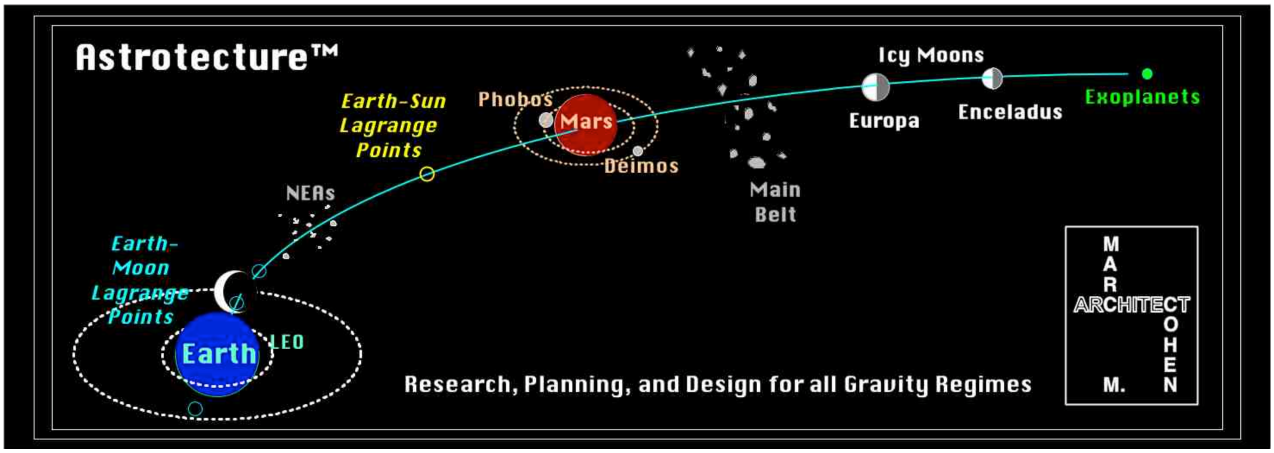


Phase I Final Report to NASA Innovative and Advanced Concepts (NIAC)

Robotic Asteroid Prospector (RAP)

NASA Contract NNX12AR04G -- **9 July 2013, REV 12 Dec 2013** ©

PI: Marc M. Cohen, Arch.D; Marc M. Cohen Architect P. C.
http://www.astrotecture.com

CoI for Mission and Spacecraft Design: Warren W. James,
V Infinity Research LLC.

CoI for Mining and Robotics: Kris Zacny, PhD, with Jack Craft and
Philip Chu, Honeybee Robotics.

Consultant for Mining Economics: Brad Blair, New Space Analytics
LLC

Astrotecture™

V Infinity Research LLC

HONEYBEE ROBOTICS
Spacecraft Mechanisms Corporation

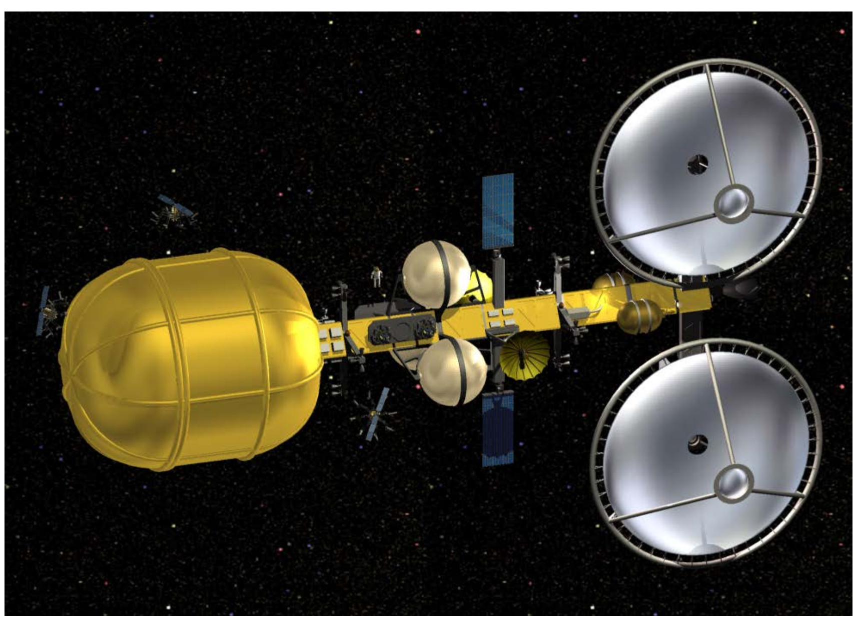


Table of Contents

Abstract

This report presents the results from the nine-month, Phase 1 investigation for the Robotic Asteroid Prospector (RAP). This project investigated several aspects of developing an asteroid mining mission. It conceived a Space Infrastructure Framework that would create a demand for in space-produced resources. The resources identified as potentially feasible in the near-term were water and platinum group metals. The project's mission design stages spacecraft from an Earth Moon Lagrange (EML) point and returns them to an EML. The spacecraft's distinguishing design feature is its solar thermal propulsion system (STP) that can provide for three functions: propulsive thrust, process heat for mining and mineral processing, and electricity. The preferred propellant is water since this would allow the spacecraft to refuel at an asteroid for its return voyage to Cis-Lunar space thus reducing the mass that must be staged out of the EML point. The spacecraft will rendezvous with an asteroid at its pole, match rotation rate, and attach to begin mining operations. The team conducted an experiment in extracting and distilling water from frozen regolith simulant.

Marc M. Cohen Architect PC – **Astrotecture®**
A California Professional Service Corporation

Postal Address:
4260 Terman Drive No. 104
Palo Alto, California 94306-3864 USA

+1 650 218-8119 mobile
marc@astrotecture.com
http://www.astrotecture.com

Executive Summary

This report presents the results from the nine-month, Phase 1 investigation for the Robotic Asteroid Prospector (RAP). The central objective is to determine the feasibility of mining asteroids. Ideally, this determination should be economic, technical, and scientific to lead to the conceptualization of initial robotic and later human asteroid mining missions.

The RAP team began its work from a deeply skeptical perspective on the viability of long-term space industrialization, including the minerals and mining sector. The team posited that for asteroid mining to become feasible, its advocates must make five arguments successfully:

1. That there are accessible, exploitable, and valuable minerals, metals, and possibly H_20 in the asteroids,
2. That a sustained market demands exists or will exist on Earth, in space, or both,
3. That the team can develop a transformational mission design to make frequent, repeated missions to an asteroid possible.
4. That the team can design, develop, and produce the innovative spacecraft necessary to carry out the mission, and
5. That the team can develop the necessary robotic mineral extraction, beneficiation, processing, and concentration technologies.

The RAP team made progress on each of these criteria, as follows.

1. Accessible Resources — The RAP team identified water as the commodity most likely to be of value for extraction and sale to customers in space for use as propellant. Platinum group metals (PGM) are the best candidates for potential sale on Earth, however the scope of the undertaking would require returning PGMs to Earth in the 10s of metric tons. Rare Earth Elements (REEs), although increasingly in demand on Earth, do not appear to be a viable candidate at this time because of the high cost and complexity of processing the ore. Additionally since the current cost of REEs extracted from the Earth is driven by the cost of the environmental remediation associated with that activity there is the very real chance that reducing those remediation costs would be a more cost effective way to increase the supply of REEs than asteroid mining. . Future potential economic resources included scientific samples, regolith for radiation shielding, structural elements such as Al, Fe, Si, and Ti, processed water for life support, and processed regolith for agricultural soil.

   With respect to where to find these resources, the RAP proposal baselined a set of telescopes in Venus orbit, looking outward from the Sun to identify and track the population of Near Earth Asteroids with far greater precision than currently available. Therefore, the RAP team was delighted when Planetary Resources LLC announced their startup in 2012, with a first phase of deploying the Arkyd space telescopes for this purpose. RAP looks forward to data from advanced versions of the Arkyd that could obtain albedo, rotation and spectrographic data for candidate asteroids.

2. Space Infrastructure Development Framework — The economic premise of RAP is that humans will develop an infrastructure for living and working in space. In this century, this

infrastructure will grow to support hundreds of people and eventually thousands of people across the Solar System. We composed a space infrastructure development framework to characterize the growth of this infrastructure both in time and in the number of people living continuously in space. These space settlers will create a demand for commodities processed and products manufactured in space. The earliest commodity for which we see this demand is water. Water exists on the Moon and in the asteroids. The Delta V to return water from an asteroid can often be less than to enter and escape from the Moon's gravity well. We believe that water from asteroids can present a comparative advantage over lunar water and an absolute advantage over water from the Earth. In addition, the space infrastructure development framework prepares to accommodate other asteroid products including structural metals (Al, Fe, Mn, Ni, Si, Ti), platinum group metals (PGM), regolith for radiation shielding, regolith to provide soil for agriculture, and scientific samples as a commodity item for education and industrial research. We are designing the RAP spacecraft to play a leading role in building this space infrastructure and supplying the people who will live within it.

3. <u>Mission Design</u> — For any interplanetary mission the orbital position of the departure and destination objects drives the energy cost of the mission. Having selected a destination there is little flexibility in selecting a departure time. Moreover, the time between mission opportunities is driven by their synodic period, which can be extremely long, i.e. decades or longer, for objects with similar orbit periods. Therefore, the RAP mission architecture encompasses a highly flexible approach to defining mission opportunities that makes uses of multi-body gravity assists, multi-revolution interplanetary transfers and deep space maneuvers to maximize the number of mission opportunities while minimizing total mission Delta V.

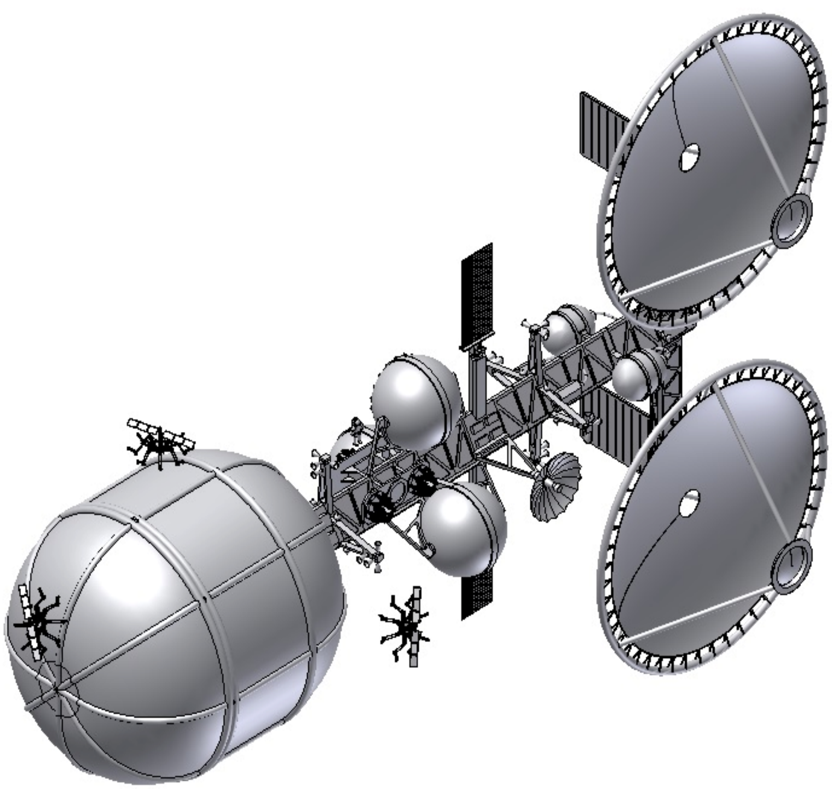
   An important part of this approach is the use of an EML point as a staging point for departing and returning mining spacecraft. This gateway approach provides significant reductions in Delta V when compared to missions departing from or returning to LEO. This benefit accrues because the EML points are located at the edge of the Earth's gravity well; a vehicle there is only loosely tied to the Earth. This strategy reduces substantially the propellant required for those missions.

4. <u>Spacecraft Design</u> — The RAP team designed a prototype prospecting and mining spacecraft. Its key features are the implementation of a solar thermal propulsion (STP) system incorporating parabolic solar concentrators that can concentrate sunlight to 10,000x the incident insolation. The design of the RAP spacecraft enables use of this concentrated sunlight in three ways. First, it provides the heat at up to 2500K to the solar thermal engine to expand the fuel out the nozzle to create thrust. Second, it provides process heat to the on board mining, extraction, processing, and refining systems. Third, it can generate about one megawatt of electricity using a Stirling cycle engine.

   Water is the preferred fuel for the RAP STP system because it has several advantages when compared to conventional propellants. It is very dense when compared to its cryogenic by-products liquid oxygen (LOX) and liquid hydrogen (LH2). Not only does it not require the complexity and cost of electrolysis followed by cryo-cooling, but also the mass of the water tanks can be much smaller than the tankage required for a comparable mass of LOX and LH2. Moreover water can be stored in flexible tanks that simplify the task of propellant management in zero gravity but which also can be launched into space in a collapsed state, which will reduce tremendously the size of the cryogenic tank fairing needed for the launch vehicle (think about the Space Shuttle External Tank). Water propellant confers a further advantage insofar as the RAP spacecraft can refuel itself while on a mission from certain asteroids, thereby reducing the propellant loading required at the outset of the mission.

5. Mining Technology — The mining operation involves rendezvous with the target asteroid and orienting the spacecraft longitudinally at its pole along its axis of rotation. The RAP spacecraft spins up to match the rotation of the asteroid and then attaches at the pole. For small asteroids of less than about 20m-diameter, the spacecraft may capture it with a system of gripping arms and airbags. For larger asteroids, the RAP will carry a set of small "spider" robots that can remove chunks of material from the asteroid and place them into the intake hopper to begin the extraction process.

   The RAP team performed an experiment to evaluate approaches to extracting water from frozen regolith. Many asteroids are covered with a frozen regolith composed of dust and particles of a range of sizes. This range of sizes poses a challenge to simply distilling the water, especially in microgravity. The team froze a slurry of regolith simulant and tested auger extraction techniques and thermal-vacuum distillation methods. The result was that the RAP team thermal-vacuum distillation of water from frozen regolith in the laboratory to TRL-4.



Executive Summary FIGURE 0. The Robotic Asteroid Prospector Spacecraft with the Containment Vessel in front in the Foreground and Robotic "Spiders" Maneuvering around it. At the Aft End Appear the Parabolic Solar Concentrators and the Two Water Propellant Tanks. Along the Truss in the Middle are the Three Payload Water Tanks, Quad Thrusters, Photovoltaics for Spacecraft Bus Power, and a Comm Antenna.

1 Introduction

The Robotic Asteroid Prospector Phase 1 proposal took the approach that the team would need to make five successful arguments in order to determine the feasibility of asteroid mining. The team succeeded in varying degrees in advanced each of these arguments. Overall, the first-year effort afforded a tremendous learning experience to the RAP team. This Introduction explains this record of success and revision.

1.1 Mineral Economics Strategy

In developing the RAP Work Plan, the team had agreed that mineral economics should play a trail-blazing role to generate the parameters within which the other three disciplines – mission design, spacecraft design, and prospecting/mining/processing must work. However, that leadership role of economics proved a non-starter. We could not find the data, the economic model, or the economic expertise to pursue that approach in a credible manner. Instead, one of the first things we learned was that the second clause of our title: Robotic Asteroid Prospector (RAP) Staged from L1: *Start of the Deep Space Economy* was vastly more ambitious than we imagined. Instead of concocting our fantasy of an economic-infrastructural demand model of human civilization expanding across the Solar System over the next century, we needed to find an alternative construct that we could validate. The best we could do is to construct a parametric model of the cost of developing and building the spacecraft, flying the mining missions, and paying for it over time at prices that the space and Earth markets could bear. This parametric model appears at the end of the Spacecraft Design chapter below.

1.2 Asteroids, Meteorites, and Metals

Our knowledge of asteroid composition comes primarily from meteorites found on Earth. The collection and analysis of these meteorites gives an extensive inventory of the minerals and metals that may occur naturally on asteroids, which in some respects are simply very large meteoroids.


Campo Meteorite Gibeon Meteorite

FIGURE 1.1 Examples of Fe-Ni Meteorites

1.2.1 Families of Metals

FIGURE 1.1 shows Fe-Ni meteorites. M-type asteroids are primarily Fe-Ni, with a distinctive emission line at 0.9µ. The sidrophilic Pt group and Au occur in Fe-Ni meteorites. M-type metallic asteroids appear to contain Pt group metals potentially worth billions or trillions of dollars at market (Lewis, 1991). The top prices for platinum group metals are $43.4k/kg for Re and $51k/kg for Pt.

1.2.2 Water from Carbonaceous Chondrites

Carbonaceous chondrites are grouped into at least 8 known groups. The two groups, notably the CM and CI, contain high percentages (3% to 22%) of water, and some organic compounds (Norton, 2002). The presence of volatile organic chemicals and water means that since their formation they have not undergone heating above approximately >200 °C. In fact it is believed that CI (which contain higher fraction of water than CM) have not been heated above 50 °C. Therefore the CI type asteroids would be the best targets for the RAP mission with a goal to acquire water.

1.3 Near Earth Objects

Near-Earth Objects (NEOs) are asteroids and asteroids that have been pulled by the gravity of nearby planets into orbits close to Earth. Asteroids are made up mostly of water ice and some dust particles, while are classified by their characteristic spectra, with the majority falling into three main groups: C-type (carbon rich), S-type (stony), and M-type (metallic). FIGURE 1.2 shows a diagram of M-type NEAs.

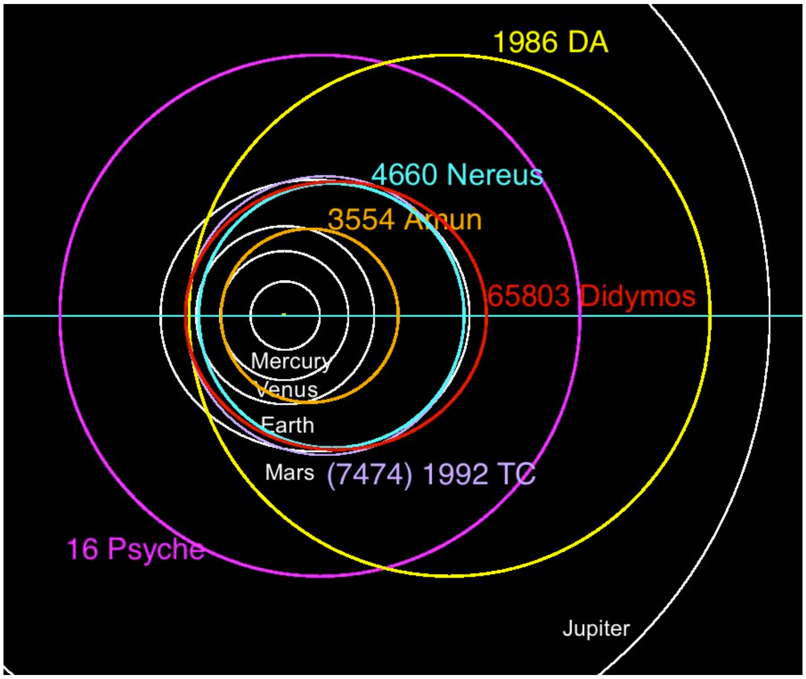


FIGURE 1.2. Orbits of M-type asteroids with major axes aligned, perihelion to the left.

NEO asteroids are also referred to as Near Earth Asteroids or NEAs in order to distinguish them from asteroids within the asteroid belt or Trojan asteroids that share an orbit with a planet or moon. NEAs can be further subdivided into groups: Atiras, Aten, Apollo, and Amor, according to their perihelion distance (q), aphelion distance (Q) and their semi-major axes (a).

Atiras orbits are contained entirely with the orbit of the Earth.

Atens are Earth-crossing NEAs with semi-major axes smaller than Earth's.

Apollos are Earth-crossing NEAs with semi-major axes larger than Earth's.

Amors are Earth-approaching NEAs with orbits exterior to Earth's but interior to Mars'.

In addition, another group, called Potentially Hazardous Asteroids are *PHAs* whose Minimum Orbit Intersection Distance (MOID) with the Earth is less than 0.05 AU (i.e. less than ~7,480,000 km) and whose absolute magnitude is 22.0 or brighter with assumed albedo of 13% (i.e. diameter smaller than about 150 m). They are called Potentially HA because the fact that they come close to Earth does not mean they will impact the Earth. There exists some threat though, and hence they are being monitored to determine the probability of their impact with Earth. Note that NEO asteroids are called Near-Earth Asteroids (NEAs) and include only short-period asteroids with orbital period of less than 200 years.

FIGURE 1.3 illustrates the Earth-Sun Lagrange Points (ESL) and the Earth-Moon Lagrange Points (ESL). The Lagrange Points will take on great importance in the Mission Design Chapter. So, finally, ***Trojans*** are asteroids captured at the triangular Lagrange points at L4 60° ahead or L5 60° behind any planet or moon in its orbit. Jupiter Trojans are well known. A few Mars Trojans have been identified and confirmed as follows:

L4 -- 1999 UJ7
L5 -- 5261 Eureka, 1998 VF31

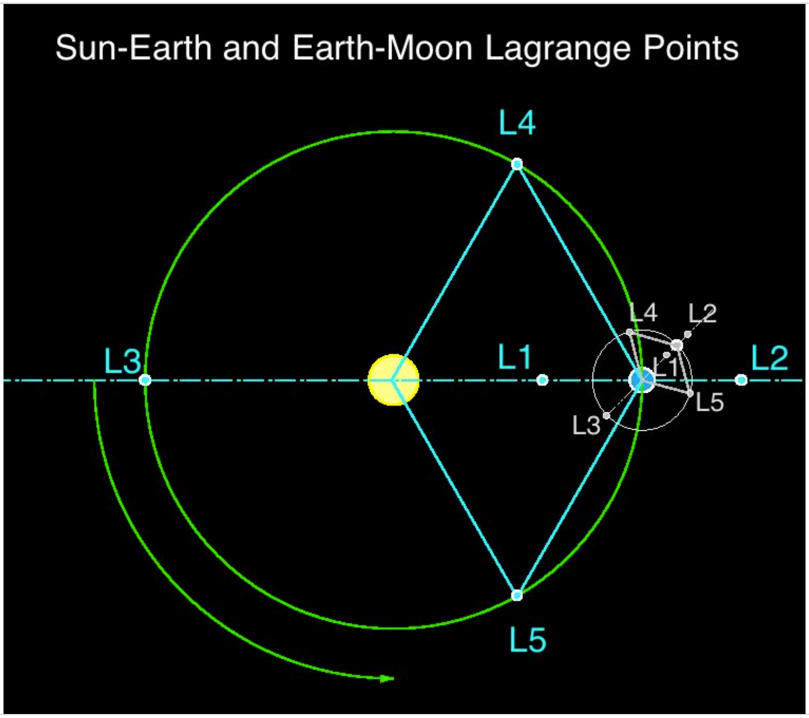


FIGURE 1.3. Earth-Moon and Earth-Sun Lagrange points.

Most asteroids formed in the cold outer planetary system, while most of the rocky asteroids formed in the warmer inner solar system (between Mars and Jupiter). Asteroids are remnants from the solar system formation that occurred 4.6 billion years ago, hence they could offer a clue to the chemical mixture from which the planets formed. The outer planets (Jupiter, Saturn, Uranus, and Neptune) formed from an agglomeration of billions of asteroids. Asteroids we see today are left over pieces from this formation process. On the other hand Asteroids are left over debris from the agglomeration of the inner planets: Mercury, Venus, Earth, and Mars.

Asteroids also pose great interest as source of raw materials. Currently, it does not seem to be cost effective to extract resources and bring them back to Earth. However, there appears to be some potential economic value in processing the resources in situ and using these processed resources in space. Raw materials from M-type asteroids could be used in developing various space structures. Water and carbon-based molecules from C-type asteroids could be used to sustain life and in generating liquid hydrogen and oxygen rocket fuel required exploring and potentially colonizing our solar system.

What makes NEAs even more enticing is that unlike asteroids within the Asteroid Belt (between the orbits of Mars and Jupiter) they come close to Earth and so could pass within a relatively easy reach by the RAP spacecraft. That proximity promises to emerge as an important consideration, given that getting to the Asteroid Belt could take several years.

Initially, the RAP team is looking primarily at asteroids in the Inner Solar System, inside the orbit of Mars. In a later phase, this attention will expand to the Main Belt, but given the scope of the present contract, the NEAs seem to be the most sensible targets. The entire commercial space and asteroid exploration community received a huge boost in this direction from the Planetary Resources LLC's project to build and fly the Arkyd space telescopes. Our PI, Marc M. Cohen met twice with Chris Lewicki, the CTO of Planetary Resources to discuss the potential for future collaboration. We have subscribed to their Kickstarter campaign and bought initial observing time on the telescope. Once we make our initial observations and understand how the Arkyd system works – assuming they can provide the appropriate and necessary instruments and sensors, the RAP team will map out our own observing campaign. FIGURE 1.2 shows some potential observing targets the inner solar system, with the orbits of several M-type NEAS plus ***16 Psyche***, the largest known M-type representing the Main Belt. ***3554 Amun***, best known because of Lewis (1991), may not be high enough density to be metallic. ***1986 DA*** may offer a better prospect in terms of confirmed composition and estimates of $20B in Pt group metals, but it orbits to the outer edge of the Main Belt. ***(7474) 1992 TC***, ***4660 Nereus***, and ***65803 Didymos*** cross from near

Earth to Mars, and so are interesting also as proto-Mars missions. The proposed repeatable trajectories and logistics from the Lagrange Points (FIGURE 1.3) to the M-type NEAs have the potential to become "revolutionary technologies."

1.3 Past Missions

The RAP team began our work with a review of all previous missions to asteroids or to fly by them. TABLE 1.1 shows a list of asteroids visited by Earth-launched spacecraft to date. Out of millions of known asteroids, Earth's spacecraft have **visited only 12 and we managed to "land" on only 2.** Obviously**,** there are thousands more asteroids to visit, many of them offering potential value for prospecting and mining **–** a veritable tabula rasa for exploration. TABLE 1.1 also shows mission cost (if available) and science returned. It shows that compared to other classes of mission, such as a Moon or Mars Lander, the science return per dollar has been relatively low. After spending billions of dollars, the space science community still does not know much about the composition and make up of 99% of asteroids. There is only so much we can learn from 'looking' at the asteroid surfaces. We can learn so much more, if we can land and analyze materials in situ and even more, if we can bring these materials back to Earth.

1.3.1 The NEAR

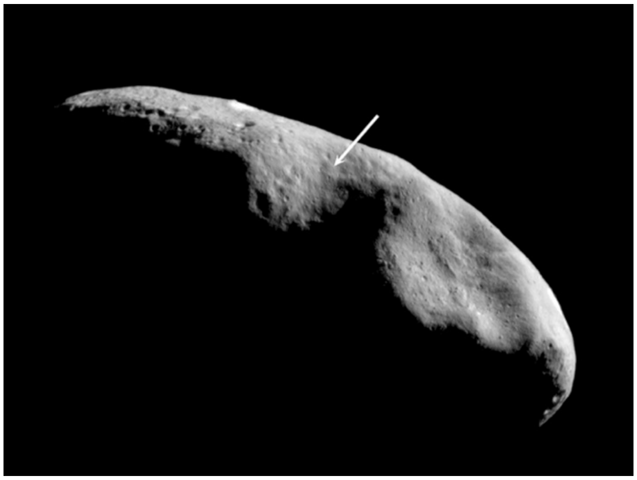
The Near Earth Asteroid Rendezvous (NEAR)-Shoemaker spacecraft was approaching the end of its life when the mission team decided to take the risk of soft landing the spacecraft on the surface of *433 Eros*. If something went wrong (i.e. the spacecraft crushed rather than soft landed, or damaged its fragile solar panels and protruding antennae) there would had been no negative repercussions for the mission since the mission was completed already and the soft landing was never the mission goal. The spacecraft was successfully brought to a 1.9 m/s touchdown onto the rocky surface hence demonstrated that soft landing on an asteroid is possible, as marked on FIGURE 1.4. Note that from a distance of 200 km, it is very difficult to resolve surface details. The spacecraft must fly much closer to resolve such details. In addition, the image taken from a distance of 250 m shows that the surface of the asteroid is quite variable.

1.3.1 Hayabusa

The second asteroid landing was with JAXA Hayabusa spacecraft; however, the spacecraft did not really land but touched the surface of 25143 Itokawa (S-type asteroid) to acquire samples. Hayabusa was essentially a *Touch and Go* mission. Hayabusa was launched in 2003, rendezvoused, landed (for a few seconds), and collected samples (1500 grains, mostly smaller than 10 microns) in 2005, and returned samples to Earth in 2010.

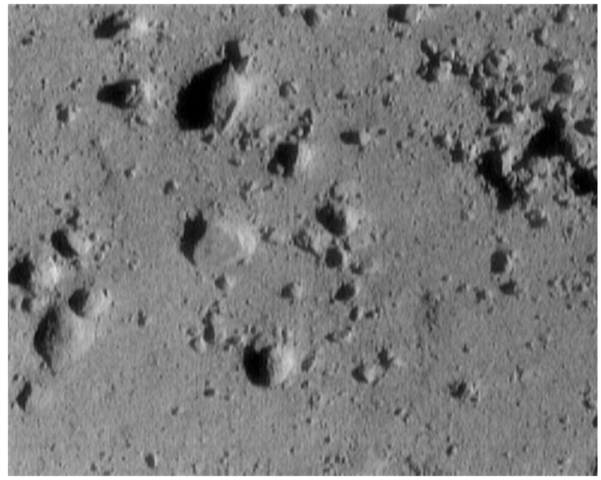
Hayabusa carried a tiny mini-lander named "MINERVA" (Micro/Nano Experimental Robot Vehicle for Asteroid). However, MINERVA was released while the probe was ascending and at a higher altitude than intended. As a result the lander escaped Itokawa's gravitational pull and tumbled into space.

TABLE 1. 1 Asteroids Visited by a Spacecraft (Missions in bold denote 'landed' missions).			
Mission and Body visited	**Agency, Launch Date**	**Mission Description (Relevant to small bodies)**	**Cost (If available)**
International Asteroidary Explorer (ICE)	NASA, 1978	Carried an X-Ray spectrometer and a Gamma burst spectrometer. Flew through the tail of the Giacobini-Zinner asteroid, and observed Halley's Asteroid from afar.	$3 Million ops-only add-on to an existing mission.
Vega 1 and Vega 2	SAS, 1984	Gathered images of Halley's Asteroid after investigating Venus.	
Sakigake	ISAS, 1985	Carried instruments to measure plasma wave spectra, solar wind ions, and interplanetary magnetic fields. Made a flyby of Halley's Asteroid.	
Suisei	ISAS, 1985	Carried CCD UV imaging system and a solar wind instrument for a flyby of Halley's Asteroid.	
Giotto	ESA, 1985	Carried 10 instruments to explore Halley's Asteroid, and provided data despite taking damage. Went on to explore asteroid Grigg-Skjellerup as well.	
Galileo	NASA, 1989	Carried 10 instruments. Flew by 951 Gaspra and 243 Ida, discovered Ida's moon Dactyl, and witnessed fragments of the asteroid Shoemaker-Levy 9 crash into Jupiter.	$1.6 Billion
Near Earth Asteroid Rendezvous (NEAR) Shoemaker	NASA, 1996	Characterized asteroid Eros using imagers, spectrometers, a magnetometer, and a rangefinder. Although not originally planned to do so, NEAR-Shoemaker landed on Eros.	$220.5 Million
Deep Space 1	NASA, 1998	Carried technology experiments. Flew by asteroid 9969 Braille and asteroid 19P/Borrelly.	$152.3 Million
Stardust	NASA, 1999	Carried instruments for imaging and dust analysis. Flew by asteroid 5535 Anne Frank, asteroid Wild 2, and asteroid Tempel 1. Returned sample material from asteroid Wild 2.	$199.6 Million
Asteroid Nucleus Tour (CONTOUR)	NASA, 2002	Carried instruments for imaging, spectrometry, and dust analysis. Spacecraft was lost.	$135 Million
Hayabusa	ISAS, 2003	Landed on the asteroid Itokawa and returned samples to Earth.	$170 Million
Rosetta	ESA, 2004	Flew by asteroid 2867 Steins and 21 Lutetia. Observed Deep Impact. Mission plans to put a lander on asteroid 67P/Churyumov-Gerasimenko.	~$1.2 Billion
Deep Impact	NASA, 2005	Carried instruments for imaging and spectrometry. Hit the asteroid Tempel 1 with an impactor and observed the collision. Will continue to study asteroids and asteroids as the EPOXI mission.	$330 Million
Dawn	NASA, 2007	Carries an imager, spectrometer, and gamma ray and neutron detector. Currently observing the asteroid Vesta, plans to move on to the asteroid Ceres.	$446 Million
Hayabusa 2	JAXA, 2014 (planned)	Plans to create an artificial crater on asteroid 1999 JU3and return samples that have not been exposed to sunlight and solar winds.	$367 Million
OSIRIS-Rex	NASA, 2016 (planned)	Plans to study C-type asteroid 1999 RQ36 and bring >60 grams of surface sample back to Earth.	$750 Million



The location of NEAR Shoemaker's landing site from an orbital altitude of 200 kilometers (see the tip of the arrow). Mosaic of images 015246034-015246840.

Courtesy NASA/JHU/APL.



NEAR Shoemaker's image taken from a range of 250 meters. **The image is 12 meters (39 feet) across**. The cluster of rocks at the upper right measures 1.4 meters (5 feet) across. Image 0157417133.

Courtesy NASA/JHU/APL

FIGURE 1.4. Images of the asteroid 433 Eros taken by the NEAR Shoemaker spacecraft.

1.3.1 OSIRIS-REx

FIGURE 1.5 shows an Apollo asteroid called 1999 RQ36. This asteroid is the target for the $750M NASA New Frontiers OSIRIS-REx mission[1] (Origins Spectral Interpretation Resource Identification Security Regolith Explorer). The goal of the mission is to return samples to Earth for further study. The asteroid is a potential Earth impactor[2]. It has a mean diameter of approximately 493 meters for which most of the information acquired came from observations at the Arecibo Observatory Planetary Radar and the Goldstone Deep Space Network. This asteroid might impact the Earth during one of the 8 close encounters between 2169 and 2199. The probability of an impact is 0.07% or less[3].

However to better determine that probability more information is required, such as the detailed shape of the asteroid to determine a magnitude of the Yarkovsky acceleration. The Yarkovsky describes the slight pressure exerted on the asteroid surface by the sun. The Yarkovsky effect pushes asteroids with prograde rotation outward in their orbit; conversely it pushes asteroids with a retrograde rotation inward in their orbits. Thus, the Yarkovsky effect can push asteroids into Earth-crossing orbits.

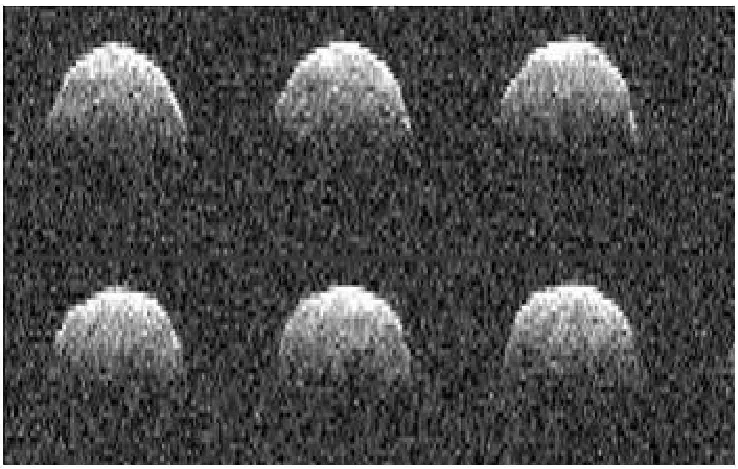
The reason for selecting 1999 RQ36 for the Osiris-Rex mission was not because of its relatively high probability to impact the Earth, but rather due to the comparatively low ΔV above Earth –escape velocity required to reach it[4]. One would think that due to its proximity and low ΔV, the mission would take a

[1] http://www.nasa.gov/topics/solarsystem/features/osiris-rex.html
[2] http://web.archive.org/web/20110721050910/http://neo.jpl.nasa.gov/risk/
[3] Andrea M., S. Chesley, M. Sansaturio, F. Bernardi, G. Valsecchi, O. Arratia, (2009). "Long term impact risk for (101955) 1999 RQ36". Icarus 203 (2): 460–471, doi: 10.1016/j.icarus.2009.05.029.
[4] http://echo.jpl.nasa.gov/~lance/Δ_v/Δ_v.rendezvous.html

short time. Not so. The spacecraft is scheduled to launch in 2016, reach the asteroid in 2019, and return samples to Earth in 2023, 7 years after the launch.

This mission will not land on the asteroid. Instead, it will be another *touch and go* mission. The sampler will be deployed from a long robotic arm, approach the surface at 0.1 m/s, and using gas fluidize regolith to collect a sample in approximately 5 seconds. The pneumatic sampler looks like a car air filter; a minimum of 60 grams will be trapped inside the filter part (some powder will also get stuck to the sticky surface of the sampler). After acquisition, the sampler will be inserted into the Earth return capsule – the same one as used on the Stardust mission. One of the main reasons for deploying the sampler from a long arm rather than landing on the surface is to protect spacecraft's large solar panels from hitting the asteroid surface and suffering damage. Little is known about this asteroid, so it is tough to design a sampling mission not knowing much about the target material.

It's interesting to note that $750M is being spent to go to an Asteroid about which researchers know so little. That`s quite risky, of course, but this is a cold reality of Asteroid missions: we will not know enough about the asteroid until we get there. Hence, the landed mission must be able to work on a wide range of potential surfaces.



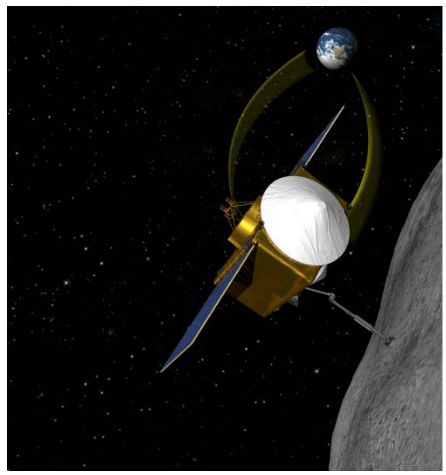


FIGURE 1.5. The $750M OSIRIS-REx Asteroid Sample Return Mission.

In FIGURE 1.5, please note: Left: Doppler imagery of the asteroid target 1999 RQ36 (by NASA's Goldstone Radar). Right: the OSIRIS-REx spacecraft. Note the long sampling arm. The spacecraft will employ a "touch and go" sampling approach – it will not land.

1.3.2 NEOs Summary

In summary, **the space exploration community has never landed <u>and</u> performed in-situ operations on a NEO (or a small body).** Why? The answer is that landing on a NEO (or any small body) is extremely difficult for many reasons, the biggest of which is that we will not really know much about a targeted NEO until we get there. There are some hazards related to sampling for example - if a NEO spins too fast, it could potentially damage large solar panels during touch down and prematurely end the mission. Therefore, we will need to make a careful selection of NEOs and other asteroids that offer a stable, single axis rotation. In addition, any anchoring, mining, and mineral processing technologies will have to be universal – i.e. work with a range of materials and surface conditions.

1.4 The Space Infrastructure Development Framework

The economic premise of RAP is that humans will develop an infrastructure for living and working in space. In this century, this infrastructure will grow to support hundreds of people and eventually thousands of people across the Solar System. We composed a *space infrastructure development framework* to characterize the growth of this infrastructure both in time and in the number of people living continuously in space. These space settlers will create a demand for commodities processed and products manufactured in space. The earliest commodity for which we see this demand is water. Water exists on the Moon and in the asteroids. The Delta V to return water from an asteroid can often be less than to enter and escape from the Moon's gravity well. We believe that water from asteroids can present a comparative advantage over lunar water and an absolute advantage over water from the Earth. In addition, the *space infrastructure development framework* prepares to accommodate other asteroid products including structural metals (Al, Fe, Mn, Ni, Si, Ti), platinum group metals (PGM), regolith for radiation shielding, regolith to provide soil for agriculture, and scientific samples as a commodity item for education and industrial research. We are designing the RAP spacecraft to play a leading role in building this space infrastructure and supplying the people who will live within it.

1.5 Asteroid Mining Technology

Asteroid mining is still in its infancy. There has been very little funding for the development of any excavation and processing technologies applicable to Asteroids. The majority of technology development effort focuses on the Moon, and to some extent on Mars. Only a small fraction of the technologies developed for these two bodies, could in fact apply to the asteroids. For all practical purposes, the excavation and processing technologies, except for a few isolated cases, are at TRL 1/2. Excavation technologies are much easier to develop and in turn, we can estimate the development cost and timelines more easily. However it is very difficult to assess costs and timelines for the development of processing technologies since they depend on fundamental chemical and physical interactions of individual atoms. It is therefore advisable to start development phases of processing and to some extent excavation technologies as soon as possible since these will take the most time. An adequate baseline funding effort over a long duration would be more beneficial and productive than a short-term funding "surge" at a higher level.

1.6 Mission Design

The purpose of the RAP Mission Design is to devise the mission architecture to enable the RAP spacecraft to fulfill its objectives of flying from a staging point to asteroids, mining valuable material, processing it on site, and returning it to the staging point. To accomplish these objectives, the RAP mission design innovates particularly in staging, trajectories, and the propulsion system. The RAP will be a "true spacecraft" intended to operate only within space, particularly in deep space outside of the Earth's atmosphere, gravity well, and protective magnetosphere. The RAP spacecraft's staging point will be an Earth Moon Lagrange point (EML) from where the added Delta V to reach many asteroids and the Martian moons is quite modest. RAPs mission design addresses the problem of the Earth's long synodic period with many NEAs by introducing gravity assist swing-bys of Venus and Mars that not only decrease the Delta V requirements but also shorten the synodic period, thereby expanding the departure and return windows. The RAP's propulsion system will innovate in using solar thermal propulsion (STP), which we believe is uniquely suited for asteroid missions. STP takes advantage of the moderate *I*sp available from this technology and in our adaptation will use water as the propellant. In this way, the RAP spacecraft can refuel itself at the asteroid for its return journey, greatly reducing both its outbound dry and wet mass. The mission design includes the concept of operations for flying to the asteroid, maneuvering around it, aligning with the asteroid's pole, matching rotation, attaching to the asteroid, and then returning to the EML staging point.

1.7 Spacecraft design

The solar thermal propulsion system provides the central organizing principle for the RAP spacecraft. This principle comes from the triplex capability that STP can provide: thermal energy to rapidly expand the propellant and generate thrust, process heat for the mining and processing operations, and solar dynamic electrical power to support the spacecraft's functions. A longitudinal truss of triangular cross-section serves as the spacecraft's main structure. This truss mounts the STP engine at the aft end with the parabolic solar concentrators to either side. At the front end, the truss supports the mining equipment. This mining equipment includes the asteroid anchoring and capture mechanisms, along with the material extraction and processing systems. Behind these "front end systems," the RAP will store the water or other materials produced in tanks mounted on the truss. The truss also furnishes the mounting accommodations for all the spacecraft subsystems where they will be easily accessible for servicing, repair, or replacement. The RAP spacecraft design emphasizes maintainability and reuse, so that each vehicle can fly multiple mining missions.

1.8 Parametric Cost Model

The parametric cost model presents an estimate for the economics of the RAP concept. This model consists of a cost estimate to develop, build, and operate the spacecraft in comparison to the revenue from selling commodities produced from mining asteroids. The cost of developing the RAP spacecraft and its mining systems will be on the order of $2.5 billion. The parametric cost model examines the market price and sales over time necessary to recoup this investment and to begin making a profit. This model shows that ~$5K/kg is an achievable price point for water returned to Cis-Lunar space from an asteroid and that lower prices will become possible after the development and production costs of the mining spacecraft have be amortized.

2 Space Infrastructure Development Framework

The RAP project's approach to understanding the prospective markets for space resources was to develop a qualitative Space Infrastructure Development Framework, shown in TABLE 2.1. To envision the start of the deep space economy, the RAP team constructed this Framework to model the values and variables of nascent space commerce. This model describes the potential market, customers, and capital funding for the development of human habitation and industry in space. This human development will include space infrastructure, colonies, settlements, stations, and mining and processing operations. The "background reference" is NASA, which is the only large-scale human spaceflight organization with sufficient openness to understand how it operates and makes decisions. This reference applies particularly with regard to the NASA budget, which serves as a meter stick against which to measure space investment.

2.1 15-Year Time Phasing.

This TABLE 2.1 is arranged along the horizontal axis into progressive phases of 15 years over 60 years from 2010 to 2070. The 15-year increment is significant for several reasons.

First, it marks the nominal period necessary to develop a major human spaceflight program. The Space Station development program took 15 years from 1984 when President Reagan announced "Space Station Freedom" until 1999, when the Russian Space Agency launched the Zarya Service Module, the first "Functional Base Block" of the International Space Station (ISS). The current Orion Multipurpose Crew Vehicle (MPCV) program began with President Bush's "Vision for Space Exploration" and Constellation Program in 2004 and now expects a first crewed flight in 2018 – a 14 year period, (assuming no schedule slips or budget decrements).

Second, 15 years constitutes the "half-life" of a standard NASA civil service career of 30 years until full retirement. This time phasing poses a significant lesson insofar as it is necessary to support and maintain a core complement of experienced civil service engineers and scientists to carry out a major development program. The continuity of this experience is vital to sustaining the agency's ability to develop new major crewed spacecraft, space stations, lunar and planetary habitats and bases.

This assertion is not an empty platitude. One example from the legacy of the Apollo Lunar program should suffice to explain how essential it is to continue the corporate memory and pass that knowledge from generation to generation – regardless of whether that corporate memory resides among government employees, major aerospace contractor personnel or "NewSpace" entrepreneurial startups.

The Apollo 17 LM Ascent Stage lifted 110kg of lunar samples plus 110kg of EVA suits for a total of 220kg of cargo up mass) in launching from the lunar surface on 14 December 1972 (Wikipedia, 2013, http://en.wikipedia.org/wiki/Apollo_17, retrieved 15 June 2013; NASA, 2011, http://www.nasa.gov/mission_pages/apollo/missions/apollo17.html, retrieved 15 June 2013). However, NASA's team of very young, inexperienced engineers who had received little or no orientation to the achievements of the Apollo Program thought they were achieving a new milestone for the Altair Lunar Lander by giving it an up-mass of 100kg as a "Constellation Driving Requirement" as a "Crew Mission Parameter" (Martinez, 2009 June 1-5, p.5; Cohen, 2009 July 14-16, p. 8). Later, in 2011, NASA management discovered this discrepancy and increased the Altair up-mass to 250kg, not including the lunar surface suits that would remain behind in the airlock module, but with the caveat that no more than 100kg would be available for scientific samples.

It bears repeating: Commercial space companies or other space organizations are not exempt or immune from the need to create and sustain the corporate knowledge. Nor is there any evidence yet that "commercial" space companies can develop deep space vehicles or habitats on their own faster than the historical NASA cycles. For example, Bigelow Aerospace Corporation licensed the TransHab patent soon after Congress cancelled the TransHab module planned for the ISS circa 1998. In 2006 and 2007, Bigelow launched the Genesis 1 and Genesis 2 subscale prototype inflatable modules to LEO on Dnepr rockets from Dombarovskiy Cosmodrome in Russia. Now, 15 years after TransHab, Bigelow is predicting a first human-occupied BA330 module in LEO by 2017 – a 20-year development cycle. Given the 15-years long development cycle, the math indicates that a typical space design and engineering professional can participate in two to three major development programs during a single career.

2.2 15-Year Investment in Total Deep Space Infrastructure

Line 2 of TABLE 2.1 shows the projected total investment in deep space infrastructure during each 15-year period. This investment can come from any source: government, commercial, or other sources. For the period 2010 to 2025, the total investment is $25B. That investment amounts to an average of $1.67B per year. Each subsequent period shows a projected doubling of investment. Deep space investment does not include the ISS because, being in LEO and within the Van Allen belts, it is not in "deep space," since it is still within the protective cocoon of the Earth's magnetosphere. This model posits a simple and perhaps simplistic doubling of investment from one 15-year period to the next.

2.3 Rate of Investment in Deep Space Infrastructure

Line 3 shows the projection for the rate of investment in deep space infrastructure occurring in the culminating year of each 15-year period. In this sense, it represents the "instantaneous" rate at that points on the curve – in effect, the slope of the tangent line. The unit of currency is stated in terms NASA Yearly Budget (NYB). This NYB unit affords a basis of comparison to what it costs to run the largest space agency on this planet. So for 2025, if the instantaneous rate of investment will be 0.2NYB, the simplistic total that year would come to:

$$\$17\mathrm{B} \times 0.2 = \$3.4\mathrm{B}$$

This metric does not distinguish directly between government (e.g. NASA) investment in deep space infrastructure and private or commercial investment. However, as this rate of investment grows over time, eventually it exceeds the actual NASA annual budget. Therefore, by definition, the commercial investment intercepts and exceeds the government expenditure.

2.4 People Living in Space Continuously

Line 4 indicates the projection for the number of people living continuously in space at the end of each 15-year increment. This population creates the source of demand for commodities, consumables, and products produced and delivered in space. This project uses the term living "continuously" instead of permanently, because that would imply that the people would not return to Earth. Rather, these numbers means that there would be a given number of berths available that will be continuously occupied by crew members or inhabitants, who would be free to rotate back to the Earth at the end of their "mission," tour, or sojourn. The Space Infrastructure Framework does not require people to move permanently to space and to live out the remainder of their lives there, although it is likely that segment of the population will grow as some fraction of the total habitation.

The first year stated is 2010, about when six people began living continuously on the ISS. The growth projection for 2025 shows a range between two numbers, from a simple doubling from six at the low end to 18 at the high end according to the following constrained growth equation, where the exponential increase of the numerator is tempered by the denominator which is 2 to the exponent of the sequential growth period.

$$Year2 = \frac{(Year1)^2}{2^{period}}, so, \frac{6^2}{2^1} = 18\ people\ living\ continuously\ in\ space\ in\ 2025$$

The exponent in the denominator is 1where the period is 1 for the first 15-year increment, 2 for the second increment, and so on. This denominator embodies a risk-constraint function that "puts the brakes" on an otherwise unfettered exponential population growth. One way to understand this risk-constraint is that presently there is at present only one contender for deep space exploration – NASA. By 2025, it is likely that as many as 4 of the NewSpace companies, will become contenders to send humans beyond LEO (e.g. Inspiration Mars/Paragon SDC, SpaceX/Virgin Galactic, Bigelow, Boeing, and MarsOne). This risk-constraint function suggests that the likelihood of any them succeeding is the inverse of the number of contenders. This insight may seem counter-intuitive, but it derives from the notion that the first private deep space mission will be so expensive that it will require a near-monopoly on investors and their investments for a period of at least several years. It also holds open the possibility of these actors forming together into larger teams than were needed for the NASA Commercial Crew and Cargo or, say, the Google Lunar-X Prize.

As this series of calculations expands out to the 5th period, the in space population ranges from 96 to 21,012, which admittedly is a wide point spread. This large range reflects the current uncertainties about how realistic the commitments are from all parties to make *homo sapiens* a true spacefaring species, and to commit the resources and effort necessary to accomplish it. Like most such estimated ranges, the likely outcome will lie somewhere in the middle. Admittedly, this analytical approach is crude and starving for data, but it helps to provide the larger *framework* to conceptualize the deep space infrastructure and the economy that will demand it.

2.5 Where Space Commodities and Products are consumed

Line 5 sets up the distinction between points of purchase in space versus back home on the Earth. The columns below them are split in two to reflect this bifurcation of the market.

2.6 Target Return on Investment (ROI)

Line 6 provides a row to indicate the target return on investment that would attract investors to the highly speculative in risk-fraught space infrastructure. The Parametric Model for RAP in section 8 below suggests a 25% ROI for platinum group metals (PGM) returned to Earth. However, there is not nearly enough data to make any entry in this row for now except "to be determined" (TBD).

2.7 Commodities Produced in Space

Below Line 6 appear rows for five characteristic and even archetypal commodities that space industry might produce as the harbinger of the deep space economy. The key point of these double columns is the distinction between Earth and in space consumption. It soon becomes clear that there would be many more commodities in demand in space than back on Earth, where the cost of return and reentry would become as prohibitive for earth customers as launch costs to consumers in space.

2.7.1 Commodity 1 Water and PGMs

Commodity 1 for in space use would consist of water as propellant for cislunar and interplanetary spacecraft. The demand for water for these applications, either to electrolyze into LOX and LH_2 or to use directly as a propellant for solar thermal or nuclear thermal engines will become tremendous. This market assumes that the water collected from carbonaceous chondrites, from regolith ice, or from chemically bound sources, can serve as propellant with little or no post-extraction processing.

2.7.2 Commodity 2 Scientific Samples and Radiation Shielding

The second tier commodities of interest consist of regolith for radiation shielding in space and the same materials excavated from the surface of the Moon, Mars, asteroids, Phobos, or Deimos for scientific samples. Once a true cislunar, NEA, and interplanetary transportation system becomes available, the problem of delivering these commodities to the customers in space will become greatly simplified. Eventually, it will not matter where the source is for these commodities and other products; the price will reflect that factor and that will be all that makes it notable to the customer.

Radiation shielding has emerged time and time again as the leading showstopper to humans living long-term or permanently in deep space. Although water offers many advantages for shielding, particularly as an amorphous material that can be stored in the environmentally controlled interior of a spacecraft, that solution comes with the downside that filling water bladders will eat up much of the habitable volume within the spacecraft. An alternative solution will be to attach regolith tiles or bricks to the exterior of a space habitat module. Attaching these ISRU-produced shielding units can provide protection not only against the constant stress of GCRs, but also micrometeoroids, and even enhance thermal stability, reducing the need for cooling and heat rejection.

These scientific samples may include revolutionary, one of a kind new discoveries by deep space exploration vehicles. However, what is more likely, they will consist of samples extracted, packaged, preserved, and returned for use in university and industry laboratories. The analogy to his type of market might be Doc Ricketts in John Steinbeck's Cannery Row, who ran a business collecting specimens of marine life and selling them to university, pharmaceutical, and other industrial labs. Of course, RAP will not expect to find any life forms, but the idea is similar: to collect, process, and sell samples to researchers who are far from the source.

2.7.3 Commodity 3 Structural Materials

A critical milestone in establishing any type of settlement, colony, or civilization arises from the ability to build its own shelter and patterns of settlement. For many decades, including throughout the period shown on the *Space Infrastructure Framework*, only manufacturers on Earth will be able to produce pressure vessels for habitats in which the crew can live and work safely, productively, and happily. However, there will be many secondary facilities, structures, and civil works that the in space population will need. These facilities range widely from a landing zone or pad made from sintered regolith to aluminum cranes or derricks to support mining, manufacturing, fabrication, and assembly operations. These abundant structural materials include aluminum, iron, silicon, and titanium, among many elements found on the celestial bodies of interest. As the space-based demand emerges for new infrastructure on the surface of the Moon, Mars, or other bodies, the opportunity to source these facilities mostly or entirely in space will arise, in competition with sourcing from Earth.

2.7.4 Commodity 4 Life Support

Commodity 1 exploits water for propellant with minimal post-extraction processing. For the life support application, the water will need considerably more processing to make it fit for human consumption or for use in drinking, hygiene, laundry, and agriculture. This refinement of the ability to provide "the universal solvent" to the deep space population will be a key to supporting and sustaining habitation permanently in deep space. Ideally, space colonies will choose to locate close to a good source of water, in much the same way as early settlers on Earth set up camp close to a river, spring, or other good source.

2.7.5 Commodity 5 Regolith for Soil

Regolith appears in Commodity 2 as a product for radiation shielding or scientific samples. However, there is another, potentially larger demand for regolith in agriculture. Under nearly all scenarios for deep space exploration, the crews will bring all the food they need with them from Earth. Sure, they may grow some vegetables in an aeroponic "salad machine" or other such relatively compact device, but none of the prospective missions come close to thinking about self-sufficiency in food. Never the less, to establish a permanent colony outside the Earth-Moon system, producing all or nearly all the food will become a necessity for self-reliance and ultimately for survival. These colonies on the surface of a body with some gravity, however slight, will not need to be limited to the aeroponics that are de rigeur in microgravity. Instead, they can take advantage of the presence of gravity to feed water and nutrients to the plants' root systems. Instead of spraying water on the naked roots, these plants should be able to take root in a solid material. Certainly regolith with no organic material – let alone humus – will hardly suffice for soil, it may be possible to process the regolith to reduce or remove the reactivity of its oxides and other constituents. Delivering regolith-derived soil for agriculture at permanent colonies or bases could furnish a valuable market.

2.8 Location and Technology Decisions

Having laid out the reasoning for the Space Infrastructure Development Framework, the RAP team took the next step to look at location and technology selection decisions. TABLE 2.2 outlines these types of decisions correlated against Customer- or Market-related factors, Supply-related factors, and Economic Feasibility-related factors.

2.8.1 Customer- or Market-Related Location Decision

Location impacts on customer demand include source and availability of product feedstocks, distance between these products and the market or end customer (typically expressed by transportation energy or ΔV), and availability as well as location of supporting infrastructure or systems nodes. Due to asteroid diversity, as well as the current level of uncertainty regarding compositions, the choice of location or orbital characteristics is felt to be of higher influence on demand due to the direct relationship to transportation energy (as expressed byΔ-V as well as frequency of transfer opportunities) to the cost of supply of said resource.

TABLE 2.1 Space Infrastructure Framework

Metric	Recent		Near-Term		Intermediate-Term		Far-Term		Very Far-Term	
1. Milestone Year (Approx.)	**2010**		**2025**		**2040**		**2055**		**2070**	
2. 15 year Investment 2013 $B in Deep Space Infrastructure			**25**		**50**		**100**		**200**	
3. Rate of Investment in NYBs* at Milestone Yr	**0**		**0.2**		**0.4**		**0.8**		**1.6**	
4. People Living Continuously in Space	**6**		**12-18**		**24-81**		**48-820**		**96-21,012**	
5. Where Consumed	**Space**	**Earth**	**Space**	**Earth**	**Space**	**Earth**	**Space**	**Earth**	**Space**	**Earth**
6. Target ROI			**TBD**	**TBD**	**TBD**	**TBD**	**TBD**	**TBD**	**TBD**	**TBD**
Commodity 1	Water		Water	PGM	Water	PGM	Water	PGM	Water	PGM
Commodity 2			Rad Shielding	Science Samples	Rad Shielding	Science Samples	Rad Shielding	Science Samples	Rad Shielding	Science Samples
Commodity 3					Structural Materials		Structural Materials		Structural Materials	
Commodity 4					Life Support		Life Support		Life Support	
Commodity 5					Regolith for Soil		Regolith for Soil		Regolith for Soil	

NYB = NASA Yearly Budgets =~$17B in FY 2013

This page is intentionally left blank.

System transportation nodes or refueling points will bring a strong mitigating influence on this cost structure and can be directly modeled when searching for optimal economic solutions. Asteroid composition tradeoffs will be addressed as they relate to sources of customer demand, including terrestrial metals markets as well as projected in-space markets for volatiles, construction and shielding materials, and other consumable items.

TABLE 2.2. Economic Trade and Analysis Matrix for Asteroid Resources.

Economic Factors	Location Decision	Technology Decision
Customer or Market-Related	• Source & Availability of Products • Distance between Product & Market (as expressed by transport cost or energy) • Availability & Location of Supporting Infrastructure (systems nodes)	• Customer Willingness to Pay (can be approximated by need or dependency of customer's ops on regular product supply) • Cost of Baseline Product Supply (typically assumed to be terrestrial – forms basis for price or value)
Supply-Related	• Source of water for propellant may start with extraction on the Moon and delivery to EML1 • Other sources for in-space propellant may be extended to include Phobos, Deimos, or the asteroids. • Propellant for returning from an asteroid may be extracted locally from that asteroid. • The source of each economically viable mineral concentrate or produce may turn on its location in terms of ease of accessibility, including synodic period, orbital inclination, and the distance to intercept.	• If an asteroid looks highly promising (e.g. has demonstrated reserves or high purity platinum family or rare-earth metals) it may influence the choice, type, and level of investment in new technology. • The choice of technology may also affect the timing of the mission and the duration of the mission design.
Economic Feasibility-Related	• Cost of Capital (with approximate degree of government leverage) • Net Present Value (NPV) at project, enterprise and architectural levels • Risk-Adjusted Return on Investment	• Cost or Intensity of Research & Development • Capital vs. Ops cost ratio (basis for value of reusability) • Return on Technology Investment (can be measured by modeling impact of trade-offs on value of project, enterprise or architecture)

2.8.2 Customer- or Market-Related Technology Decision

Technology impacts on customer demand include the customer's willingness to pay for the asteroidal product under examination, as well as the cost of the equivalent baseline (or Earth-derived) product including the cost to transport that Terrestrial product to its point of use in space. Customer willingness to pay can be approximated by "need" or "dependence" of the customer's operations on regular product supply, which is a direct function of the technology choices. For example, expendable or "open-loop" life support systems will use and deplete a steady stream of Oxygen as a function of crew size. Closed-loop life support systems (a different technology choice) will recycle much of the gas, dramatically reducing Oxygen demand for breathing. The anticipated technology choice is assumed to be a function of technology maturity as well as perceived risk to the customer application. The cost of a baseline product supply (typically assumed to be terrestrial) will form a basis for price or value of the in-space resource. While terrestrial commodity precious metals are the best example of this (and are technology neutral with respect to the customer demand side), the cost of providing Earth-derived propellants for refueling vehicles in space does have a strong technology influence.

2.8.3 Supply-Related Location Decision

Location impacts on cost of supply are related to the energetic proximity as well as frequency of resource availability. Strong influences are anticipated due to transportation system choices (esp. propellant options), the degree of transportation system reusability, operational considerations and level of annual systems maintenance. Reusability, operations, and maintenance assumptions as well as direct refueling unit costs will determine the cost of supply. The capital replacement schedule will also play an important role in maintaining steady supply given the attrition and need to replace used equipment.

2.8.4 Supply-Related Technology Decision

Transportation choices will dominate the technology impacts on cost of supply, but critical factors will also include product extraction and refining costs. The RAP project will test this conjecture. Cost of supply will rise in direct relation to the requirements for custom or unique technology. The availability of COTS (commercial off-the shelf) solutions will mitigate this somewhat (presumably some subset of terrestrial mining technology will be usable in space e.g. drilling bits). In general, the development v. capital cost ratio will always drive the price of custom hardware upward. Once new technology is put into production, the development costs can be amortized over the production lifetime.

2.8.5 Feasibility-Related Location Decision

Location impacts on economic feasibility will utilize standard decision criteria or metrics including cost of capital, net present value, and risk-adjusted rate of return. Effective cost of capital can be reduced due to government leverage in the form of public-private partnerships, technology development programs, and guarantees on the purchase of products. For this reason, cost of capital will be an input variable in the feasibility modeling. Net Present Value (NPV) at project, enterprise and architectural levels as well as risk-adjusted Return on Investment (ROI) are the most common decision variables used in making business decisions, and will form the bottom line outputs for determining feasibility.

2.8.6 Feasibility-Related Technology Decision

Technology impacts on economic feasibility include cost of research & development, capital vs. operating cost ratios for various systems, and return on technology investment. Capital vs. operating cost ratios will form the basis for evaluating the value of reusability, and will be an important secondary output of the economic modeling work. Finally, return on technology investment can be estimated by modeling the direct influence of systems-level trade-offs (such as chemical vs. solar thermal propulsion) on costs at the project, enterprise, or architecture level.

2.9 Mineral Economics

Platinum group metals (PGMs) may prove to be a diversion from the main future market for asteroid materials: the customers that will be located in space. These potential customers may be much more interested in utilitarian metals such as Ti, Al, Ca, H_2O, Fe, Ni, and Mn than platinum group metals, and they will consume those products in locations beyond low-Earth orbit. The early need to find construction materials, propellant, and radiation shielding drives this demand. It is possible that later demand will arise for PGMs, REEs, and other exotic elements or materials for in-space industrial use.

2.9.1 Terrestrial PGM Utilization

We hold open the possibility that there may be a way to make a profit on returning platinum group metals (PGMs) to Earth, but doubt that it would be feasible as a first asteroid mining venture. TABLE 2.3 shows the distribution of PGM uses among the market segments on Earth today. The current market price of refined platinum is about $51,000/kg, and slightly less for rhodium.

TABLE 2.3. PGM Demand in Five Market Segments (data from Butler, 2012)

Element	Autocatalyst	Electronic	Industrial	Investment	Jewelry
Ruthenium	0.0%	62.1%	37.9%	0.0%	0.0%
Rhodium	93.1%	0.0%	6.9%	0.0%	0.0%
Palladium	71.4%	0.0%	29.3%	-6.7%	6.0%
Osmium	0.0%	0.0%	99.9	0.0%	0.0%
Iridium	0.0%	54.8%	45.2%	0.0%	0.0%
Platinum	38.4%	0.0%	25.3%	5.7%	30.6%

2.9.2 Market Segmentation

The RAP project anticipates two primary marketplaces for asteroid materials: customers in space and customers on Earth. This natural segmentation comes from the energy required to lift ‘competing’ products into space. Earth demand is known. Space mining will compete against an established, low-cost source on Earth for every element under consideration. We did not find any current economic scenario in which it might be profitable to mine resources in space and sell them to the market on Earth based on the current high cost of space access as well as the per-kg average cost of spacecraft manufacturing. Sale of some products on the Earth may be the icing on the cake, but it is not the cake itself. The preliminary conclusion is that the only feasible path to long-term asteroid mining profitability appears be through the space market.

2.9.3 PGM Market Elasticity

The platinum family metals comprise five primary market segments: autocatalyst, electronic, industrial, investment and jewelry. The jewelry and investment segments drove almost 40% of the platinum market in 2011. (See Table 1 below – after Butler, 2012). A market has zero elasticity if the price does not vary with changes in supply or demand. For platinum and palladium, industrial demand is steady and growing, but the investment and jewelry segments are highly elastic (note the negative value of palladium investment demand in 2011 indicating dumping of investment bullion into the autocatalyst segment). If the bullion price goes down, the bottom typically falls out of the investment segment. For future asteroid PGMs, it will be important to look at potential in-space uses to offset the reduced relative contribution of investment in total demand.

FIGURE 2.1. Setting the Capstone on the Washington Monument, 06 DEC 1884, Contemporary Artist's View (showing the point sharper than it actually is).

2.9.4 Processing Improvements in Extractive Metallurgy

Processing improvements in extractive metallurgy could transform markets exponentially, and in ways that are difficult to foresee. Look at the example of Aluminum since the early 19th century. When the Washington Monument was completed in 1884, the design incorporated an aluminum capstone. See FIGURE 2.1 for an illustration of this "high tech" accomplishment. The capstone was the largest refined Al casting free of pinhole porosity made up to that date, and weighed 2.85kg, 97.5% Al content. The final cost of the pyramid was $225, at a time when, as George J. Binczewski (1995, pp. 20-25) points out, the average worker on the Washington Monument was paid $1 per day. In 1885, the total production of Al by Aluminum Company of America was 28.3kg. The aluminum capstone showed the world that the United States was a technology leader, but at that time the potential uses for aluminum were limited to novelty items.

Adam Smith (1776), in his "An Inquiry into the Nature and Causes of the Wealth of Nations" provides a conceptual basis that can be utilized to analyze the basis for future wealth in a developing solar system. One important conclusion is that free markets facilitate economic growth rates. Smith wrote that gold does not have utilitarian uses; the cost of gold derives not just from the cost of one person seeking it and producing it but also the cost of the efforts of all the people who seek but do not find gold. Conversely, Smith asked what could be more useful than water? However, because water was abundant everywhere on the large damp island of Britain, he concluded the price was and should be very low. In space this is exactly the opposite. Water is valuable due to its utility in propellant, life support, and radiation shielding. Smith's framework explains why water will be so valuable in space.

2.9.5 Abundance and Scarcity

The effects of abundance and scarcity will apply in space mining. For example, at this time the market for radiation shielding in space is zero because the Van Allen Belts and the Earth's magnetosphere shelter the ISS and all current human spaceflight is in LEO with the same protection. The market demand for radiation shielding will stay zero until there is an L2 station, and missions to the lunar surface and beyond. Then, we may meet that demand by providing water from the moon or an asteroid for amorphous shielding, or solid material from an asteroid for solid shielding. One 10 to 20m-diameter asteroid could probably satisfy the market for solid shielding for a long time.

3 Mining and Processing Technology

The RAP approach to prospecting, mining, and processing flows from the preceding TABLE 2 *Space Infrastructure Development Framework*. Our mining strategy takes the form of FIGURE 3 the Hierarchy of Resources and Markets. It lays out the tactical approach to each type of likely candidate resource and the corresponding markets on Earth and in space. The Hierarchy of Resources and Markets shows that in general, the resources obtainable from Asteroids can be divided into 4 broad categories: free water, bound water, metals, and regolith.

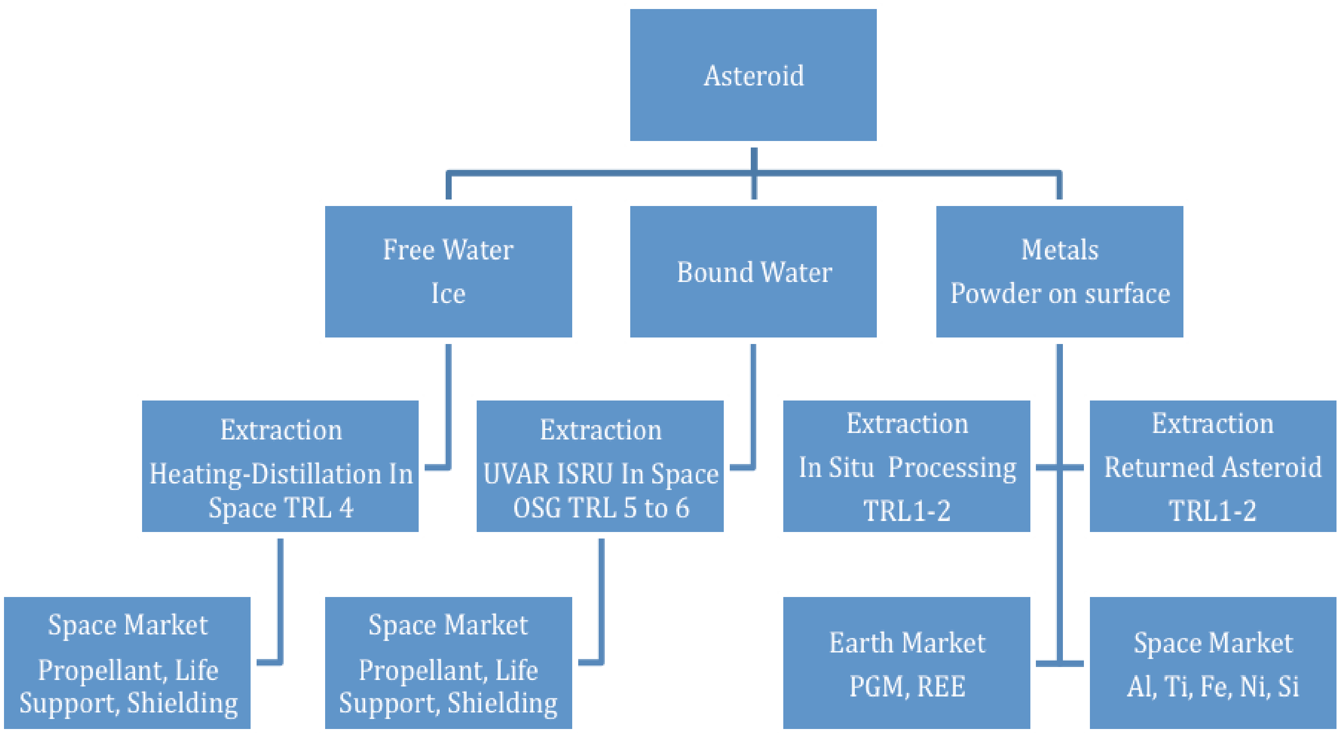


FIGURE 3.1. The Hierarchy of Space Resource Extraction and Markets.

It is a standard practice for terrestrial mines to organize mining operations around the main mineral product, while collecting bonus revenues from 'byproducts' of lesser concentration. In a similar vein, we will not travel all the way to an asteroid to mine just one resource. ***But neither will we be able to develop a "universal mining toolkit" that can extract and process any and all ores that we find on an asteroid or anywhere else.*** We will need to match particular technologies to specific deposits in selected locations. How do we align target body, the type of deposit, the mining technology, and what are the market and the price? On the other hand, the technology to grab and return different types of asteroids will be similar.

We also need to change the way we think about valuable commodities, and recognize the influence of location on value. Value on Earth does not equate to the same relative value in

space. A simple analogy is this: What's worth more to a person is stranded in the middle of a desert: a gallon of water or an ounce of gold?

3.1 Mining Functions

Mining of all types must generally follow these steps: prospecting, excavating/mining, processing (e.g. comminution), extraction, and storage. Comminution is an energy intensive step and hence it should be avoided by mining pulverized regolith instead of small rocks or boulders. TABLE 3.1 explicates these steps.

Free water and bound water would be used in the space environment for life support, radiation protection, and propellant either as electrolyzed LOX/LH2 or as liquid water for Solar Thermal Propulsion). Based on the RAP teams work during Phase 1, the extraction and processing technology associated with this frozen water falls at of TRL 4 component or subsystem laboratory test. .

Metals may be used in space to make structural components for spacecraft and spacecraft subsystems, or brought back to Earth and sold. The most ready-to-hand approach would be to extract regolith dust or powder, feed it into a 3D printer, and then sinter it into various components for spacecraft (e.g. fuel tanks), structures (e.g. trusses), and habitats. Eventually, the technology will evolve to where it is possible to manufacture pressure vessel primary structure that is equal to aluminum, steel, or titanium counterparts made on Earth. The technologies necessary to mine minerals or metals, to extract metals from minerals, to de-alloy metals (from M-type asteroids), or to mine regolith and sinter it, are all at very low TRL. The cost to develop such technologies is not currently known (this topic is dealt with in detail further in this section). However, we believe that initially, low-hanging fruit could be pursued to establish a sustainable market. That low-hanging fruit is water.

Extracting free water is relatively easy – water ice can be sublimed and captured on a cold finger. Water extraction from hydrous minerals requires more heat, and so is relatively easy to achieve. In addition, methods of recovering of bound water from lunar regolith have been developed. Some of these methods could also be applicable to asteroids.

Table 3.1. Mining – Top Level Functions		
Steps	**Explanation**	**Asteroids**
Prospecting	Finding and then defining the extent, location and value of the ore body	It could be assumed that mineral concentration will be relatively uniform on asteroids, unless some minerals are preferentially present in fines or coarse fraction
Excavating	Mining minerals out of the ground • Pneumatic • Magnets (iron-nickel dust, nano-phase iron) • Auger, scoop etc.	3 options possible: Bagging an entire asteroid, delivering excavated resource into extraction plant, excavating and extracting in situ
Extraction	Minerals: Extraction of valuable metals from their ores through chemical (since most metals are present in ores as oxides or sulfides, the metal needs to be reduced to its metallic form) or mechanical (crushing, grinding, and washing that enable the separation of valuable metals or minerals). Metals: If in alloy form (Nickel-Iron) – need to de-alloy (tough to do)	If carbonaceous chondrite – water plus mineral extraction (titanium from ilmenite, nano-phase iron) If metallic - difficult to de-alloy, hence use iron-nickel dust for 3D printing (structures can be much weaker in space than at 1g and no launch loads)

3.2 Asteroid Mining Approaches

FIGURE 3.2 shows our approach to asteroid mining. In general, we envisage that small, inexpensive, fully robotic Asteroid Reconnaissance Probes or ARProbes, whether large or small, will initially study an asteroid, from orbit. These probes would be deployed from the RAP mother spacecraft, fly to a nearby asteroid, acquire samples from various locations, and fly back to the mother spacecraft. The regolith sample from each spacecraft would then be studied to determine the exact fraction of volatiles and minerals. This information would then be used to determine the optimum extraction process and establish initial value of the extracted material.

After the initial reconnaissance step, the actual mining will proceed. The mining approach (and in turn the mining spacecraft) will largely depend on the size of the asteroid to be processed. If an asteroid is less than approximately 20m in size, RAP could potentially capture an asteroid and process it in situ. There is an inherent limitation to capturing and mining such a small object. Assuming that the asteroid is a carbonaceous chondrite with the maximum expected water content of about 13%, the extractable water might not amount to a full return payload or even approach it. However, if the target asteroid proves large enough to yield a full payload of water, it will likely be much too large to capture. In this scenario, the RAP mother spacecraft would deploy a team of smaller mining robot spacecraft, called Spiders. These Spiders could deliver feedstock material back to the RAP mother spacecraft for processing. More advanced Spiders might process material in-situ and deliver only processed material to the RAP. FIGURE 3.3 shows a concept for this asteroid mining system: with Spiders and ARProbes.

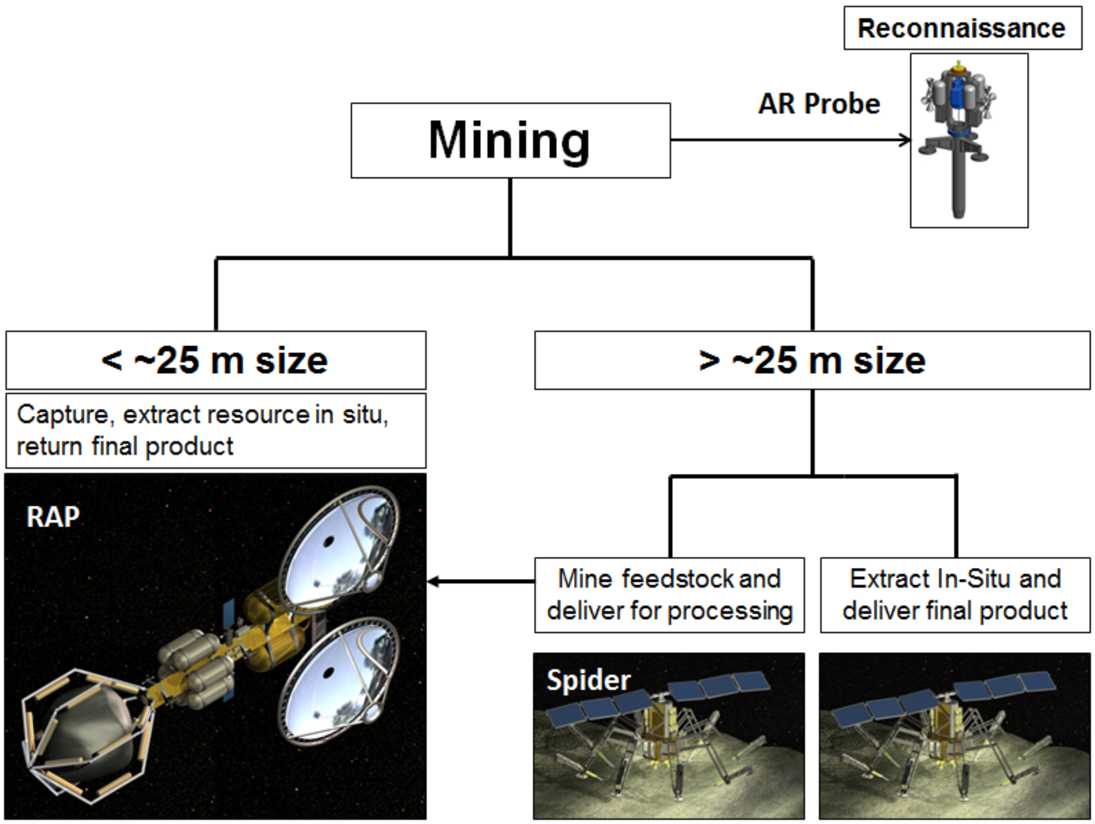


FIGURE 3.2. Asteroid mining approaches.

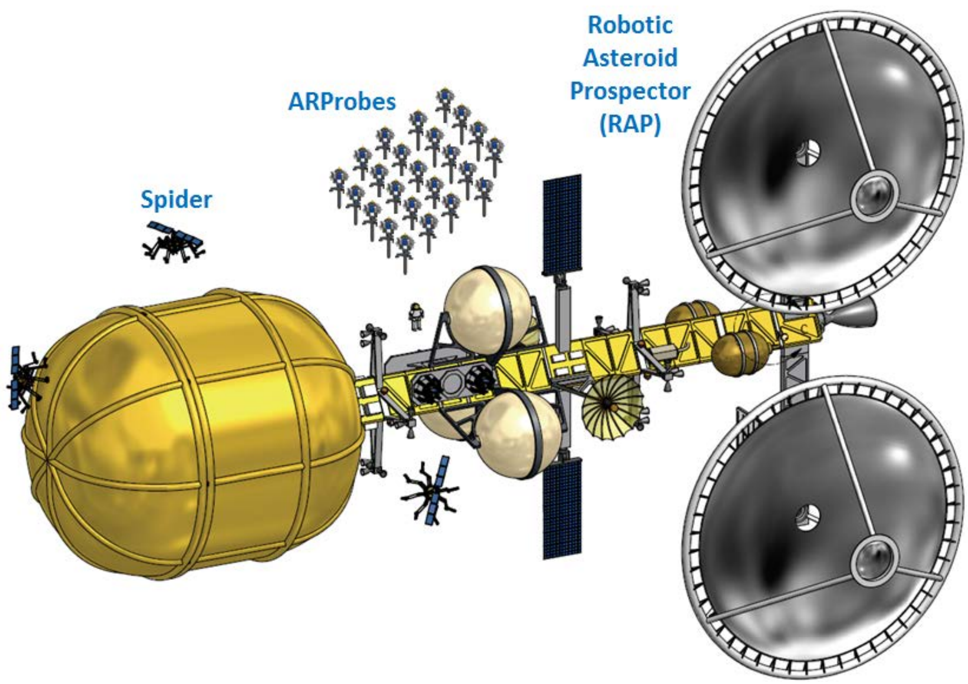


FIGURE 3.3. The RAP Spacecraft carries Spiders and ARProbes.

3.2.1 ARProbes

FIGURE 3.4 shows the Asteroid Reconnaissance Probe or ARProbe. Such a probe is used to acquire small samples for analysis and determination of the regolith mineral grade of fraction of water presence, whether it is a free water or bound water. ARProbes are fully robotic systems. They are small and inexpensive and hence the RAP spacecraft could carry 10s or 100s of them.

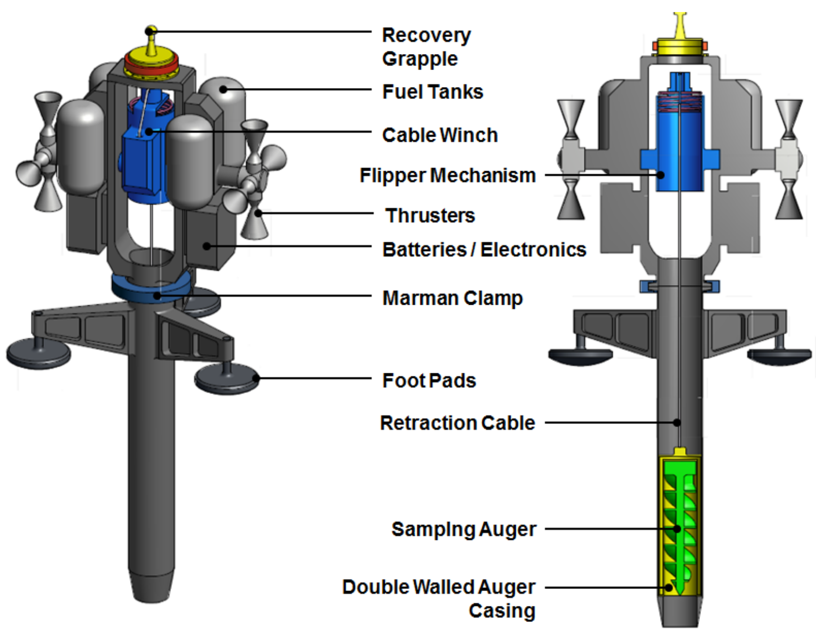


FIGURE 3.4. Asteroid Reconnaissance Probe (ARProbe)

The notional concept of operations for sample return from the surface of an asteroid is described in the following figures. The sequence begins with the parent spacecraft approaching the targeted asteroid. The parent spacecraft may spend several weeks to months orbiting the asteroid and analyzing it with remote sensing equipment, including cameras and stereo mapping. During this time, a potential landing site will be chosen based on the data available. Following landing site selection, the ARProbe will be detached from the spacecraft as shown in FIGURE 3.5 then uses its built in attitude control system to align itself for the descent towards the asteroid. It is envisioned that the attitude control system will initialize a spin stabilization routine to both stabilize the descent, and also potentially aid in penetration of the probe into the asteroid surface. Impact velocity is controlled using an onboard Guidance, Navigation and Control (GNC) system.

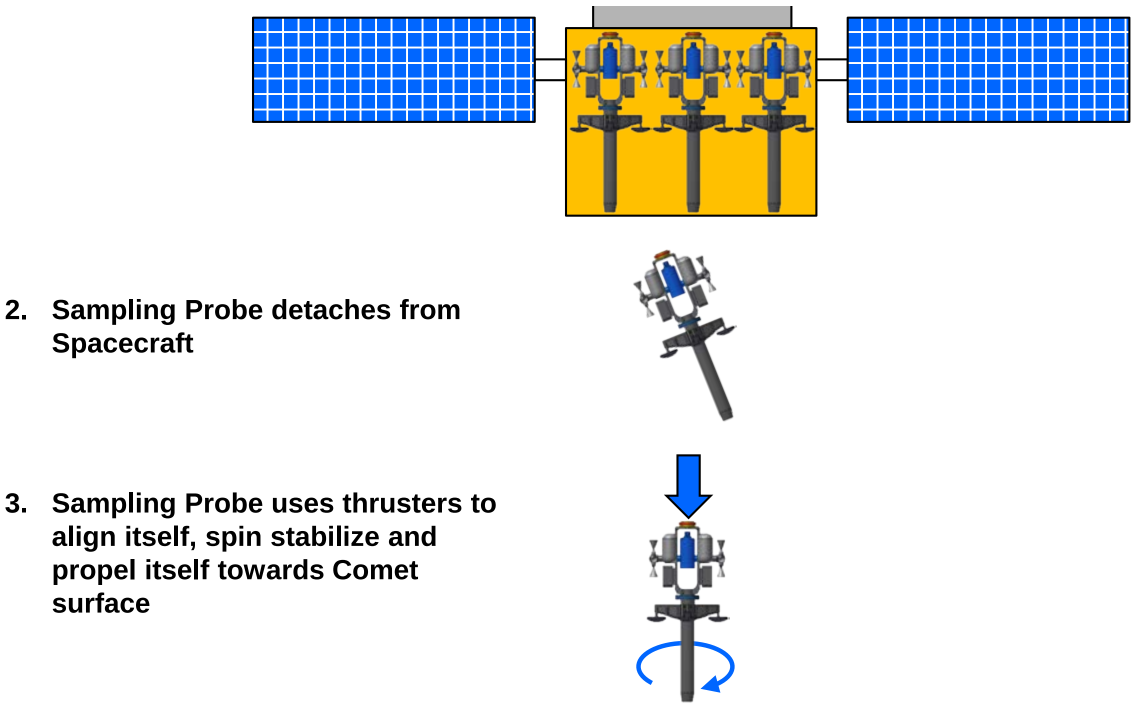


FIGURE 3.5. Concept of Operations – AR Probe Descent

Once the ARProbe reaches the surface of the asteroid, it impacts. The momentum of the descent is determined based on results of laboratory testing, as well as imaging of the asteroid surface by the parent spacecraft to determine surface characteristics. If the asteroid is determined to be a loose conglomerate of fine particles, the velocity of the probe can be commanded as relatively low. If the asteroid is determined to be denser than expected, high velocities can be commanded to insure penetration and sample acquisition. This architecture allows for a broad variety of asteroid surface characteristics. This flexibility is highly desirable due to the relatively unknown physical properties of asteroid surfaces.

Immediately following impact, the ARProbe sends a signal to the parent spacecraft alerting it to its successful impact and sending a system status update. The ARProbe then uses a cable winch mechanism to pull the auger and casing out of the asteroid and into the upper stage of the probe (Flipper Mechanism). When the auger and casing are fully retracted into the upper stage, a spring is compressed, maintaining tension on the cable.

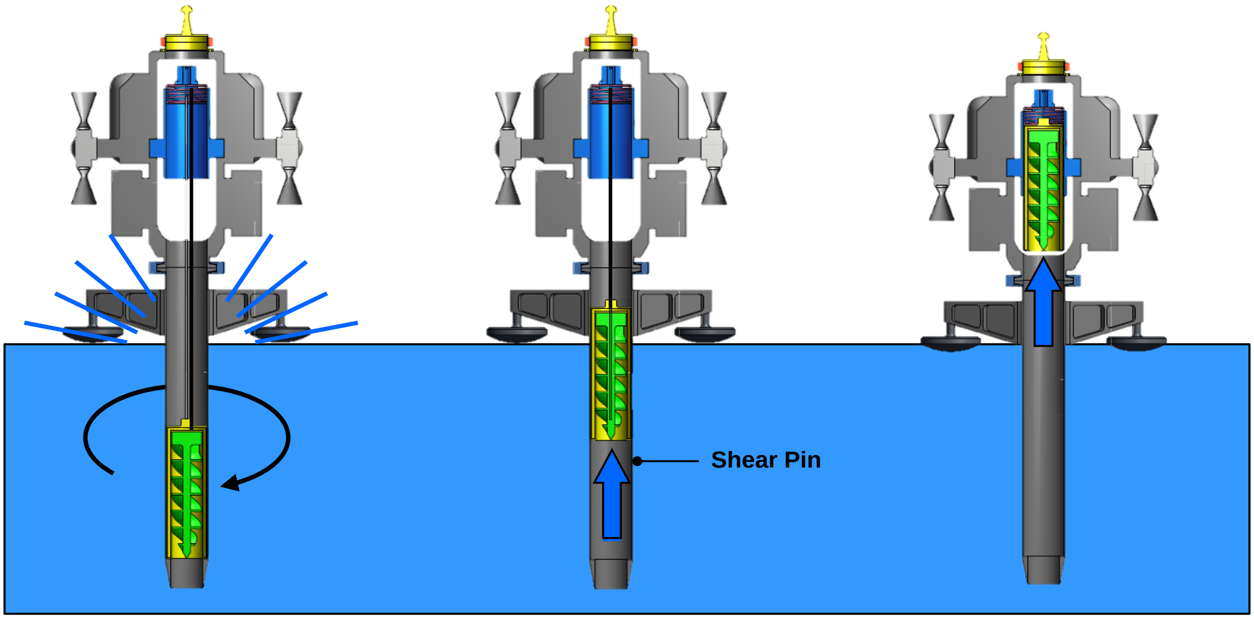


FIGURE 3.6. Concept of Operations – ARProbe Impact and Sample Acquisition

Once the casing is fully retracted into the upper stage and the spring is preloaded, the Flipper Mechanism rotates the entire canister and winch mechanism by 180 degrees. As soon as the Cap to the Casing is confirmed to be closed, the upper stage (Asteroid Ascent Vehicle) is detached from the lower probe body and travels to the parent spacecraft as shown in FIGURE 3.7.

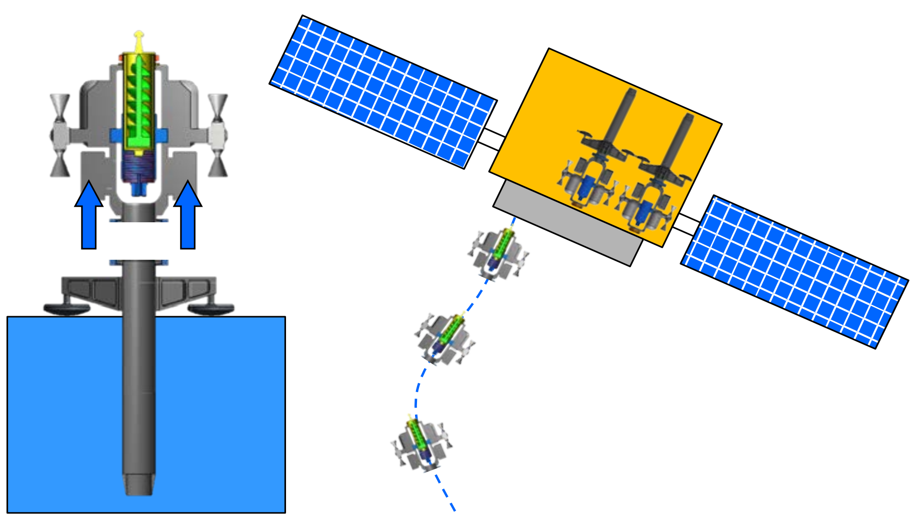


FIGURE 3.7 Concept of Operations – Upper Stage Release of ARProbes and Rendezvous with a Parent Spacecraft

3.2.2 Spiders

Spiders have numerous flexible legs with deep fluted augers mounted at the end of the legs. The augers are pointed inwards at a high oblique angle. This particular geometry allows the numerous Spider legs to anchor itself through a bracing action. In fact, the anchoring strength will only get stronger once the deep fluted augers are deployed and engage the subsurface to capture the surface regolith. The spiders can be used to acquire loose regolith and either process it in situ or deliver it to the RAP spacecraft for processing.

The bracing system uses two or more multi-mode anchors, positioned at an oblique angle to the surface as shown in FIGURE 3.8, resulting in a net force component along the asteroid surface. This resultant force braces the spacecraft to the surface. The advantage of this approach is that during the anchors' deployment only the force component in a vertical direction has to be overcome by, for example, firing rocket thrusters in the opposite direction.

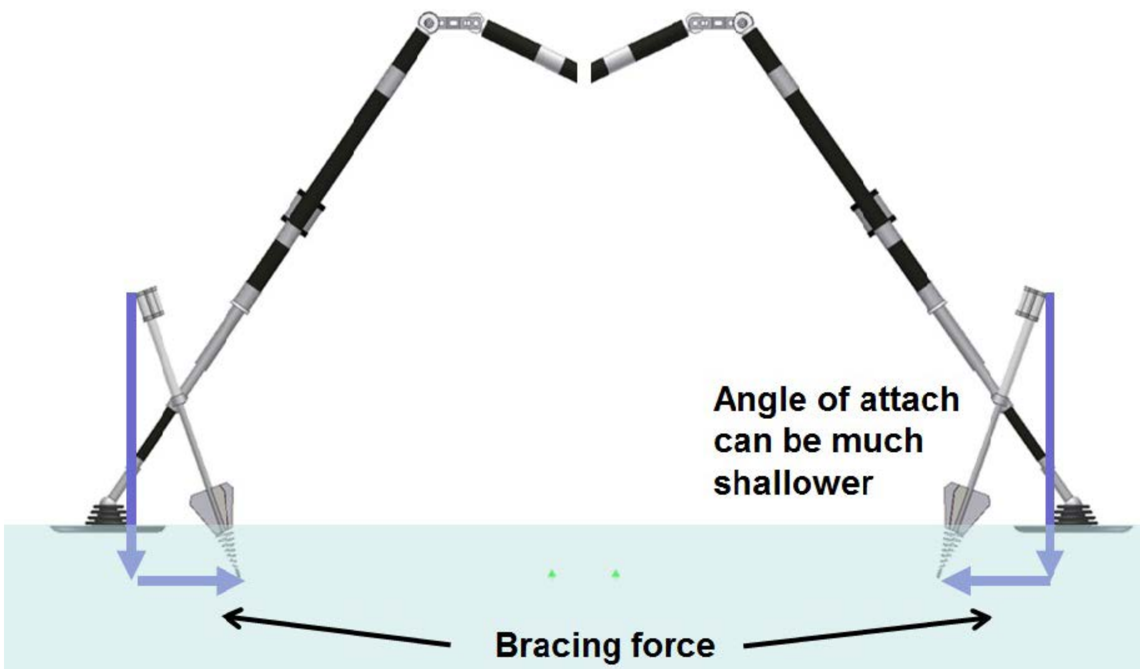


FIGURE 3.8. The bracing anchor engages the surface at an oblique angle.

3.3 Water Extraction

As mentioned earlier, water can serve several purposes. It can be used as fuel (LOX/H2 or for Solar Thermal propulsion), for human consumption, and for radiation protection. The market for radiation protection is finite – that is, once the shield has been filed with water, there is no longer need for additional water. The market for water for human consumption is not very large. This limit is because 95% of water can in fact be recycled. Only the market for fuel may be sustainable, because once fuel is burned or used up, it has to be replaced by fresh supply.

There are two ways to extract water. First, an asteroid can be placed inside a bag and sealed. Heat from the solar concentrators could be funneled in the form of light directly into the bag. The inside of the bag would be coated with reflective compound to help evenly spread the radiation, while the outside could be painted black to absorb solar radiation (however, this needs to be traded against losing heat faster from the black rather than reflective surface). Other forms of heat might include microwave or heat from radioactive sources. Once water starts to sublime, it will create its own pressure. The pressure difference between the bag and the condenser (cold finger) where the water is captured does not have to be high to channel the vapor into the condenser. Water extraction tests conducted by the RAP team in a vacuum chamber showed that a pressure difference of approximately 500 Pa was sufficient (Zacny et al., 2012). It should be noted that heat recovered from the condenser unit could be pumped back into the bag to enhance extraction efficiency.

The second method of water extraction is to capture regolith that contains ice onto deep auger flutes FIGURE 3.10 shows the several steps required to capture water. When the auger digs fully into the frozen regolith, it acquires the load of material for processing. Once fully loaded, the augur withdraws from the hole it drilled in the regolith and then retracts into a reactor. Heat from a supplemental RTG or an electrical heater could then heat up the icy-soil to 0 °C. Further heating will allow ice to sublime and be captured on a cold finger. This technology has already been proven in vacuum and shown to work well (Zacny et al., 2012).

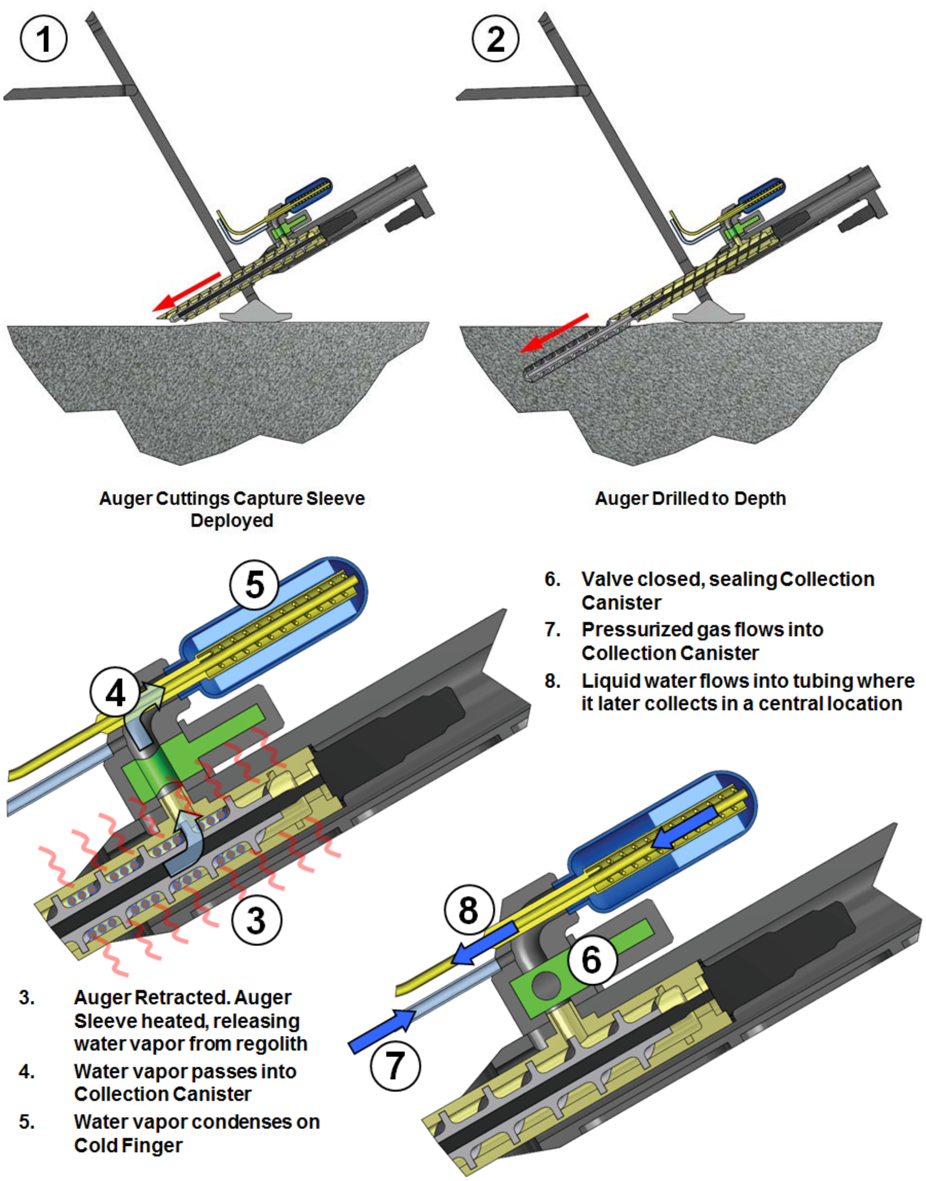


FIGURE 3.10 Water Extraction and Condensation Process

FIGURE 3.11 shows vacuum chamber tests to extract frozen water that the RAP team conducted at Honeybee Robotics. These tests used analog simulants with various water fractions to demonstrate that water extraction efficiency can be as high as 90% at 80% energy efficiency (i.e.

energy used relative to the energy required to heat up ice and sublime water vapor). FIGURE 3.11a shows the general experimental setup.

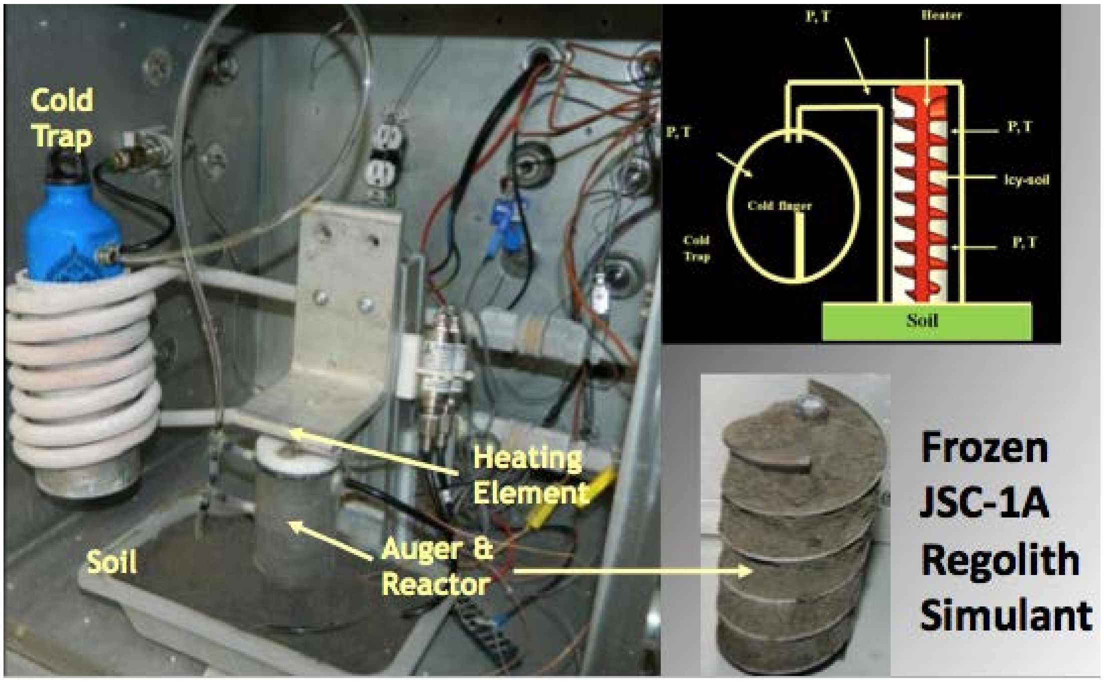


FIGURE 3.11a. The laboratory set up for the Frozen Regolith Extraction experiment.

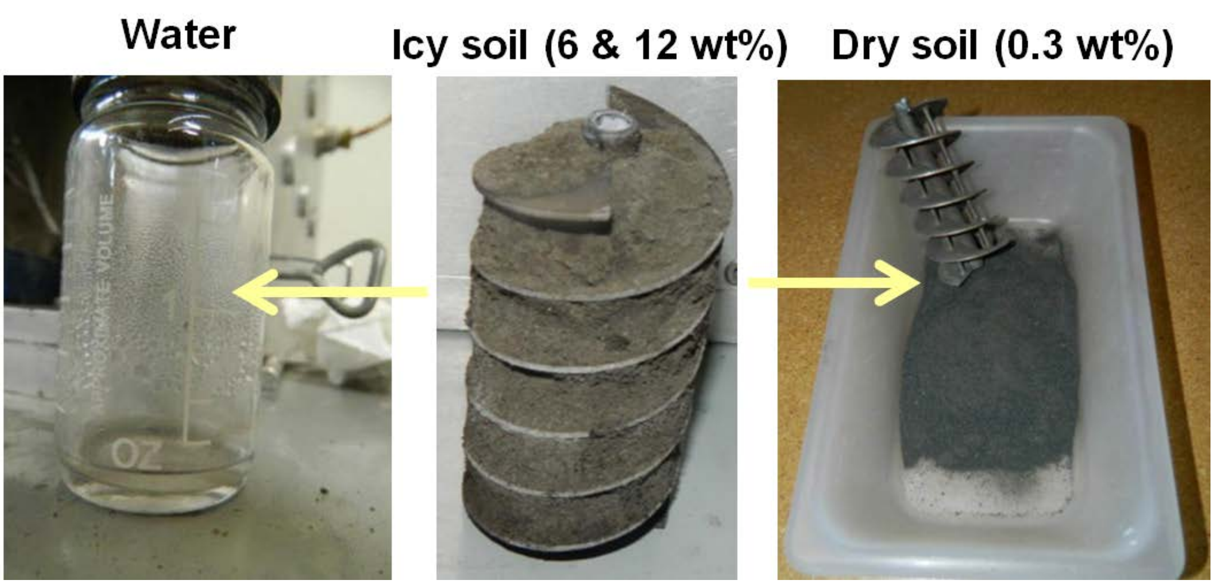


FIGURE 3.11b. Water extraction process showing the auger and separation of liquid water and solid debris.

The energy required to sublime 100kg of water with 90% energy efficiency is 83kWh. TABLE 3.2 shows the results of a feasibility study to determine various parameters for gathering water extraction from an asteroid. In particular, variables include the mass fraction of water in the asteroid, efficiency of a reactor (fraction of water captured inside a reactor), and required mass of water for a particular mission. To make the calculations easier to understand, the table uses 100kg of water as an initial parameter – that is, the mission is required to extract 100 kg of water. In order to recover so much water, the RAP spacecraft (bagging) approach, would require an asteroid to be at least 3 meter in diameter. This approach assumes the asteroid has a 10wt% water and 50% of that is recovered. Our estimations did not include time required to do that – this is a fraction of energy that solar concentrators can deliver to the bag.

On the other hand, a Spider with 8 reactors, each 20cm diameter and 1m long, would require 3.3 days to capture 100kg of water (assuming the same wt.% water and efficiency cycle). This sizing assumes 1 hour per each drill/capture regolith/extract water cycle. It should be noted that Spiders are deployed on larger asteroids while RAP captures smaller (sub 25 m) asteroids.

TABLE 3.2. Water extraction calculations

		Water weight % in asteroid					
		1%	**2%**	**5%**	**10%**	**20%**	**22%**
Percentage of Water Recoved from an Asteroid							
Percentage of water captured inside reactor		50%	50%	50%	50%	50%	50%
Percentage of water recovered from an asteroid		0.50%	1%	2.50%	5%	10%	11%
RAP (Asteroid Bagging Option)							
Required mass of asteroid	tons	20	10	4	2	1	0.9
Bulk density of asteroid	g/cc	1	1	1	1	1	1.0
Required volume of asteroid	m3	20	10	4	2	1	0.9
Diameter of required spherical asteroid	**m**	**3**	**2**	**2**	**1**	**1**	**1**
Spiders (for larger asteroids)							
Auger (Reactor) diameter	m	0.20	0.20	0.20	0.20	0.20	0.20
Auger (Reactor) length	m	1	1	1	1	1	1
Auger (Reactor) volume	m3	0.03	0.03	0.03	0.03	0.03	0.03
Number of Augers/Reactors (i.e. Spider legs)		8	8	8	8	8	8
Number of spiders		1	1	1	1	1	1
Total volume captured	m3	0.25	0.25	0.25	0.25	0.25	0.25
Total mass captured	ton	0.25	0.25	0.25	0.25	0.25	0.25
Total water captured	ton	0.00	0.00	0.01	0.01	0.03	0.03
Number of auger mining cycles		80	40	16	8	4	4
Time per cycle	hr	1	1	1	1	1	1
Duration of water capture	hrs	80	40	16	8	4	4
Duration of required water capture cycles	**days**	**3.3**	**1.7**	**0.7**	**0.3**	**0.2**	**0.2**

FIGURE 3.12 shows the required diameter of an asteroid for the RAP spacecraft concept and the duration of Spider cycles (for Spiders) required to recover 100 kg of water as a function of the water weigh % in the asteroid regolith. As the fraction of water in an asteroid increases, the required mass of an asteroid, or duration of an 8-legged Spider cycle, decreases. Note the maximum water fraction assumed for this analysis is 22% (Norton, 2002).

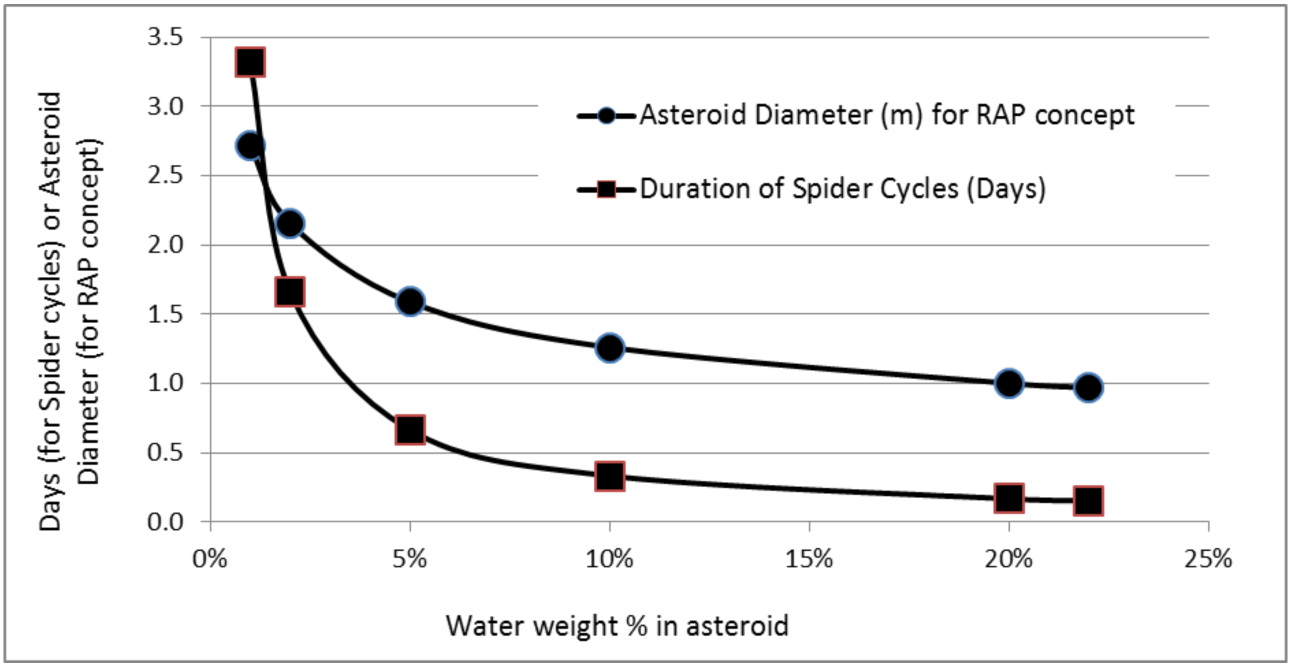


FIGURE 3.12 Diameter of an asteroid for the RAP spacecraft or duration of Spider cycles required to recover 100 kg of water as a function of water wt% in asteroid regolith. Water extraction efficiency assumed at 50% in both cases.

3.4 Metal Mining

Metal recovery is extremely challenging. Gravity and water are key components of most terrestrial technologies. In addition, on Earth, majority of metals are in the form of various minerals and oxides and have to be liberated and purified though a number of chemical processes. For example, ilmenite is a crystalline iron titanium oxide (FeTiO3). Titanium can be extracted from the ilmenite by the following set of reactions:

$$FeTiO3\ (s) + 3Cl2\ (g) + 3C(s) \text{->} 3CO\ (g) + FeCl2\ (s) + TiCl4\ (g)$$

$$TiCl4(g) + 2\ Mg\ (s) \text{->} 2MgCl2\ (l) + Ti(s)$$

Terrestrial metal extraction and refining technologies took centuries developed to a level that makes them economically viable. For example, until Hall and Héroult developed a process for the production of aluminum, the metal was more expensive than silver. However, it took much experimentation to arrive at the best formula. The paragraph below bears witness to the complexity of the aluminum extraction process[5]. The recipe does not include many details that go into making sure this process works, however, it does give a 'taste' of complexity.

> *In the Hall–Héroult process alumina, Al_2O_3, is dissolved in an industrial carbon-lined vat of molten cryolite, Na_3AlF_6 (sodium hexafluoroaluminate), called a "cell". Passing a direct electric current through it then electrolyzes the molten mixture of cryolite, alumina, and aluminum fluoride. The electrochemical reaction causes liquid aluminum metal to be deposited at the cathode as a precipitate, while the oxygen from the alumina combines with carbon from*

[5] http://en.wikipedia.org/wiki/Hall%E2%80%93H%C3%A9roult_process

the anode to produce carbon dioxide, CO_2*. The liquid aluminum is taken out with the help of a siphon operating with a vacuum, in order to avoid having to use extremely high temperature valves and pumps. The liquid aluminum then may be transferred in batches or via a continuous hot flow line to a location where it is cast into aluminum ingots.*

The electrolysis process produces exhaust that escapes into the fume hood and is evacuated. The exhaust is primarily CO_2 *produced from the anode consumption and hydrogen fluoride (HF) from the cryolite and flux. HF is a highly corrosive and toxic gas, even etching glass surfaces.*

Unfortunately there is no rule of thumb or a golden formula to determine how much investment and how many years will be required to develop certain metallurgical process. This uncertainty is probably one of the major problems in extracting metals from Asteroids. Even if the target is an M-type asteroid, a process would have to be developed to de-alloy the asteroid (M-type asteroids are made primarily of an iron-nickel alloy). Metal alloy separation into individual metals is a complicated task, and not feasible in space environment at this time. No simple, robust, lightweight, automated system has yet been identified. Potentially, one could process the existing asteroid alloy into steel or other alloys by adding different components to it.

In addition to the complexity in developing any metallurgical process, the fact that the technology has to work in microgravity and possibly in a low pressure environment means that it will have to be tested in LEO. Flying any experiment to the International Space Station is extremely expensive and such a process would add years and millions of dollars to the R&D cost and schedule.

We also spoke to Brent Hilscher, Senior Process Engineer at Hatch, a service provider to the mining industry. He reviewed many current technologies and concluded that none of these were ready for in situ space mining. They would essentially need to be morphed into new technologies and be tested in space. He also confirmed our findings regarding the uncertainty (budget/schedule) to develop new technologies. Potential new approaches might include plasma heating and vapor deposition/selective condensation of various elements. New technology utilizing ionic liquids could also be promising. In summary, it is extremely difficult to develop a metallurgical process for terrestrial environment, and it will be even harder to do so for the space environment. Hence, it is anyone's guess whether mining REE or PGM would be profitable.

However, the space environment may offer some advantages. The spacecraft have to be robust to withstand launch loads at the top of a rocket. However, once in space, most components do not see major loading, save for a few (e.g. pressurized fuel tanks). Hence, space structures could, in fact, be made weaker since they would not need to withstand launch loads. In that case, a 3D printing technology might as well be suitable for the space environment. In 3D printing, a feedstock material in the form of fine powder is heated up and melted and then printed layer-by-layer to form a structure. The feedstock to a space 3D printer could be fine asteroid regolith. Such regolith could be relatively easily mined using magnetic or even pneumatic mining techniques. Both approaches are relatively easy, and at high TRL.

A concept of mining using magnetic rakes has been proposed before as well. Magnetic systems, just as pneumatic systems, minimize moving parts and hence enhance robustness and reliability, which is vital when dealing with small and often abrasive dust. Since M-type asteroids contain magnetic kamacite and taenite, the magnetic mining could be very efficient. Of course other types of asteroids could also be targeted for metallic powders. Pneumatics can provide a method not only to collect but also beneficiate the loose particulates from an asteroid, with gas recovery to enhance efficiency. Our tests revealed that 1g of gas at 7 psia could loft over 5000g of regolith in vacuum at high velocities. Unlike a magnetic system, a pneumatic system would work with any types of powder and it will provide required cut off particle sizes.

4 Mission Design

The focus of the Mission Design activities for RAP is to devise a mission architecture that can profitably return asteroid material to Cis-Lunar space. To meet this goal, the mission and spacecraft must be scalable to a variety of mission ΔV budgets, mission durations and cargo mass requirements. The system design must be resilient against performance shortfalls in the propulsion system and other spacecraft subsystems. The system must also reduce the total time required for any individual asteroid mining/recovery mission so that 'cost of money' impacts can be minimized.

Innovative mission design will prove crucial to achieving the above goals and conducting a successful asteroid mining enterprise. To achieve this transformational mission design, it will be necessary to challenge many of the familiar assumptions about staging, trajectory design, and the roles of spacecraft design and propulsion technology. The Mission Design consists of several elements including the Concept of Operations (ConOps), the trajectory, and ΔV budget.

One of the challenges of doing Mission Design for asteroid mining missions is that there exist a large number of potential destinations; each comes with its own ΔV budget and launch/arrival space. A mission design for a specific object will define what is needed for that mission to that specific asteroid but it will not answer the more general question of accessibility of other asteroids. Given the limited time and budget available for the RAP study we judged it impractical to make a survey of all potential asteroid mining missions. Instead, we developed a basic Delta V budget, based upon previous analyses of missions to Near Earth Asteroids. Next, we developed a spacecraft design that would allow us to tailor the performance of the mining spacecraft easily for each specific mission. By conceptualizing a system whose performance can grow as needed, we designed a spacecraft that can accommodate many – if not most – of the early asteroid mining missions that we might fly. This Robotic Asteroid Prospector (RAP) spacecraft will be capable of further growth for more challenging missions later.

The three major drivers on the asteroid mining mission architecture are:

1) The type of propulsion system used for the Interplanetary Vehicle (IPV) –
    a. Low thrust electrical,
    b. High thrust chemical,
    c. Solar thermal, or
    d. A hybrid combining high and low thrust systems.
2) The Earth-Moon orbital location from where the vehicle departs and returns.
3) The source of the propellants for the IPV.

Chemical propulsion offers the virtue of simplicity and a long experience base. However it suffers from low performance compared to electric propulsion. The best performing chemical propellant combinations require the ability to store cryogenic propellants for extended durations.

Solar Thermal propulsion can offer an *I*sp twice that of chemical propellants. Provided it uses hydrogen as the reaction mass, and depending on the size of the solar collectors, Solar Thermal Propulsion can offer moderate thrust levels. This *I*sp and thrust will allow performance comparable to a Nuclear Thermal propulsion system without entanglement with the operational and political issues of launching a nuclear reactor into space. If we use liquid water as the reaction mass, then the *I*sp will be comparable to that of a traditional chemical system but the propellant, i.e. water, has the virtue of being dense and non-cryogenic. Cryogenic LH2 is 71kg/m^3 compared to water at 1,000kg/m^3. This system also opens the possibility of a hybrid in which Solar Thermal does the initial departure from Cis-Lunar space; then the solar dynamic collectors would drive a Solar-Electric propulsion system. The Solar Thermal propulsion system can also make available large amounts of process heat for extracting and processing resources at the asteroid.

Electric propulsion offers specific impulse values an order of magnitude higher than chemical propulsion but at the cost of low thrust and the need for a large electrical power supply. However, this "cost" becomes a virtue for a mining mission since that large solar-electrical power system can also support energy intensive mining operations. The low thrust of an electrical propulsion system can require significant increases in mission duration as a result of the long time that the vehicle must spend spiraling out from Low Earth Orbit (LEO) prior to escaping from the Earth. However, we can effectively eliminate this problem by staging the mission from an EML point.

The low thrust of a Solar Electric system significantly increases the ΔV needed for the mission, which erodes some of the benefits available from such a system. Unfortunately the best propellants for use in a Solar-Electric propulsion system are not likely to be easily available from the asteroids and thus all of the propellant required for returning asteroid material to Cis-Lunar space will have to be carried out to the asteroid from the Earth. However, using Solar Thermal, we can solve this problem by using water extracted from asteroids as the fuel for the return leg of the mission.

Staging the mission from a Lagrange point, EML-1 for example, substantially reduces the ΔV requirements for the IPV and thus reduces the size of that vehicle. Staging from a Lagrange Point further creates the option for departing via a lunar swing-by trajectory that would use a lunar gravity assist to eject the IPV from cis-lunar space with associated reductions in departure ΔV. However, the Lagrange Point option does increase the ΔV required to get the IPV to the staging orbit when compared to a LEO staging option. In addition to this issue, there are timing and orbital geometry issues to consider when using the Lagrange points as a staging point for an interplanetary mission.

The greatest benefit from staging at a Lagrange point arises when a reusable vehicle will serve to undertake multiple missions. In this case, the Lagrange points offer large net ΔV savings over LEO staging since the ΔV required to capture to a Lagrange point is much less than the ΔV required to capture to LEO. Additionally, since the ΔV to depart from a Lagrange point is much less than the ΔV to depart from LEO the propellant for the next IPV mission will be greatly reduced.

The largest driver on the RAP architecture will be the source of the propellant for the IPV. The conventional approach to this need would be to lift the propellant from the Earth to the staging point. Lifting a payload to LEO requires a ΔV of approx. 10 km/s; sending that payload on a low energy escape trajectory requires an additional 3.2 km/s of ΔV, yielding a minimum of 13.2 km/s to put the spacecraft onto its interplanetary trajectory. Thus, for asteroid and Mars missions, as much as 80% of the IMLEO at the start of the mission would be propellant.

For missions staged from EML-1, it takes 10 km/s to get the propellant and equipment to LEO, 3.8 km/s to deliver that material to EML-1, but only an additional 0.14 km/s for a low energy departure from cis-lunar space. This total ΔV of 13.94 km/s is slightly higher than for the departure from LEO; superficially it would seem to argue against Lagrange point staging. However, the picture changes completely if the Earth's moon is the source of propellant for the IPV. It only takes 2.5 km/s of ΔV to deliver propellant from the lunar surface to EML-1 - which is 11.3 km/s less than the ΔV required from the Earth's surface. Since the ΔV for a low energy escape trajectory starting from EML-1 can be as low as 0.14 km/s, the amount of propellant produced on the Moon and delivered to EML-1 is substantially smaller than the propellant that would have to be launched from the Earth for a LEO staging option. Avoiding the need to launch propellant from Earth's surface reduces the total higher-cost mass launched from the Earth. Since subsequent missions could reuse the IPV after it returns to EML-1, they would only have to pay the modest propellant cost of moving propellant from the lunar surface to EML-1, avoiding the cost of repeatedly launching propellant from the Earth to LEO and then from LEO to EML-1.

4.1 Rationale for Operations from the Earth Moon Lagrange Point

The economic success of asteroid mining will depend on RAP's ability to minimize the transportation cost of flying to the asteroid of interest and returning extracted materials to cis-lunar space. Establishing a staging base at one of the Earth Moon Lagrange points, to which a commercial effort can deliver propellant from the Moon or some other in-space source to fuel asteroid mining spacecraft appears critical to achieving this goal.

4.1.1 Lagrange Libration Points

The efficacy of a Lagrange point for staging operations is driven by seven factors.

- First, a spacecraft parked at a Lagrange Point is only loosely coupled to the Earth and needs only a modest propulsive maneuver to depart for an interplanetary destination.
- Second, a spacecraft departing from a Lagrange Point can use an Earth swing-by maneuver to significantly reduce the ΔV required to achieve a given departure C3.
- Third, in a similar vein, the ΔV required to capture back into orbit at a Lagrange Point is much lower than the ΔV required to capture into LEO, especially if this capture maneuver uses a Lunar swing-by maneuver.
- Fourth, if the spacecraft is recovered back to a Lagrange Point, then it is in a position to go out on another prospecting or mining mission without having to pay any extra energy costs to get that spacecraft to the Lagrange Point staging location.
- Fifth, staging from a Lagrange Point also opens up the opportunity for using a Lunar gravity assist to aid in the departure and return maneuvers and will reduce the total mission ΔV by a meaningful amount.
- Sixth the ΔV cost of delivering propellant to a Lagrange point from the Moon or other in-space sources is a small fraction of the cost of Terrestrially-derived propellant delivered to LEO or a Lagrange Point.

- Seventh, and finally, by using an intermediate phasing orbit during the Earth departure maneuver it will be possible to adjust the orientation of the departure trajectory for a small cost in Delta V. This maneuver will also allow us to accommodate a normal launch period by controlling the amount of time that is spent on the phasing orbit.

4.1.2 Delivering Propellant to a Lagrange Point

Delivering propellant from the Earth's surface to LEO requires carrying it through a 10km/sec ΔV increment, after accounting for the various losses incurred by the launch vehicle during the ascent to orbit. To deliver propellant, or any other cargo, from LEO to EML-1 requires an additional ΔV of ~3.5 km/sec. However to deliver propellant to EML-1 from the Lunar surface only requires carrying that propellant through a 2.5 km/sec ΔV increment. The ΔV increment for transporting propellant from a NEA to a Lagrange point is only somewhat higher than the ΔV increment for transporting propellant from the lunar surface and opens up another avenue for providing the propellant needed to support asteroid mining. This large difference in ΔV for transporting propellant to the staging orbit from in-space sources translates into a potentially huge cost savings for operations out of EML-1. Similar savings are possible for operations from the other Earth Moon Lagrange points.

FIGURE 4.1 below shows a graphical depiction of the Earth's gravity well. It shows, qualitatively, that a spacecraft parked at an Earth-Moon Lagrange point is almost completely out of the Earth's gravity well and thus will only take a modest ΔV to venture into interplanetary space.

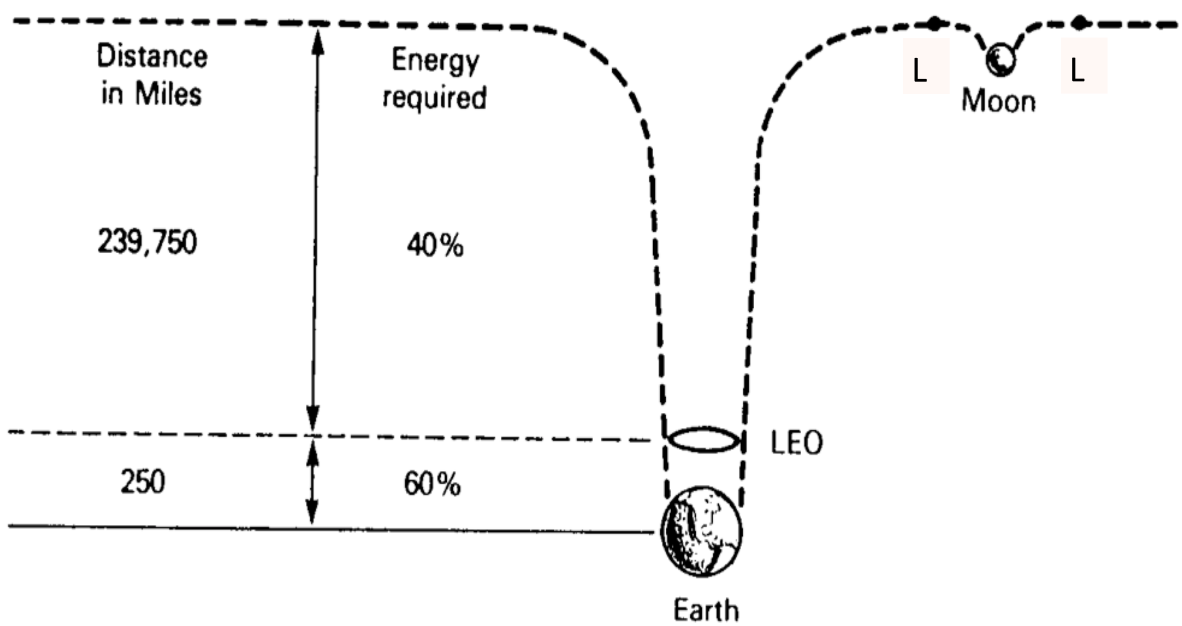


FIGURE 4.1. Graphical Depiction of Earth's Gravity Well.

4.1.3 Lagrange Point Benefits Summary

The benefits of operating from a Lagrange Point using in-space propellants are:

- ***Operations from a Lagrange point will require significantly less ΔV than operations from LEO, and***
- ***The cost of delivering propellant to that staging location will be lower than the cost of providing the propellants for operations in LEO.***

- ***Therefore, an asteroid mission staged from a Lagrange Point using propellants derived from in-space sources will be significantly less expensive than an equivalent missions staged from LEO using terrestrially derived propellants.***

4.2 Earth Departure Options

FIGURE 4.2 shows a schematic illustration for several options for departing from cis-lunar space. The direct departure from LEO is the most straightforward since it is the departure mode that every interplanetary spacecraft launched to date has used. ***The direct departure from EML-1 is similar but with the lower ΔV requirement.*** At the appropriate time, the spacecraft makes a propulsive maneuver that increases its velocity and places it on a hyperbolic orbit. The timing of this departure is chosen so that the asymptote of the outbound hyperbola is aligned with the heliocentric velocity vector of the desired interplanetary orbit.

The departure via Earth swing-by is a bit more complicated. In this mode, the spacecraft performs a Lagrange Point departure maneuver that places itself onto an elliptical orbit targeted for a 300 km periapsis with the apoapsis of the orbit located at EML-1. When the spacecraft reaches the periapsis of this orbit it performs a second maneuver that places it onto the desired departure hyperbola and the spacecraft is on its way to the targeted asteroid. As in the Direct Departure case, the timing of these maneuvers is used to align the asymptote of the outgoing hyperbolic orbit with the required heliocentric velocity vector for the interplanetary transfer trajectory.

We can achieve additional departure time flexibility by using an intermediate phasing orbit before initiating the transfer to the Earth flyby. In this mode, the spacecraft is placed onto an orbit that takes it beyond the orbit of the Moon.[6] At the apoapsis of this intermediate orbit the spacecraft will have a very low velocity in an Earth-centered coordinate system. This spacecraft can change the inclination of its orbit, and thus the inclination of the targeted interplanetary orbit, for a very small ΔV cost. Additionally, by adjusting the period of the intermediate orbit it will be possible to control the date when the spacecraft departs Cis-Lunar space, thus allowing us to provide a launch period that meets normal mission design practices.

Please note that these departure options can be "played in reverse" when the spacecraft is returning from its asteroid mining mission. This "reverse play" will allow us to use lunar gravity assists and deep space maneuvers to minimize the ΔV cost of returning the spacecraft to its EML point staging base while maintaining a return period that meets normal mission design requirements.

4.2.1 Impact of Staging from an EML Point on Departure C3

As noted above, the ΔV for departing cis-lunar space from an EML is lower than for departures from LEO because the Lagrange Points are located at the edge of the Earth's gravity well. However the numerical benefit of such a departure is a function of the C3[7] of the spacecraft's heliocentric orbit. The ΔV required

---

[6] This orbit can be a direct transfer or it can utilize a gravity assist from a lunar fly-by.
[7] C3 is the measure of the energy of the orbit and is the square of the orbit's V Infinity - i.e. the velocity the spacecraft would have when it reaches an infinite distance from the Earth in the absence of any other gravitational forces.

to inject a spacecraft onto a hyperbolic orbit having a specified C3 when starting from an initial orbital radius of R and with an orbital velocity V_i is given by the following equation.

$$\Delta V = \sqrt{\frac{2\mu}{R} + C3} - V_i$$

If the initial orbit is circular, then V_i is just the circular velocity for radius R. If the injection is being performed from the periapsis of a non-circular orbit, then V_i is the periapsis velocity of that elliptical orbit. We can quantitatively assess the impact of operating out of a Lagrange point by calculating the ΔV as a function of C3 for the hyperbolic departure orbit for various staging locations and departure options. The results from this calculation appear in FIGURE 4.2.

As expected, low C3 direct departures from L1 or L2 take less ΔV than direct departures from LEO. However as the C3 increases the benefits of operating from the Lagrange point decrease until they become non-existent for higher C3 departures. There are few missions to Near Earth Asteroids that require outbound C3 values greater than 20 km^3/s^2, so they would benefit from operations out of a Lagrange point using direct departures.

However we can obtain a valuable additional benefit from doing a departure maneuver that includes an Earth swing-by maneuver. In doing this swing-by, the spacecraft performs the departure maneuver at the periapsis of an elliptical transfer orbit, and as such, makes this velocity change when the spacecraft is moving much faster than it would have been moving at the Lagrange point. The ΔV for the Earth swing-by departure includes both the ΔV required to perform the departure maneuver and the ΔV required to put the spacecraft onto the transfer ellipse. Since kinetic energy increases as the square of the vehicle's velocity, it should be obvious that the proportionate energy gain from a given ΔV is higher if that maneuver is done when the vehicle is moving at a higher velocity. FIGURE 4.3 clearly shows this effect. For example, to achieve an outbound hyperbolic orbit with a C3 of 5 km^3/s^2 requires a ΔV of 3.43 km/s when departing from LEO, but just fewer than 2 km/s when directly departing L1 and only 0.98 km/s when departing from L1 using an Earth swing-by maneuver. This performance gain for swing-by departures continues for increasing C3 values for the swing-by departure.

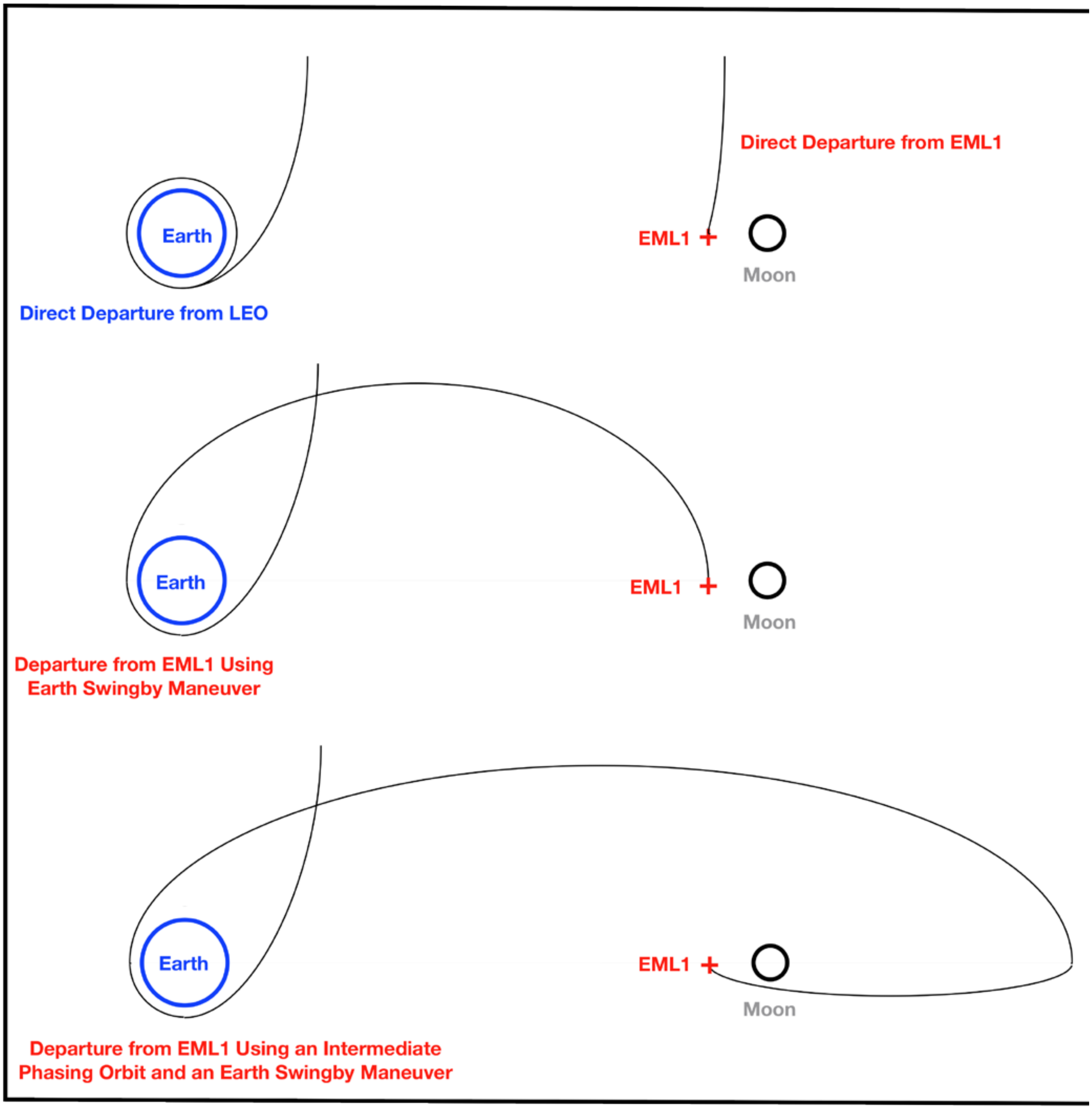


FIGURE 4.2. Schematic Illustration of Options for Departing Cis-Lunar Space

Very low C3 departures do not benefit from the swing-by maneuver because the gains from the swing-by are not as large as the ΔV cost of first inserting onto the transfer ellipse. However, most asteroid missions would require departure orbits with C3s high enough to justify the use of a swing-by departure over a direct departure from the Lagrange Point.

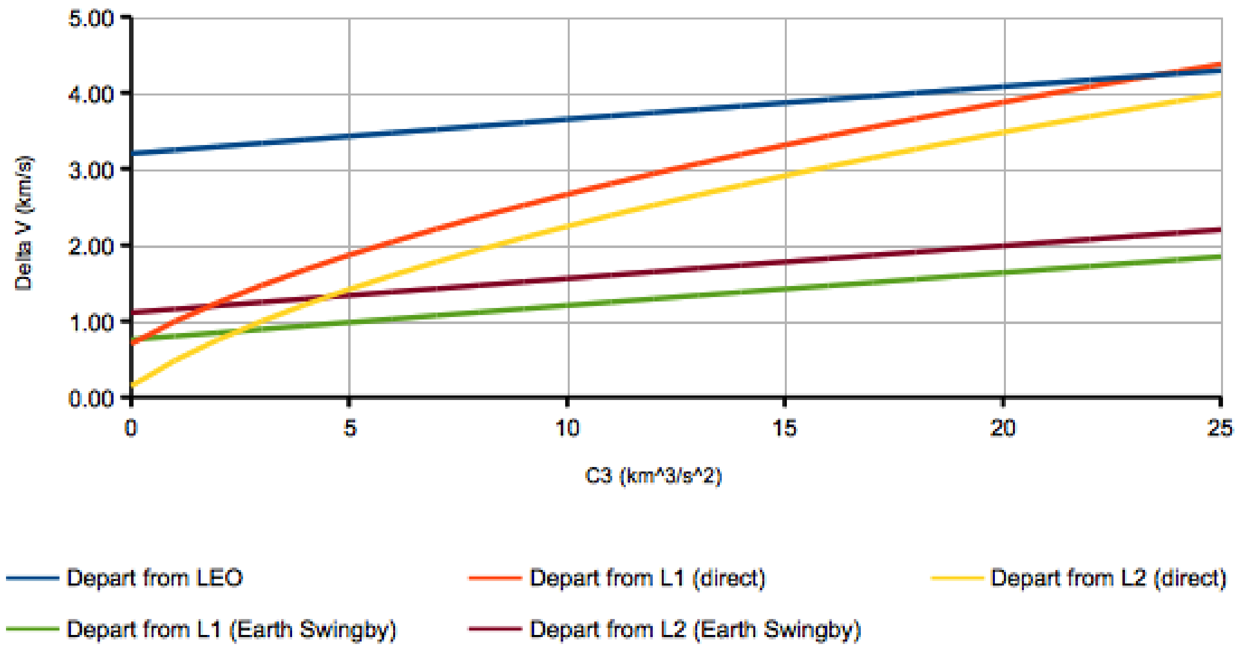


FIGURE 4.3. Departure ΔV from Cis-Lunar Space for Various Departure Options

4.2.2 DeltaV Reduction -- System Sizing Case

It will help to see the importance of the ΔV reductions made possible by operations from a Lagrange Point by looking at a simple system sizing case. We will calculate the weight of a system capable of transporting a 10,000 kg payload on a round trip mission to a typical asteroid and compare the results for staging from LEO and EML-1.

We will assume a departure C3 of 12 km^3/s^2, which is a typical value for many NEA missions. Departure from LEO will use a direct mode whereas the EM-1 departure will use an Earth swing-by maneuver. When departing from LEO the ΔV for post-departure maneuvers, including propulsive capture back at LEO, is 6,000 m/s, which is sufficient to perform many missions to various NEAs. For a departure from EML-1, the post-departure ΔV budget is 3,600 m/s, which reflects the reduction in ΔV associated with returning to EML-1 using an Earth swing-by maneuver vs. returning to LEO. We will assume chemical propulsion with an *I*sp of 450 seconds and a two-stage propulsion system with a propellant fraction of 0.9 for each stage. Based on these assumptions we can now calculate the system weights for the two departure options.

The system staged from LEO will have an initial mass of 157,478 kilograms of which 132,730 kilograms is propellant. The system staged from EML-1 will have an initial mass of 36,626 kilograms of which 23,963 is propellant. The dry mass of the LEO based system will be 14,748 kg whereas the EML-1 based system will have a dry mass of 2,663 kg. In other words, the required propellant and system dry mass for the EML-1 based system is 18% of that for a system based in LEO. This 82% savings in system mass will translate into substantial cost savings for propellant transport - especially since the ΔV increment to get propellant from the lunar surface is significantly less than the ΔV increment to get propellant into LEO from the Earth's surface. The large difference in system dry weight can also translate into measurable cost savings for procuring the spacecraft.

4.3 Options for Returning from an Asteroid and Reducing Delta V

When returning from an asteroid there will be many techniques that can be used to reduce the Delta V required for that phase of the mission and to create return opportunities that would not be available for conventional direct returns. This reduction will expand the launch/arrival space for these round trip missions and reduce the impact of the potentially long synodic period for such missions. Since these techniques will be depend strongly upon the specific geometries and timings for individual asteroid missions, it is not possible to provide general recommendations that will apply to all missions. Rather we will simply note some of the approaches that could be used and indicate that in Phase II of the RAP study we will provide demonstrations of the efficacy of these techniques for a variety of specific cases.

A dominant factor in the design of trajectories to return material from a Near Earth Asteroid is the potentially long synodic period for conjunctions between the asteroid and the Earth. This factor has the impact of greatly increasing the time between successive launch and return opportunities. The Synodic Period between the Earth and an asteroid is given as S below where E is the period of the Earth's orbit and A is the period of the asteroid's orbit.

$$\frac{1}{S} = \frac{1}{\left|\frac{1}{E} - \frac{1}{A}\right|}$$

We can see in the above equation that the Synodic period approaches infinity as the period of the asteroid's orbit approaches the period of the Earth's orbit. This means that launch/return opportunities between Earth and a specific NEA can be separated by many years or even decades if the orbit period of the NEA and the Earth are nearly identical. This can create a challenge for returning materials in a timely and economically useful time frame.

4.3.1 Venus and Mars Synodic Periods

However there is a possible solution to this problem. Although an asteroid may be in an orbit that has a long synodic period with respect to Earth, that object must have a much shorter synodic period with respect to Venus and Mars. This effect appears in FIGURE 6. FIGURE 6 shows that Near Earth Asteroids will, in general, have a synodic period with Venus or Mars that is less than two years and in some cases can approach one year.

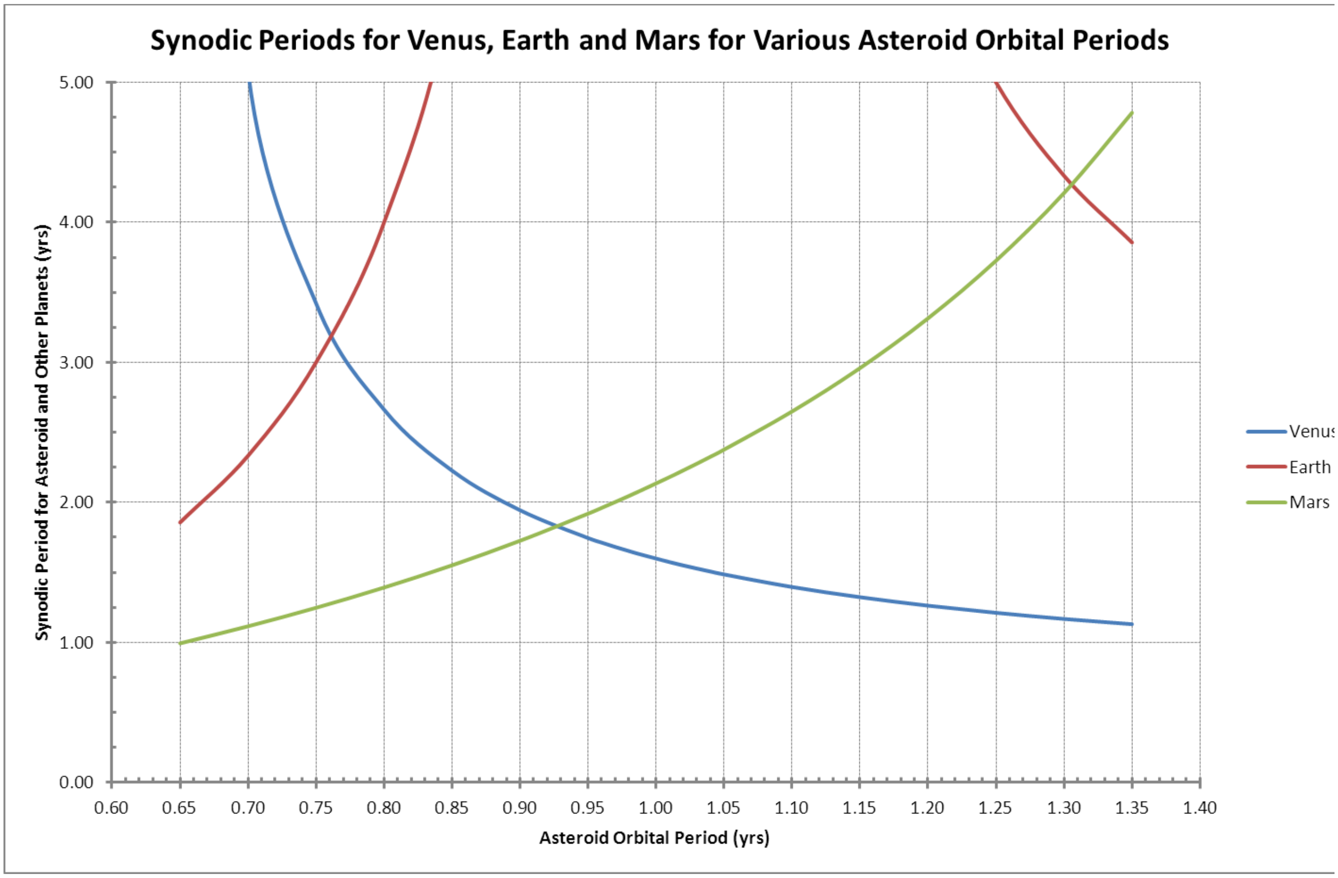


FIGURE 4.4. Synodic Periods for Earth, Venus and Mars for Various Asteroid Orbital Periods

This phenomenon creates the opportunity to take advantage of the frequent asteroid departure periods from Venus or Mars, allowing the spacecraft to transfer from the asteroid to a Venus or Mars flyby trajectory and then execute a gravity assist at the planet (either powered or ballistic) that would set up the spacecraft for a subsequent return to the Earth. The post-encounter trajectory would not need to be explicitly targeted for a direct return to Earth. Instead, we could use this trajectory to adjust the size or orientation of the spacecraft's orbit so that a subsequent deep space maneuver could complete the targeting for a return to Earth.

Each asteroid mission will be different so it is impossible to draw a general conclusion about the availability and efficacy of such gravity assist maneuvers for a specific mission before completing the analysis for that mission. However, we expect that given the power and flexibility of powered gravity assists for altering spacecraft interplanetary trajectories, this approach will open up and expand the mission space for asteroid exploration and mining in a useful and valuable way.

4.3.2 Solar Lagrange Point Transfers

There is another option for opening the Earth return space for these missions and that is to use trajectories targeted for the Lagrange points of the Earth-Sun, Venus-Sun or Mars-Sun systems and then take advantage of a low energy transfer from those Lagrange points to the Earth Moon Lagrange point being used as a staging base for the asteroid missions. The Delta V for making a transfer between Lagrange

Points can be quite low and it offers the opportunity to return materials from an asteroid for a minimal expenditure of energy. But more importantly there are five Solar Lagrange points for any planet and thus when looking for return opportunities targeting such a Lagrange point as an intermediate transfer point would give us 15 potential destinations, five each for Earth, Venus and Mars. It is much more likely that there will be a departure opportunity for one of those destinations at any given time than it is that there will be an opportunity for a direct return to Earth.

Combined with deep space maneuvers, gravity assists and multi-revolution transfer trajectories the return space for bringing material back from an asteroid is quite large. All of these techniques can also be applied when searching for outbound departure opportunities and will thus significantly increase the size of the launch/arrival space for many candidate asteroids. This increase in departure opportunities will in turn increase the number of candidate objects for any asteroid mining venture. In fact the resulting mission design space grew so large that it quickly grew beyond the scope of Phase 1 for the RAP study. We hope to address it in the next phase of this investigation.

4.4 Delta V Budget for Sizing Asteroid Mining Missions

Even though there will be a great deal of variation in the Delta V budget and mission timing for asteroid mining missions, we need to define a preliminary Delta V budget for sizing the spacecraft and performing the supporting analysis. The following table provides such a budget for a round trip mission based on various previous mission analyses. The Delta V for the Earth Departure maneuver is based on the requirements for a mission departing from LEO and not EML-1, since this would allow the initial deployment of the RAP spacecraft to be from LEO -- with subsequent missions starting from EML-1. Including the higher Delta V requirement for the LEO departure in the sizing budget provides us with significant margin that we can apply to the post-departure maneuvers when operating exclusively from EML-1. This budget is a good starting point to generate size estimates for the spacecraft and for the supporting economic analysis, but the budget for specific missions will need to be calculated on a case-by-case basis.

TABLE 4.1. Delta V Budget for Sizing RAP System

Maneuver	Delta V (km/s)	Comments
Earth Departure	3.50	Based on LEO departure to account for initial deployment. Excess Delta V can be applied to subsequent maneuvers.
Asteroid Arrival	1.25	
Asteroid Departure	1.35	Propellant for this maneuver and Earth Arrival can be provided by water mined from the asteroid
Earth Arrival	2.50	Does not include use of Lunar Gravity assist to reduce Delta V requirements

4.5 Mission Types

There are four types of missions that could fly as part of the RAP program. This section briefly enumerates them. These four mission types will be discussed in more detail below however the vast majority of the effort during the RAP Phase 1 contract was focused on the integrated Mining/Retrieval mission type.

1) Prospecting: This mission would return relatively modest amounts of asteroid material back to the Earth for study and analysis. It would not depend upon the use of asteroid In-Situ Resource Utilization (ISRU) to produce the propellants needed to return the vehicle back to the Earth.

2) Mining/Retrieval: This mission would transport mining/processing equipment to an asteroid, extract ~150- 300 Metric tons of useful material from that asteroid and then return to Cis-Lunar space with the mining equipment and the recovered asteroid material. This mission would use asteroid ISRU to create the propellant needed to return the spacecraft and its cargo back to the Cis-Lunar space.

3) Processing: This mission would deliver mining/processing equipment to an asteroid for extended resource extraction operations. This equipment would be left on the asteroid and a separate vehicle would be used to return asteroid resources to Cis-Lunar space.

4) Transport: This mission would involve a spacecraft that would shuttle between the EML staging base and an asteroid where mining/processing has been previously emplaced. Asteroid ISRU would to produce the propellant needed to return this spacecraft and its payload back to the EML staging base.

4.5.1 Prospecting Mission

The purpose of this mission is acquire detailed information about the available resources on an asteroid and their physical state as well as testing the spacecraft and mining/extraction equipment that would be used on an operational mining mission. The three specific objectives for this mission are:

> 1) Serve as a test flight for the mining spacecraft while testing and validating all -- or at least most -- of the technologies for the mining spacecraft without flying a mission that requires all of those technologies be fully operational.
>
> 2) Provide a flight test for the solar thermal propulsion system along with a demonstration of all of the maneuvers needed for operating to and from an EML point along with Prox-ops at the asteroid.
>
> 3) Return sufficient material from the asteroid so that those materials can be used to test and validate the systems for doing asteroid resource extraction and processing on an industrial scale.

This mission will serve to characterize a NEA and demonstrate/validate the spacecraft design and the equipment that will be used for mining. But it will not depend on the successful operation of that mining equipment to provide the propellant for returning the spacecraft back to the Earth.

Superficially, this mission would seem to be closely related to a traditional NASA sample return mission. However, that perception is not correct since this mission focuses on industrial-scale processing of asteroid materials, not on scientific investigations that a few grams of asteroid material can satisfy. To address the questions related to industrial scale processing of asteroid materials we will need to bring back industrial-scale quantities of asteroid materials. This mission will require a much larger vehicle than typical for scientific sample return missions.

The spacecraft for this prospecting mission would be based on the RAP spacecraft designed for the Mining/Retrieval mission. It could be a full sized prototype of that vehicle or it could be a sub-scale version. This mission would target an asteroid with a diameter of 15 meters or less. It would carry a suite of remote sensing instruments to characterize the asteroid's composition and structure. It would use small secondary spacecraft to get a selection of samples from the asteroid and store them for return to Earth. The prospecting spacecraft would demonstrate the maneuvers needed for synchronized rotation with the asteroid followed by capturing the asteroid in the containment vessel.

With the asteroid safely within the containment vessel, robots would take more samples. Having the asteroid secured to the spacecraft will make it easy to retrieve samples by using effectors on the end of an RMS type arm operating within the containment vessel. After gathering sufficient samples, the spacecraft would then demonstrate water extraction. The spacecraft would store recovered water without further processing so that studies back on the Earth could characterize the various volatile components along with particulates. Some of the water would be processed to the purity level needed for use as propellant but would not be used for that purpose; rather it would be stored and returned to Earth to verify that the system generates water of suitable quality for propulsion applications.

This prospecting mission would acquire several tons of processed materials as well as pristine materials. These samples would be returned to the EML staging base along with the equipment used to extract and process those materials so that studies of the mining equipment can verify their suitability for operational use.

Following return of the spacecraft and samples to EML-1, we could retrieve it by sending a human crew to meet the spacecraft at the EML. The astronauts would study the spacecraft and assess its status, recover samples of equipment for study back on the Earth and retrieve the samples. Some of the samples would be returned with the crew and the rest could be recovered using a cargo vehicle. A Dragon launched on a Falcon Heavy would be suitable for crew and cargo transport to EML-1.

The mass of the returned asteroid samples will not play a dominant role in sizing the spacecraft for the Prospecting Mission. That determination derives from the fact that this mission returns the entire spacecraft, along with the mining equipment, back to Cis-lunar space and that mass is much larger than any reasonable mass of scientific samples. For example, if we assume:

1. A spacecraft dry mass of 25 tons,
2. Chemical propulsion,
3. A DeltaV budget as shown in Section 6.4, and
4. A return payload of 2kg of asteroid samples,

Then the spacecraft will have an IMLEO of 150,745kg. Increasing the returned payload to 2,000kg causes the IMLEO to increase to only 158,924kg. Although perhaps counter-intuitive, this finding is a result of the fact that even though there was a three order of magnitude change in the mass of samples being returned there was virtually no difference in the dry mass of the spacecraft at the time of departure and at the end of the mission the dry mass of the spacecraft for the 2ton payload case was only 7% larger than for the case with a payload of 2kg.

A sub-scale prototype of the RAP spacecraft might fly this mission, in which case this smaller vehicle could fly later prospecting missions to other asteroids. On the other hand, using a sub-scale vehicle for this mission would necessitate designing a second size of spacecraft for the operational mining missions. It might be more cost effective to use a vehicle sized for operational missions to perform the prospecting missions since this would eliminate the cost of designing a second vehicle even though the larger vehicle would be greatly oversized for the Prospecting missions. This question demands a trade study on the sizing issues in Phase 2.

Moreover, the development cost of a purpose-built small spacecraft will not cost significantly less than the price of the larger Mining/Retrieval spacecraft. Many spacecraft functions are independent of spacecraft size – such as communications and computer control systems. Other spacecraft systems that scale with size - such as ACS thrusters - have costs with only a small dependency on size. Propellant mass does vary with spacecraft size, but that has only a small impact on the development costs. There is a minimum cost to design a spacecraft – independent of size; cutting the size of the spacecraft in half does not cut the cost in half.

The biggest cost difference will appear in actual operations since a larger spacecraft will have a higher launch cost and will undoubtedly cost more to build. But those costs are only one part of the total cost for the project and they are likely to be overwhelmed by the development costs – which are largely independent of size. Moreover, it will advantageous to get flight test data on the actual vehicle that will be used for operational mining missions prior to the start of commercial operations.

There do not appear to be any cost savings or great benefits associated with doing a traditional sample return mission to an asteroid that would return only grams of samples. However, by doing a mission that demonstrates asteroid mining technologies, without requiring that those technologies be fully operational to provide the means of acquiring the propellant needed for returning to Earth, will allow us to test our spacecraft in a realistic environment and return sufficient samples to further develop technologies for asteroid mining and resource extraction.

4.5.2 Mining/Retrieval Mission

The purpose of this mission is to acquire economically exploitable amounts of useful materials from an asteroid and return them to cis-lunar space. This mining/retrieval mission combines the functions of resource extraction with those of resource transportation into a single mission. Since Solar Thermal Propulsion would serve to transport the spacecraft to and from the asteroid and the solar concentrators for that system would provide process heat for extracting water from the asteroid.

The spacecraft for this mission will transport the resource extraction/processing equipment to the asteroid and position the “business end” of that equipment on the asteroid. The spacecraft would then extract water from the asteroid as both a commodity to be returned and sold and as propellant for the return trip. If the target asteroid has a radius of 15-20m or less, it would be possible to design a containment vessel that will completely encase the asteroid to simplify the process of recovering water from that body. This containment vessel approach was the option upon which the Phase 1 RAP study focused. This approach offers many advantages for mining small asteroids although other options will be needed for mining larger asteroids.

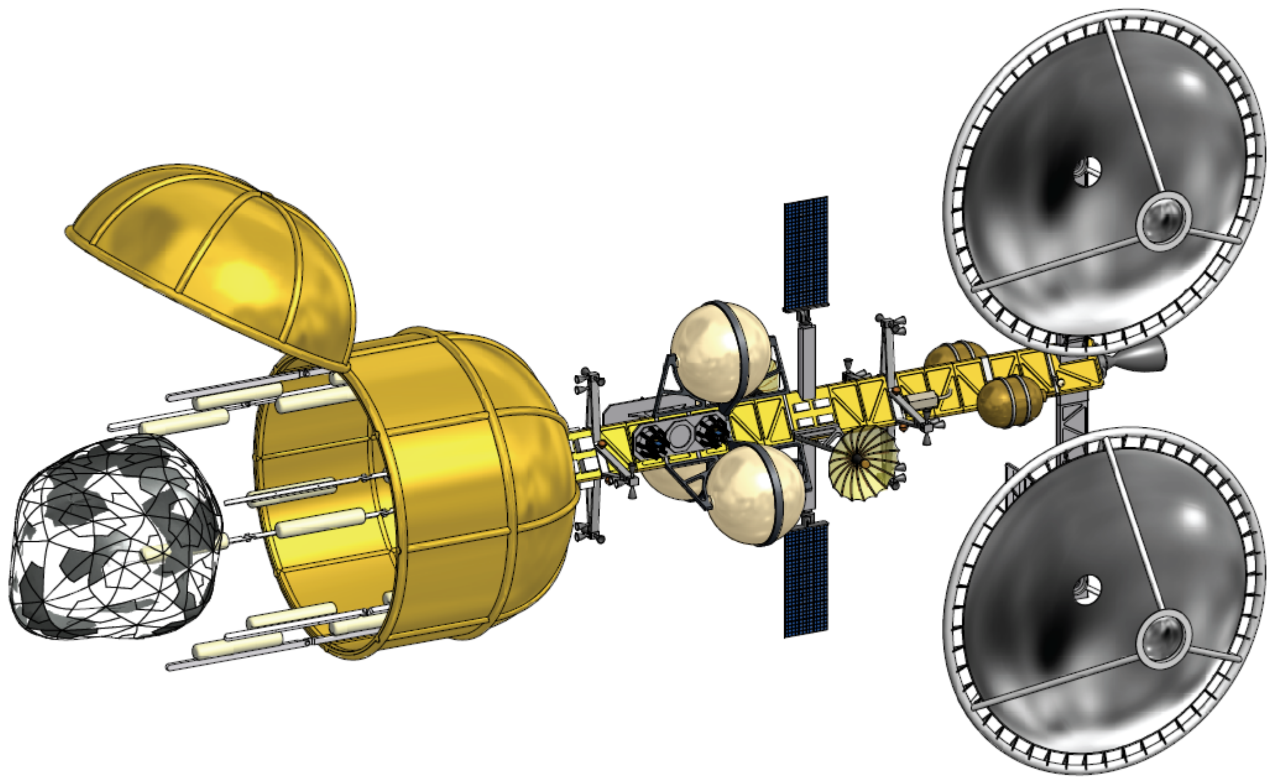


FIGURE 4.5. RAP Spacecraft Approaches Asteroid Along Its Polar Axis and Matches Rotation with the Asteroid

This mining/retrieval mission starts with the RAP spacecraft at the EML staging base and fueled with propellant, i.e. water that was delivered either from a Moon base or by previous asteroid mining missions. The departure starts with the spacecraft inserting itself onto an elliptical transfer orbit that takes it from the EML base to an Earth flyby at a distance of a few hundred kilometers. At the periapsis of this transfer orbit the spacecraft will perform a propulsive maneuver that inserts it onto the interplanetary trajectory targeted for the asteroid. The spacecraft traverses interplanetary space and a few months later arrives at the asteroid where it performs a propulsive maneuver that matches the spacecraft's orbit to that of the asteroid.

While in orbit around the asteroid the spacecraft will use remote sensing instruments to study the asteroid, characterize its rotation and prepare for the asteroid capture maneuver. When this phase of the mission is done the asteroid will maneuver so that it can approach the asteroid along it polar axis. While approaching the asteroid the spacecraft will spin around its longitudinal axis and match rotation rates with the asteroid. This alignment and synchronization appears in FIGURE 4.5.

Once the spacecraft has matched rotation rates with the asteroid and maneuvered so that the asteroid is within the containment vessel the asteroid will be secured to the spacecraft using a combination of mechanical linkages, air bags, nets or other devices as shown in FIGURE 4.6.

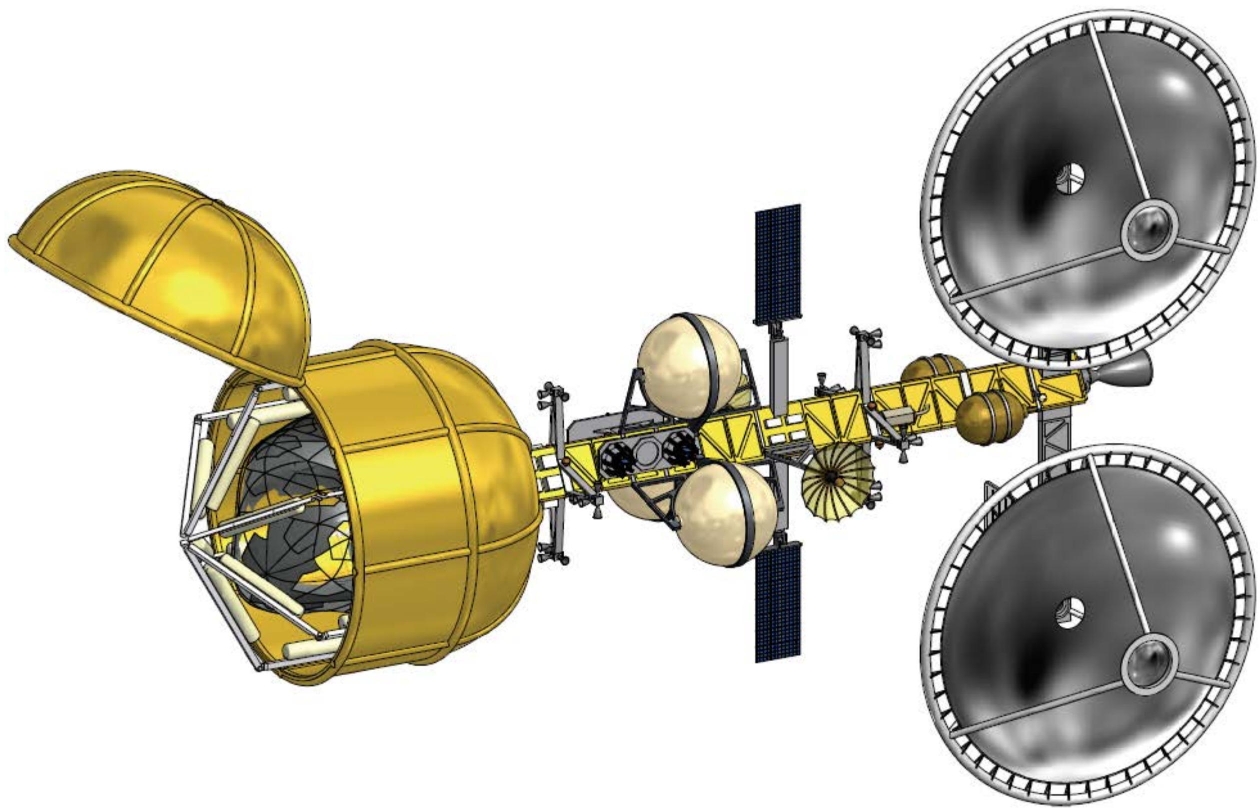


FIGURE 4.6. RAP Spacecraft Secures Asteroid Within Containment Vessel

Once the asteroid is secured within the containment vessel the front hatch of the containment vessel will be closed and secured as shown in FIGURE 4.7. Mining operations can now be commenced. Process heat provided by the solar collectors will be directed onto the asteroid, which will cause water to be released which will be caught within the containment vessel as a low pressure gas. Controlling the amount of heat applied to the asteroid will control the amount of water sublimated. The water vapor released from the asteroid will be condensed using cryo pumping and the resulting condensate will be delivered to the water storage tanks on the spacecraft. This process will continue until the water storage tanks on the spacecraft, along with the propellant tanks for the trip back to the EML staging base, are full. Once that has happened the spacecraft will be ready to return home with its economic bounty.

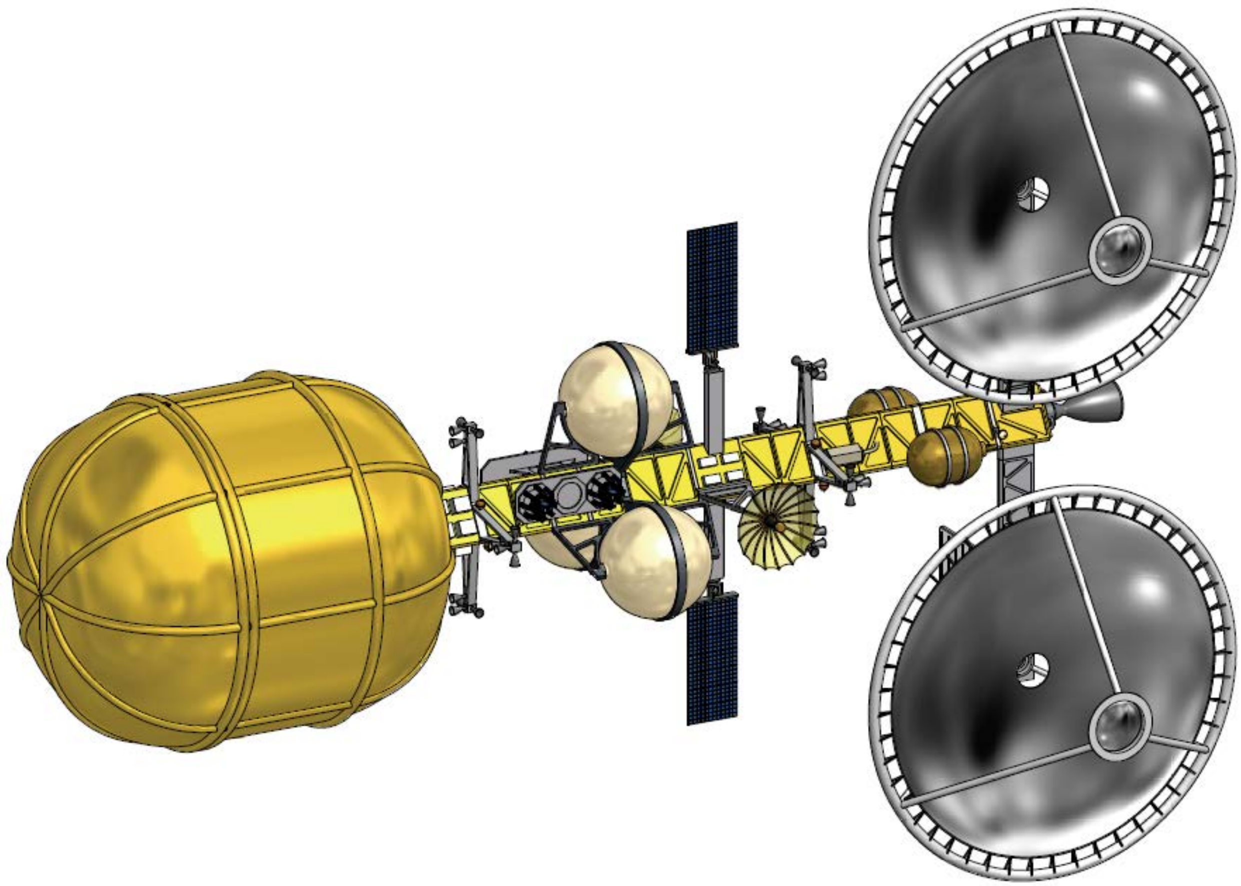


FIGURE 4.7. Containment Vessel Closes Around Asteroid and Resource Extraction Commences

After completing the mining operations, the RAP spacecraft will vent the residual gas within the containment vessel and open its hatch. The restraining linkages will release and the spacecraft will back away from the asteroid. Once the spacecraft is free of the asteroid, the containment vessel will close, and the spacecraft prepare for its return to Earth. When the departure period for the return to Earth opens, the spacecraft will perform a maneuver to insert itself onto a return trajectory for the EML staging base. Upon arrival back in Cis-Lunar space the spacecraft will insert itself into a loose capture orbit around the Earth and will then adjust its orbit through a series of propulsive maneuvers and, potentially, lunar flybys so that it returns to the EML staging base. Once there, the water from the payload tanks will be transferred to a storage facility and the spacecraft will be serviced for its next mission.

4.5.3 Processing Mission

The purpose of this mission is to extract economically meaningful amounts materials from an asteroid and prepare them for transport back to the EML staging base by a dedicated Transport Mission. This mission would start with the delivery of an autonomous extraction/processing plant to the asteroid. This facility would be emplaced at the asteroid so that it could continuously extract and process various products. These materials would be packaged for return to the EML staging base. When a sufficient quantity has been produced, a dedicated transport mission would fly to pick up those materials and return them to the

EML staging base. This mission type is noted here but was not analyzed in detail during Phase 1 of the RAP Study.

4.5.4 Transport Mission

The purpose of this mission is transport the material extracted by a dedicated processing mission back to cis-lunar space. In this mission the spacecraft would shuttle between the EML staging base and the asteroid being mined and would not transport significant amounts of payload out to the asteroid. It is expected that the propellant needed to return this vehicle and its payload back to the EML staging base would be provided by ISRU at the asteroid. This mission type is noted here but was not analyzed in detail during Phase 1 of the RAP Study.

4.5.5 Mining the Martian Moons

There is a fifth mission that we should mention although strictly speaking, it falls outside the scope of this Phase 1 NIAC study. The Martian moons appear to be compositionally similar to carbonaceous class asteroids. This similarity suggests that the RAP spacecraft could mine the water to provide propellant and life support consumables. An independent study of this possibility by our Co-I for Mission Design has shown that the asteroid mining technologies identified in this NIAC study could be used to support human missions to Mars. This application of the RAP technologies, mission design, and spacecraft design would result in significant reductions in the size of the spacecraft needed for such missions as well as large reductions in the associated mission costs.

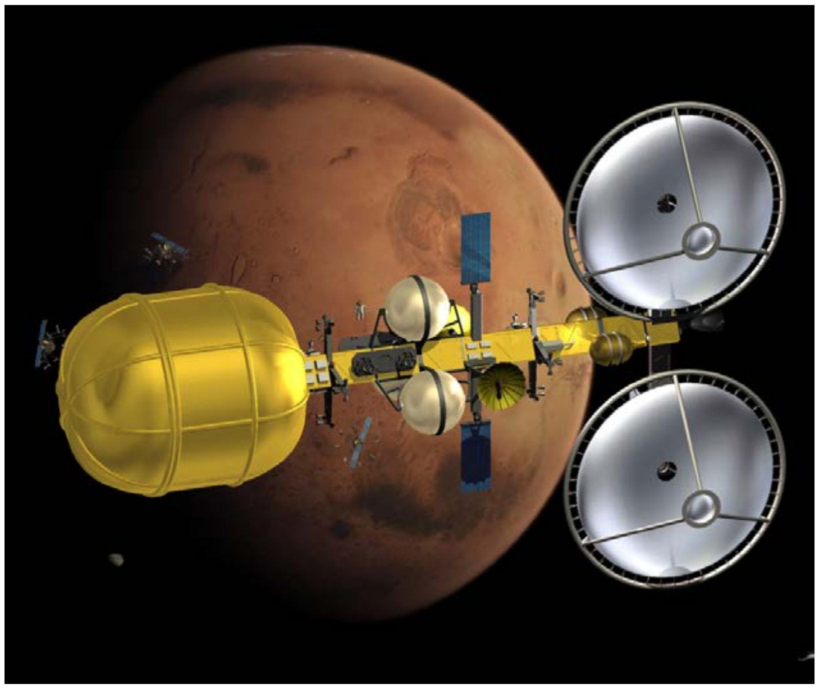


FIGURE 4.8. Mining the Moons of Mars

4.6 Preliminary ConOps for Mining/Retrieval Mission

The RAP team has developed a ConOps for both robotic and crewed/robotic missions. The scope of the current phase of the study only covered the robotic ConOps, however for the purpose of this report, it is important to show the continuity and consistency from the robotic to the crewed/robotic missions. A key commonality is that the principle RAP mission begins at a platform stationed at an EML point and concludes at an EML, presumably the same one from which the mission started.

4.6.1 Preliminary ConOps – Robotic

- Robotic spacecraft are launched into LEO and checked out.
- Vehicle is transferred to EML1.

    - Transfer may be performed by solar electric tug or spacecraft may be fueled with sufficient propellants so that they may 'self-ferry' to EML1.
    - Electric tug has performance benefits, self-ferry allows thorough checkout of both vehicles prior to departing for the asteroid and does not require development of an additional vehicle.
- Spacecraft are fueled with propellants from Moon.
- Depart EML1 via an Earth swing-by.
    - At the opening of the launch period the spacecraft enters an elliptical Earth orbit with a 300 km altitude periapsis.
    - At periapsis of the elliptical orbit (apoapsis at L1), the spacecraft performs a departure maneuver that inserts them onto an interplanetary transfer trajectory targeted for the asteroid.
- Spacecraft arrive at their destination and insert into orbit around the asteroid.
- Robots and prospecting modules deploy to the asteroid surface to begin prospecting and later mining.
- Samples and/or processed materials are loaded into the spacecraft for the return to Earth.
- When the launch period for the return to Earth opens the spacecraft perform a maneuver placing them onto the return trajectory that targets a 300 km minimum distance at Earth encounter.
- When the spacecraft reach Earth they perform an initial orbit insertion putting them into a highly elliptical orbit.
- Spacecraft performs maneuvers to rendezvous back at EML1.
- Payload returns to Earth via the most economical transfer system available.
- Interplanetary spacecraft remain at L1 and are serviced and prepared for their next mission.

4.6.2 Preliminary ConOps – Crewed and Robotic

- Robotic and piloted spacecraft are launched into LEO and checked out.
- Both vehicles are transferred to EML1.
    - Transfer may be performed by solar electric tug or spacecraft may be fueled with sufficient propellants so that they may 'self-ferry' to EML1.
    - Electric tug has performance benefits, self-ferry allows thorough checkout of both vehicles prior to departing for the asteroid and does not require development of an additional vehicle.
- Spacecraft are fueled with propellants from Moon.
    - Water for Solar Thermal Propulsion system doubles as radiation shield for crewed vehicle.
- Crew arrives on a fast trajectory in an Orion-class vehicle, and then transfers to the interplanetary spacecraft.
- Depart EML1 via an Earth swing-by.
    - At the opening of the launch period the spacecraft enter an elliptical Earth orbit with a 300 km altitude periapsis and an apoapsis at L1.
    - At periapsis of the elliptical orbit the spacecraft performs a departure maneuver that inserts them onto an interplanetary transfer trajectory targeted for the asteroid.
- Spacecraft arrive at their destination and insert into orbit around the asteroid.
- Robots and/or the EVA crew deploy to the asteroid surface to begin prospecting and later mining.
- Samples and/or processed materials are loaded into the spacecraft for the return to Earth.
- When the launch period for the return to Earth opens the spacecraft perform a maneuver placing them onto the return trajectory that targets a 300 km minimum distance at Earth encounter.

- When the spacecraft reach Earth they perform an initial orbit insertion putting them into a highly elliptical orbit to apoapsis at EML1.
- Both Spacecraft perform maneuvers to rendezvous back at L1.
- Crew returns to Earth using an Orion-class vehicle.
- Payload returns to Earth via the most economical transfer system available.
- Interplanetary spacecraft remain at L1 and are serviced and prepared for their next mission.

5 Spacecraft Design

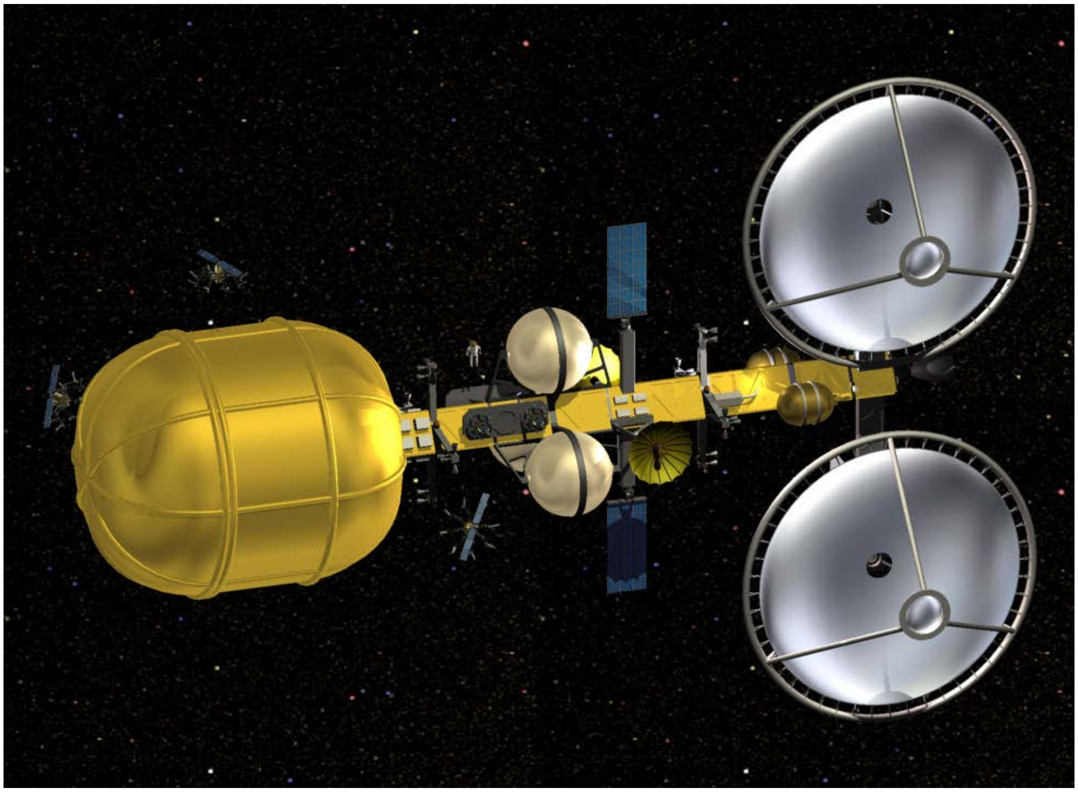


FIGURE 5.1. RAP Spacecraft Reference Configuration

The RAP spacecraft in FIGURE 5.1 will be required to fly multi-year missions to selected asteroids, support mining operations at those asteroids and then return 150 metric tons, or more, of processed materials back to Cis-Lunar space. This problem poses is a challenge in spacecraft and mission design but is one that is well within the current state of the art for astronautical engineering. The majority of the functions that the spacecraft will be called upon to perform, such as propulsion, power, GNC, communications and thermal control, are functions that have been performed by all spacecraft that travel into deep space. The big difference for the RAP spacecraft lies is in the scale of the spacecraft and its payload. Previous sample return missions have transported a few grams – or at most a few kilograms – of samples back to Earth from deep space destinations whereas the RAP spacecraft will return hundreds of metric tons. This size difference can be daunting, but it does not represent a fundamental difficulty, just a challenge to the scale of the spacecraft systems normally used for deep space missions.

The central challenge for the design of the RAP spacecraft is to devise a system that performs the required functions in a reliable and economic fashion. This challenge will drive all of the design choices when putting together a design for the RAP spacecraft. Consider just a few examples.

There are many possible target asteroids for RAP missions; each of them will have a unique Delta V budget and associated required propellant mass. We expect that there will be a large variation in these requirements over the mission design space. It will be critical that the spacecraft accommodate these variations without requiring a major redesign for each mission. One approach to accommodating this need for flexibility and adaptation would be to add or subtract propellant tanks as required but if we do that then the propellant tanks cannot be used as primary structure. This insight suggests that the spacecraft's primary structure should be a truss that can carry propellant tanks sized for the specific mission being flown. This arrangement allows easy reconfiguration of the spacecraft but does come at the cost of a small inefficiency since the truss structure is providing a function that the propellant tanks might provide instead.

Since reducing the mass of the spacecraft is important for maximizing mission performance and minimizing costs, it is desirable for the spacecraft subsystems to support more than one function. The preceding paragraph showed an example of a case where that was not possible but there are other cases where multi-function systems can be incorporated into the spacecraft design. For example, a propulsion system than can provide power and/or thermal energy in support of mining operations as well as the normal propulsion functions is greatly to be preferred over a propulsion system than can only service the propulsion needs of the spacecraft. This precept is one of the factors that drove the RAP design to use Solar Thermal Propulsion for the primary propulsion system for this spacecraft. This system would use solar concentrators to heat a working fluid to provide propulsion and once at the asteroid those same solar concentrators could be used to collect and direct solar energy onto the asteroid for use in extracting water from the asteroid.

The development and production cost of the RAP spacecraft will be substantial. To achieve good operating economies, it will be necessary to spread those costs over the largest possible mission base and operating lifetime. Therefore, we must design the spacecraft for multiple missions. That is why we need a spacecraft that can easily be reconfigured for a variety of missions as opposed to being redesigned to accommodate each new mission. Moreover, to amortize the manufacturing cost of the spacecraft it will be important that the spacecraft be reusable with minimal servicing between missions. Our economic analysis shows that a reconfigurable spacecraft that can fly multiple missions is essential for getting the costs of asteroid mining down to a level that make economic sense.

This team has created a design for the RAP spacecraft that is capable of flying these missions. Although this design does require engineering development work on a new propulsion system virtually all of the other elements of this spacecraft design are completely conventional if not off-the-shelf. Most of the engineering challenges for the RAP project will be focused on the payload carried by this spacecraft, i.e. the mining/processing equipment, since those aspects of the system design deal with functions that have not done before in deep space.

5.1 Spacecraft Requirements

The following are the system level requirements for the RAP spacecraft.

- Provide the services needed to perform the mission such as communications, command sequence processing, thermal control, power generation, guidance, navigation and control, and propulsion.

- Provide the Delta V needed to fly a round-trip asteroid mission and be able to scale up the system to various Delta V requirements without requiring a major redesign of the system.

- Provide systems for attaching the spacecraft and mining equipment to the target asteroid.
- Provide support for the systems used to extract and process resources from the asteroid.
- Provide the systems used for returning processed asteroid resources back to Cis-Lunar space.
- Be reusable.
- Be capable of operating from an EML point.
- Be capable of operating for mission durations lasting between one and five years.

5.2 RAP Spacecraft Reference Design Description

FIGURE 5.2 shows an overview of the configuration of the RAP spacecraft and highlights some of the key attributes of the spacecraft which are described more fully below while FIGURE 5.3 provides dimensions for various spacecraft components.

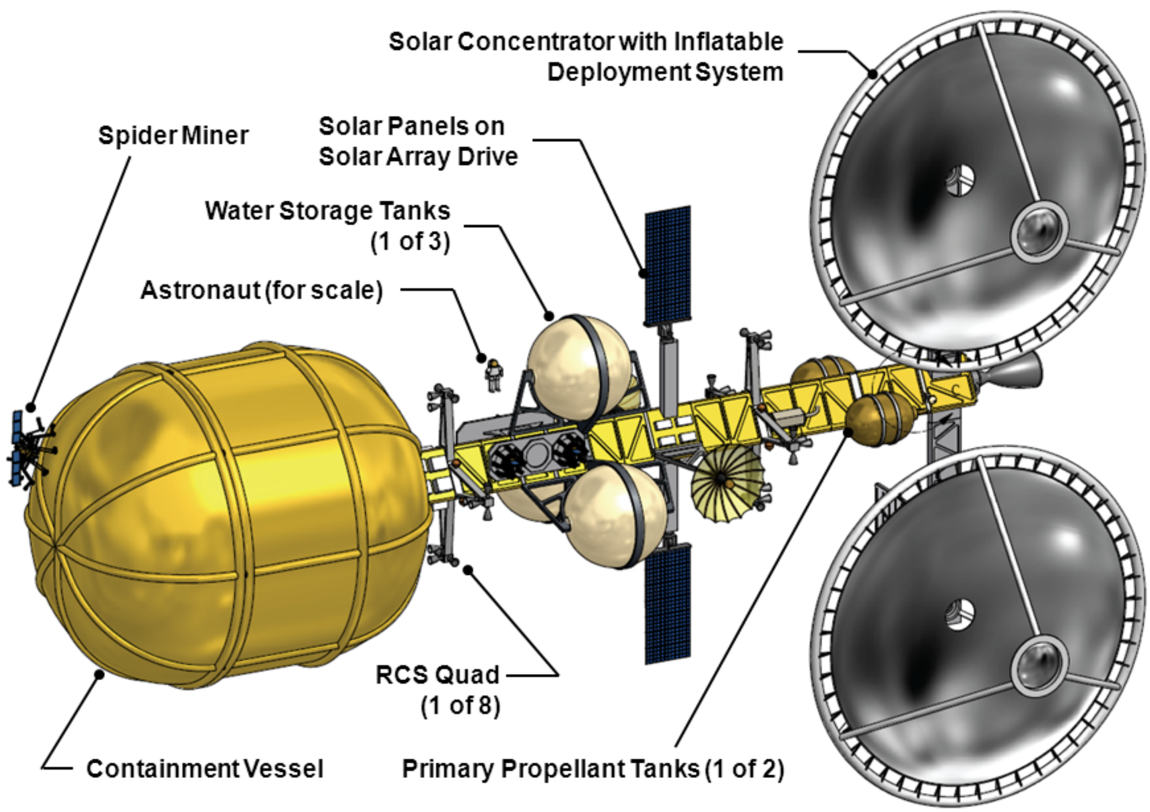


FIGURE 5.2. RAP Spacecraft Reference Configuration Showing Containment Vessel

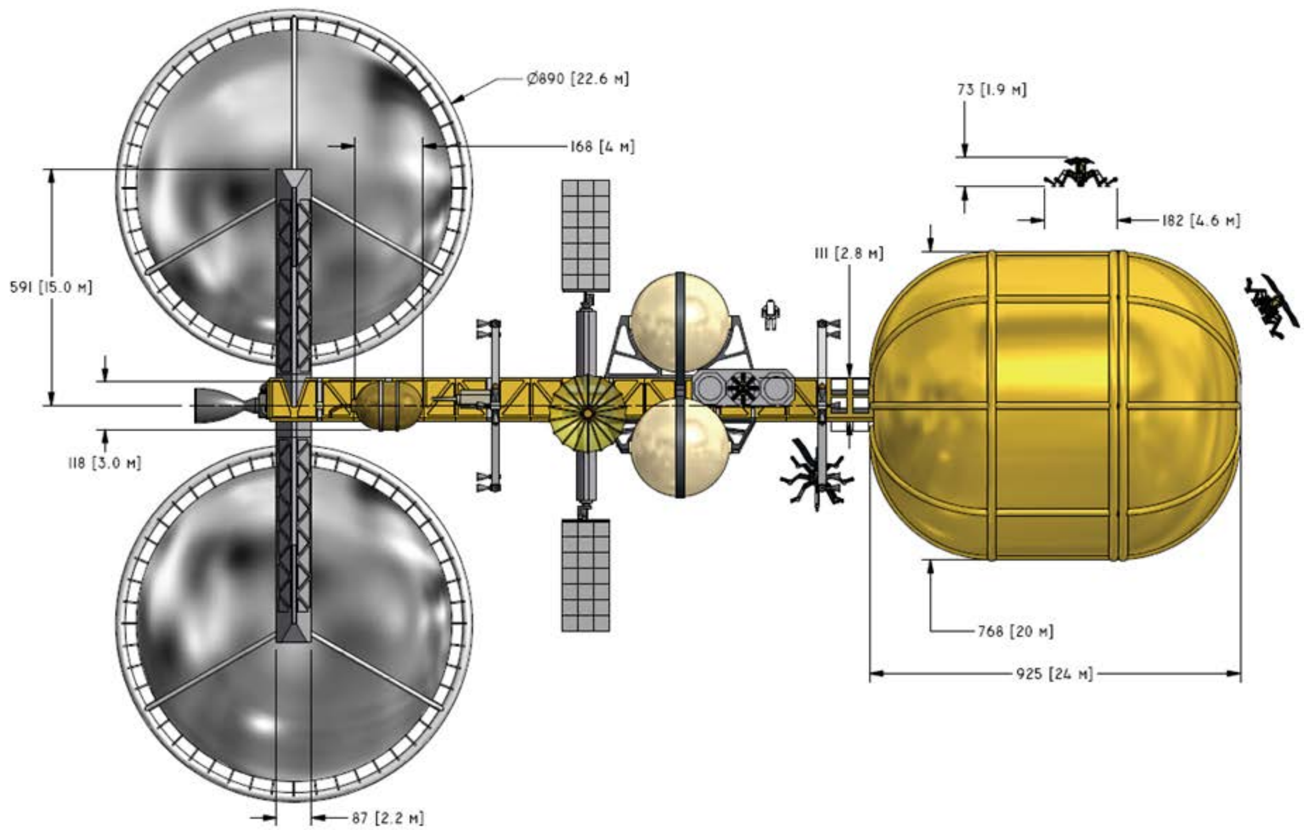


FIGURE 5.3. RAP Spacecraft Reference Configuration with Dimensions

5.2.1 RAP Spacecraft Truss Structure

The structural backbone of the RAP spacecraft is a graphite composite truss of triangular cross section. It is 40 meters long and three meters wide. This truss provides the sufficient real estate to mount all of the various spacecraft systems and sub-systems. Graphite composite was selected for the truss because of the low weight of composite structures and because these materials have a low coefficient of thermal expansion. This later property will help provide a rigid and stable structure when the spacecraft is required to hold a fixed orientation relative to the direction of solar illumination.

There were two driving factors in selecting a truss structure for the RAP spacecraft.

1) <u>The need to be able to scale up the propulsion system for a variety of Delta V budgets.</u> We can easily achieve this scaling by adding extra propellant tanks or by replacing the existing tanks with larger ones. This scaling would not be possible if the propellant tanks served as the primary load bearing structure.

2) <u>The need for a clear and unobstructed path to direct concentrated solar flux</u> from the Solar Collectors to the Containment Vessel. The operational plan for the extracting water from the asteroid requires that solar energy be directed onto the asteroid in the containment vessel thus heating it and driving out the water. The truss structure provides an unobstructed path for such energy transmission through the interior of the truss.

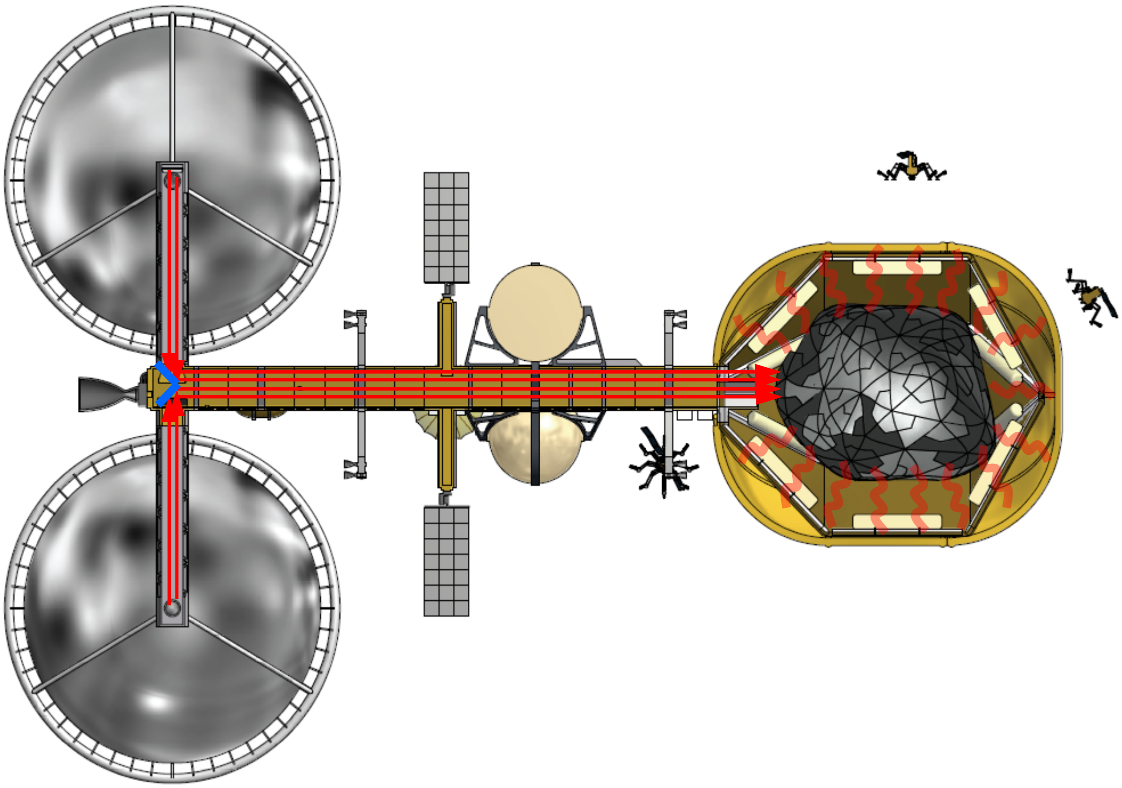


FIGURE 5.4. Distribution of Concentrated Solar Flux Energy from the Solar Collectors

FIGURE 5.4 shows how the solar collectors incorporate a secondary mirror that can to redirect the concentrated solar flux away from the engine and toward the asteroid that is being held within the Containment Vessel. This light will pass through the truss, enter the Containment Vessel through a series of windows and then be directed onto specific parts of the asteroid through the use of other mirrors or fiber optic. Pointing the solar collectors obliquely off the sun line can modulate the amount of energy being directed toward the asteroid.

5.2.2 RAP Spacecraft Subsystem Hardware

FIGURE 5.5 shows how the subsystem hardware is mounted to the exterior of this truss. Each subsystem box - such as the radio, power conditioning equipment, computer, GNC, and others – is connected to a structural mounting plate by several attach fittings with additional connections for power, data and thermal control. A hinged non-structural MMOD shield covers these boxes to provide for protection from in-space debris and to moderate the thermal environment.

The conventional approach for configuring a spacecraft is to package the subsystems in individual boxes and then mount these within the spacecraft. The RAP approach turns this approach inside out by taking those subsystem boxes and mounting them on the outside of the spacecraft. This approach allows each subsystem box to be configured as an easily accessible orbital replaceable unit (ORU) in order to simplify servicing of the vehicle between missions.

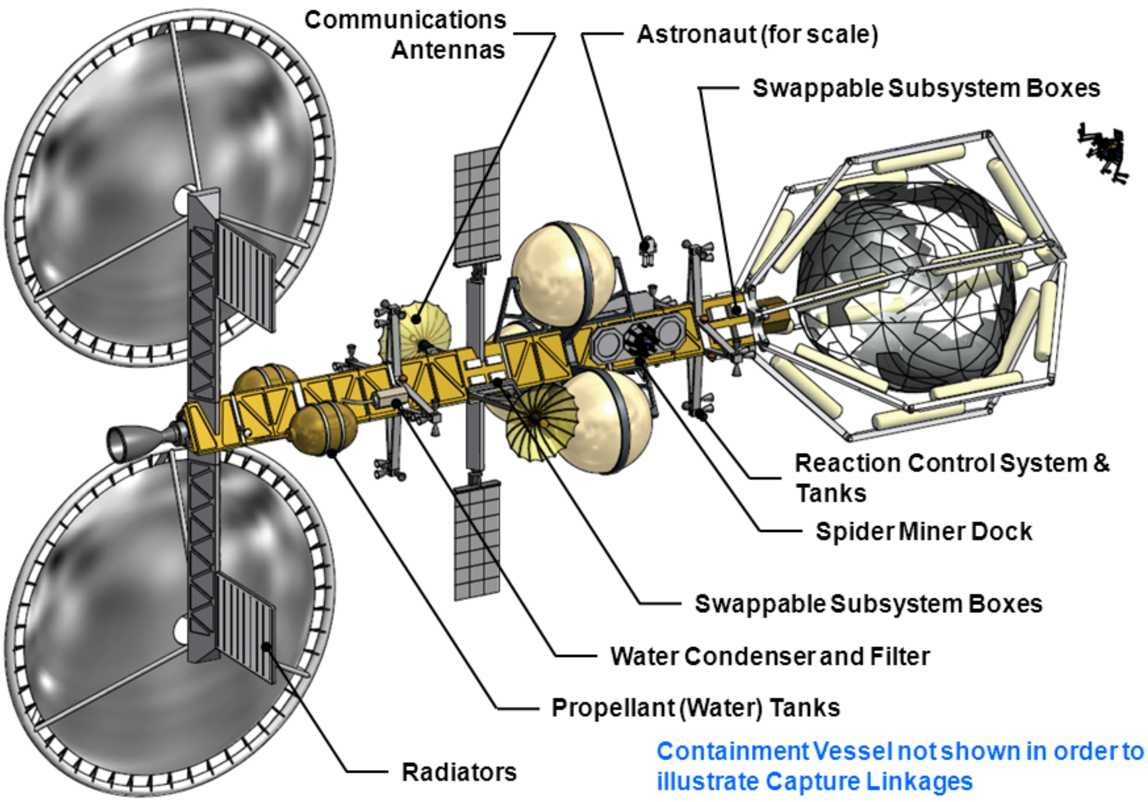


FIGURE 5.5. RAP Spacecraft Reference Configuration with Containment Vessel Removed

If a subsystem fails, or is flagged as having a problem by the vehicle health monitoring system, an astronaut can replace it once the spacecraft returns to the EML staging base. At that time the hinged MMOD shield will be raised, the power, data and thermal control fittings disconnected, and the attach fittings released. The ORU can then be removed and a new one put in its place. The attach fittings will be reconnected, the power, data and thermal control fittings reattached and the hinged MMOD shield lowered back into place. The various fittings on the ORU will be designed to accommodate operations by a space suited astronaut or a telepresence robotic device.

Having three completely independent sets of subsystem hardware enhances the reliability of the spacecraft - any one of these hardware strings will be capable of handling all of the required spacecraft functions. Putting each set of subsystem hardware at different locations on the spacecraft enhances the system reliability. This prevents an accident that damages one hardware box from damaging the redundant hardware.[8] Additionally the redundant subsystems will be wired so that the failure of one hardware box in a string does not require switching to the other string. Rather, the failed hardware element from one string can be taken off-line and be replaced by the corresponding element from one of the other hardware

[8] The reader will recall that in the Apollo 13 accident, an explosion that might have only destroyed one cryogenic oxygen tank instead destroyed the other tanks on the spacecraft through fratricide. Had those other tanks not been located next to the tank that exploded they would have survived the original explosion and the crew would have been at much less risk.

strings. This will maximize the flexibility for dealing with hardware faults and insure that the failure of one subsystem does not make the other redundant hardware on its string unavailable for use. Furthermore, within each subsystem box there will be component level redundancy for critical hardware elements. The RAP spacecraft will thus be able to accommodate the total failure of two complete subsystem strings and still be able to complete its mission.

A series of thruster quads located on booms provide attitude control. Mounting these thruster quads there increases the torque they can produce and thus reduce the attitude control propellant requirements. The propellant for these thrusters is currently baselined as a hypergolic bi-prop although other alternatives can be considered in Phase 2 of the RAP study. The location of the thruster quads represents a trade between available torque and plume impingement on containment vessel and the solar concentrators. The numbers and locations of the thrusters allows for translational motion along any axis independent of the attitude of the spacecraft and allows attitude maneuvers to be performed without introducing any translational motion to the spacecraft.

Having eight thruster quads provides a substantial amount of redundancy in the attitude control system. With this system there are four thrusters available for translational motion along any axis. If one of those thrusters fails it is still possible to translate along that axis using two thrusters although the acceleration will be lower. If one of those two thrusters subsequently fails it is still possible to translate along any axis but now the spacecraft will have to first perform an attitude adjustment to allow translation along the selected axis. Thus, the design for the RAP attitude control system, allows for effectively un-degraded translational performance with one failed thruster and degraded performance with two failed thrusters. In a similar vein the system would be tolerant of two thruster failures for any rotational axis and could tolerate even more failures if unbalanced rotational maneuvers are allowed.

A pair of articulated flat panel photovoltaic (PV) solar arrays provides electrical power for the spacecraft bus generating a few tens of kilowatts of power. That level of electrical power is more than adequate for powering the basic spacecraft functions. The use of solar panels to provide spacecraft power might be perceived as reducing the benefits of using a Solar Thermal Propulsion system which could also incorporate a Solar Dynamic Power system. However those systems suffer from weight and reliability issues and are not needed for low power applications. Since direct solar energy is used to provide process heat for mining operations there is no need to generate large amounts of electrical power to support mining operations. Thus PV solar arrays are the preferred system for generating spacecraft power. If the mining/processing approach changes to one using electrical power rather than direct solar heating then there could be a benefit associated with using a Solar Dynamic power system rather than a photovoltaic system and we would have to revisit this system design decision. However the current approach to mining/processing focuses on the direct application of solar thermal energy and does not require the use of electrical power levels that would justify the weight of a solar dynamic power system.

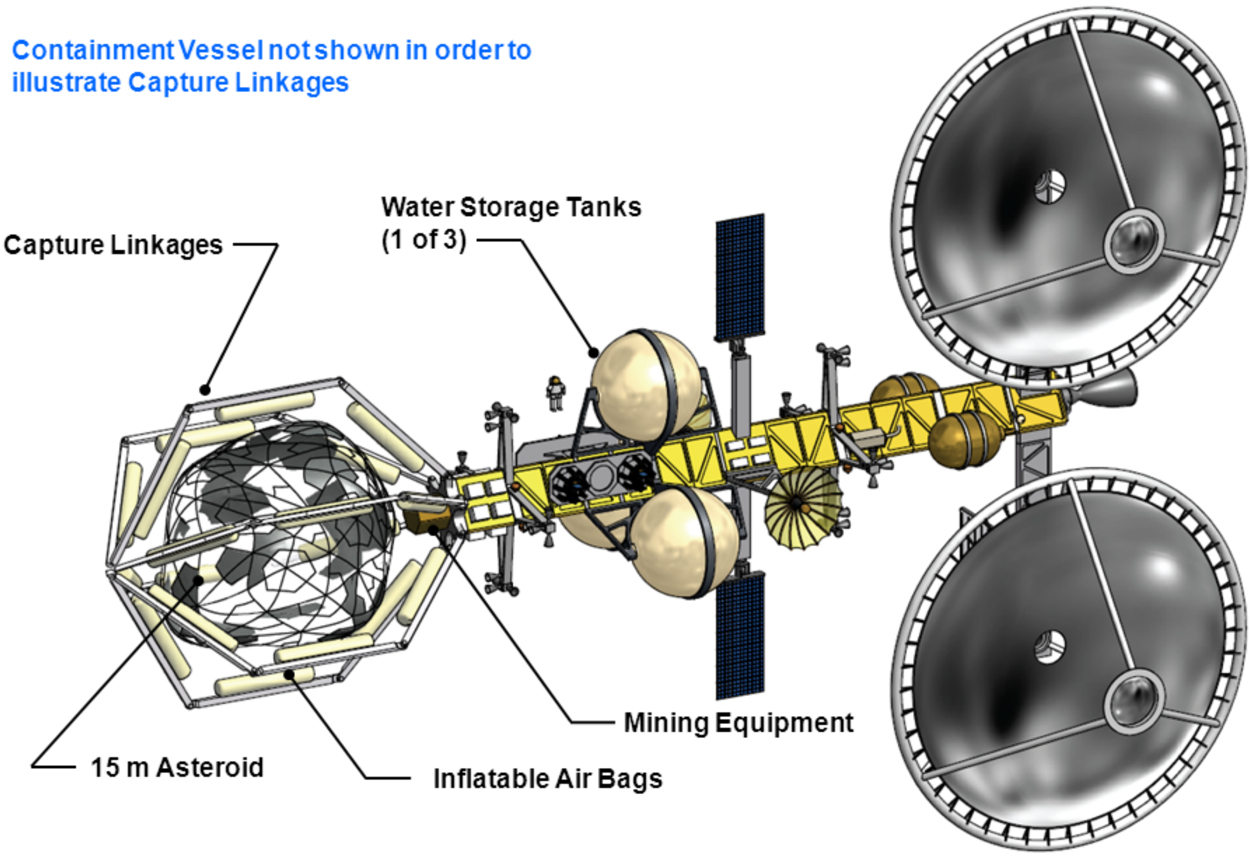


FIGURE 5.6. RAP Spacecraft Reference Configuration Showing Asteroid Restraint System

5.2.3 Propellant Tanks

As shown in FIGURE 5.6, there are three 6m-diameter water storage tanks located in the forward section of the truss. These tanks are sized to hold several hundred tons of water, which is the amount that may be extracted from a water-rich 15m to 20m meter asteroid. This water is the payload that the RAP spacecraft will return to cis-lunar space. These tanks are plumbed and filtered so that some of this water can also be used as reaction mass for the return journey. The spacecraft is designed so that the size of the water storage tanks can be easily adjusted to accommodate missions with various Delta V budgets and return payload requirements. These tanks are inflatable so that they can be easily packaged for being launched into space.

5.2.4 Anchoring to the Asteroid

The mining/processing equipment is located at the forward end of the structural truss. There are many approaches that can be taken for configuring this equipment and so it is important to understand that this configuration is being shown as a point of departure for discussion and analysis and not as a representation of a definitive design.

A major issue for the design of the spacecraft is how the spacecraft will anchor itself to the asteroid. The immense variation in size, structure, physical and chemical composition for likely asteroid destinations exacerbates this problem. The design presented is suitable for attaching the spacecraft to asteroids with a maximum diameter of 20 meters. This dimension represents a good size for early mining missions since an asteroid of this size may contain between 350 and 1,500 metric tons of water that is recoverable, depending upon the asteroid's shape, intrinsic water content, and the location of its water deposits.

The resource extraction occurs within an inflatable containment vessel, as described in the mining technology chapter. This structure is given its shape by s series of inflatable ribs and truss members give this structure its shape, which does not depend upon the gas pressure within the containment vessel to hold its shape. The forward dome of the containment vessel is hinged and can open to allow the asteroid to pass inside the spacecraft. A series of mechanical arms, or linkages, attached to the spacecraft's structural truss are used to hold the asteroid in place.

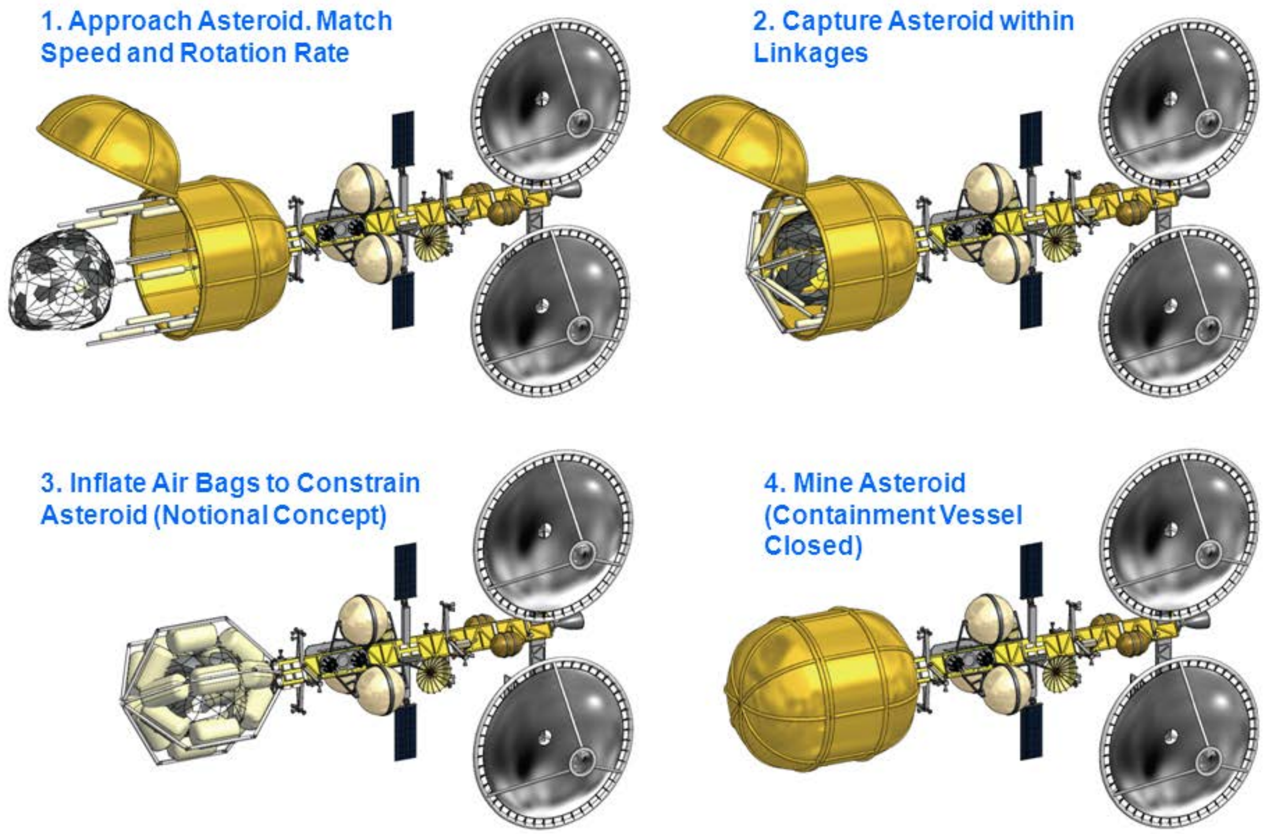


FIGURE 5.7. Notional Approach for Anchoring the RAP Spacecraft to an Asteroid

The strategy to secure the spacecraft to the asteroid, shown in FIGURE 5.7, is to approach it along its polar axis with the forward hatch of the Containment Vessel open, match rotation rates with the asteroid, and then engulf it with a series of linkages or arms. Airbags then inflate to constrain the asteroid and restrict its ability to move relative to the spacecraft. After securing the asteroid, the containment vessel hatch closes and mining/processing can commence. This strategy of engulfing the asteroid in the containment vessel applies to asteroids up to 20m in diameter. Larger than that, and the RAP spacecraft cannot envelope it, but must instead anchor at the pole by mechanical means.

Heating the asteroid by directed, concentrated solar energy will release water vapor. This water can be collected according to the methods described in the previous chapter on mining technology. The recovered water can be transferred to the water storage tanks for use in powering the return to Cis-Lunar space, or stored as a recovered resource to sell on the in-space market at the end of the mission. The containment vessel will also keep the debris from the water extraction process from escaping. It will then be possible to use other systems such as remote manipulator system (RMS)-like arms within the containment vessel to manipulate the asteroid materials and perform other mining/processing activities.

5.2.5 Solar Thermal Propulsion

The primary propulsion system for the RAP spacecraft is solar thermal propulsion (STP) using water for its propellant mass. This STP system has many advantages for this type of mission but the biggest one is that it uses a propellant that is available at the asteroid for fueling the return to cis-lunar space. This significantly reduces the mass of the spacecraft when departing cis-lunar space. It also allows the spacecraft to return an arbitrarily large mass of asteroid material by adjusting the amount of water that the spacecraft takes on from the asteroid for the return voyage. Additionally the high density of water as a propellant means the propellant tanks will be significantly smaller than tanks for carrying a similar mass of LOX/LH2. Using liquid water also allows us to avoid the issues of storing and conditioning cryogenic LOX/LH2 in space for extended periods of time.

The propellant tanks, as shown in Figure 5.8 and the preceding figures, were sized based on the expected *I*sp of the Solar Thermal Propulsion system and the previously discussed DeltaV budget for the asteroid mission. It is important to note that these tanks are sized for the outbound trip only since the propellant for the return trip is acquired at the asteroid. Thus, when the spacecraft arrives at the asteroid the primary propellant tanks will be essentially empty. Eyes that are accustomed to the size of LOX/LH2 tanks for conventional vehicles will find that the propellant tanks for the RAP spacecraft appear small but their small size is a result of the reduced Delta V budget thanks to asteroid ISRU and the fact that the density of water is more than an order of magnitude higher than the density for LOX/LH2.

Another benefit of using water as the reaction mass for the Solar Thermal Propulsion System is that it is relatively easy to move the water from the tanks to the engine even when the vehicle is not under acceleration. This transfer is accomplished by having the small startup tanks shown in FIGURE 5.8 located close to the base of the main tanks. These tanks consist of an inner flexible bladder with an outer rigid tank made of metal or composites. Water that is in the startup tank can be expelled from the tank by introducing a pressurant gas into the space between the rigid tank and the flexible inner bladder. The pressure from this will force the water out of the bladder and into the engine.

Once the engine is producing thrust and the vehicle is under acceleration, the water in the main tanks will settle to the bottom of the those tanks where it can be forced into the engine using conventional pumps. By plumbing the system so that the water flows through the startup tank as it flows from the main tank to the engine we can insure that the startup tanks are always filled with water. During the startup process closing a valve between the startup tank and the main tank will insure that the water will flow into the engine and not the main tank when the water is expelled from the startup tank. Systems such as this are in use today for hypergolic propellants but are not useable with LOX/LH2 propulsion systems since there are no materials that can be used for a flexible bladder at cryogenic temperatures. If a working fluid other than water were to be used this solution would then require the addition of an auxiliary propulsion system to provide a settling impulse before the main propulsion system could be used and this would introduce unwanted cost, complexity and reliability issues.

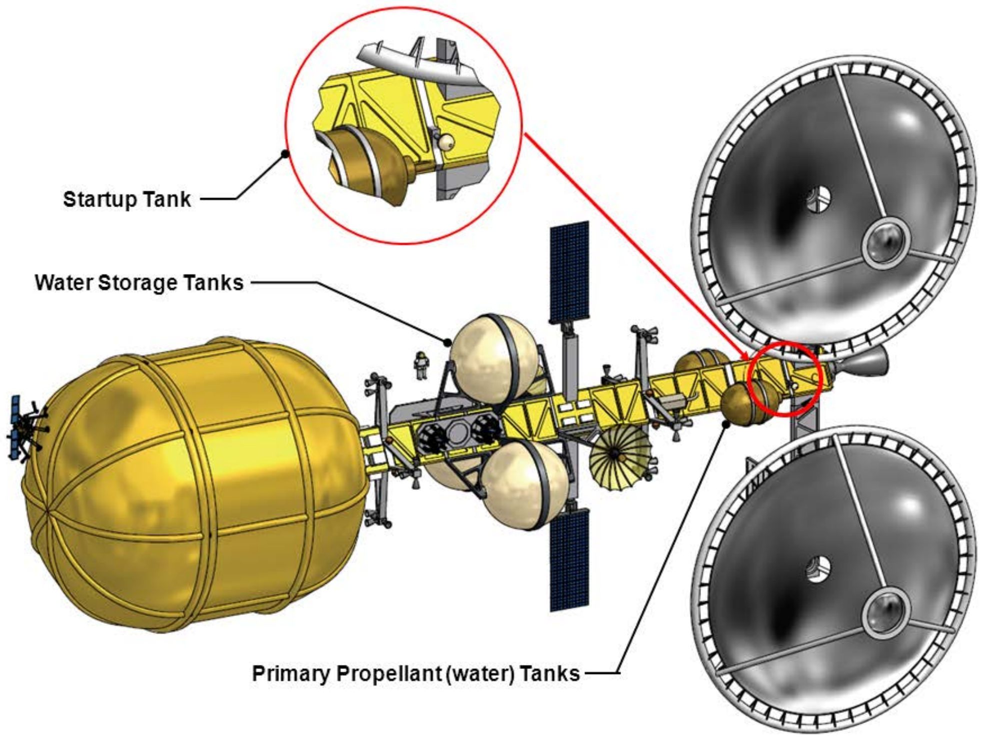


FIGURE 5.8. RAP Spacecraft Startup Tanks

One of the most prominent features of the spacecraft is the pair of Solar Collectors used for the Solar Thermal Propulsion System. The collectors gather approximately one Mw of solar energy and concentrate it onto a heat exchanger which heats the reaction mass, in this case water, and generates a high temperature gas that provides the reaction force that drives the spacecraft. {It would not be inaccurate to describe this as a steam powered rocket.} This engine will provide significantly more thrust than an electric engine although less than from a conventional chemical engine. The thrust level achievable from an engine of this type is large enough to allow interplanetary transfers that are significantly faster than those provided by electrical propulsion albeit at the cost of higher propellant mass requirements.

Using water as the propellant mass for STP will allow an *I*sp comparable to most chemical propellants but not as good as LOX/LH2. This difference arises because the maximum allowable temperature for the heat exchanger will be lower than the combustion temperature of LOX/LH2. Most proposals for STP systems such as this one baseline using LH2 as the reaction mass, in which case the *I*sp would be comparable to that of a Nuclear Thermal Rocket. However, that would require long-term cryogenic storage of LH2 and would not make the best use of the ISRU resources available from the Moon or the asteroids.

Although the Solar Collectors are rather large, they are not massive. It will be possible to build these collectors using Mylar films, which will keep the mass of the mirrors down to a few hundred kilograms or less. It should be noted that the design of the mirrors shown in the various figures is just a place- holder to indicate the size of the mirrors and to aid in making preliminary mass estimates. There are many trades that we will need to perform before we can select the final configuration for these solar concentrator mirrors.

The sunlight energy gathered by the solar concentrators will focus at the focal point of the concentrator, from which a series of mirrors will redirect it to the engine's heat exchanger. It will be necessary to provide cooling for the intermediate mirrors in this series so that these mirrors do not over heat and deform or become damaged. There is a final mirror in this optical train that can be used to redirect the solar energy from the engine's heat exchanger to the asteroid processing equipment in the Containment Vessel. All this heating of mirror surfaces means that the STP design must account for energy losses from the sunlight flux.

As the spacecraft moves away from the Sun, the amount of solar insolation drops, so that the thrust of the engine decreases but the *I*sp does not. This effect occurs because the operation of the engine is a balance between the energy reaching the STP engine heat exchanger via the solar concentrators and the heat that is being removed by the propellant as it passes through the heat exchanger. Once the heat exchanger achieves its working temperature, the reaction mass passing through it will remove the excess energy being deposited there and prevent the temperature of the heat exchanger from rising. Thus, the amount of reaction mass that can flow through the heat exchanger is set by the amount of energy being put into that device. If the amount of energy directed to the heat exchanger declines then the mass flow through the engine will correspondingly decline. But this change will not alter the operating temperature of the heat exchanger. Since the thrust an engine produces is a function of the mass flow rate through the engine this reduction in propellant flow rate will change the thrust the engine produces. But since the *I*sp, for a fixed exhaust gas composition is a function of temperature the Isp will not change since the temperature has not changed. This means that as the spacecraft moves away from the Sun, the thrust of its engine will decline but the *I*sp will not change. This reduction will only cause a small operational impact on the design of the missions for this spacecraft, since we can compensate for the reduced thrust by using longer burn durations.

5.2.6 RAP Spacecraft Summary

Our design for the RAP spacecraft constitutes a complete system for interplanetary exploration, at least within the "inner planets" extending out to Mars and its Moons. In this region, the sunlight is sufficient to power the solar thermal propulsion engine and the associated capabilities. FIGURE 5.9 shows the "standard" orthogonal views of the RAP reference configuration. This image shows additional details for the relative sizing of the elements without distortion from the earlier perspective views. Additional features that appear more distinctively here include the thermal control system radiators, the quad thrusters on booms, the relative size of the propellant and payload tanks, and the clear range for the solar concentrators to swivel 360° on their alpha joints.

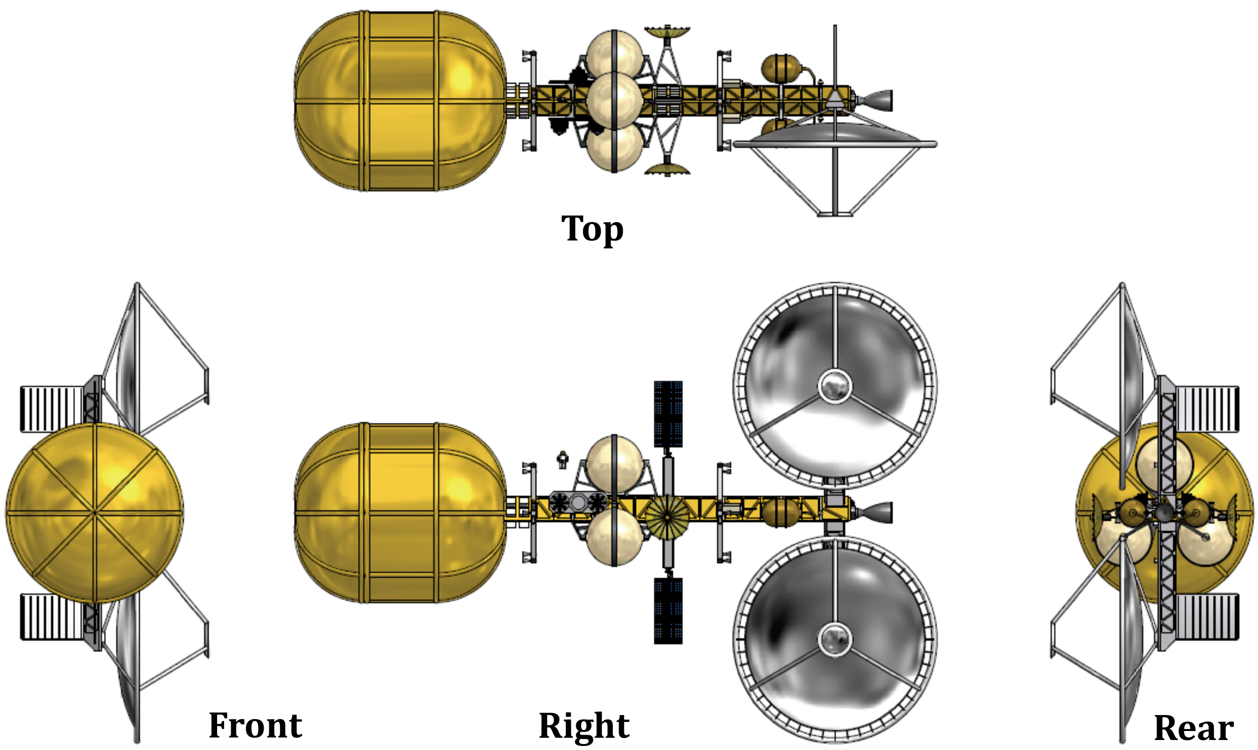


FIGURE 5.9. RAP Spacecraft Reference Configuration, Standard Views

6. Parametric Cost Analysis

6.1 Overview

The key to making asteroid mining a reality will be to demonstrate the ability to deliver useable products to their final destination for a lower cost than competing systems. Initially, those competing systems will all operate from Earth. For the Phase 1 RAP study, this business case means delivering water to an EML point for a lower cost than delivering water – or some other propellant – there from the Earth. To verify that our system achieves this goal requires us to estimate the total system cost for this venture. This question is unfortunately very complicated, and many factors that cannot be well established at this time, go into this calculation.

Determining the cost of a venture of this sort requires more than just estimating the development and operational cost for the system. It requires estimating the size of the market for the product, devising a strategy for setting the selling price of the product, estimating reductions in system development costs achieved by leveraging investments in previous spacecraft, details of how the venture will be financed, interest rates from external economic markets. These factors and many other can be established only by creating a business plan that we could share only under the protection of a draconian Non-Disclosure Agreement (NDA). Moreover, since this asteroid mining project would move forward as a commercial venture and not a traditional NASA project, it is likely that conventional cost estimation methodologies would not accurately estimate the development and operations costs. Recent studies of the development costs for the Falcon and Dragon vehicles from SpaceX suggest that traditional cost estimating techniques would have over-estimated the development costs for those projects by two to four times.

We need a way to get a first look answer to the question, “Does asteroid mining make economic sense?” For Phase 1 of the RAP study we did not try to find a definitive answer to this question but instead used a simplified model to get an answer that can – hopefully - serve as an existence proof to the proposition that asteroid mining makes economic sense. If this simplified approach indicates that mining asteroids and returning their material to EML-1 costs less than sourcing those materials from the Earth, then we can have some confidence that we will be able to make asteroid mining work as an economic proposition. Additionally, it is possible to use this first look analysis as a way of setting limits on the costs of various parts of the project, and as a way of setting targets for the development and operations cost of this venture.

6.2 What is the Price Point to Make Asteroid Mining a Success?

Before we begin discussing the expected cost of returning material from a Near Earth Asteroid we need to establish the price point for materials delivered from the Earth to EML-1. The measure of economic success for asteroid mining is whether it costs less to deliver water – or some other resource – to an EML point than it costs to deliver that resource there from the Earth. Thus, we need to estimate what it will cost to deliver materials from the Earth to EML-1 using launch vehicles likely to be operational during the timeframe of interest.

In this context we are talking about the delivery of bulk cargo and not the delivery of small discrete payload elements such as are carried to the ISS under current commercial cargo contracts. This distinction is important because operational decisions that NASA made concerning how to deliver cargo to the ISS resulted in cargo transport systems where most of the launch mass is consumed by the cargo transport vehicle with only a small part of the launch mass being the actual payload delivered to the ISS. For example, the Falcon 9 carries a payload of approximately 9,360 kg to the ISS altitude and inclination, but the Dragon spacecraft is able to deliver a maximum payload of only 3,310 kg to the ISS. Thus, the payload to the ISS is 35% of the launch capability of the Falcon 9. Moreover, we can estimate the cost of that payload by noting that NASA's Commercial Resupply contract with SpaceX purchased 12 launches for a total cost of $1.2B. This contract translates into a cost per kilogram of just over $40K, if each of those flights were to carry the maximum payload that a Dragon can deliver. This number stands in stark contrast to the $5,770/kg cost for using a Falcon 9, based on a launch cost of $54M as quoted on the SpaceX web site. This price shows us that the cost to deliver bulk cargo can be significantly lower than the cost of delivering supplies to the ISS but to do this we need to develop a payload transportation system that minimizes the mass of the transportation system and maximizes the fraction of the launch weight that is the actual payload.

For the RAP study we are concerned with the cost of delivering bulk cargo, i.e. water in multi-ton lots, to EML1. To do this we note that the Falcon Heavy is capable of lifting ~50 tons to LEO for a cost of $125M. If we assume that this 50-ton payload consists of:

1. A payload of water to be delivered to an EML,
2. A carrier to hold the water,
3. A LOX/LH2 stage, and then

We can estimate the cost of bulk cargo delivery to EML1.

We will assume that the payload carrier, which is nothing more than a tank to hold the water, would have a structural fraction of 10% and that a LOX/LH2 stage has a propellant fraction of 90%. The propellant fraction for the stage is well within the current state of the art and the structural fraction for the payload carrier provides mass margin for adding rendezvous, prox/ops, and berthing aids needed for docking with the staging base at EML-1. Then, based on the Delta V required to deliver that payload to, we would need a LOX/LH2 stage with a mass of 3,193 kg carrying 28,733 kg of propellant. This stage would deliver a payload of 18,074 kg to EML-1 of which the payload carrier would constitute 1,807 kg. This gives us a net payload to EML1 of 16,267 kg. Furthermore we estimate that adding that LOX/LH2 stage to the Falcon Heavy would add an additional $75M to the launch cost for the mission.

Given the above assumptions and calculations, we estimate that the cost of delivering bulk cargo from Earth to EML-1 is $12,295. This target number is the "Figure of Merit" that asteroid mining must beat to become an economic success.

6.3 Parametric Cost Model Results

In order to perform a parametric economic analysis of asteroid mining, we studied previous projects and used their costs as a guide to estimate the development, production and operations costs for various elements of the RAP system. We assumed that the cost of developing and operating this system would be charged a very conservative 2.5% annual interest rate on all incurred expenses.
We assumed that:

1. There would be four asteroid mining/retrieval mission spacecraft, and
2. That their production would be phased so that
3. One new spacecraft would become available each year,

. . . until all four ships were available. We allocated four years for each asteroid mining mission with three years budgeted for flight operations and one year for refurbishment/servicing. We then calculated the costs of developing and operating this system of four mining spacecraft over 25 years and estimated the revenue from water sales. This narrative gave us a time history of revenue, expenses and profits that showing that asteroid mining offers the potential to become an economic success.

The cost of the spacecraft was based on previous interplanetary spacecraft since the core spacecraft functionality mirrors them. We assumed that the spacecraft would have a development cost of $500M exclusive of the costs of the mining equipment and the Solar Thermal Propulsion system. Given that a number of interplanetary missions were designed/built/flown for $500M, this amount does not seem to be an unreasonable ballpark estimate for the spacecraft development costs, especially since many of the subsystems can derive from existing systems. We assumed further that the production cost of the spacecraft would be $300 based on its estimated mass. We placed the refurbishment cost of the spacecraft between missions at 10% of the procurement cost, in constant-year dollars at the time of the original procurement (value not reduced by inflation. In order to be conservative we did not assume any ‘learning curve’ benefits for the production cost of the second and subsequent vehicles.

The development cost of the Solar Thermal Propulsion system was placed at $1B. This estimate was reasonable, given the fundamental simplicity of the system and the absence of cryogenic fluids and other complicating design attributes. Recently, SpaceX showed that new propulsion systems could be developed for much lower costs than conventional cost models indicate. We believe that using commercial methodologies to develop this system makes these numbers a reasonable cost goal. We estimated a production cost of $300M for this propulsion system and a refurbishment cost of 10% of the production cost.

The greatest uncertainty in our costs comes from the mining/resource extraction systems. However the initial resource to extract is water and it is a relatively simple resource to acquire. We have identified and, in some cases, even demonstrated several potential systems for this purpose. We will use an estimated development cost of $1B for the mining/extraction systems to deploy on the initial asteroid mining spacecraft while noting that further analysis may force us to a higher number. As with the other systems we estimate a production cost of $300M for this equipment and a refurbishment cost of 10% of the production cost.

Given the above commitments and constraints, we estimate that the development cost for this spacecraft will be $2.5B and each spacecraft will cost $900M. We allocate five years for the development program for this vehicle and assume that it will take three years to manufacture each spacecraft. Further, we estimate that it will cost $375M to launch the spacecraft and that it will be initially assembled in LEO and will then self ferry itself to EM-1 prior to departing on its initial mission. Each launch from EML-1 would cost $50M. TABLE 6.1 shows this preliminary costing exercise.

The operations cost for the first spacecraft would be $40M per year. This expense covers the cost of the people actually operating the spacecraft and its mining equipment along with support for the “back room” personnel who provide domain expert knowledge necessary for on-going operations. However we found that adding a second spacecraft would not double the annual operations budget because most of the time the spacecraft would be in cruise mode, so they would not require the full time attention of the flight control team. Thus, each additional spacecraft added $2.5M to the annual operations budget.

We baselined the complete RAP mining system to return 150 metric tons of salable water to EML-1 along with sufficient water to support the next outbound journey of that spacecraft. We applied the revenue from the sale of that water against the accrued costs of the project, which reduced the cost base used for calculating the annual interest expense for the project. However, to estimate the revenue from water sales and their impact of the project's expense and revenue history, it became necessary to estimate the selling price of the water. We made this "chicken and egg" calculation before we knew what the cost of delivering water to EML-1 would be. To solve this conundrum, we staked the selling price of water from the asteroid mining venture at $8K per kilogram. This price falls comfortably below the cost of delivering water to EML-1 from the Earth and so insures that the asteroid mining venture can capture that market in the face of terrestrial competition. If it had turned out that this number was too low for the venture to be profitable, then we would have been required to iterate the calculations using a larger selling price. {It wasn't.}

TABLE 1. First Approximation Cost Estimate for the RAP Spacecraft & Missions

	2013 $M	1st S/C	2nd S/C	3rd S/C	4th S/C
Non-Recurring Development Costs $M	**2,500**				
Spacecraft	500				
Propulsion	1,000				
Mining Technology	1,000				
Recurring Costs 2013 $M		**1550**	**860**	**695**	**630**
Spacecraft		300	150	125	125
Propulsion		500	250	200	175
Mining		500	250	175	150
Integration		100	90	80	75
Operations		150	120	115	110

Given the above estimates, assumptions, and calculations it becomes possible to get a first order estimate of the cost of doing asteroid mining and returning water to EML-1. The asteroid mining venture starts with a five-year development program that results in buildable designs for the RAP spacecraft, its mining equipment, and the associated mission design pluses and minuses for selected asteroids. The three-year construction phase starts with the first mining spacecraft after which new ships are started on each successive year. Over a period of six years, we would construct four RAP spacecraft and launch them into space. The first spacecraft goes out on its initial mining mission at the start of year nine of the project. The second and following spacecraft launch at the start of each successive year. After all of the spacecraft have been commissioned, we will have four missions underway with three spacecraft either en route, mining or returning, with the fourth spacecraft being serviced in preparation for its next mission. These mission durations may vary dramatically, but we baselined four years as the nominal mission duration.

The first sale of asteroid-derived water occurs at the end of year 11 of the project when the first spacecraft returns from its first mission. Each following year will see the return of one spacecraft to the EML staging base and the subsequent sale of its cargo of water. Over the 25 years that we investigated for this study, there would be four complete missions by two spacecraft and three complete missions by the other two spacecraft. These 14 mining missions will return 2.25 million kilograms of water to EML1. We plan to sell this water for $18 Billion at a profit of $6.2 billion. When we amortize the cost of the venture over the total amount of water returned, our analysis gives us a cost per kilogram of $5,205 by the end of year 25. This price compares quite competitively with the $12,295 per kilogram price for water delivered to EML-1 from the Earth.

FIGURE 6.1 shows a time history for the total expenses, revenue and profit for this asteroid mining venture. The expenses start growing gradually during the first five years of the project when the spacecraft development is occurring. Each year of development incurs an interest expense for the monies spent during the previous years, and this interest adds to the cost of the project. In years seven through twelve, the expenses climb at a higher rate as the project incurs the cost of building and launching multiple spacecraft.

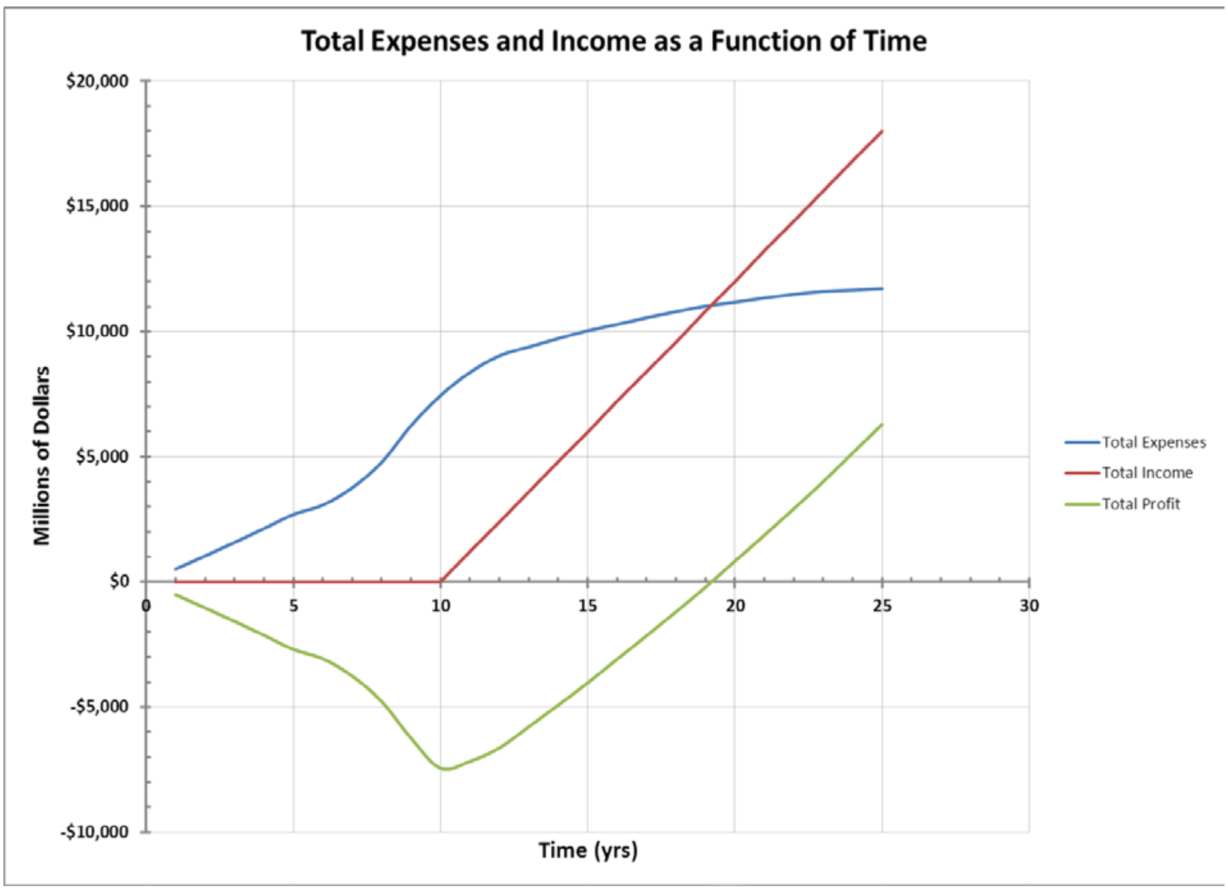


FIGURE 6.1. Total Expenses and Revenue as a Function of Time

However, starting in year 11, revenue starts to flow into the project, as sales from the first completed mission are booked against the accrued expenses. This revenue begins to reduce the cost base used for calculating the interest expense to the project and this helps to reduce the growth rate of the integrated expenses. By year 19 the integrated revenue and income for the project equals or exceeds the integrated expenses for the project – including all interest expenses. This break-even point means the project has now recovered all of the costs associated with its development and operations. From this time on, the project operates with a net profit.

It should be obvious that as the project progresses and the initial development and production costs become amortized, then the price that for returned water can go down. We will call this amount the “full cost recovery” price. FIGURE 6.2 illustrates this effect. When the first mining missions return their payload to EML1, the full cost recovery price for their water would need to be very large if the project was to break even at that time. As more and more missions return with their economic harvest the price required to break even declines. To achieve full cost recovery at the time of the first mission would require that the water sell for more than $55K per kilogram.

Thus, when the model sets a fixed price for selling the water returned to EML-1, this calculation means that for the first few missions we will be sell our product at a loss. We’ll eventually make it up on volume. Over time, the revenue from those sales helps to retire the company’s debt and this price allows the company to compete against terrestrial sources for water. But by year 15, the full cost recovery price for the water would be just slightly higher than the cost of terrestrial water and in year 16 the price of the water, based on actual costs, would drop to less than the cost of water from Earth. The full cost recovery price for water equals the selling price in year 19 – or sooner -- and that ends the practice of selling our product at a loss.

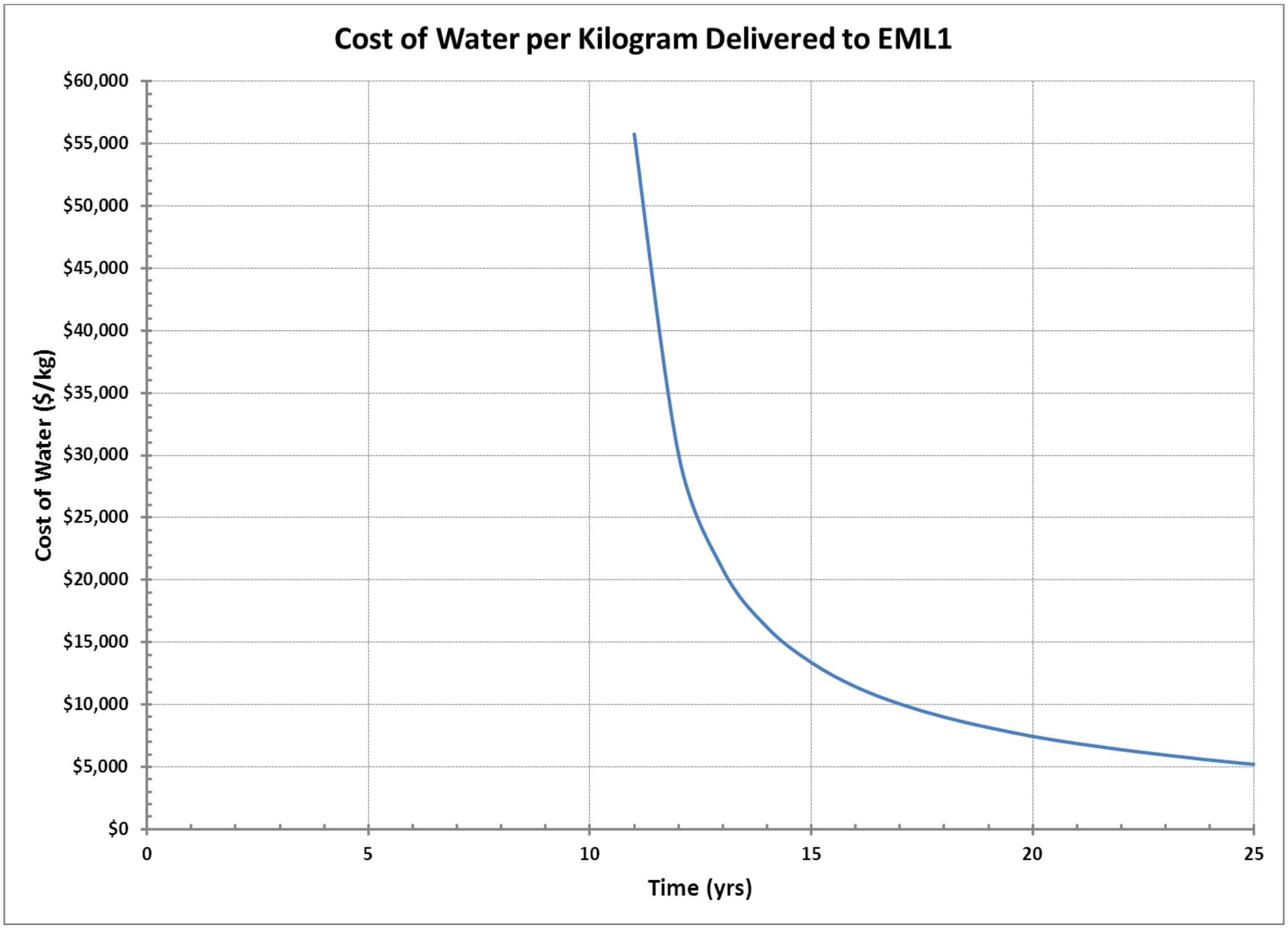


FIGURE 6.2. Full Cost Recovery Price of Water per Kilogram Delivered to EML1

It is useful to consider how this project will incur expenses, revenue, and income on an annual basis, as is shown in FIGURE 6.3. Here we see that the project goes "cash flow positive" on an annual basis in year eleven, i.e. in the first year that water gets returned from an asteroid. This means that after year eleven there will not be a need to get outside financing for this venture and that the on-going operations, as well as the paying down of the previously incurred expenses plus interest, can be financed from the annual operating revenue derived from the sale of asteroid water at EML1.

It is also instructive to note how the annual expenses decline as the venture progresses. The annual expenses peak in year nine when the project is paying for the construction of several new mining ships as well as sending out its first asteroid mission. But after that time the annual expenses decline since the project is moving out of the construction phase and into the operational mining phase. Annual expenses continue to decline sharply through year thirteen when all of the ships have been build and launched into space. Annual expenses continue to decline as the company's debt gets retired and the annual interest expenses decline. The annual expenses continue to decline after the company has become profitable because now the company is able to earn interest on the cash on hand that it has as a result of the profits it has made. This projection models the normal cash asset management practices of all big businesses and represents a potentially significant revenue stream from which an asteroid mining venture can benefit.

Once the company becomes profitable and has retired the debt on its initial expenses, it will have many options for pricing the returned water production. At that time they will be able to price the returned water not on the total money spent - since that money will have already been repaid - but rather they will be able to price the water based on the operational costs for flying an asteroid mining mission. The concomitant development of the solar thermal propulsion engine will help to create a fleet of STP-powered spacecraft exploring the solar system, all of which will be in the market to buy water in space.

After year-19 the company will have repaid all of its initial R&D investments. The mining ships will be in space and fully paid off and amortized. The cost of an asteroid mining mission at that time would be $260M; that mission would return $1.2B worth of water that would have a full cost recovery price of $1,733 per kilogram. The company could make a profit by selling the water at any price above that level.

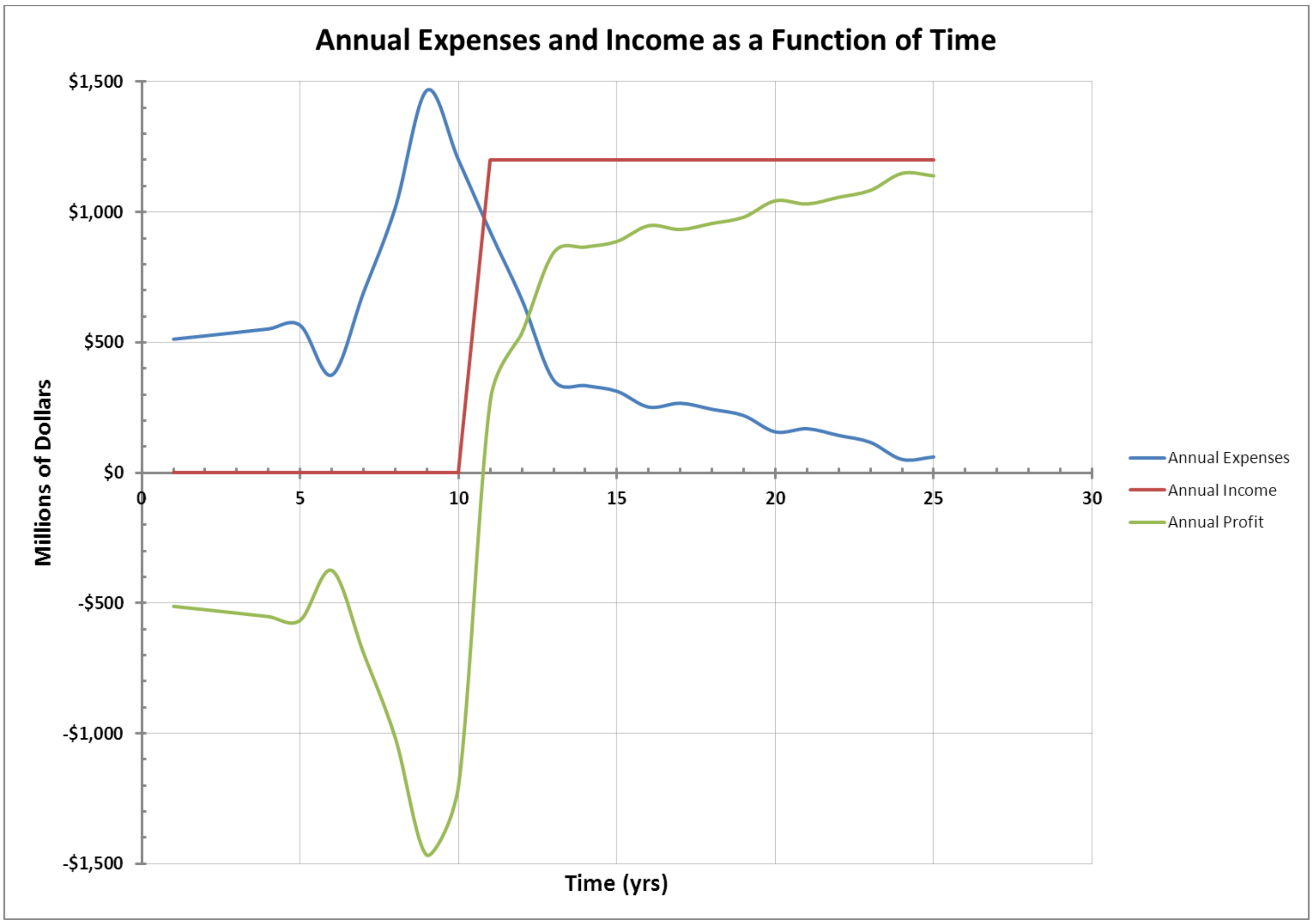


FIGURE 6.3. Annual Expenses and Revenue as a Function of Time

6.4 Parametric Cost Model Summary

We developed a parametric cost model for the RAP system that allows us to examine the first order economic feasibility of asteroid mining. The results provide a favorable answer that that asteroid mining will be economically viable. We estimated that the initial cost for water delivered to EML-1 from an asteroid is on the order of $5,205/kg and that compares quite favorably with the $12,295/kg cost of delivering water from the Earth to EML1.

There appears to be a "first mover" benefit for the business of asteroid mining. Any company that succeeds in developing an asteroid mining enterprise – and can return its product in a timely manner -- will be in a position where they can set prices of not just water, but all other asteroid materials.

We acknowledge that these cost estimates are preliminary and that a more detailed cost model is needed. But the fundamental conclusions drawn from this analysis seem to be quite resilient. If the spacecraft development and production costs are doubled the cost of asteroid water remains below the cost of Terrestrially sourced water. If the amount of water returned per mission is doubled, something the RAP spacecraft design would be capable of doing, and then the cost of asteroid water would be halved.

7 Conclusion

The NIAC Phase 1 project enabled the RAP team to arrive at these fairly dispositive findings. Some of these findings contradicted or negated our going-in assumptions about the related issues or topics.

7.1 Findings on Mission Parameters

The findings from the RAP study arise from a complex of analyses and design exercises. These topics include economics, mining material, propulsion systems, ISRU, asteroid prospecting and mining mission classes, trajectories, spacecraft design, and mining technologies

7.1.1 Economic Feasibility

There is no economically viable scenario we could identify that depends solely upon returning asteroid resources to LEO or the surface of the Earth. To be economically feasible, asteroid mining will depend predominantly upon customers in-space who are part of the space industrial economy and infrastructure. It is possible that secondary markets on Earth may provide “icing on the cake” for platinum-group metals.

7.1.2 Rare Earth Elements are not an Economic Option

Although the initial RAP proposal included REEs as a potential option, there is not economically advantageous basis for returning REEs from Space to the Earth. REEs on Earth are not rare at all; rather the economics of producing them have favored cheap labor and high tolerance of environmental degradation, which has given China a near monopoly. New, environmentally safe processing technologies can make REEs much more widely available, although perhaps not less expensive.

7.1.3 Solar-Electric Propulsion is not Economically Viable for Sustained Asteroid Mining

Solar-Electric Propulsion is an available, reliable, and relatively low-cost propulsion system for deep space. However, if an asteroid mining venture is based on conventional, commercial bank financing, the time-cost of money – the interest to be paid over the years the spacecraft is in transit – kills the whole project. Therefore, a different and higher-performing technology is required.

7.1.4 Structural Metals of Interest

Initially, we proposed to focus on the platinum group metals. However, from the perspective of the in-space infrastructure industry and economy, the production of structural materials will emerge as far more important as an early market. Therefore, aluminum, iron, nickel, manganese, and other familiar elements may be far more useful and in demand among deep-space customers. This finding does not rule out the possibility that there may also be a deep-space use for platinum-group metals or that there may be a secondary market on Earth for platinum-group products that can “piggy-back” on an Earth-return lander.

7.1.5 Four Classes of Asteroid Mission

We identified four classes of asteroid prospecting and mining mission: Prospecting, Mining/Retrieval, Processing, and Transportation.

a. Prospecting – This mission launches either from LEO or an EML and does not depend on ISRU to create the propellant for the return to Cis-Lunar space. It could return 1-2 tons of asteroid material to an EML staging base.

b. Mining/Retrieval – This mission launches from an EML staging base and returns there using ISRU derived propellant from the asteroid. It delivers mining/processing equipment to the asteroid, stays with that equipment till the mining is finished and then returns the mining equipment plus 150 metric tones, or more, of asteroid material to the EML staging base.

c. Processing. This mission launches from an EML staging base and delivers mining/processing equipment to an asteroid. This equipment stays at the asteroid generating processed materials that will be returned to Cis-Lunar space by a separate spacecraft.

d. Transportation. This mission launches from an EML staging base and goes to an asteroid where a Processing Mission has been created refined material for return to the Earth. This mission goes the asteroid, picks up those materials and then returns them to the Earth.

7.1.6 Stage Operations from an Earth Moon Lagrange Point

Operating from an Earth Moon Lagrange Point significantly reduces the Delta V needed to fly round-trip missions to Near Earth Asteroids – or other deep space destinations. This reduction in Delta V budget reduces the size of the spacecraft and minimizes the mass that has to be lifted into space and the associated launch cost. Additionally, operations from an EML point offer several benefits when compared to operations from LEO when it comes to dealing with the specific trajectory geometries required for meeting launch window constraints. However, to take advantage of these benefits requires that propellant for missions departing from a Lagrange point comes from a non-Terrestrial source, such as the Moon or the asteroids.

7.1.7 Use Gravity Assists Whenever Possible

Significant reductions in the total Delta V needed to fly asteroid mining missions can be provided by using Gravity Assist maneuvers whenever possible. Using an Earth Swing-by maneuver, a form of Gravity Assist, can reduce the Earth Departure Delta V by several km/s depending upon the C3 required for the mission. Gravity assists from Mars or Venus have the potential to reduce the Delta V required for returning to Cis-Lunar space and can increase the time period during which an Earth return is possible.

7.2 Findings on Spacecraft Design

As a result of this study we were able to draw a number of conclusions relating to the design of the spacecraft and its supporting technologies. These conclusions are presented below.

7.2.1 Asteroid ISRU is Essential for the Success of Asteroid Mining

The use of asteroid ISRU is a powerful technique for reducing the initial mass of the spacecraft and can be considered to be an enabling technology for making asteroid mining economically viable. This insight has many ramifications for the design of the system extending from propulsion and propellant choice to the selection of candidate asteroids for specific mining missions. Moreover, the ability to take on an arbitrarily large amount of propellant for the return to Cis-Lunar space means that arbitrarily large amounts of asteroid resources can be returned without significantly altering the initial mass of the spacecraft. The need to produce propellant at the asteroid suggests that the best initial asteroid targets would be C or S class asteroids that might contain both water and valuable metals.

7.2.2 Water is a Superior Propellant Choice for Asteroid Mining Missions

If the candidate asteroids are ones that can provide water – and this is likely since water is the best candidate product for early asteroid mining missions – then this water is a prime candidate for use as the propellant for a Solar Thermal Propulsion system. The advantages for water as a propellant include its high density, non-cryogenic storage temperatures and its ability to be used directly as a propellant without requiring that it be broken into its constituent molecules or being liquefied to cryogenic temperatures.

7.2.3 Liquid Hydrogen is a Poor Propellant Choice for Asteroid Mining Missions

Although a Solar Thermal Propulsion system using Liquid Hydrogen as a reaction mass offers a temptingly high Isp this propellant would be a poor choice for asteroid mining missions. This is because any ISRU Hydrogen obtained from an asteroid would be extracted from water. Breaking down that water to extract the hydrogen would result in the creation of 8 kilograms of waste oxygen for each kilogram of hydrogen propellant that was extracted. It is noted that the superior Isp of Hydrogen, when used in a Solar Thermal Propulsion system, would reduce the propellant mass requirements by roughly 75% when compared to a water based Solar Thermal Propulsion system. However, when both of these factors are considered it can be seen that using hydrogen as a propellant will more than double the mass of water that has to be extracted from the asteroid to generate the propellant needed for the return trip. Using the water directly as a propellant would require extracting less water from the asteroid or, alternatively, would allow more of the extracted water to be returned to Cis-Lunar space and sold for a profit. The case against hydrogen gets even stronger when one factors in the cost and complexity of first extracting the hydrogen by electrolysis, liquefying the hydrogen and cooling it to cryogenic temperatures, dealing with the challenges of keeping the hydrogen cold for extended periods and dealing with the design impacts caused by the low density of liquid hydrogen.

7.2.4 Solar Thermal Propulsion is Uniquely Suited to Asteroid Mining Missions

Solar Thermal Propulsion is uniquely suited for asteroid mining missions because of the ease with which it can directly use available asteroid ISRU resources for the propellant for returning to Cis-Lunar space. This greatly simplifies the system design. Additionally, this system can be used to generate large amounts of process heat to support mining operations or to produce large amounts of electrical power when operating as a solar dynamic power system,

7.2.5 Building the Spacecraft Using a Central Truss Enhances Mission Flexibility

Given the large number of possible asteroid destinations and their diverse Delta V budgets it is important that the spacecraft be able to accommodate that diversity without having to be redesigned for each mission. One approach to doing this is to design the spacecraft around a central load-bearing truss so that propellant tanks can be added as necessary to accommodate changing Delta V budgets. This approach also makes it easy to devise a configuration allowing solar energy to be transmitted from the Solar Thermal Propulsion system in the rear of the spacecraft to asteroid mining/extraction equipment located at the front of the vehicle.

7.2.6 Spacecraft Subsystems Should Perform Multiple Functions

Reducing the mass of the spacecraft plays an important part in reducing the cost of the asteroid mining venture. One powerful way of accomplishing this goal is to find ways for the spacecraft's subsystems to support multiple functions. For example, the Solar Thermal Propulsion system can provide the obvious function of propulsion but it can also provide process heat for mining/resource extraction and could serve as the basis for a Solar Dynamic Electrical Power system. Finding ways to get other subsystems to serve such double duty will help reduce the cost and mass of the spacecraft.

7.2.7 The Mining Spacecraft Must be Reusable

By its nature an asteroid mining mission requires that the spacecraft return to the staging base from where it departed. It would be wasteful and uneconomic if the spacecraft was subsequently discarded without flying another mission. Although there will be additional design/manufacturing costs associated with a reusable mining spacecraft as well as a non-trivial servicing cost it is clear to this team that those costs are greatly outweighed by the cost savings associated with a reusable spacecraft.

7.2.8 Design the Mining Spacecraft for Ease of In-Space Servicing

Given that the mining spacecraft will have to be reusable it is then important that it be designed to simplify the task of servicing it between missions. This is why the RAP spacecraft was designed with all of its subsystem boxes located on the exterior of the truss and built as ORUs with simplified mechanical/power/data/thermal connections to the spacecraft bus. Moreover, designing those ORUs so either astronauts can service them or TeleBots creates the ability to service the spacecraft remotely and thus save the cost of transporting astronauts to the EML staging base. By preserving the option for servicing by astronauts this means that if the servicing tasks exceed the capabilities of the TeleBots then human astronauts can be dispatched to the EML staging base to accomplish the more challenging servicing tasks.

7.3 Mining Technology

Technology for asteroid mining and processing is at very low TRL. Extracting of asteroid material is easier to do since it is more or less a mechanical/physical process and could be accomplished via various means such as magnetic, pneumatic, mechanical (scoop, rake, drill) etc. These approaches could be modeled and tested in zero-g flights and vacuum chambers.

7.3.1 Water Extraction

Processing of regolith to extract free-water is at TRL 3-4 and has been demonstrated in vacuum. Since water extraction is driven by water vapor pressure and not gravity, vacuum-chamber tests at 1g could be applicable to asteroid environment as well.

7.3.2 Metal Extraction

Metal extraction or de-alloying technologies are at TRL 1. Metallurgy (the science of extracting and purifying valuable resource) does not have parametric equation that would allow determining time and cost for developing new extraction technology. Hence, it is extremely difficult or even impossible to determine how much and how long and ‘if’ a technology could in fact be developed for extracting and purifying metals on asteroid. Since potential technologies would have to be tested in zero-g, the cost and schedule for such a technology development would be substantially higher.

7.3.3 3D-Printed Structures from Regolith Fines

Collecting regolith fines and printing 3D structures in zero-g would be feasible since 3D printing relies on physical (melting/sintering) rather than chemical process. Such technology has been in development for many years and zero-g prototypes could be developed and tested in zero-g planes and outside of ISS (i.e. vacuum and zero-g) at relatively low cost.

7.3.4 Do Not Return the Slag

There is no model of mining economics that shows a profitable economic return on returning all the slag from an asteroid to the Earth-Moon system. This assertion applies to the popular notion of retrieving a whole asteroid to wherever. Certainly, it may be technically doable and there may be some scientific value or astronaut training benefit in retrieving an asteroid, but unless someone finds one made of solid "unobtainium;" it is unlikely to ever be profitable.

Precious metal ores occur naturally at very low concentrations; that low concentration is a large part of what makes them rare and valuable. In terrestrial mining, the whole point of beneficiation, concentration, and the other processes on-site at the mine are to separate the ore from the slag, so that the expense of transport applies only to the useful ore. The same principle applies to Space Mining. Therefore, we do not see an economic model for returning a whole asteroid to the Earth-Moon System, with the possible exception of capturing one for use as a testbed for mining technologies. This finding carries profound implications for the “Asteroid Return Missions” (ARM) that have been enjoying a popular vogue in the “NewSpace” arena and media (but not in Congress) as we go to press. This finding means that there is no realistic economic return in an ARM, and probably not much of a scientific exploration return.

7.4 Business Case

7.4.1 Asteroid Mining Makes Economic Sense

A first order calculation of the cost of returning water from a Near Earth Asteroid to a staging base at EML1 yields a cost of $5,205 per kilogram, which compares quite favorably to the $12,295 cost of delivering water there from the Earth using a Falcon Heavy. Once all of the initial costs of establishing the asteroid mining enterprise have been retired and the cost of the returned water can be based solely on the operations cost of asteroid mining then that cost could fall to $1,733 per kilogram. Several techniques exist that could reduce these costs by a factor of two or more.

7.4.2 Competition in Asteroid Mining

The first company to engage in asteroid mining will be competing against the fundamentally high cost of materials delivered to high orbit from the Earth. The second company to engage in this business will be competing against an asteroid mining company whose costs for delivering products to cis-lunar space will be lower than the Earth-launched default. However, the key discriminator for successful competition will be how long it takes an asteroid mining company to deliver their product from the production site to market, which translates into how long they are paying interest on their initial debt-load. The choice of propulsion system will play a dispositive role in determining this competitive edge. Any space mining company will need to invest heavily in the mining technology. The big difference comes in propulsion technology. RAP’s choice would be to pay this interest on the capital cost to develop the solar thermal propulsion system, instead of paying collective decades of interest waiting for solar electric spacecraft to return to the Earth-Moon system.

7.5 Next Steps

The Phase 1 results influence the RAP direction for Phase 2 in several key respects. We will sharpen our focus on the mining technology, mission design, and spacecraft design and integration. At the same time, we will substitute our subscription to the Arkyd Public Science telescope to pursue identifying promising targets for prospecting. With respect to the notion of the "deep space economy," we find that we cannot pursue it in a meaningful way given the knowledge, expertise, resources, and time that will be available during Phase 2. Instead, we will drill down on generating a more accurate cost model for the RAP spacecraft and its mission operations. As part of this modeling, we plan to obtain a NASA Air Force Cost Model (NAFCOM) license to generate standard, comparable costing data.

11. Bibliography

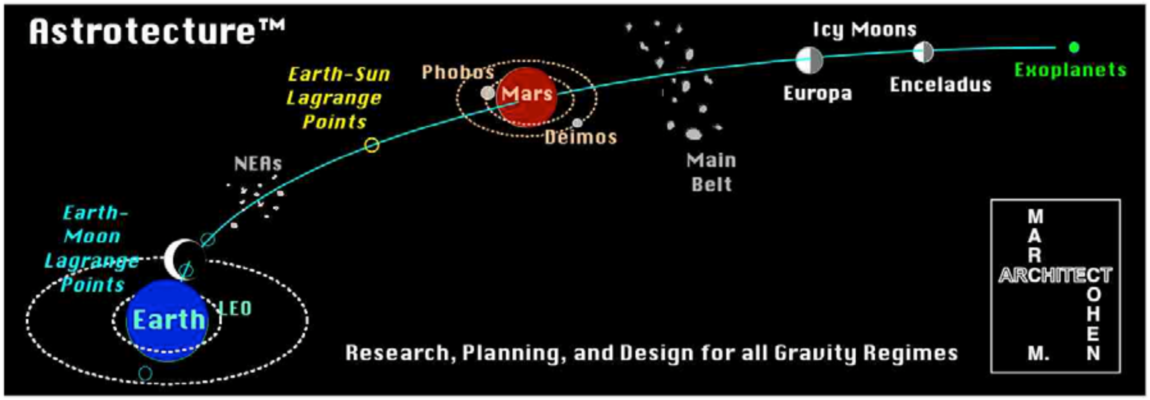
ANSI/AIAA (2012). Guide to the Preparation of Operational Concept Documents, *American National Standard*, ANSI/AIAA G-043A-2012. Reston VA: American Institute of Aeronautics and Astronautics.

Wikipedia (2013) Apollo 17, http://en.wikipedia.org/wiki/Apollo_17, retrieved 15 June 2013;

Beauford, R.E. (2011). "Rare Earth Elements: A key to understanding geological sorting processes in the solar system", 27 April 2011, retrieved 1-10-13, http://rareearthelements.us/ree_in_space.

Binczewski, George J. (1995). The Point of a Monument: A History of the Aluminum Cap of the Washington Monument, *Journal of Metals* (*JOM*), 47 (11), pp. 20-25. http://www.tms.org/pubs/journals/jom/9511/binczewski-9511.html, retrieved 5 JAN 2013.

Brophy, et al.: Asteroid Retrieval Feasibility Study. Keck Institute for Space Studies, California Institute of Technology, Jet Propulsion Laboratory (2012),



Robotic Asteroid Prospector (RAP)
Phase 1 Final Report to NIAC
Appendix A

9 July 2013
Revised 10 March 2015

Astrotecture®
Marc M. Cohen, Arch.D

Marc M. Cohen Architect P. C.
Astrotecture®
+1 650 218-8119
marc@astrotecture.com
http://www.astrotecture.com

Table of Contents

1 Introduction

The purpose of this Technical Appendix is to identify and discuss the issues and problems that arose during the Robotic Asteroid Prospector (RAP) team's work on our Phase 1 NIAC contract. This discussion will lead hopefully to a disposition of how to address and handle such issues in the future – in Phase 2 – and in developing privately funded contributions to the RAP effort. The central issue that arose concerns the technology maturity of the solar thermal propulsion system that is central to the concept.

The NIAC-funded RAP project did not include the design of a Solar-Thermal Propulsion (STP) engine. For the purpose of RAP, an STP engine would ideally be a commodity item that our team could purchase off the shelf like any other mass-produced engine (e.g. RL-10). Our approach in Phase I was not to design an STP engine, but to design the spacecraft to accommodate a variety of potential STP engine designs. To identify these designs, we performed a literature search and commissioned our patent counsel to perform a patent search that included STP engine designs. These searches turned up a several designs for solar thermal engines in a variety of configurations. In this Technical Appendix, we review those patents and discuss their implications for the design and engineering of the RAP spacecraft. We also show and discuss a selection of the conceptual sketches that we drew to try to understand the issues of installing an STP engine in the RAP spacecraft, and what might be the implications for the design of that engine for an optimal installation.

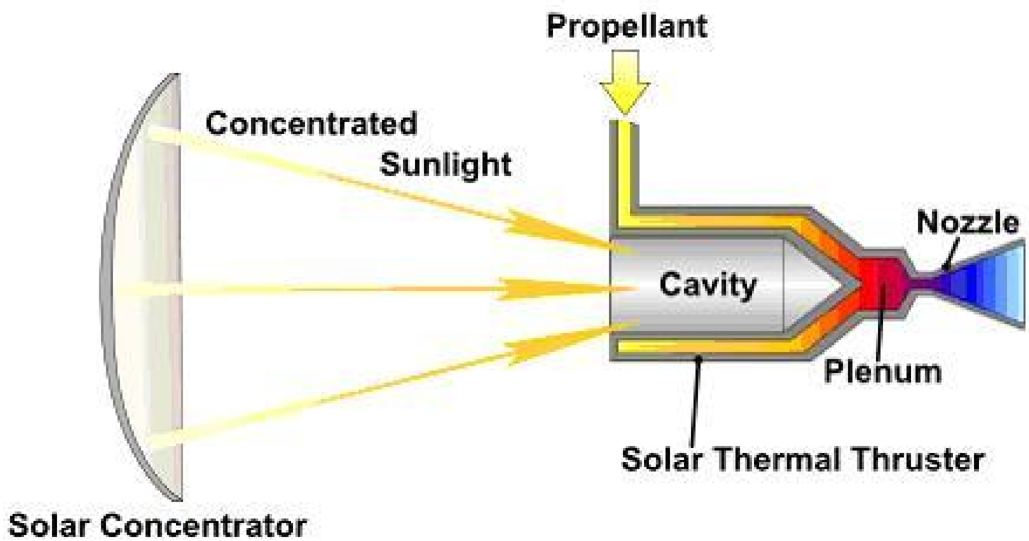


FIGURE 1.1. Schematic of a Solar Thermal Engine,
Courtesy of the USAF, Solar Orbit Transfer Vehicle (SOTV) Program.

In addition to this search, through the good offices of the NIAC executives, we have been able to identity and begin communication with the people at MSFC who worked on the *Shooting Star* Solar-Thermal Propulsion (STP) Engine program, circa 1998-2002. This contact has been very helpful to our team. What was perhaps most useful was to learn the names of everyone and the keywords they used in their publications. These clues allowed the PI to go to the librarians at Ames to search for reports and other documents on the restricted library server, from which they could print hard copies. We have collected everything we can find on Solar-Thermal engines and are now in the process of figuring out where we go from here. The key features of a solar thermal engine appear in FIGURE 1.1. Beyond these contacts with participants in the Shooting Star program, we have been able to identify areas of expertise and specialization where we need to learn more, and to find new experts or advisors in areas that Shooting Star did not cover, or did not cover sufficiently.

1.1 What is Novel in the RAP Concept?

The RAP spacecraft is specifically conceived and designed as a true spaceship that would operate only in the vacuum of space, and would be capable of finding its propellant in space as well. In this way, once launched the RAP could function largely independent of Earth resources. Because it uses concentrated sunlight for propulsive energy, electrical power, and mining process heat, it should be possible to achieve valuable economies of agglomeration and scale in its design and operation. The RAP team finds itself in a somewhat awkward position insofar as we proposed the STP engine as a baseline but find that none have been completed or operated to date. The RAP concept offers many innovations. These generally non-obvious points of novelty include:

- The triplex solar energy implementation
- The grid of light to distribute power
- Multiple orientations for installation of an STP engine or engines
- The diverter mirrors serve as first stage heat exchangers, which also cools them,
- The alpha-joint passes concentrated sunlight through it.
- The alpha-joint allows the spacecraft to keep the solar concentrators always pointed toward the sun.
- The spacecraft attaches to the pole of a rotating asteroid to commence prospecting and mining ops.
- Obliquity of the solar concentrators.

1.2 The STP Design Point of Departure for RAP

What is most important for RAP, from all the design data identified and reviewed in this appendix, is that although these designs and inventions accomplish a few of the necessary tasks, none of them do all the tasks that RAP requires. Several of the design precedents provide a single, credible optical path to direct the concentrated sunlight flux into the STP engine, but none of them allow for alternative vectors to accomplish all the other functions that the RAP team identified as essential to our objectives. These objectives include:

- Use of H_2O as the STP propellant,
- Two stage phase-change heating of the propellant,
- Deflecting concentrated sunlight flux to the mining module for process heat, and
- Generating megawatt-scale electrical power through the use of Sterling Cycle or Rankine-Brayton Cycle engines.

What this literature review and patent search accomplishes for the RAP project is to help define where we will need to go from here with respect to the RAP spacecraft design at the conclusion of our Phase I.

1.3 Literature Review for STP Engines and Spacecraft

The scholarly, scientific, and technical literature on STP engines and the spacecraft they power is rather limited. Many publications repeat Stewart and Martin's (1996, p. 752) misconception of a spacecraft powered by superheating a very light gas (e.g. hydrogen H_2, ammonia, NH_3) without oxidation or combustion: "Because the thrust level generated is relatively low, the solar thermal concept can only be used as an upper stage to provide orbital transfers."

Beyond such stated misconceptions, the leading challenge to understanding the design, engineering, and prospective operations of these kinds of concepts is to find sufficiently complete documentation. In some cases, the government researchers and their contractor associates publish certain aspects or parts of their work, but often what they publish is incomplete and misleading in the sense that it is very difficult to ascertain the degree to which the authors or the sponsors succeeded in meeting their objectives. In other cases, the lack of technical detail in the form of drawings or specifications can leave the reader struggling to grasp these key details of the concept.

The NASA Shooting Star Project (circa 1998-2002) accomplished important advances that unfortunately they were not able to integrate or put into practice due to funding limitations and curtailment. The key features of the Shooting Star were:

- A parabolic solar concentrator
- A focusing "solar window" in the form of a sapphire lens
- An internal heating chamber/heat exchanger where the propellant can be heated by direct gain, and
- A nozzle to direct the thrust from the expanding superheated propellant.

1.3.1 Parabolic Solar Concentrator

FIGURE 1.2. The Glenn Research Center Pathfinder 3 Inflatable Solar Concentrator

Slade, Tinker, Lassiter, and Engberg (2001) published the test results for an inflatable structure to deploy and support the parabolic concentrator for an STP engine, modeled closely on the US Air Force's Solar Thermal Upper Stage. FIGURE 1.2 shows this experimental model called the Pathfinder 3 Inflatable Test Article. It consisted of inflatable struts and two circular mounting rings, one of which was a "torus/lens assembly." The degree to which this "lens" may have contributed stiffness to the model is not clear, nor is the extent to which this test article represented a primary mirror. This work occurred concurrently with the development of the Shooting Star, at NASA MSFC where the Shooting Star work was underway, but the article does not mention a direct connection. The article goes into minute detail on how the researchers set up their test regime and what they measured. Their main conclusion is that an inflatable structure's dynamic range varies substantially between an ambient atmosphere and vacuum, and they conclude that they need to perform much more vacuum testing. FIGURE 1.2 shows an inflatable solar concentrator, but it is not clear if it is the same as the ones tested at NASA MSFC or similar to them.

Leigh and Tinker (2003) present a computational model of "an Inflatable Concentrator for Solar Thermal Propulsion." They review the earlier physical testing of the inflatable test article, and then describe a computational dynamic model using the MSC/NASTRAN finite. Much of the discussion concerns how to configure the flange thickness and stiffness and the film properties, and then how to model them to obtain results that reflect the test article results.

In a companion piece to Leigh and Tinker (2003) in the same issue of the Journal of Spacecraft and Rockets, Smalley, Tinker, and Taylor describe the modeling of an "inflatable 5m antenna/concentrator with tapered composite struts and support stand." This concentrator differs from the one discussed in the two preceding articles in two key respects: it clearly includes an inflatable parabolic dish concentrator and it is not all inflatable. The struts are rigid composite material. The purposes of the article is to "discuss the methodology for dynamically characterizing a large 5m thin-film inflatable reflector and to discuss the test arrangements and results." It is heartening to see that this team addressed the design of an inflatable reflector, even thought the study of the Solar Thermal Upper Stage (STUS)/Integrated System Upper Stage (ISUS) -like inflatable structure did not mention the concentrator or reflector.

FIGURE 1.3 shows the archetypal ISUS solar thermal booster stage. This solar orbital transfer vehicle (SOTV) would ferry satellites from LEO to GEO. As the patent search showed, most STP applications to this role fly solar-inertial, with the fixed solar concentrators always pointing toward the sun. However, in the ISUS/STUS/SOTV sequence of design concepts from the Air Force Research Laboratory (AFRL), there is at least the option for these oblong parabolic concentrators to be mounted on gimbals capable of pointing them to track the sun.

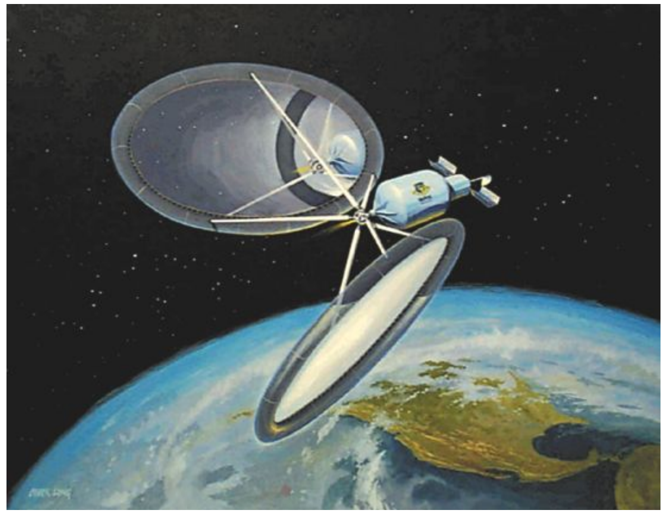


FIGURE 1.3. Archetypal STUS/ISUS type booster spacecraft, with inflatable primary parabolic solar concentrators.

How these solar concentrators would be supported by the three rigid struts, how the struts would connect to the spacecraft frame, and how the system would control and point them, remain one of the incomplete areas of this search. Several of the authors who worked on these projects (ISUS/STUS/SOTV) or related efforts, conclude that the inflatable concentrator technology still has a long way to go to maturity, but it is not clear if they make this assessment solely in the context of the three narrow rods that must connect somehow, somewhere to the inflatable thin film structure. Inflatable structures with other applications and less idiosyncratic mechanical or structural connections do not appear to result in such negative assessments. Instead, this lack of clarity about what is really the problem raises the question of whether the expectation of connecting the inflatable to a predetermined rigid structure is the problem and not the “maturity” of the inflatable technology.

Fortunately, in the case of the AFRL’s alphabet soup of ISUS, STUS, and SOTV, there was a parallel (and somewhat obscurely coordinated) effort in the European Space Agency (ESA) that released somewhat more complete documentation on the archetypical concept. The ESA/ESTEC version is called the STOTS, the Solar Thermal Orbital Transfer Study. The revelation in the STOTS literature is the way that the solar concentrators focus the concentrated sunlight into the solar window on two sides of the STP main thruster (main engine). The STOTS drawings show the gimbal arrangement around the STP engine and its “solar windows,” although it remains uncertain what type of secondary concentrator they may plan to use in those solar windows (see the discussion below for further elaboration of these terms).

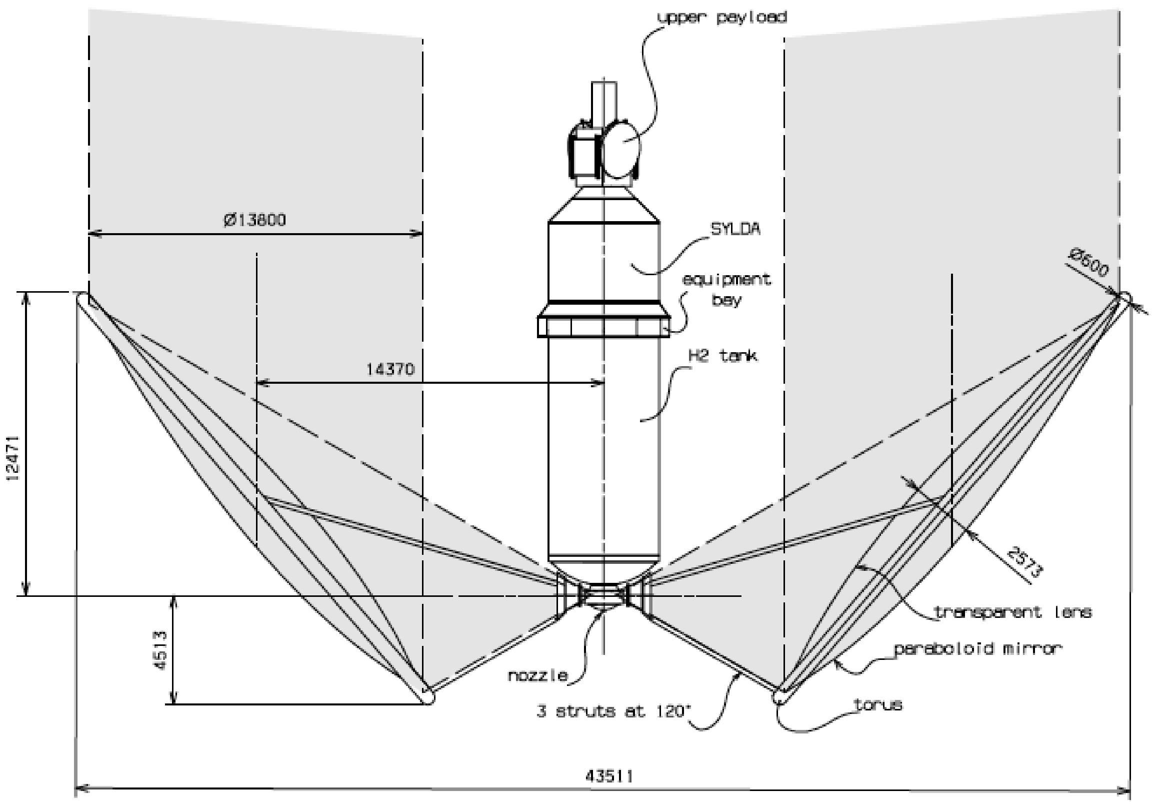


FIGURE 1.4. European Space Agency version of an ISUS called the STOTS. Courtesy of EADS Launch Vehicles.

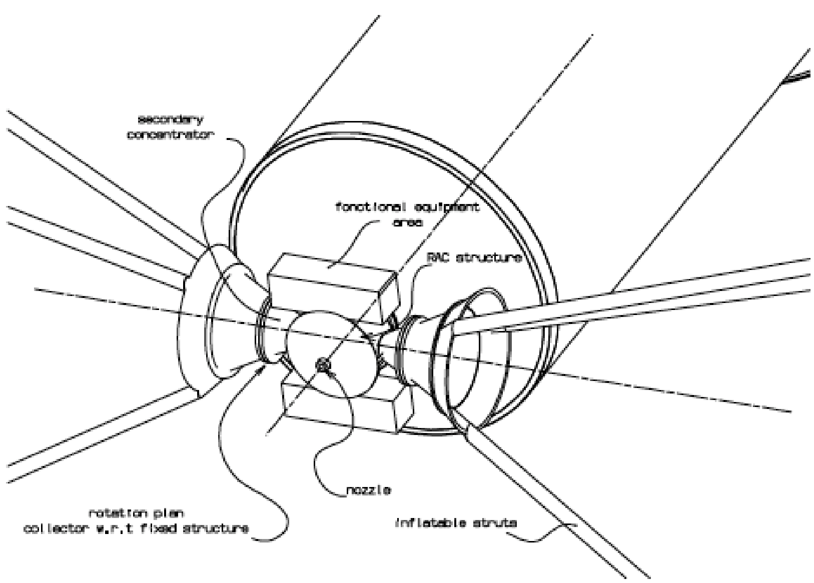


FIGURE 1.5. Detail drawing from the ESTEC STOTS project showing the "rotation plan[e]" *sic* on the lower left.

FIGURE 1.4 shows the geometry and dimensions of the STOTS spacecraft study, with an indication of where the solar window or secondary concentrator would receive the sunlight between the convergences of the rigid struts. This gimbal arrangement would serve as a kind of limited "alpha joint," allowing the primary solar concentrators to point toward the sun and track it to some limited extent.

FIGURE 1.5 shows how it would work. This detail drawing from the ESTEC STOTS project shows that indeed the solar concentrator support structures are intended to rotate about the axis of the solar window, with the "rotation plan[e]" labeled.

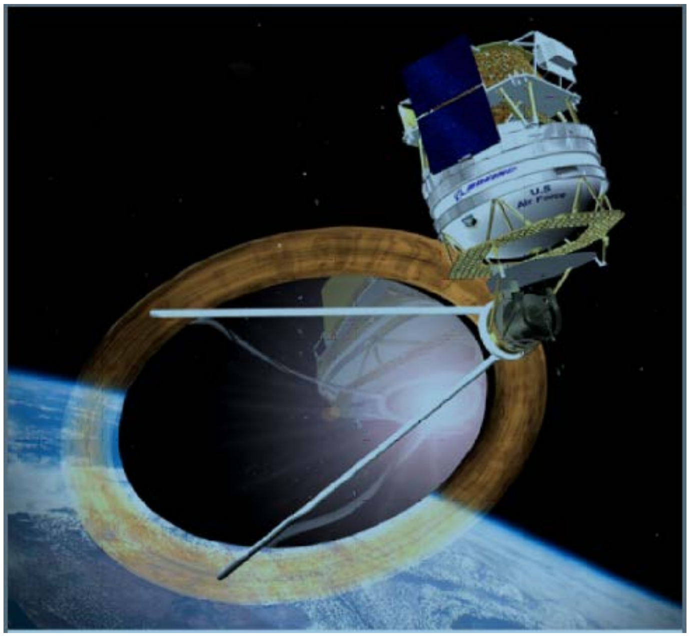


FIGURE 1.6. Nakamura, Kerch, et al. AFRL SBIR Study for an "Off-Axis Inflated Concentrator Solar Thermal Propulsion System."

In Nakamura, Krech, et al, a later AFRL SBIR study published in 2005, the designers/researchers replaced the direct reflection/concentration method of transmitting the concentrated sunlight flux with fiber optic cables, which they called "wave guides." These "flexible optical waveguide transmission lines" would enter the inner cavity-heating chamber of the STP engine to heat the propellant as it passes through the heat exchanger. Nakamura, Krech, et al referred to the engine – or the placeholder for the engine – as the solar receiver housing to which they connected four wave guide cables. For propellant, they ran their tests with helium and argon at ambient Earth atmospheric pressure and temperature. Even so, their tests achieved an Ar gas temperature of 1357K and a graphite receiver wall temperature of 1502K. FIGURE 1.7 shows a diagram of how the optical wave-guide cables would distribute the concentrated solar flux to the points of use in the main engine thruster and in the attitude control thrusters. Note the "optical switches" that can divert the sunlight flux between the main engine thruster and the attitude control thrusters.

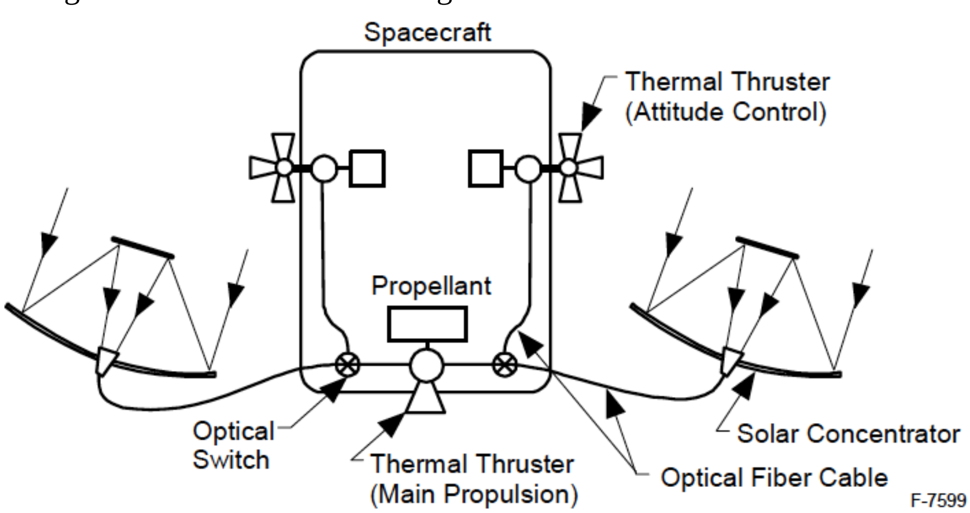


FIGURE 1.7. Nakamura, Krech, et al., Diagram of an STP spacecraft that uses fiber optic cable "wave guides" to transmit the concentrated sunlight flux.

1.3.2 Secondary Concentrator Lens

The solar window or lens or concentrator to direct the concentrated sunlight into the heating chamber or surface of the STP engine is a small but essential component. Robert E. Lee filed the first patent in 1977 (reviewed below) that described such a secondary concentrator for an STP engine. Since that patent for a Solar Space Vehicle (1982), there have been several efforts to refine this component.

Wong and Macosko (1999) published what may be the most comprehensive treatment in Refractive Secondary Concentrators for Solar Thermal Applications, for the Shooting Star STP engine program. Wong and Macosko were the first authors to suggest the "triple threat" of using solar thermal technology for power generation and for a solar furnace in addition to propulsion.

What they mainly address is the design and testing of the sapphire secondary concentrating lens in a facility at Glenn Research Center. Wong and Macosko make these key points:

> *Although researchers have been proposing lightweight, thin film and inflatable primary concentrators, "no primary concentrator technology is capable of achieving the system requirements alone, thus necessitating the addition of a secondary concentrator to the energy delivery system.*

This secondary concentrator consists of a sapphire lens in the "solar window" with an extended refractive flux extractor section that extends well into the inner heating chamber of the STP engine. The sapphire lens of the solar flux extractor transmits greater than 90% of the sunlight that is concentrated 10,000x from the primary solar concentrators into the inner heating chamber of the Shooting Star. "The flux extractor serves as a light pipe to efficiently deliver the energy to the point of extraction." This flux extractor appears in FIGURE 1.8, although it is labeled zirconium oxide, not the sapphire (aluminum oxide) that the authors advocate in this publication.

The design needs to address the issue of thermal shock from "going on-sun." The article does not discuss potential control mechanisms to ensure that the secondary concentrator or the engine heating chamber do not heat up too quickly and suffer fractures when "going on-sun." The authors appear to argue that sapphire is preferable to other materials, including zirconium oxide, because of superior performance under thermal shock.

This engine operates at greater than 2000K, cavity wall temperatures, and ideally would operate at over 2500K. Wong and Macosko assert that the refocusing of the solar energy entering the secondary concentrator's flux extractor by refraction and total internal reflection "are essentially loss-less mechanisms."

1.3.3 Solar Thermal Propulsion Engine Design

In the technical literature and patent filings there are two primary design strategies for STP engines. A solar thermal engine does not combust or "burn" the fuel with an oxidizer. Instead, the rapid heating of the propellant provides the propulsive thrust. The most common application proposed for STP is to boost satellites from low Earth orbit (LEO) to geosynchronous Earth orbit (GEO). During the 1990s, the Air Force Research Lab (AFRL) supported research for the Integrated Solar Upper Stage to provide this boost from LEO to GEO. Nearly all the STP spacecraft for Earth orbit transfer are designed to fly exclusively in solar-inertial mode, with their solar collectors or concentrators always pointed directly and fixedly toward the sun, without any gimbaling or pointing mechanism to adjust for different directions of travel. Such gimbaling or pointing would be necessary for any spacecraft that escapes Earth orbit to travel to the Moon or beyond the Earth-Moon system.

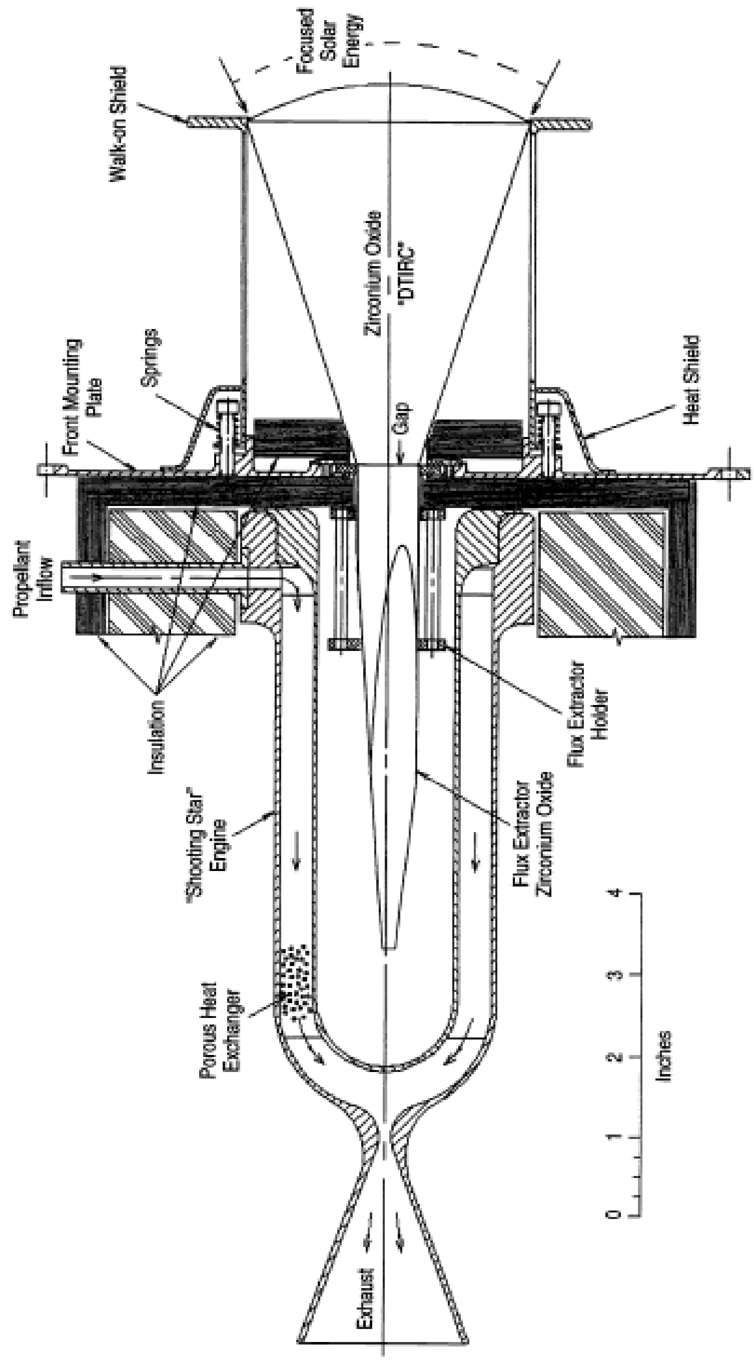


FIGURE 1.8. Longitudinal Section through The Shooting Star Engine with the flux extractor installed.

The first type of STP proposed was direct gain, where concentrated or focused sunlight directly heats a heating chamber that transfers heat to the propellant, causing it to expand rapidly, thereby creating thrust. In two publications Stewart and Martin (1995, 1996) describe their Dual Fuel Solar Thermal Propulsion. Their two fuels are ammonia (NH_3) and liquid hydrogen (LH2), which the authors propose to use differentially in various stages of the boost from LEO to GEO. Stewart and Martin's discussion underscores the limitations of using a very low-density fuel. They argue this limitation means that solar thermal propulsion is suitable only for an orbital transfer upper stage.

The RAP team rejects this view. Our selection of water as a high-density fuel makes possible much higher thrust. One of the disadvantages that Stewart and Martin identify is the loss of propellant from boil off and other causes, which they estimate at 26%. Using water that does not require cryo-cooling or incur nearly the same level of boil off would substantially reduce these losses of propellant.

One point that Stewart and Martin make that is absent from most other publications and patents is that their solar concentrators appear to be pointable to track the sun: "Two parabolic collectors are mounted on a rotation and gimbal system . . ."

Wassom (2001) describes Thiokol's work on an STP engine and elliptical solar concentrator concept during the same period as the NASA MSFC Shooting Star engine. He reports the results of testing on a 2m x 3m prototype of the inflatable concentrator. Interestingly, in the next section, he describes a water-based propulsion system that uses water for propellant feedstock, but electrolyzes it on board the spacecraft into LH2 and LOX.

Several of the patents reviewed below include direct gain solar engines, namely Lee (1982); Shoji (1988); Malloy, Rochow, and Inman (1995); and the three patents from BWX Technologies, Inc. (2001-2004) for the "Solar Thermal Rocket." As the BWX team points out repeatedly, these types of engines would need to rely on fairly large concentrating mirrors to direct the concentrated sunlight into the heating chamber. These concentrators may typically range in size from 25 to 50m, which is may prove quite bulky and awkward for a satellite payload that may be only a meter in diameter by two meters long. The limitation of this approach was that for a spacecraft in orbit about the Earth, the sunlight is only intermittent, so the thrust cannot be continuous.

Because a continuous burn in Earth orbit is not possible in direct sunlight, the trajectory designs emphasize various maneuvering and boosting strategies. The preferred strategy is to perform burns only at perigee or apogee for the maximum effectiveness. However, it is neither practical nor feasible to collect sufficient thermal energy just at those moments of the "burn" (although no propellant is combusted) to achieve the desired thrust to take the spacecraft incrementally into a higher orbit. Therefore, it becomes advantageous to be able to store the thermal energy acquired during flight between apogee and perigee.

Kennedy and Palmer (2002) make the most thorough case for an STP system that relies upon thermal storage to provide the thermal energy when needed and where needed. They begin by making the case for STP from first principles, and provided the diagram in FIGURE 1.8 to identify the STP performance range in comparison with other propulsion technologies. They describe the Air Force Research Lab's follow on to the Integrated Solar Upper Stage (ISUS), the Solar Orbit Transfer Vehicle (SOTV).

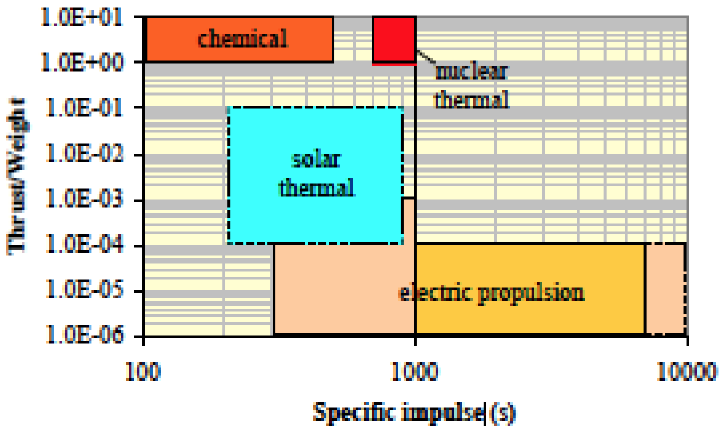


FIGURE 1.8. Performance of solar thermal propulsion systems vis-à-vis chemical, electric, and nuclear thermal propulsion options. Courtesy of Kennedy and Palmer, AIAA-2002-3928[9]

They present a design intended to meet the requirements of this new program through the use of thermal storage on the spacecraft. Kennedy and Palmer discuss the limitations of these USAF systems that would use LH2 as a fuel, which has a density of 71kg/m^3. In this respect, LH2 compares unfavorably to liquid H_20, which has a density of 1,000kg/m^3, and does not require the mass or complexity of cryo-cooling to maintain its temperature. Instead Kennedy and Palmer suggest a variety of alternative propellants, including "storables," in which they include hydrazine, ammonia, and water to operate at a "peak temperature" of 2500K. The engine would rely on rigid fixed optics that would not be pointable.

However, Kennedy and Palmer also suggest a wider range of potential missions than a solar-inertial spacecraft could perform. They describe three such candidate missions (p. 3):

1. *Near-Escape*, including Near Earth Object (NEO) flybys, . . . very highly eccentric earth orbits, and LaGrange point and earth-trailing orbiters. . . with delta-V requirements of as little as 770 m/s;
2. *GEO Insertion*, with relatively long-duration firings at apogee and delta V requirements on the order of 1,700 m/s; and
3. *Other Body Capture*, to include lunar orbiters and interplanetary missions, with delta-Vs ranging from 1,100 m/s to 4,000 m/s and higher. . . .

By "other body capture," these authors mean capture of the STP spacecraft into orbit around other bodies, not capturing an asteroid.

Kennedy and Palmer provide one of the few well-informed discussions of high temperature material selection for the thermal storage and engine design. For their thermal storage system, they selected Boron Nitride (BN) "BN-coated boron carbide particles contained within a BN 'can.'" They also present a treatment of propellant selection in relation to material selection. First they rule out most propellant containing oxygen to avoid oxidation problems. Then they state (p. 7):

[9] Kennedy and Palmer's publication is a work of the US government and not subject to copyright in the United States.

> *Water, while more stable than either N2O or H2O2, is likely to severely corrode graphite, BN, or B4C systems. While refractory carbide and oxide coatings could permit the use of water as a propellant, the authors opted for simple, uncoated structures to reduce cost.*

This assessment suggests that water may be viable as a propellant for STP engines, provided it is possible to overcome the material science challenges that high temperature, high pressure steam poses. The three BWX patents all describe combined direct gain and thermal storage STP engines, discussed below in the patent search section.

1.3.4 Water Propellant

Our colleagues in the NIAC Class of 2012, MSNW LLC, describe a project on their website, http://msnwllc.com/space-propulsion, for the "Electrodeless Lorentz Force Thruster," which they claim can use water as a propellant. We requested further information from them, but they declined to provide anything beyond these two paragraphs on their website:

> The ELF thruster, funded by the Department of Defense, utilizes Rotating Magnetic Field (RMF) and pulsed-inductive technologies that promise radical advances in space propulsion. The ELF creates, forms, and accelerates field-reversed plasma toroids to high velocity. It has demonstrated the ability to efficiently utilize complex propellants such as Martian Air, Liquid Water, and Hydrazine.
>
> The ELF enables a broad range of high-power propulsion missions. Fundamentally, this technology has significantly greater thrust and power densities than any realizable propulsion technology. The ability to operate on in situ propellants will enable very eccentric orbit propulsion, re-fuelable orbital transfer vehicles, deep space return missions, and even direct drag makeup for extremely low orbits. At current power levels, this thruster technology minimizes system mass, size, and cost, while increasing overall mission flexibility. Finally, extending this technology to higher densities and powers that have been demonstrated in the laboratory, there are mission applications in high-altitude, air-breathing, hypersonic flight and beamed-energy upper stage propulsion that are not feasible with traditional technologies. Please see technical publications below for a complete description of experiments, thruster specifications, and results.

2 Review Of Patent Search On Robotic Asteroid Prospector, Including Solar Thermal Propulsion (STP) Systems

The patent search conducted by Levine Bagade Han turned up about a half dozen concepts for an STP engine and in some cases the spacecraft it serves. For the most part, these precursors are not relevant to RAP, mainly because their intended applications are quite limited. They do not contemplate specifically any deep space missions beyond orbital boost or reboost, notably from low Earth orbit (LEO) to geosynchronous orbit (GEO). Some show solar concentrator arrays to generate the concentrated sunlight, but there is very little information about the design of the concentrators; they are almost like Mr. Potato Head ears stuck on as an afterthought.

The Patent search turned up 10 previous patents, of which a few were directly relevant to RAP. However, of these, none were relevant to the RAP spacecraft configuration and they were only minimally relevant to the installation of an STP engine within the RAP. These patent search results were:

2.1 Early Solar Thermal Concepts

This first group of three patents describes the early concepts for solar thermal propulsion.

2.1.1 Solar Space Vehicle

4,354,348 (Lee, 19821019) discloses a solar space vehicle (column 1, lines 4-6). Lee shows a solar-thermal vehicle that consists of two concentric spheres. The outer sphere contains an array of lenses that focus incident sunlight on the inner sphere, which serves as the heating chamber, labeled "boiler," for water as the fuel. Since the boiler is made of copper, the heating of the water to make steam for space propulsion cannot exceed about 812°K, the melting point of copper, and in practice must be conservatively less to avoid local hot spots. The invention includes a "means for delivering said generated steam to the environment outside the outer sphere" to provide thrust, from steam in this temperature range. This disclosure does not show any reflective or parabolic solar concentrators, only the lenses mounted in the outer sphere.

2.1.2 Solar Thermal Propulsion Unit

4,781,018 (Shoji, 19881101) discloses a solar thermal propulsion unit (column 1, lines 39-58).

The design of the spacecraft for which this engine is intended looks very much like the USAF ISUS. It incorporates solar concentrators, the STP engine, and a payload that appears destined from LEO to GEO. However, this invention appears to be the only one that suggests powering of interplanetary probes as an application of the invention. This disclosure shows a reflecting parabolic solar concentrator that appears to be a kind of parabolic trough with an elliptical front perimeter. Shoji et al enumerate or essential parts of any STP engine for their invention:

(a) A nozzle including an exhaust outlet;
(b) A heating chamber;
(c) A "solar window;"
(d) A solar radiation absorber assembly retained within the heating chamber

In addition, they include the non-enumerated parts:

- A propellant storage and delivery system;
- Means for delivering propellant to the absorber assembly; and
- Means for collecting, concentrating and delivering solar radiation to and through the solar window and into the solar radiation absorber assembly, the absorber assembly receiving concentrated solar radiation for heating the propellant in the absorber assembly to generate thrust.

The De Laval nozzle mentioned prominently and repeatedly in this patent is simply the hourglass-shaped nozzle that Robert Goddard first applied to rocket engines circa 1920 and is still the most common form of nozzle. The choice of propellant for this nozzle and engine seems quite obscure, mentioned in passing as "one known solar propulsion concept utilizes and alkali metal seed added to the propellant." The other propellant mentioned is LH2.

The "solar window" appears to be a simple glass or other translucent material that closes in the heating chamber. This choice of materials limits the temperature of the heating chamber to the melting point of glass about 1227°K. For a quartz window, the allowable temperature may rise 1327°K. These numbers to not take into account the temperature resistance of any caulking or "O-rings" to keep the heat in the heating chamber. Inside the heating chamber are a series of "graded porous material" in the shape of disks or wafers to afford the convective heat transfer from the sunlight hitting the porous material to the LH2. This porous material would consist of "zirconium carbide, tantalum carbide, hafnium carbide, and a combination thereof."

The inflatable parabolic solar concentrators appear to be held in position by a few thin rods or rigid wires. ***There is no indication that these parabolic concentrators are articulated or pointable in any way.***

The intended fuel is cryogenically maintained liquid hydrogen. This propellant requires a great deal of energy to maintain it in liquid condition. The patent states somewhat idiosyncratically that this engine "delivers high specific impulse values (1000 1 bf sec/1 bm)." This use of non-standard nomenclature is rather a puzzler.

One innovation is that this STP engine appears to be the earliest to address the 90° bend problem between the vector of the incoming concentrated sunlight and the direction of thrust. The concentrators would be inflatable

2.1.3 Hybrid Solar Rocket Utilizing Thermal Storage For Propulsion and Electrical Power

5,459,996 (Malloy III et al., 19951024) discloses a solar rocket system collecting solar energy and using it for both propulsion and electrical energy (column 1, lines 48-60). Malloy III et al emphasize the central role of the thermal storage system as the key to the propulsive burns or to a thermal to electric diode-based conversion system. The summary states that it is an aim of the invention "to provide a system having independent pointing mirrors for focusing sunlight into a thermal storage receiver at any spacecraft." The patent mentions mounting the mirror segments on a "rotation bearing" at the end it attaches to the spacecraft, but there is no clean line of sight or optical focus from the mirror segments to the "receiver" of the concentrated thermal energy. There is no suggestion that these mirrors would point anywhere except toward the sun from an Earth orbit, flying in solar-inertial mode.

Finally the summary states "It is another object of the present invention to provide a system which [sic] utilizes a variety of propellants in order to strike a balance between tankage volume and high specific impulse." The preferred embodiment mentions "hydrogen, ammonia, methane, or other suitable propellants." What is ironic about this assertion is that none of these propellants may truly be "suitable" because they all require an oxidizer and a combustion chamber, and there is no mention of an oxidizer. However, the use of thermal storage allows the engine to fire selectively at the desired points in the orbit, and not be bound to fire only when in direct sunlight (e.g. on the night side of the orbit).

The engine operates by a two-stage process of heat transfer into storage and out of storage into the propellant. The engine accumulates thermal energy (heat) from the concentrated sunlight in the boron or boron carbide thermal storage capacity. This storage can hold the heat for some period of minutes or hours, typically to be discharged for apogee or perigee burns by transferring heat "by forced thermal convection" through a second heat exchange process into the propellant.

This spacecraft is designed to boost payloads from low earth orbit (LEO) to geosynchronous orbit (GEO). The point of the rotating bearing at the base of the mirror leaves is not explicated. This spacecraft appears to fly in a solar-inertial mode, with the mirror segments always pointing toward the sun except perhaps for minor adjustments that the "rotation bearing" would allow. There is no suggestion this spacecraft or STP engine would serve the purpose of tracking the sun on a deep space Earth departure.

2.2 BWX Technologies Inc. Solar Thermal Rocket Patents

From 2001 to 2004, BWX Technologies filed three patents for their "Solar Thermal Rocket." In all three cases, it was a "hybrid" engine that accommodated both a direct gain heating section and a thermal storage section. The order of issue of the patents differs from the apparent order in which the inventors developed the concept, so that the first patent describes a variant of the core concept that appears in the second patent.

2.2.1 Solar Thermal Rocket (1)

6,290,185 (DeMars et al., 20010918) discloses a solar thermal rocket (column 2, lines 55-66).

This patent's background to the invention presents an excellent summary of the challenges that the RAP spacecraft faces in implementing STP. In addition to the usual discussion of heat storage and direct gain, it addresses thermal shock and heat loss from the engine.

This patent is much like the other BWX Technologies Inc. patents, particularly Westerman 20020205, US Patent 6,343,464, except that instead of treating the concentrated sunlight heating of the direct gain interheater and the thermal storage section as completely separate, it connects the two on the downstream side. This design takes the propellant that is heated initially by direct gain in the interheater and then passes it though the thermal storage unit to increase the heating of the propellant. It is one of the few designs that recognizes the potential problem of thermal shock in the engine, although it discusses it only in the context of the second stage heat exchanger in the thermal storage unit. The inference is that the direct gain interheater provides the thermal tempering by heating up the cryogenic propellant from 25K to whatever temperature the thermal storage device can handle. This design also makes a provision to reduce radiant heat loss from the STP engine through the addition of a “secondary solar concentrator,” although neither the drawings nor the text explain how this plug-ring device installed in the engine or thermal storage aperture actually collects sunlight to concentrate. The operating temperatures in this STP engine variant are modest compared to claims in other inventions. This interheater would achieve propellant heating up to 1200K and the solar thermal storage heating would reach 2400K. This design accounts for the 90° bend with a nozzle that appears to stick sideways out from the primary heating chamber.

2.2.2 Solar Thermal Rocket (2)

6,343,464 (Westerman et al., 20040205) discloses a solar thermal rocket (column 1, line 65 through column 2, line 11).

In this version of the invention, the engine would first heat the propellant in the thermal storage section and then second in the direct gain interheater. This sequence of heating the propellant simply switches the sequence promulgated in the other BWX Technologies Inc. patent 6,290,185 DeMars et al.

This patent propounds a dual system for generating thrust that incorporates both a direct gain interheater subsystem and a thermal storage subsystem. It includes an “optical switch” that allows the concentrated sunlight from the solar concentrators to switch its pointing between these two subsystems. This engine would use liquid hydrogen (LH2) as its propellant. The patent argues that the lower mass or density of the LH2 presents an advantage compared to conventional or “storable” propellants. This advantage is dubious at best because it means that the size of the tankage for the cryogenic LH2 must be much larger than for storables and the low mass density must translate ultimately into lower overall thrust. The inventors make the argument for the thermal storage subsystem as an alternative to very large solar concentrators, which they estimate (elsewhere) in the range of 25m to 50m. The patent drawings show the concentrator as only the most vague cartoon, not even physically connected to the “solar thermal rocket.” This concentrator has the ability to switch pointing of the concentrated sunlight between the two subsystem receivers, but it does not have the ability to move to track the sun. It appears to be intended only for LEO to GEO boosts while flying solar inertial.

2.2.3 Solar Thermal Rocket (3)

6,574,951 (Miller et al., 20030610) discloses a solar thermal rocket (column 2, lines 33-54).

This variation of the BWX Technologies Inc. design adds a “preheater” adjacent to the solar energy secondary concentrator. The non-pointable solar concentrator directs sunlight into this secondary concentrator orifice. The text proclaims that the engine design reduces complexity by combining the “receiver/absorber/exchanger” (RAX) into a single, simpler assembly, and that this assembly is modular in the sense that multiple copies can be added together to make a larger space vehicle. This invention claims the ability to achieve 750 seconds of I*sp*. Most of the drawings and the text is devoted to describing an improved thermal storage section.

2.3 Specialized Retrieval Spacecraft and Solar Thermal Patents

Following the filing of the BWX opus, four inventors acquired patents that touch on specific aspects of spacecraft that bear a superficial similarity to the Robotic Asteroid Prospector. Two of these patents describe spacecraft intended to retrieve objects in space and two provide narrowly focused details of how to accomplish specific tasks using concentrated solar energy.

2.3.1 Method of Making a Thruster Device for Solar Radiation

6,745,466 (Frye et al. 20040608) discloses a solar thruster for spacecraft (column 4, lines 9-16).

James M. Shoji, who was the sole inventor on the Solar Thermal Propulsion Unit, (1988) is a co-inventor on this patent. This patent provides a detailed design for a direct gain solar thermal propulsion thruster. The main innovation is the spiral wound tube that runs around the inside perimeter of the frustoconical shaped heating chamber. Because of its very long total length, this spiral heat exchanger provides the maximum surface area for the concentrated sunlight to heat the propellant to thrust-generating temperatures. What may be most important about this design is the simple way it can be manufactured and assembled, without the need for critical weldments or difficult seal conditions. This thruster design could theoretically substitute for the heating chambers in Shoji (1988) or in the Shooting Star Engine.

2.3.2 System for Capturing and Recovering Free-Flying Objects in Space

7,168,660 (Bischof et al., 20070130) discloses a system for capturing free-flying objects in space (column 1, lines 61-65).

This patent is a rare European (DE) entry into the innovative spacecraft arena. It does not employ solar thermal propulsion, or even mention a specific type of propulsion as part of the claims. Instead, it focuses almost entirely on a “spring loaded system” for throwing and closing a net to capture “free-flying objects.” The definition of a “free-flying object” is not quite clear. It appears to refer to satellites or other spacecraft, but the context for using the net seems to imply that the free-flying object may be tumbling out of control. It clearly does not apply to an asteroid because an asteroid is not “free-flying,” but in a natural solar orbit. The only part of the claims that are relevant are the vanilla assertions that the “system” comprises “a space platform, and at least one capture device mounted on said platform; wherein said capture device comprises: . . .”

2.3.3 In Orbit Space Transportation and Recovery System

7,624,950 (D'Ausilio et al., 20091201.) discloses space systems with objectives and missions including replenishment of fuel, producing propellant from an asteroid and processing ice present on an asteroid (column 9, line 51 and column 10, lines 12, 15).

The first and most obvious difference between this invention and RAP is that it is designed specifically for Earth orbital operations. The inventors even trademarked the acronym IOSTAR™, incorporating the “In Orbit” into its title. The most distinguishing characteristic is the recovery system, which includes a variety of grasping devices to grab onto satellites that presumably have handles, fixtures, or other protrusions or cavities to allow grasping by the deploying or recovering vehicle. The dominant grasping target would be the mounting ring that attached the satellite to the upper stage booster rocket that placed it in its final orbit.

The propulsion system is nuclear electric, capable of producing 500KW of sustained electrical power. There is no solar electric or solar thermal component to the power or propulsion system. The primary fuel, at least for maneuvering, is hydrazine. The spacecraft incorporates an extensible boom to separate the high radiation and heat-producing nuclear power plant from the grasping end of the system, presumable so as not to damage the satellite

This patent includes three tables that consist of long enumerations of items that read like a patent troll's shopping list. Table 1 presents a list of "Electric Propulsion Alternatives" but it does not include solar thermal. Table 2 lists nearly all the propellants that have been used in spacecraft (and a few that have not). It includes H_2O, but the patent does not give any specifics about how it might be used. Table 3 lists 39 "Objectives and Missions." Of these 39, three mention an asteroid:

- Use a laser to divert an asteroid,
- Produce propellant from an asteroid, and
- Process ice present on an asteroid by electrolysis to form hydrogen and oxygen.

These mission objectives show all good intentions, but neither the drawings nor text show any details of how to accomplish these objectives, nor do the claims mention them. In fact, the claims focus almost exclusively on the nuclear electric power and propulsion system and its associated cooling loop and propellant subsystems.

2.3.4 System and Method of Solar Flux Concentration for Orbital Debris Remediation

2012/0261514 (Boone, 20121018) discloses a system for concentrating solar radiation onto orbital debris (paragraph [0011]).

This invention is almost entirely about a method of deploying a solar concentrator from a tightly packaged payload into a large, lightweight structure. This structure incorporates two focusing systems, presumably powerful lenses, to focus concentrated sunlight on orbital space debris within a specified size range of "Category 2" from 1 to 10 cm, and to "vaporize it." This patent does not discuss or claim anything pertaining to the platform spacecraft design, propulsion, or power system. The patent claims the use of several types of lenses or optical devices in its optical focusing design: Fresnel, pinhole array, photon sieve, and a "plurality of isosceles panels." It makes no mention of parabolic or elliptical concentrators, mirrors, or lenses.

3 Forensic Archeology of the Shooting Star Engine

The Shooting Star engine appears to represent the only attempt to build and test an STP engine. Although the Shooting Star program lasted only about four years, it did establish a kind of baseline (although perhaps not a benchmark) for the design, development, testing, and engineering (DDT&E) of STP technology. This follows on the selective literature review and patent search review above that touches on the highlights of key aspects of STP designs and precursors.

There is a fragmented variety of literature and imagery available on the STP in general and on the Shooting Star more specifically, but putting together a coherent story has been elusive. FIGURE 2.1 shows the disassembled pieces of an early version of an STP prototype that appears to pre-date the Shooting Star, although it may in fact include some Shooting Star parts. The official caption on the Marshall Image Server states:

Components of the Solar Thermal Propulsion Engine	
MSFC-9701564	Feb 01, 1997
Researchers at the Marshall Space Flight Center (MSFC) have designed, fabricated, and tested the first solar thermal engine, a non-chemical rocket engine that produces lower thrust but has better thrust efficiency than a chemical combustion engine. This photograph shows components for the thermal propulsion engine being laid out prior to assembly. MSFC turned to solar thermal propulsion in the early 1990s due to its simplicity, safety, low cost, and commonality with other propulsion systems. As part of MSFC's Space Transportation Directorate, the Propulsion Research Center serves as a national resource for research of advanced, revolutionary propulsion technologies. The mission is to move the Nation's capabilities beyond the confines of conventional chemical propulsion into an era of aircraft-like access to Earth-orbit, rapid travel throughout the solar system, and exploration of interstellar space.	

This caption appears to be the only “official” documentation that NASA started work on STP about 20 years ago.

3.1 Where We Started with STP

When we wrote the RAP NIAC Phase 1 proposal a year ago, we had access to the “popular” literature plus a few conference papers and journal articles. From what we found at that time, it appeared that the Solar-Thermal engine technology was developing well along and commensurable with competitors such as solar-electric nuclear-electric, and nuclear-thermal. We thought that although a solar-thermal engine was not available off the shelf, there was enough progress that we could pick up where others had left off. However, it turns out that the solar-thermal engine technology is not nearly as advanced as it had appeared. That is the main realization from this deep-dive into the literature and history. It might be more accurate to describe this effort as forensic archeology. I believe we have identified all the key pieces of hardware from 1998-2002, who made it, what they did with it, what testing may or may not have been completed, and where those artifacts may exist now.

FIGURE 3.1. Components of the early Solar Thermal Propulsion Engine, presumably pre-Shooting Star Program. Courtesy of NASA MSFC.

3.2 Where We Are Going with STP

However, please do not mistake our current retrospective attitude toward Solar-Thermal propulsion for “backing off.” On the contrary, we have been engaged in a ferocious debate whether we should use the *Shooting Star* as a jumping-off point or we should go directly to a new “clean sheet” start. So, we are “digging in” to make sure Solar-Thermal can work for RAP, regardless of whether we eventually choose to adopt the *Shooting Star* template or create an entirely new engine. The following discussion treats the MSFC *Shooting Star* concept as a kind of exemplar for what we need to accomplish to address all the known and as yet unknown challenges. The design of the STP engine and its subsystems must integrate smoothly into the RAP spacecraft and its operations. This discourse led to our own detailed design sketches the RAP spacecraft to incorporate an STP engine without needing to go into the specific design of an engine itself.

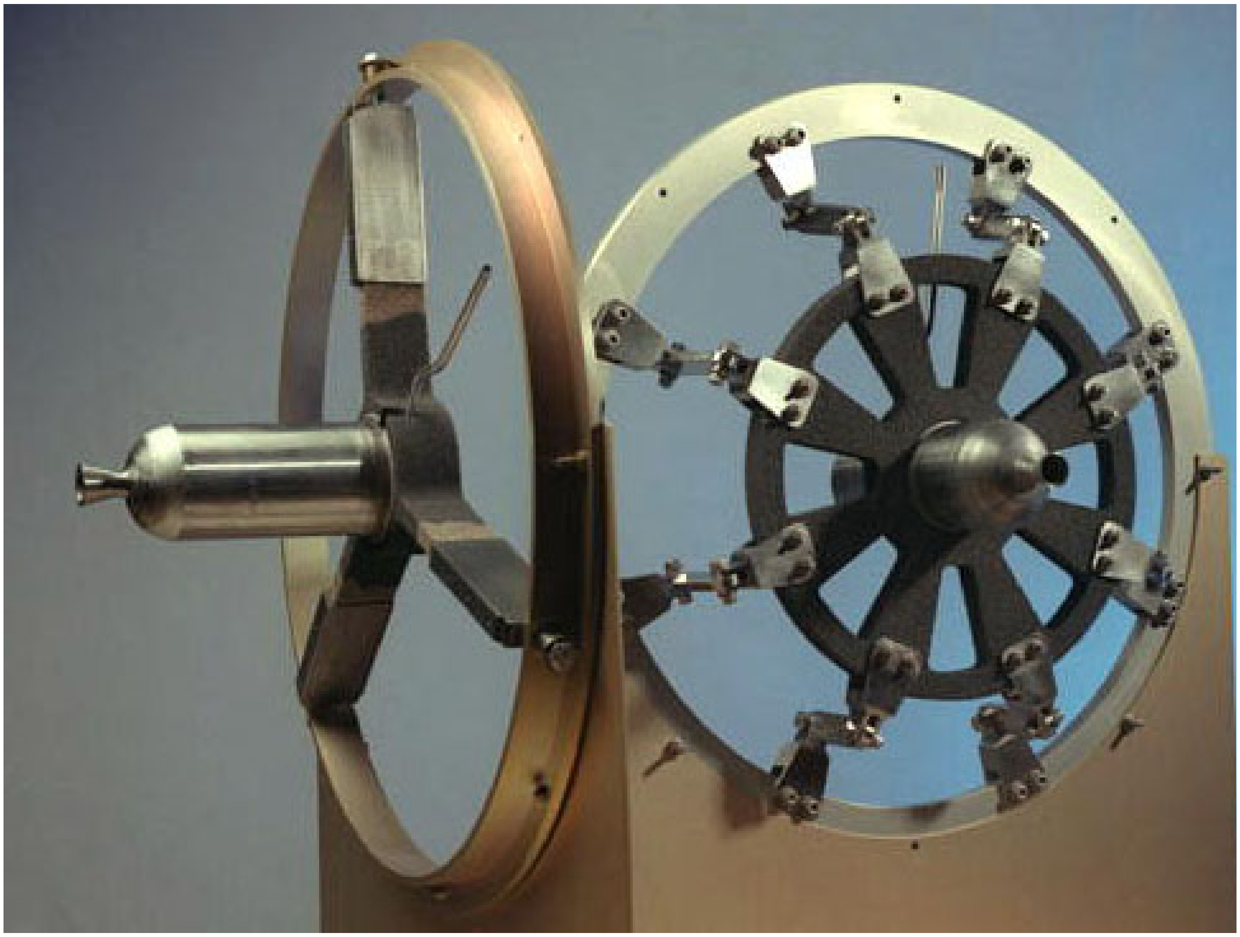


FIGURE 3.2. Two Versions of the Shooting Star Engine Mounted on Rhenium Foam Wheel Support Structures, Courtesy of Ultramet Corporation, Pacoima California.

3.3 The Shooting Star High-Water Mark

Our key revelation is that these two critical elements were never united in practice. Glenn Research Center (GRC) made the lens; in their publications, they show results from thermal-shock testing and advocate the need for thermal-vacuum testing that apparently they could not accomplish. In the absence of the sapphire lens, the MSFC team tested the Shooting Star with an electric carbide resistance heater. This approach enabled them to test some basic properties of the Shooting Star Engine at high heating. That testing was essential to demonstrate the materials properties of a rhenium chemical vapor deposition-formed (CVD) pressure vessel lined with rhenium foam to create the heat exchanger. However, the MSFC team did not achieve the central objective: to heat the engine through insolation by light concentrated to 10,000x the intensity of normal sunlight. The literature we read a year ago glosses over this omission.

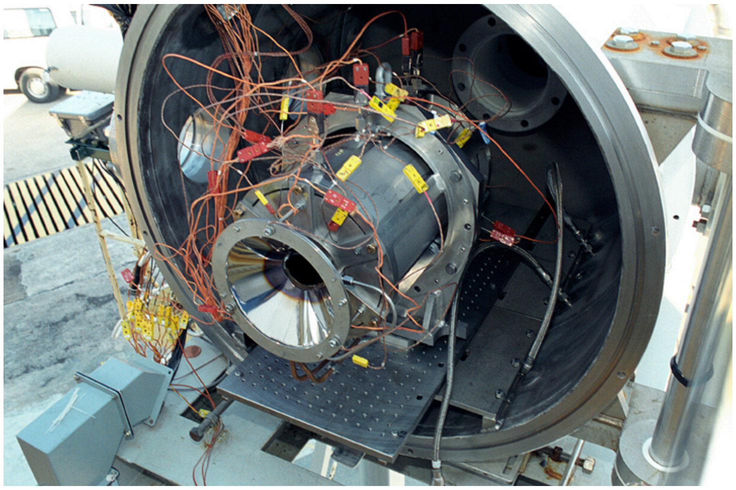


FIGURE 3.3. Solar Thermal Engine Test Facility at NASA MSFC with an STP Engine Supposedly Mounted in it. Courtesy of NASA MSFC.

When the team at MSFC tested the preferred version of the Shooting Star, they mounted it in a rhenium foam "wagon wheel" style frame, as shown in FIGURE 3.2. They produced two or three engine/rhenium foam wheel ensembles. The reason for this design and material selection was that the whole engine would get so hot – e.g. it would radiate such intense heat – that it would need to be isolated from any surrounding structures by the high-temperature Rh foam. Presumably, the team chose the foam for its comparatively low conductivity but high loss via convection properties.

In conjunction with the Shooting Star Engine, NASA built a Solar Engine Test Facility at MSFC. Although the photo in FIGURE 3.3 shows this facility, it does not appear that it was ever actually used to test a Solar Thermal Engine.[10] The caption provided with the photo on MIX states:

Solar Thermal Propulsion Test Facility	
MSFC-9906910	November 1, 1999
Researchers at the Marshall Space Flight Center (MSFC) have designed, fabricated, and tested the first solar thermal engine, a non-chemical rocket engine that produces lower thrust but has better thrust efficiency than a chemical combustion engine. MSFC turned to solar thermal propulsion in the early 1990s due to its simplicity, safety, low cost, and commonality with other propulsion systems. . . .This photograph shows a fully assembled solar thermal engine placed inside the vacuum chamber at the test facility prior to testing. The 20- by 24-ft heliostat mirror (not shown in this photograph) has a dual-axis control that keeps a reflection of the sunlight on the 18-ft diameter concentrator mirror, which then focuses the sunlight to a 4-in focal point inside the vacuum chamber. The focal point has 10 kilowatts of intense solar power.	

3.3.1 Heat Gain and Heat Loss

FIGURE 3.4 shows a longitudinal section through the Shooting Star engine. It shows the absorber cavity (the primary heat exchanger heating chamber), where the incident sunlight flux would heat the inner wall and the propellant that flows behind it. FIGURE 3.4 also shows the other surfaces of the engine, which can all act as points or surfaces of heat loss. The Shooting Star engine employs all three modes of heat transfer: conductive, convective, and radiative. The dynamic process starts with the radiative heat gain in the inner cavity, where the concentrated sunlight (in

[10] Two employees at MSFC who worked on the STP program told the RAP PI that this facility was never actually used to test an STP engine, although it has been used for other purposes.

the visible light portion of the spectrum) insolates the heat-absorbing cavity wall. This inner wall then transfers its heat gain to the adjacent rhenium foam, primarily by conduction, we believe, although there would also be some re-radiation from the "backside" of the rhenium inner chamber wall. The rhenium foam heats up to a high temperature in the 2000-2500k range. The foam transfers this heat, mainly by convection and conduction to the propellant that is passing through the heating chamber. The question of heat loss arises for the outer wall of the STP engine. The rhenium foam would conduct heat to the outer wall, which would then reradiate heat in the infrared portion of the spectrum. Achieving the right balance will be important between heat gain, convective/conductive "cooling" of the engine by the propellant, and heat loss that occur simultaneously.

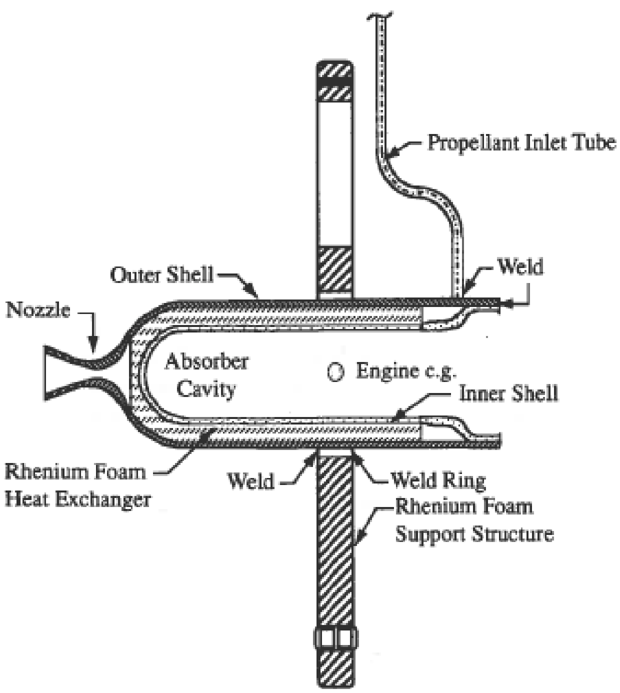


FIGURE 3.4. Longitudinal Section through the Shooting Star Engine "Wagon Wheel" test configuration concept. (From Tucker, Salvail; 2002).

3.3.2 Radiative Loss, Gain, or Insulation?

In our analysis of the Shooting Star design, one feature raised a big question mark: Why does all the solar heating occur only in the inner cavity? The outer surface area exceeds that of the inner cavity, so therefore the rate of conductive heat loss would exceed the rate of conductive heat gain. What enables the heat exchanger to operate is the amount of heat transferred by convection through the foam to the propellant. Kennedy and Palmer (2002) at the Surrey Space Center in the UK presented an alternative design concept for a Solar-Thermal engine. Their "Micro STP" engine employs an inner cavity for the concentrated light somewhat similar to the Shooting Star, so it appears that they recognized this same potential issue. They attempt to solve the external conductive heat loss problem by encasing it in a massive block of thermal insulation.

However, we recognized that there might be an advantage to heating both the internal and external surfaces of the engine, however, that will require a different arrangement of mirrors and lenses. The lenses offer the advantage of very little heating or energy loss, however reflectors or mirrors incur substantial heating and energy loss with each reflection. What is more challenging is that the mirrors will require active cooling. One of the articles we reviewed asserts that no reflective material will retain its reflective properties above 1500K. The maximum potential design operating temperature for the rhenium inner cavity is about 2500K (although the MSFC literature speaks of up to 3000K). So the baseline for operating temperatures will need further attention. In addition, the lesson we draw from these heating physics and dynamic factors would be to try to minimize the use of diverter mirrors while relying more extensively on custom designed lenses.

FIGURE 1.8 shows a more detailed and close-up longitudinal section through the Shooting Star STP engine from Soules, Buchele, Castle, and Macosko (1997, p. 244). This cross section is instructive in providing the dimensional scaling of the Shooting Start test article (despite the fact that it underwent severely constrained and incomplete testing). Please note that the width of the "porous heat exchanger" where the propellant receives heat is no more than about three inches (7.5cm). The flux extractor therefore is quite thin, on the order of less than one inch in width, and much less at the distal end closest to the thruster nozzle. The propellant piping appears entering the heat exchanger in only one location, but presumably there would be multiple fuel feeds of this type to ensure an even distribution and expansion of the fuel in the heat exchanger. The inside and outside surfaces of the heat exchanger appear to be made from the same material, the same thickness, etc., so there is no suggestion of trying to insulate the outside surface against radiative heat loss.

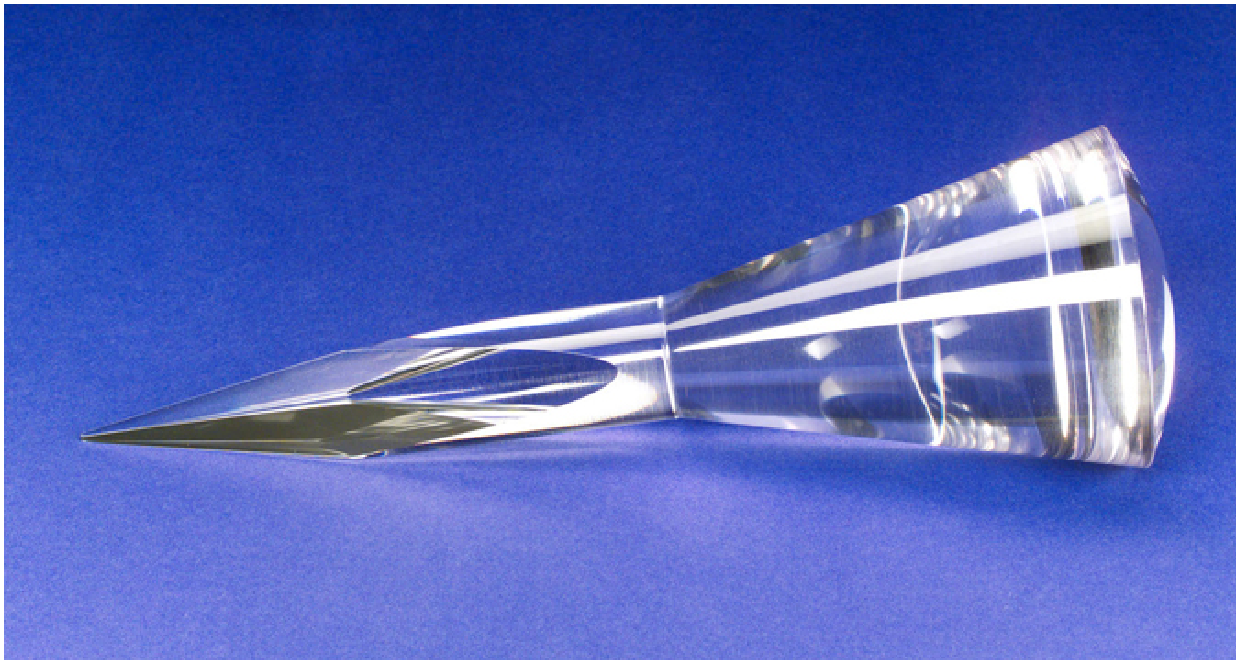


FIGURE 3.5. Sapphire Lens Flux Extractor for the Shooting Star STP Engine.
Courtesy of Wayne Wong, NASA Glenn Research Center.

Another feature of all these longitudinal sections through the Shooting Star engine is the cross section through the large blocks of material that presumably represent blocks of insulation. However, none of the drawings are sufficiently complete to determine if there was a serious intention of insulating the heating chamber/heat exchanger against heat loss from the outer surface. Certainly, the two engine-rhenium wheel ensembles in FIGURE 3.2 do not reveal anything about surrounding insulation.

3.3.3 Propellant

Another discrepancy worth noting is that the propellant for the "hot" test differed from the original and published design. The engine was designed for H2 (not clear if it was meant to be liquid or gas), but MSFC tested it with N2. We imagine that there may be more features that differed historically from the planned project.

FIGURE 3.5 shows a photograph of the Sapphire Flux Extractor Lens in pristine condition, never having been installed in the Shooting Star prototype. Although it looks like a single casting or crystal, in fact it is assembled from two pieces, the lens that fits in the solar window of the heating chamber and the flux extractor that extends deep into the cavity. FIGURE 3.6 shows a different version of the Shooting Star longitudinal section that labels the joint between the two pieces as the "Diffusion Bond." In FIGURE 1.8, it is labeled simply as a "Gap."

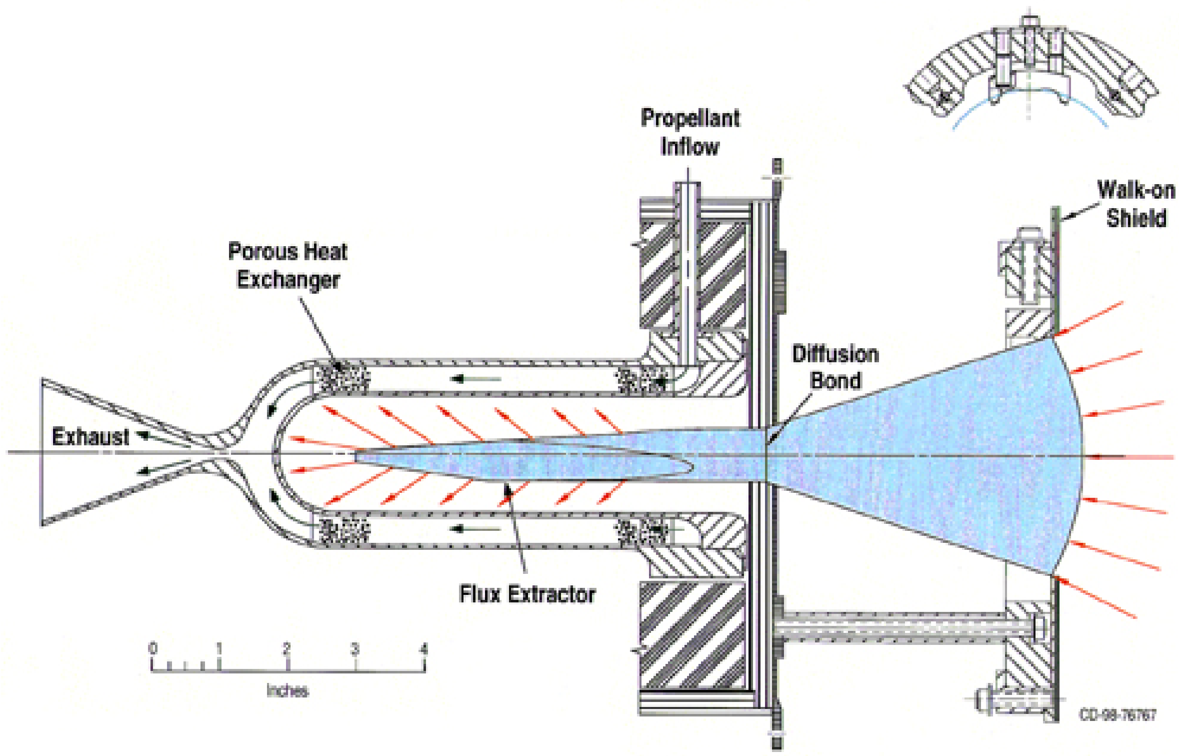
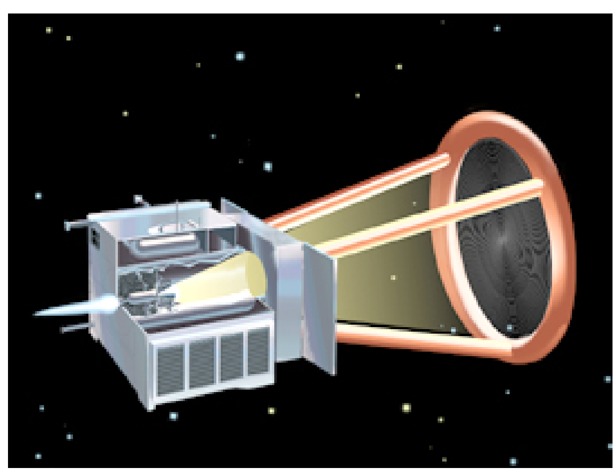


FIGURE 3.6. Longitudinal Section through the Shooting Star Engine, showing where the "Diffusion Bond" Joint occurs in the Sapphire Flux Extractor.
Courtesy of Robert P. Macosko and Richard M. Donovan

3.3.4 Shooting Star Flight Test



The Shooting Star Program was intended to include provisions for a flight test. Not much detailed information is available, but FIGURE 3.7 the one such image we could find, shows that the inflatable solar concentrator is positioned rigidly in line with the axis of the engine. That geometry would appear to mean that the engine could "fire" only while it is flying in a straight line directly away from the sun. Please note the small white thruster plume on the side opposite the concentrator/reflector.

FIGURE 3.7. Artist's Rendering of the Shooting Star Flight Experiment Concept. Courtesy of NASA MSFC.

Meanwhile, we have been working on details of the RAP spacecraft configuration design and the PI has been doing analytical drawings to understand how to adapt, modify, and apply the Shooting Star template to the RAP spacecraft. We have not been able to close the design solution yet: far from it, we are on the upward swing of a very steep learning curve that we did not imagine would be necessary when we started RAP. Some of our insights can be surprising, given the preceding discussion. For example, needing to cool diverter mirrors may not be all bad; perhaps we can use the diverter mirror cooling system as the "stage one" heat exchanger to accomplish the propellant phase change from liquid water to steam.

Using water as the propellant raises a host of material science questions at high temperature. Water is highly corrosive and the roster of materials that can resist it erosive and corrosive effects, especially at high temperature, appears rather limited. In this area, the Shooting Star precursor does not help. One of the articles states unequivocally that rhenium is the only material that could serve to make this high temperature engine, which we question.

4 The Solar Thermal Energy "Triple Threat"

RAP poses the "triple threat" of solar thermal concentrated sunlight flux powering solar thermal propulsion, powering a solar furnace-type mining processing system, and generating megawatt electrical power. It is easy to state this triplex capability, but far more challenging to implement it. The primary challenge concerns reflecting or deflecting the concentrated, but still incoherent (i.e., not coherent like a laser) light with mirrors. These mirrors require cooling to keep them below, 1500K above which they will lose their reflectivity and suffer heat damage. The sequence of possible use is also important. TABLE 7.1 shows that there are six possible permutations for the sequence in which the RAP system makes the concentrated sunlight available to the point of use.

TABLE 4.1. Permutations in the Use Sequence of Concentrated Sunlight

	First Use	Second Use	Third Use
Sequence 1	Electricity	Propulsion	Process Heat
Sequence 2	Electricity	Process Heat	Propulsion
Sequence 3	Propulsion	Electricity	Process Heat
Sequence 4	Process Heat	Electricity	Propulsion
Sequence 5	Propulsion	Process Heat	Electricity

Sequence 6	Process Heat	Propulsion	Electricity

Each sequence of application of concentrated sunlight implies functional, geometric, and operational distinctions for the design of the entire RAP spacecraft. It will especially pose issues for the design of the *Grid of Light* distribution system. TABLE 4.1 shows megawatt electricity generation, as the first ordering unit of the six permutation sequences – because it is the lowest priority in the triplex. This layout is intentional to draw attention to the implications for the propulsion and process heat applications.

FIGURE 7.1. Parabolic Solar Concentrator focused on an electrical generator in the secondary reflector position.

4.1 Electrical Power Generation

Although the STP engine is the first priority user of concentrated sunlight, in fact the first opportunity to use it will arise for the electrical power generator. This generator may be a Stirling cycle, Brayton cycle, or other type of power device that converts intense incident solar energy heat into power. The logical place to locate this device would be in the secondary reflector mount position, as shown in FIGURE 7.1. Since this secondary reflector will also need to deflect the concentrated sunlight along the pathway to the other uses, it must be possible to vary and control the relative amounts of sunlight that it distributes to them. That division of the insolation means there will need to be some kind of shutter or aperture control. The more the aperture is reduced for generating electrical power (and accounting for energy losses along the way), the more concentrated sunlight would available for deflection toward the STP engine and on to the mining processing point of use. FIGURE 7.2 shows a close-up of a state of the art three-phase solar thermal Stirling engine.

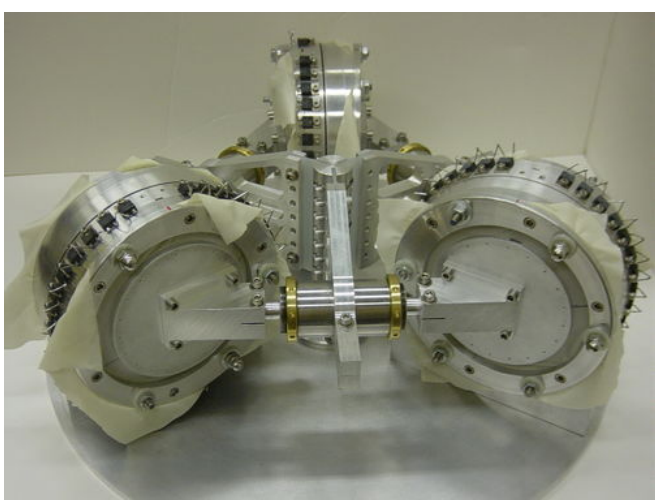


FIGURE 4.2. Three Phase Solar Thermal Stirling Cycle Engine,

Courtesy of Artin Der Minassians and Zachary Mills Norwood (Professor Seth R. Sanders)
UC Berkeley Department of Electrical Engineering and Computer Science

4.2 Obliquity of the Optics

Wong and Macosko (1999) discuss the problem of thermal shock. The need they recognize is to avoid cold propellant hitting a hot surface or intense sunlight flux hitting a cold surface too quickly. For this reason, the operational design of RAP requires that these exposures should occur gradually and under precise control. The need is to heat the propellant gradually to first phase change water to steam and then to superheat the steam up to 2500K or more. Similarly for the introduction of intense sunlight flux, the need is to gradually heat the receiving and reflecting surfaces.

The primary means of accomplishing this operating control to ensure gradual changes heating and cooling is to control the directness of the concentrated sunlight flux. The most direct way to achieve this control is to gradually rotate the parabolic solar concentrators at the alpha joints to slowly increase the sunlight deflected into the grid of light to distribute it to the first stage and second stage heat exchangers. If an actively cooled reflector (mirror) serves as the first stage heat exchanger to heat the propellant from liquid to steam, then it would experience gradual heating as the primary concentrator slowly aligns with the sun to gradually heat its surface and the propellant that cools it. A similar process would occur with the apparatus for heating the cavity of the heating chamber – the second stage heat exchanger – so that the propellant gradually reaches its operating temperature and pressure. In this way, the gradual heating of the engine cavity will protect secondary concentrator lens, the flux extractor, and the structural materials of the engine from thermal shock.

4.3 Directing Sunlight into the STP Engine

The fact that two of the STP engine designs reviewed in the patent search take pains to direct the nozzle thrust at 90° to the incoming sunlight vector shows how significant a challenge it is. In the RAP STP engine, this 90° bend will pose a similar concern because the RAP design requires pointable concentrators that can only direct the concentrated sunlight laterally from the sides (e.g. port and starboard, if such nautical terms are appropriate) to the secondary concentrator lens. The Shooting Star engine *could* have incorporated such a secondary, but the challenge remained unaddressed concerning how to focus a lateral source of concentrated sunlight directly upon it.

4.4 Directing Sunlight to the Mining Process Furnace

The application of process heat to the asteroid mining operation is the third leg of the triple threat, but it is the most unique in the sense that it embodies the fundamental purpose of RAP and its mission. The challenge for this solar furnace is that because it sits at the farthest end of the spacecraft from the solar concentrators, the concentrated sunlight has the farthest to go to reach it and probably the light will need to bounce off of more diverter mirrors than for the electric power or the STP applications. This distance may also bring the inverse square law into play, diminishing the energy of the sunlight flux as it travels to this point of use. We will need to calculate the amount of heating necessary to enable the solar furnace to function effectively for mining processes.

4.5 How Much Energy is Available?

One of the essential studies that we need to make for the RAP installation of an STP system is how much energy will be available and how much will be required for a given size of system. This estimate may seem simple on the surface, but what we do not understand yet are:

- The energy losses from passing through lenses,
- The energy losses from reflecting from mirrors or being deflected,
- The effects of reflector or mirror obliquity to the main direction of "beaming" the light,
- The energy losses from the inverse square law for distance transmission, and
- How efficiently the light that finally reaches the inner heating chamber wall will convert to heat, and how much of that heat transfers through the heat exchanger to the working fluid of the propellant.

We will need to understand how the increases in power and the increases in losses scale with the complete system, and where it is possible to achieve efficiencies as systems grow larger or shrink smaller.

5. Discussion: Architectural System Options

This chapter looks past the Phase 1 Final Report to Phase 2. What is more important, it looks beyond Phase 2 to future questions concerning exploration issues for RAP and the architectural design and system options. This Chapter has two parts: the discussion about what we know and do not know about exploration for RAP, and the discussion of the spacecraft and system architectural issues beyond Phase 2.

5.1 Synthesis

This technical appendix describes the several points of novelty in the RAP spacecraft, plus some aspects of the further design, engineering, operations, and technology development to make it a reality. In many cases, the discussion presented these issues as sets of options or priorities. It is not the purpose of the technical appendix to identify preferred design solutions, but rather to lay out the problems for which fruitful future work will be necessary.

FIGURE 5.1 shows a more detailed drawing of RAP to indicate an example of some of these choices and options. In terms of the alpha joint – beta joint discussion in section 6 above, this drawing portrays the simplest configuration, Option 1, with alpha joints only that provides process heat half the time when attached to the pole of an asteroid. FIGURE 10.1 shows the STP engine in the position to receive concentrated sunlight flux at either side of the engine mounting. In this alignment, the engine installation could point the flux directly to secondary concentrator lenses or to a first stage mirror-heat exchanger. FIGURE 10.1 also shows both the propellant tanks close to the engine and the payload tanks. Unlike the drawings in the main part of the final report where the propellant tanks are much smaller than the payload water tanks, in this one they are about the same size. The design of the RAP spacecraft will require many such design iterations to look at the many possible combinations of the sets of options to arrive at an optimal combination of subsystems and other choices.

5.2 Knowns and Unknowns for RAP as an Exploration Mission

This section presents an overview of the Robotic Asteroid Prospector Project going forward toward the Phase 2 proposal. It is philosophical insofar as it discusses the state of our knowledge for staging exploration missions, which is what RAP will be in the most fundamental sense: what we know, what we do not know, and what we know only partially or incompletely. We take this moment before completing the Phase 2 proposal to contemplate our progress toward acquiring this knowledge, verifying its relationship to the requirements, and validating its ability or hopes to meet those requirements.

Towards the end of this Socratic dialogue, we consider some analyses and studies to undertake after completion of Phase 2.

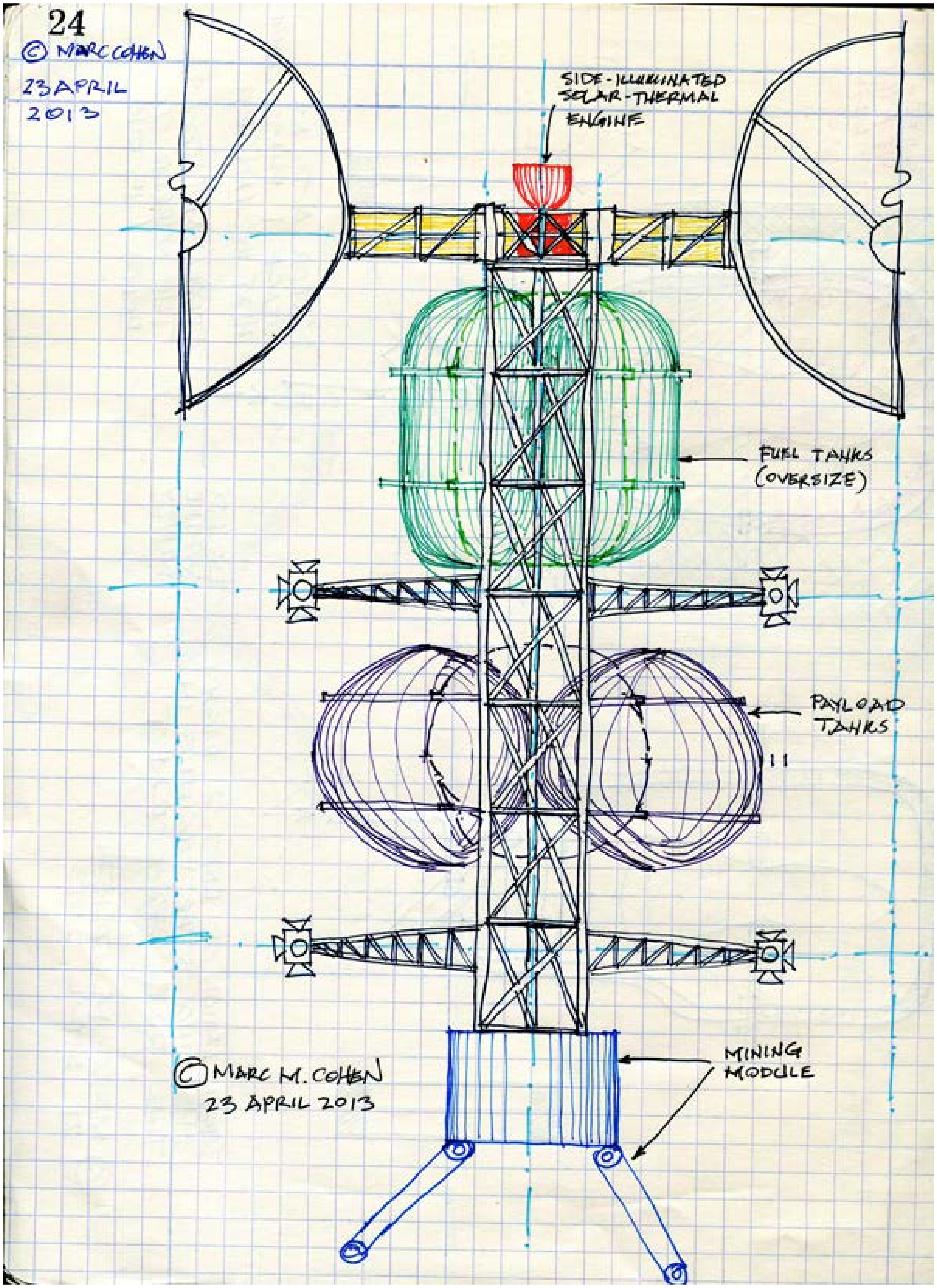


FIGURE 5.1. RAP Spacecraft Sketch: Truss, Engine, Alpha Joint Configuration.

TABLE 5.1 lays out our taxonomy of this knowledge, however inadequate it may be at this time. There are four intersections: known knowns, known unknowns, unknown knowns, and unknown unknowns. Although this matrix may appear to be a detour into a netherworld of Rumsfeldian jargon, in fact it displays some important points of design logic:

- The known knowns derive from the familiar, deterministic world of physics and engineering, and catalog what progress we make toward building the RAP spacecraft as a physical artifact with dynamic operating capabilities.
- The known unknowns are the things we need to learn or discover along the developmental pathway that we can predict, including the concerns about whether we are being too optimistic about system performance and the probability of technical and operational success.
- The unknown knowns pertain primarily to what we do not know about the space environment in which RAP will operate; we know that we must go out and discover these facts.
- Finally, the unknown unknowns are those factors such as synergetic system events or space environment conditions that we cannot predict – that we cannot seek, but will discover only by serendipity. With regard to synergy, Buckminster Fuller coined the term “synergy” to describe a system in which the actions, characteristics, or performance of the whole are unpredictable by the parts or by the sum of the parts.

The point of this discussion is not to try to package our knowledge into cute little squares, but to show how RAP will embark on a series of exploration missions of vast scope and immense challenge.

5.2 RAP Configuration Concept

The design parameters of the RAP spacecraft derive from a matrix of architectural system options, shown in TABLE 5.2. This matrix consists of two axes, the horizontal Decision Axis and the vertical Relationship Axis. The Decision Axis consists of two columns: Architectural and Technology. The vertical Relationship axis consists of three rows: Structural-related, Functionality-related, and Mission Design-related. The sequence of these rows are in the opposite order from which one might expect from reading the RAP Phase 1 Final Report. This order is reversed to examine the spacecraft design and technology assumptions and decisions first before tracing the thread back to functionality and mission design.

The purpose of this simple but far-reaching analytical matrix is to provide a kind of system-level traceability for why the RAP team made our design decisions. The six intersections or squares within TABLE 5.2 indicate the issues – or at least the types of issues – we considered together as sets of interacting concerns.

TABLE 5.1. Asteroid Mining: What we know and don't know		
	KNOWN	**UNKNOWN**
KNOWN	***Known Knowns (Standard Aerospace & Mining Design Engineering).*** • Orbital mechanics, • Low Energy trajectory design, • Ion engines, • Spacecraft structures, • Power systems, • Thermal control systems, robotics, • Terrestrial mining technologies, • Planetary drilling and abrading technologies,	***Known Unknowns (Design Engineering)*** • High-energy trajectory designs for "go-anywhere, go anytime" missions? • Designs exist for Solar Thermal Propulsion (STP) Engines, but can they be efficient and reliable? • STP using water fuel (the "Space Steam Engine")? • Can a mining system be made to operate largely autonomously? • What crew supervision would be optimal?
UNKNOWN	***Unknown Knowns (Astrodynamics and Planetary Science)*** • M-type steroids contain a wealth of metals, but where are they? • In what concentrations do these metals occur? • What are the best solutions to matching synodic periods?	***Unknown Unknowns*** • Are there undiscovered/ hitherto unknown material resources in the asteroids that could have commercial value? • Are there planetary protection issues associated with mining asteroids that at the present are considered devoid of life?

5.2.1 Structure-Related Architectural Decisions

TABLE 5.2 summarizes the RAP structural decision options. The primary structure consists of a longitudinal truss with a cross-Tee mounting the alpha joints for the solar concentrators. This large-scale truss technology has been well tested in space on the ISS. The RAP truss will be triangular instead of octagonal, and smaller in cross-section and in length than those on the ISS, but many of the structural and utility connection technologies (e.g. power, thermal) can derive from this legacy.

The main longitudinal truss carries the main solar-thermal engine at its aft end, with the tanks for H_2O, both as propellant and as payload. The propellant tanks mount aft, close to the STP engine that will use the propellant. The payload tanks mount forward, close to the mining equipment that will extract and process the water from the asteroids. The triangular surfaces of the truss provide the mounting areas for all the spacecraft subsystems such as avionics boxes, GN&C, antennae and communications gear, and video equipment -- except the Grid of Light or other method of distributing the concentrated solar flux to the points of use. The main truss also carries the high-pressure air tanks for pneumatic prospecting and mining operations forward, along with the avionics. The reaction control system units on booms or mass, along with their RCS propellant tankage, are mounted forward and aft on the main truss.

This upper truss carries a cross-T at its aft end, in which the solar-thermal engine will be installed. This upper T truss deploys the solar concentrators in close proximity to the STP engine, the first priority point of use. The "T" section incorporates a "bilateral" swivel joint – or pair of joints, serving as an *alpha* joint for the solar concentrators to track the sun, primarily while the spacecraft is in flight.

TABLE 5.2. Architectural System Options for the Mining/Robotics Trade and Analysis

	Architectural Decision	Technology Decision
Structure-Related	• Types and sizes of trusses • Deployable or assembled in-space • Load paths • "Industrial" accommodation of mining modules • Mounting Solar-Thermal Arrays • Structure & Spacecraft Dynamics	• Architectural Systems • Materials, mechanisms, fasteners, connectors, welding, and bonding. • Processing Systems • Waste Heat rejection • Integration of Robotics
Functionality-Related	• Integration of Structures & Engines • Mining & Processing • Communications • Anchoring to the Asteroid • Movement around the Asteroid • Stow refined ores	• Utility Systems • Data System Architecture • Control Systems • Surface mobility? Drilling anti-torque
Mission Design-Related	• Launch packaging • Staged deployment of trusses and Engines, • Docking to L-1 platform to fuel, landing, and docking at Asteroid, Return to L-1 to off-load samples and concentrates.	• Flight Dynamics of Power Systems • RCS for zero-gravity fine maneuvering • Propellant & Pneumatic Gas tanks.

5.2.2. Structure-Related Technology Decisions

The selection of technology for the RAP structure will focus upon ways to reduce cost, reduce mass, to enhance reliability, and to help manage risk. These foci mean that the study will consider existing and future structural technologies, including materials, fabrication techniques, deployment mechanisms, rotational or swivel mechanisms, connectors, fasteners, and attachment techniques including welding and bonding. The design space for the truss should involve a multi-variable analysis and optimization between geometry and technology choices.

These truss structures will need to meet a range of requirements to be defined for supporting:

- Propellant and pneumatic gas tanks,
- Large rotating mechanisms in the form of alpha and beta joints for the solar concentrators, and
- Attachments for:
    - Power cables,
    - Data cables,

- Pneumatic tubing,
- Cold gas tubing, and
- Thermal loops.

Another area of structure-related technology consideration involves the robotic system themselves (e.g. Spiders and ARProbes) that the RAP spacecraft will deploy and that will also have a role in deploying and operating the spacecraft. This application of robotics may influence the best ways to assemble, deploy, erect, extend, or unfold the spacecraft structures. Compactness of structures poses great advantages for launch packaging, with the caveat that it should not impair the ability to construct the most capable spacecraft. The payoff is that instead of trying to launch a 40m long, rigid, and fully integrated truss, some automated or robotic assembly, deployment, extension, or unfolding of the structural elements may greatly help to reduce the bulk and volume of this launch package.

5.2.3. Functionality-Related Architectural Decisions

The RAP spacecraft will need to perform a wide variety of mission functions and perform them under a variety of potential conditions in deep space. Several integration matrices will be needed to define this functionality. These matrices include the data system (including programming for robotics) that in many ways provides the central operating principle for the RAP mission. Structures provide another matrix, as discussed above, but with the further proposition that they afford a consistent and complete system of physical integration for attaching, connecting, mounting, and operating the diverse functions.

As proven during the rescue of Apollo 13, ***power is everything,*** and so it comprises a further integration matrix, one that includes power conditioning, distribution, load shedding, and all the other features of a small industrial utility plant. The solar thermal power generation options play into this power integration matrix, largely because of the several options for STP heat, fiber optic waveguide transmission, and solar dynamic generation with a Stirling-cycle engine raise a host of potential choices. Finally the thermal process fluid system provides a kind of integration utility: thermal energy source for the solar-thermal engine(s), cooling, and possibly as a heat transfer medium for the prospecting and mining extraction and refining processes.

Communications within the RAP spacecraft and back to the humans in the loop – wherever they are – are part of the data system. This data system carries all capacity for teleoperations, uploading programming, and downloading engineering and science results, either in the form or raw bits and bytes, as reduced data, or as usable findings. Comm includes the antennae that must be capable of pointing toward Earth or to a relay satellite at any time during the RAP's mission. These antennae connect to the structural, data, and power matrices; a key study topic should address the range of motion required by these antennas.

Once the RAP spacecraft arrives "on site" at a target asteroid, it must go into orbit around that body or in some other fashion station keep with it. Subsequently the RAP spacecraft will have to perform a variety of maneuvers, such as:

- Inspecting the surface and subsurface over all areas of the asteroids for promising mineral deposits,
- Aligning with the pole of the asteroid and then matching rotation rates,

- Touching down or landing on the asteroid, most probably at the pole,
- Bringing the prospecting modules in contact with the candidate deposit area,
- Attaching the spacecraft to the surface of the asteroid, and once attached, and
- Rotating and swiveling the spacecraft to track the sun and point the antennae.

As the prospecting modules perform their work to abrade, drill, and excavate the asteroid material for further analysis and test processing, the prospecting/mining system consumes high thermal energy either via hot fluid loop or high voltage/high current electrical power. This high-energy transmission system passes from the diverter mirrors at the top of the upper T-truss and then down the main truss. From this junction, it redirects energy from where it would otherwise power the solar-thermal engine and instead to the prospecting modules at the "forward" end of the truss, now anchored to the asteroid. The system architecture must integrate these functions to deliver the utility service where it is needed with a minimum of interference and duplication that does not enhance systemic redundancy and reliability.

Finally, the RAP architecture provides stowage for mineral samples collected during prospecting activities and ultimately cargo stowage for commercially valuable ores and concentrates to return to the Earth-Moon system for refinement and tertiary processing. The volume and mass of prospecting samples are likely to exceed a few kilograms. However, the mass of commercial quantities will be measurable in mTons and the volume in tens of cubic meters to carry either ore concentrates or refined products.

5.2.4. Functionality-Related Technology Decisions

The major functionality-related technology issues concern the design and capabilities of the utility subsystems at the points where they deliver their service. There are many aspects to these issues. For example, the data systems serve to operate both the spacecraft and the prospecting/engineering payload(s). That duality may seem obvious, but the service requirements, feedback, and "sampling" rates may prove to be very different. Embedding an integrated vehicle health monitoring system (IVHM) or a real-time automated diagnostic system (RAD) in the data management operating system may be vital to achieving mission success. In a later phase, we will perform an analysis to determine if those prospecting, mining and processing functions can be integrated into the primary spacecraft data system of if that will require an independent data system.

Another function that ties into many other subsystems is the control system for flight, guidance, and navigation. Ample technology exists off the shelf to control a spacecraft on journeys of many AU. However, the job of maneuvering around a very small object in micro-G, landing on it, attaching to it securely, operating high-torque drilling equipment while maintaining stability, then detaching from the surface and moving to another drilling site are a new challenge. The technologies selected for these more complex, and ideally autonomous operations, will require careful functional, operational, and physical integration with the spacecraft architecture and the overall mission design.

5.2.5. Mission-Related Architectural Decisions

The principal set of mission-related architectural decisions address launch packaging, staging – including firing of engines, and the deployment of trusses, prospecting modules, and other equipment. Referring back to the ConOps in the Mission Design chapter of the RAP Phase 1 Final Report, an example of the relationship between staging and structural deployment concerns the question of "self-ferrying" from LEO to an EML. If the RAP spacecraft relies upon a commercial carrier to deliver it from LEO to the EML platform (as assumed in the proposal), then it keeps its trusses, solar arrays, and radiators all stowed until reaching that EML. The RAP spacecraft would then deploy those subsystems when docked to the EML platform, or astronauts working at that platform might assemble those systems into their operational configuration. However, if it is more advantageous to self-ferry, spiraling out from LEO in the manner of an orbital transfer vehicle (but faster than the ones powered by solar-electric), then the RAP spacecraft must deploy and activate those subsystems and their related elements in LEO, arrive at the platform, and maneuver itself to docking with everything deployed, under its own power. The scenario in which docking in space is not required occurs under the conditions that the RAP spacecraft is capable of fully self-deploying in LEO and there is no EML platform to which to dock. However, without an EML platform, the problem remains of where the RAP spacecraft will return to discharge its cargo.

5.2.6. Mission-Related Technology Decisions

The mission-related technology decisions are the most sensitive to the time or rather the era of spaceflight and exploration in which we consider them. Many, if not most, space technologies advance rapidly, so rapidly that mission and spacecraft and mission designers often feel tempted to wait until the "next best thing" comes along with better performance, lower cost, or both. However, for RAP, this tendency toward procrastination does not apply; in fact it cannot apply. The reason is that our central enabling technology, solar thermal propulsion has not been making the great advances for which we would hope, for at least the past decade. Instead of the more familiar "wait, wait, wait and then hurry-up" decision, it poses an existential dilemma. If the RAP team does not take up the gauntlet of the STP challenge, this promising technology, which for our mission is essential, will languish for decades more. Therefore, the take-away message from RAP Phase 1 is the necessity to advocate for solar thermal propulsion and to motivate both the public and private sectors to invest in it as the best choice for a true spacecraft: a spacecraft that operates only in space and depends on the resources it can find there.

6. Conclusion

Nearly all of the precursors are specifically designed to serve the orbital boost/reboost market. A few are more generic in the sense that the patent does not show them installed in a spacecraft configuration. Two engine designs (Shoji and the three BWX patents) show the propulsive flow making a 90° turn, from which it may be possible to infer that the inventors anticipated a spacecraft configuration in which the insolation would reach the engine at right angles to the velocity vector. However, they do not show any structure around the engine that might support that conjecture, or the logical extension that the solar concentrators may be pointable to track the sun.

6.1 Solar-Inertial Flight Mode

All the STP engines that show an indication of where their solar concentrators are located do not show any articulation or rotation mechanism. From this apparent absence of an "alpha joint" (α–j) in which solar collectors rotate about an axis to track the sun, we can infer that the design assumes the spacecraft will always fly solar-inertial, with immobile solar concentrators or arrays that are mounted rigidly on the spacecraft. To keep the concentrators pointing toward the sun, the spacecraft must maintain its flight attitude always in more or less the same position.

The assumption of flying solar-inertial and limiting the application of the STP engine to boost and reboost constitutes a kind of illogical disjunct union or negative chicken and egg construct. If the spacecraft design baselines solar-inertial mode, then it cannot do anything "useful" beyond going up and down. It is not suited for a mission that requires maneuvering, change of plane, cislunar or interplanetary injection. Because RAP commenced from the requirements to fly to an asteroid, rendezvous with it, maneuver, and land on it, the requirement for a versatile propulsion system has always been integral to the whole concept and mission.

6.2 Precursor Insolation and Heat Gain

Except for Lee's earliest STP patent in 1982, all the known designs employ an internal cavity for the heating chamber. The 1982 patent shows an outer structural spherical shell around the inner spherical heating chamber. The outer shell mounts an array of lenses that focus sunlight on the heating chamber, presumably from almost any direction. This design seems to assume that it would be possible to keep adding insolation and heat to the engine, without it ever being rejected anywhere, except through the propulsive thrust. The inner spherical "boiler" of this STP engine would be make of copper, so obviously it is operating below the 1358K melting point of copper.

The other designs for STP engines all specify higher temperature metals than copper. Rhenium and rhenium-coated graphite, plus carbide- or boron-nitride- coated materials, including coated rhenium are potential candidates, but the material science literature cited in the STP publications appears rather murky. Kennedy and Palmer discuss these material options in detail. What emerges from the material science discussions is the challenge of finding or creating a material that can handle the high temperatures of 2500K or preferably greater, generated by steam that is very corrosive and erosive of nearly all candidate materials.

Since all the other precursor designs and the Shooting Star require an inner heating cavity, the problem that arises is how to distribute sunlight evenly into a deep cavity. The Shooting Star sunlight flux extractor appears to be the only serious attempt to address this problem. None of the patents reviewed show a solution to this design problem.

6.3 Installing a Solar Thermal Engine

For the purpose of RAP, the STP engine would ideally exist and be available as a commercial off the shelf (COTS) commodity item -- just like other mass production engines. For RAP, the job would then be how best to install the engine to accomplish the asteroid mining mission. For all the prospective engine designs above, there are several common features that this installation would accommodate. Designing, building, and testing an STP engine design is far outside the scope of the NIAC contract or its budget in Phase 1 or Phase 2. However, the team felt that we could do very valuable work investigating how to install and operate an STP engine, once it becomes available.

6.3.1 Insolation

The universal property is insolation from the sun to concentrator mirrors or lenses (or both), depending on the particulars of each design. The concentrator must either use its own optical geometry to direct the concentrated light to the heating chamber, or employ a system of mirrors and/or lenses to direct the light to the heating chamber. Each time the STP engine installation reflects the light with a mirror, there is substantial loss of energy and corresponding heating of the mirror assembly. According to Wong and Macosko, there is no material that can retain a reflective surface above about 1500K, so it would become necessary to cool the mirror assembly and discard that waste heat.

6.3.2 Propellant

All candidate STP engines require a propellant fluid or gas to provide the mass, that when super-heated and under pressure, the expansion of the propellant provides the thrust out the engine nozzle. At first glance, almost any gas or fluid that can be super-heated to sufficient temperature can serve as the propellant. However, there are profound differences among propellants of differing composition. Whereas xenon generally serves as the fuel of choice for solar-electric engines, the most commonly recommended propellant for STP is H_2, usually as in the cryogenic liquid phase (LH2) but sometimes as gaseous hydrogen (GH2), simply compressed in a tank.[11] In selecting a propellant, there is a universe of considerations:

- Availability on Earth and in Space
- Difficulty in preparing the fuel
- Energy content
- Cost
- Stability
- Toxicity
- Density (Volume required to afford a given mass)
- Complexity
- Power to maintain it in a usable state
- Retention against off gassing.

A detailed analysis is beyond the scope of the current discussion. The RAP team chose H_2O as the propellant of choice because of its:

[11] It is important to distinguish this compressed but not cooled GH2 from the cryogenically cooled LH2 that fueled the Space Shuttle.

- Ready availability on Earth
- Relative abundance in space (lunar and Mars poles, potentially under the Mars surface, and carbonaceous chondrite asteroids)
- Simplicity in preparation
- Low cost
- Non-toxicity
- High Density
- No need to cool this propellant; heating it to keep it liquid is far simpler than cooling to keep it from boiling.

6.3.3 Thermal Management

Because the STP engine will operate at extremely high temperatures, thermal management becomes a central issue. The main strategies among the precursors are:

- Direct radiative loss,
- Thermal storage for intermittent (apogee, perigee) propulsive "burns,"
- Active cooling and heat rejection through a radiator, and
- Thermal Insulation.

The Shooting Star engine exemplifies the direct gain radiative approach, but it also allows omni-directional radiative heat loss. Basically, the mounting of the engine in the center of the wagon wheel structure affords omnidirectional radiative heat loss. The Rhenium foam structure of the "wagon wheel" mounting provides conductive isolation during testing, although absent the atmosphere, there would be no convective losses.

References

Brockmeyer, J. W.; Fortini, A. J.; Williams, B. E.; Tuffias, R. H. (1998). High-Efficiency Open-Cell Foam Heat Exchangers for Actively Cooled Propulsion Components (AIAA-98-3441), *34th AIAA Joint Propulsion Conference. DOI: 10.2514/6.1998-3441.*

Calabro, M. (2003). Solar Thermal Upper Stages, *International Conference on Recent Advances in Space Technologies, 2003. RAST '03*. 20-22 Nov. 2003.

Fiot, D.; Corberand, F.; Bravais, P.; Bernasconi, M. C.; Parkinson, R. C.; Durante, N.; Le-couls, O; et al. (2002, 5 June). Solar Thermal Upper Stage Technologies for Future Launcher Generation Program STOTS: Executive Summary, *ESTEC Contract 14.022/99/NL/PA.* Les Mureaux France: EADS Launch Vehicles.

Kennedy F. G.; Palmer, P. L. (2002). Preliminary Design of a Micro-Scale Solar Thermal Propulsion System, (AIAA 2002-3928). *38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 7-10 July 2002, Indianapolis, IN. DOI: 10.2514/6.2002-3928.*

Kontragool, Bancha; Wongwises, Somchai (2003, July). A review of solar-powered Stirling engines and low temperature differential Stirling engines, *Renewable and Sustainable Energy Reviews, 7 (2003) doi: 10.1016/S1364-0321 (02) 00053-9.* pp. 131-154.

Lacy, J. M.; Carmack, W. J.; Miller, B. G.; Raines, R. D. (2001). Results of Risk Reduction Activities for the SOTV Space Experiment Solar Engine, AIAA-2001-3989. Reston VA: AIAA.

Leigh, Larry M.; Tinker, Michael L. (2003, January). Dynamic Characterization of an Inflatable Concentrator for Solar Thermal Propulsion, *In, J. Spacecraft and Rockets, Vol. 40, No. 1 (2003), doi: 10.2514/2.3935.* pp. 24-27.

MSNW LLC (2012, 2013). Space Propulsion: The Electrodeless Lorentz Force Thruster, http://msnwllc.com/space-propulsion, last checked and retrieved, 3 July 2013.

Nakamura, Takashi; Krech, Robert H.; McClanahan, James A.; Shoji, James M.; Partch, Russell; Quinn, Skylar (2005, 10-13 July). Solar Thermal Propulsion for Small Spacecraft - Engineering System Development and Evaluation, AIAA-2005-3923. *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Tucson AZ. Chapter DOI: 10.2514/6.2005-3923.* Reston VA: AIAA.

Partch, R.; Frye, P. (1999, 20-24 June). Solar orbit transfer vehicle space experiment conceptual design, AIAA-99-2476. *Chapter DOI: 10.2514/6.1999-2476,* Reston VA: AIAA.

Scharfe, David; Young, Marcus (2010 March 24). A Study of Solar Thermal Propulsion System Enhancement via Thermal Storage and Thermal-electric Conversion, AFRL-RZ-ED-TP-2010-110. Edwards AFB California: Air Force Research Laboratory.

Smalley, Kurt B.; Tinker, Michael, L.; Taylor, Scott W. (2003, January-February). Structural Modeling of a Five-Meter Thin-film Inflatable Antenna/Concentrator, *In, J. Spacecraft and Rockets, Vol. 40, No. 1 (2003), doi: 10.2514/2.3935.* pp. 27-30.

Slade, Kara N.; Tinker, Michael L.; Lassiter, John O.; Engberg, Robert (2001, May). Dynamics of an Inflatable Structure in Vacuum and Ambient Conditions, *In, AIAA Journal, Vol. 39, No.5.* pp. 894-901.

Valentian, D. (2002, 16 Sept). Solar Thermal upper stages technologies (STOTS) Executive Summary. *ESTEC Contract N° 14023/99/NL/PA.* Foret de Vernon, France: SNECMA Moteurs.

Sippel, Martin; Kauffman, Jens (2008). Solar Thermal Propulsion for Upper Stages. In, Bruno Claudio; Accettura, Antonio G. [Eds], *Advanced Propulsion Systems and Technologies Today to 2020, AIAA Progress in Astronautics and Aeronautics, vol. 223.* Reston VA: AIAA. pp. 201-222.

Soules, Jack A.; Buchele, Donald R.; Castle, Charles H.; Macosko, Robert P. (1997). Design and Fabrication of a Dielectric Total Internal Reflecting Solar Concentrator and Associated Flux Extractor for Extreme High Temperature (2500K) Applications. *SPIE Vol. 3139* • 0277-786X/97 (SPIE-ILL-102408599). *Nonimaging optics, maximum efficiency light transfer IV: 27-28. doi:10.1117/12.279219.*

Stewart, Jesse F.; Martin, James A. (1995, July). Dual Fuel Solar Thermal Propulsion for LEO to GEO Transfer: Ideal Rocket Analysis, *AIAA 95-2840.*

Stewart, Jesse F.; Martin, James A. (1996). Dual Fuel Solar Thermal Stage: Ideal Analysis, *In, AIAA J. Spacecraft, Vol. 33, No. 5: Engineering Notes.* pp. 752-754.

Tinker, Michael L. (1998, April). Passively adaptive inflatable structure for the Shooting Star Experiment, (AIAA-98-1986). *39th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference and Exhibit*, NASA, Marshall Space Flight Center, Huntsville, AL. *In, Chapter DOI: 10.2514/6.1998-1986.*

Tucker, Stephen; Salvail, Pat (2002). Solar Thermal Engine Testing, In, El-Genk, Mohamed S. (Ed.) *Space Technology and Applications International Forum—STAIF 2002.* American Institute of Physics Conf. Proc. V.608. Albuquerque NM, 3-6 FEB 2002.

Westerman, K.; Miles, B. (1998). Testing of a Receiver-Absorber-Converter (RAC) for the Integrated Solar Upper Stage (ISUS) Program, Space Technology and Applications International Forum, American Institute of Physics, pp. 375-380.

Wong Wayne A.; Macosko, Robert P. (1999, July). Refractive Secondary Concentrators for Solar Thermal Applications, (NASA TM—1999—209379). Cleveland OH: NASA Glenn

Wassom, Steven R. (2001) Revolutionary Propulsion Concepts for Small Satellites, AIAA-2001-xxxx. 15th Annual AIAA/USU Conference on Small Satellites, Reston VA: AIAA.

http://kiss.caltech.edu/study/asteroid/asteroid_final_report.pdf

Butler, J. (2012). "Platinum 2012", Copyright Johnson Matthey PLC, retrieved on January 10, 2013, http://www.platinum.matthey.com/publications/pgm-market-reviews/archive/platinum-2012/.

Haskins, Cecilia, Ed (2010). Systems Engineering Handbook: A Guide for System Life Cycle Processes and Activities, INCOSE TP-2003-02-03.2. San Diego CA: International Council on System Engineering.

International Aluminum Institute (2013). World Aluminum Data, http://www.world-aluminium.org/statistics/alumina-production/#data, retrieved 5 JAN 2013.

Lewis, John S. (1997). Mining the Sky, Boston MA: Addison-Wesley Publishing Co.

Long, K.R.; Van Gosen, B.S.; Foley, N.K., and Cordier, D., (2010). "The principal rare earth elements deposits of the United States—A summary of domestic deposits and a global perspective: U.S. Geological Survey Scientific Investigations Report – 2010" Available at http://pubs.usgs.gov/sir/2010/5220/.

Mariano, A.N.; Cox, C.A.; and Hedrick, J.B.; (2010). "Economic Evaluation of REE and Y Mineral Deposits," Presentation at the 2010 Annual Meeting of the Society for Mining, Metallurgy & Exploration [SME], Phoenix, Arizona, available at http://www.smenet.org/rareEarthsProject/SME_2010_Mariano.pdf, 33 p.

NASA (2011). Apollo 17, http://www.nasa.gov/mission_pages/apollo/missions/apollo17.html

Norton, O. Richard (2002). The Cambridge Encyclopedia of Meteorites. Cambridge: Cambridge University Press. pp. 121–124. ISBN 0-521-62143-7.

Walters, A.; Lusty, P.; Chetwyn, C.; and Hill, A., (2010). "Rare Earth elements: Commodity profile", British Geological Survey. Available online at www.mineralsuk.com.

Zacny, Kris; Chu, Phil; Avanesyan, Arshak; Osborne, Lars; Paulsen, Gale; Craft, Jack; Oryshchyn, Lara A.; Sanders, Gerald B., Mobile In-Situ Water Extractor for Mars, Moon, and Asteroids In Situ Resource Utilization, AIAA Space 2012, 11-13 September 2012, Pasadena, CA