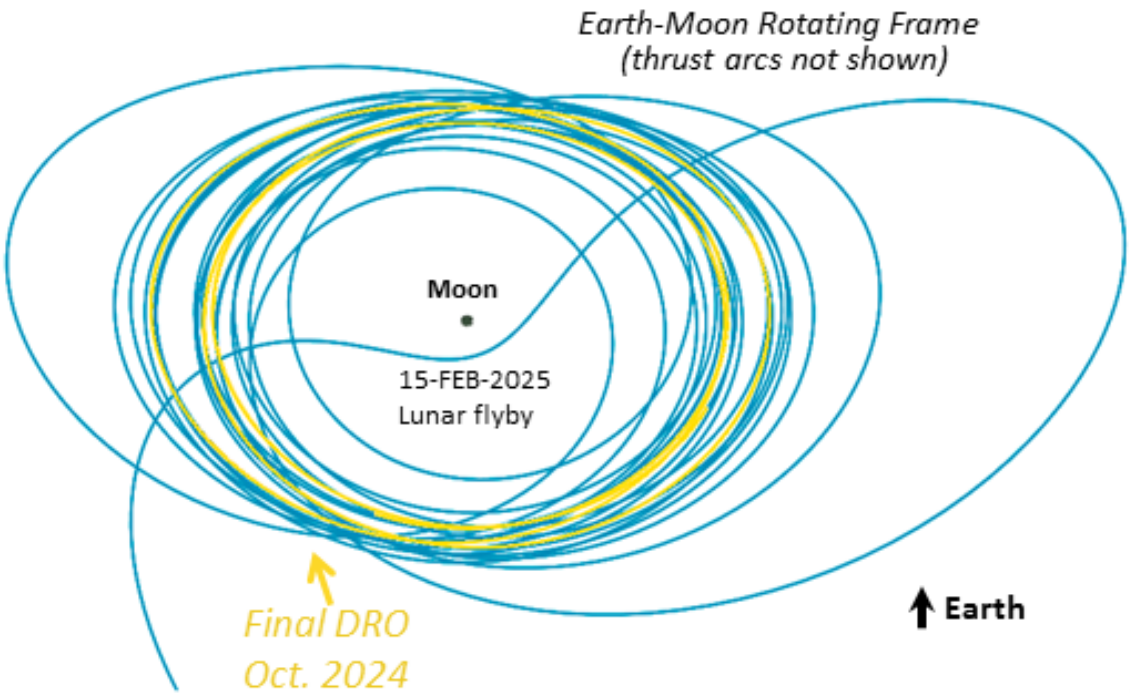


Figure 7: Distant retrograde orbits in translunar space are stabilized by both the Earth and the Moon and are an attractive destination for the retrieved asteroid.

Phase III: Asteroid Rendezvous & Capture

In the current mission scenario, 60 days are allocated after asteroid rendezvous for the characterization and capture of the asteroid. The ARRM team has developed an inflatable capture system design that is beyond the scope of this paper, which focuses on mission design, to describe. The current ARRV capture system concept is designed to accommodate a range of asteroid parameters sufficient to cover the uncertainty of the target asteroid's dimensions and spin state prior to arrival. The capture bag concept is also designed to contain any lose rubble or regolith that might be knocked off of the asteroid in the capture process.

The ARRV spacecraft would image the asteroid as it approaches to within 50 m. Once it is within 50 m, the ARRV would perform a number of small excursions around the asteroid to image the entire sunlit surface at a



variety of solar phase angles. This would allow the determination of the asteroid spin state, volume, surface morphology, and construction of a 3D model of the asteroid for capture planning. This characterization would be used by the flight controllers to update the pre-planned capture sequence. After the asteroid is captured it would be despun and the mass prosperities of the combined spacecraft and asteroid would be determined with a sequence of small test maneuvers. This entire operations sequence is currently estimated to take 30 days if all tasks proceed nominally. An additional 30 days of margin are allocated for contingency operations, spacecraft safe mode events, and to allow design margin for updates to the capture sequence.

Figure 8: Twenty maneuvers are used to trim the DRO Storage Orbit for long term stability.

Table 3: Possible Return Candidates

Asteroid	Asteroid Diameter Estimate	Asteroid V_∞	Current Best Return Mass	Return Date	Notes
2007 UN12	4–12 m	1.2 km/s	490 t	Sep-20	
2008 EA9	7–20 m	1.9 km/s	130 t	Nov-20	
2013 EC20	< 4 m	2.6 km/s	120 t	Mar-21	Discovered 2013 and characterized to be < 50 t by ARRM observation campaign.

Phase IV: Asteroid Fetch Leg

After the asteroid is acquired and cinched up to the spacecraft, the SEP system would be used to perform a change to the asteroid's trajectory that targets it to an LGA that will place it into Earth orbit. For 2009 BD, this requires 165 m/s starting from June 11, 2020 for an LGA on June 9, 2023. This 3-year fetch leg is shown Figure 5 in both inertial and Earth-Sun rotating reference frames. There is no thrusting for the last 14 months of this trajectory, with the possible exception of small trajectory correction maneuvers to fine-tune the aim-point of the capture LGA.

Phase V: Lunar Endgame

The Earth-Moon capture LGA places the asteroid in a loosely captured Earth orbit that would eventually escape the Earth-Moon system if left alone. The "Endgame" phase would use low-thrust arcs, solar perturbations, and lunar flybys to transition from this initial capture orbit to a long-term stable storage orbit. Figure 6 shows this leg for the 2009 BD example in this paper. It requires 42 m/s and 8 months. It ends on Feb 15, 2024 with an LGA that would place the asteroid into the long-term storage orbit.

Phase VI: Lunar Asteroid Safe Storage Orbit (LASSO) Phase

The endgame places the asteroid into its long-term storage orbit (see Figures 7 and 8). Currently the baseline storage orbit is a Lunar Distant Retrograde Orbit (DRO).[8-10] This is a 70,000 km orbit of the Moon that revolves clockwise, in the opposite sense of the Moon's orbit around the Earth. This orbit is actually a 3-body orbit and is dynamically in orbit around both the Earth and Moon simultaneously. DRO orbits are known to be very stable, but in this application the asteroid would initially enter the DRO in an unstable region around the Earth-Moon L1 and L2 points in order to minimize ΔV. For the example trajectory, 16 m/s in 20 small maneuvers over 8 months would be used to trim this initial DRO into one that is stable for over 100 years.

2010 UE51	5–15 m	1.2 km/s	130 t	Oct-22	
2009 BD	4–10 m	1.2 km/s	590 t	Jun-23	Area/Mass known; Spitzer fb 2013
2011 MD	6–17 m	1.0 km/s	800 t	Jul-24	Rotation rate known; Spitzer fb 2014
2008 HU4	6–16 m	0.5 km/s	1600 t	Apr-26	Earth flyby 2016
2012 TF79	8–23 m	0.3 km/s	170 t	Mar-27	Spitzer flyby 2016
2006 RH120	3–9 m	1.0 km/s	490 t	Oct-28	A/M known; rot. rate known
2012 LA	8–21 m	1.5 km/s	230 t	May-29	Spitzer flyby 2018
2011 BL45	9–26 m	1.4 km/s	1400 t	Aug-29	Spitzer flyby 2015
2008 UA202	3–9 m	1.9 km/s	310 t	Oct-29	

Preliminary analysis suggests that although the DROs at the beginning of this phase would be unstable their motion is bounded and they may not be able to escape the vicinity of the Moon. If this is verified through future analysis the orbit trim period at the beginning of this phase may be shortened or possibly even replaced by a single maneuver at the end of the Endgame.

Phase VII: Crew Access

The final DRO period and inclination would be chosen to provide monthly access for Orion launches and to minimize the number of orbits when the Moon shadows the DRO (such orbits would be skipped for Orion missions). The Orion could access this orbit (for 2009 BD) as early as March 2024 while the DRO is being trimmed for long-term stability.

III. Identifying Retrievable Asteroids

Asteroids with v-infinities that have a magnitude less than 2.1 km/s and a declination less than 15° may be captured by one or more LGAs,[11-12] provided they can be targeted to the proper Lunar flyby aim-point. For asteroids with higher v_∞ declinations, Earth flybys could be used to lower the declination. For asteroids with higher v_∞ magnitudes, low thrust double Earth-Moon flybys could be used to lower the v_∞ magnitude to an acceptable range. Any object with a v_∞ less than 2.6 km/s is considered to be possibly retrievable (depending on its mass). In order to include asteroids that do not cross the Earth's orbit, but may still be retargeted to a capture LGA, the v_∞ magnitude is calculated from Tisserand's parameter[13] and objects with perihelion less than 1.03 AU and aphelion greater than 0.97 AU are also considered. Of the known NEAs, 32 objects meet these criteria and have natural Earth close approaches in the 2020s.

Retrieval trajectories for these objects have been calculated and are shown in Table 3. These cases represent the current best return masses for these asteroids assuming a 40-kW SEP system with 3000 sec Isp and launch on a Falcon Heavy. The trajectory search used to generate this table was not exhaustive, and better trajectories may exist. Twelve asteroids have been found with return masses greater than 100 t, and seven in the years 2020 through 2026. Three of these have opportunities for further characterization of the asteroid: 2009 BD, 2011 MD, and 2008 HU4. 2013 EC20 was discovered in March 2013 and characterized by the ARRM observation campaign. It is retrievable if the ARRV could launch by January 2013 on a Falcon Heavy or SLS.

Table 4. 2009 BD propellant loads for other reference configurations.

	Spiral	Heavy Lift	Direct Drive (1)	Direct Drive (2)
Asteroid	2009 BD	2009 BD	2009 BD	2009 BD
Launch Vehicle	Atlas V	Fal. H or SLS	Falcon Heavy	SLS
Launch Mode	Spiral to Moon	Direct to BD	Direct to BD	Direct to BD
Launch Time	Early 2017	Mid-Late 17	Mid 2017	Mid-Late 17
SEP Power	40 kW	40 kW	40 kW	40 kW
SEP *Isp*	3000 sec	3000 sec	2000 sec	2000 sec
Xe Margin	6%	6%	6%	6%
Missed Thrust Margin	365 kg	365 kg	540 kg	540 kg

Hydrazine	300 kg	300 kg	300 kg	300 kg
Launch Mass	11616 kg	10000 kg	10690 kg	12000 kg
Asteroid Mass	400000 kg	400000 kg	400000 kg	400000 kg
Spiral ΔV	4583 m/s	0 m/s	0 m/s	0 m/s
Got To ΔV	3940 m/s	3940 m/s	3750 m/s	3835 m/s
Fetch ΔV	165 m/s	165 m/s	114 m/s	121 m/s
Endgame ΔV	42 m/s	42 m/s	42 m/s	42 m/s
LASSO ΔV	16 m/s	16 m/s	16 m/s	16 m/s
Total ΔV	8746 m/s	4163 m/s	3922 m/s	4014 m/s
Total Xe Propellant	6009 kg	4341 kg	5431 kg	5856 kg
Total Margined Xe	6757 kg	4988 kg	6329 kg	6780 kg
Maximum SC Dry Mass Allowed on Trajectory	**4559 kg**	**4712 kg**	**4061 kg**	**4920 kg**

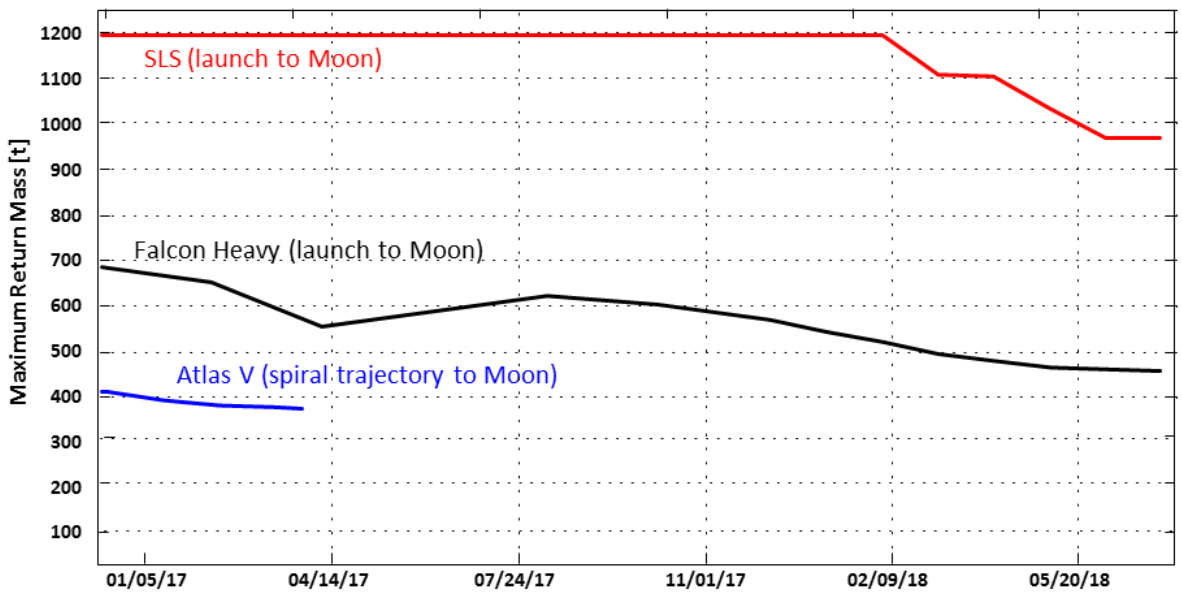


Figure 9: Return mass as a function of launch date and launch vehicle using 2009 BD's orbit as an example with a 40 kW, 3000 sec Isp SEP system.

IV. Other Storage Orbit Options

The DRO was chosen for the storage orbit because it is known to be long term stable, could be entered by the ARRV for relatively low ΔV (10–60 m/s), and would be accessibly by the Orion. Other weakly captured Lunar orbits such as Lunar Circulating Eccentric Orbits could potentially also meet these criteria, but have not yet been verified to do so. If long-term orbit stability is less important (because the asteroid is not an Earth entry hazard), other orbits such as Earth-Moon L1/L2 Halos, and Earth orbits with multiple Lunar flybys might also be considered.

V. 2009 BD Trajectories for Other Example Configurations

Figure 9 shows the maximum return mass for an asteroid in 2009 BD's orbit for Atlas V 551, Falcon Heavy, and SLS launch vehicles as a function of the launch date. All curves assume a 4500-kg dry mass spacecraft, 3000-sec I_{sp}, 60% propulsion system efficiency, and 40 kW to the SEP system. The Atlas V curve drops below 400 t by February 2017, but the SLS and Falcon heavy curves stay above 400 t well into late 2018. Later launch dates are possible for

lower asteroid mass or if the SEP I_{sp} is lowered. For example, with a 325 t 2009 BD and an I_{sp} of 2700 sec, Atlas V launches as late as June 2018 are possible. Table 4 provides the propellant loads and launch masses for 400 t, 2009 BD missions varying launch vehicle, launch date, and spacecraft I_{sp}.

VI. Conclusion

The mission design presented in this paper demonstrates the astrodynamic feasibility of the ARRM. A detailed trajectory design is presented for the asteroid 2009 BD with launches possible on an Atlas V, Delta IV Heavy, Falcon Heavy, or SLS launch vehicle. The mission design remains feasible across a range of launch dates and spacecraft design options. One asteroid (2013 EC20) has been characterized and could be retrieved with launch on a Falcon Heavy or SLS by Jan. 2018. Three other asteroids (2009 BD, 2011 MD, and 2008 HU4) may be retrievable after further characterization with either the Spitzer Space Telescope or during an Earth flyby.

The ARRM mission design team is continuing refinement and performance optimization of the trajectory design for 2009 BD and developing detailed trajectories for other asteroids including 2011 MD, 2013 EC20, and 2008 HU4. Contingency trajectories for missed thrust periods during the interplanetary trajectory, low-thrust spiral, and the endgame will also be examined. The ARRM Observation campaign is also searching for new potential mission targets and trajectories will be developed for these asteroids as they are discovered.

Acknowledgments

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References

[1] J. Brophy, L. Friedman, and F. Culick, "Asteroid Retrieval Feasibility", DOI: 10.1109/AERO.2012.6187031, IEEE Aerospace Conf., March 2012.

[2] D. Landau, J. Dankanich, N. Strange, J. Bellerose, P. Llanos, M. Tantardini, "Trajectories to Nab a NEA (Near Earth Asteroid)", Paper AAS 13-409, 2013.

[3] J. Sims, P. Finlayson, E. Rinderle, M. Vavrina, and T. Kowalkowski, "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design", Paper AIAA 2006-6746, Aug. 2006.

[4] G. Whiffen, "Mystic: Implementation of the Static Dynamic Optimal Control Algorithm for High-Fidelity, Low-Thrust Trajectory Design", Paper AIAA 2006-2356, Aug. 2006.

[5] C. Ocampo, J. Senent, "The Design and Development of COPERNICUS: A Comprehensive Trajectory Design and Optimization System", Paper IAC-06-C1.4.04, Oct. 2006.

[6] M. Micheli, D.J. Tholen, G.T. Elliott, "Detection of radiation pressure acting on 2009 BD", New Astron. 17, 446–452, 2012.

[7] D. Farnocchia, S.R. Chesley, D. Vokrouhlicky, A. Milani, F. Spoto, W.F. Bottke, "Near Earth Asteroids with measurable Yarkovsky effect", Icarus 224, 1–13, 2013.

[8] D. Benest, "Effects of the Mass Ratio on the Existence of Retrograde Satellites in the Circular Plane Restricted Problem IV. Three-dimensional Stability of Plane Periodic Orbits", Astronomy & Astrophysics, Vol. 54, 563-568, 1976.

[9] R. Brouke, "Periodic Orbits in the Restricted Three-Body Problem With Earth-Moon Masses," Technical Report 32-1168, Jet Propulsion Laboratory, California Institute of Technology, 1968.

[10] T. Lam, "Exploration of Distant Retrograde Orbits Around Europa," Advances in the Astronautical Sciences, Vol. 120, 135-153, 2005.

[11] D. Landau, T. McElrath, D. Grebow, N. Strange, "Efficient Lunar Gravity Assists for Solar Elec- tric Propulsion Missions," Paper AAS 12-165, Feb. 2012.

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

[13] Roy, A. E., Orbital Motion, 2nd ed., Adam Hilger, Bristol, England, U.K., 1982, pp. 129–130.