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DEC 13 1999

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NEP for a Kuiper Belt Object Rendezvous Mission

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Abstract. Kuiper Belt Objects (KBOs) are a recently-discovered set of solar system bodies which lie at about the orbit of Pluto (40 AU) out to about 100 astronomical units (AU). There are estimated to be about 100,000 KBOs with a diameter greater than 100 km. KBOs are postulated to be composed of the pristine material which formed our solar system and may even have organic materials in them. A detailed study of KBO size, orbit distribution, structure, and surface composition could shed light on the origins of the solar system and perhaps even on the origin of life in our solar system. A rendezvous mission including a lander would be needed to perform chemical analysis of the surface and sub-surface composition of KBOs. These requirements set the size of the science probe at around a ton. Mission analyses show that a fission-powered system with an electric thruster could rendezvous at 40 AU in about 13.0 years with a total ΔV of 46 km/s. It would deliver a 1000-kg science payload while providing ample onboard power for relaying data back to earth. The launch mass of the entire system (power, thrusters, propellant, navigation, communication, structure, science payload, etc.) would be 7984 kg if it were placed into an earth-escape trajectory ($C=0$). Alternatively, the system could be placed into a 700-km earth orbit with more propellant, yielding a total mass in LEO of 8618 kg, and then spiral out of earth orbit to arrive at the KBO in 14.3 years. To achieve this performance, a fission power system with 100 kW of electrical power and a total mass (reactor, shield, conversion, and radiator) of about 2350 kg. Three possible configurations are proposed: (1) a UZrH-fueled, NaK-cooled reactor with a steam Rankine conversion system, (2) a UN-fueled gas-cooled reactor with a recuperated Brayton conversion system, and (3) a UN-fueled heatpipe-cooled reactor with a recuperated Brayton conversion system. (Boiling and condensation in the Rankine system is a technical risk at present.) All three of these systems have the potential to meet the weight requirement for the trip and to be built in the near term.

INTRODUCTION

Kuiper Belt Objects (KBOs) are a recently-discovered set of solar system bodies which lie at about the orbit of Pluto (40 AU) out to about 100 astronomical units (AU). The Harvard Minor Planet Center reports that as of September 6, 1999, 191 KBOs ("transneptunian objects") have already been discovered (<http://cfa-www.harvard.edu/cfa/ps/lists/TNOs.html>). These are typically 100 to 300 km in diameter, and there are estimated to be approximately 100,000 KBOs 100 to 1000 km in diameter (Malhotra, 1999, Jewitt, 1999), and possibly as many as 10,000,000,000 KBOs with a diameter greater than 1 km. Clearly, this is a significant fraction of the solar system which has not yet been explored. KBOs are postulated to be composed of the pristine material which formed our solar system and may even have organic materials in them. A detailed study of KBO size, orbit distribution, structure, and surface composition could shed light on the origins of the solar system (Malhotra, 1999) and perhaps even on the origin of life in our solar system. A rendezvous mission including a lander would be needed to perform chemical analysis of the surface composition of KBOs. Since cosmic rays and solar wind have altered the chemistry of the outer 10 m of the KBOs, the lander also would need to have a drill to obtain a deep core sample. These requirements set the size of the science probe at around a ton. To deliver a one-ton payload to rendezvous with a KBO in 10-15 years requires acceleration to and from high velocities (about 25 km/s). A fission reactor with electric propulsion can deliver such performance in the near future without the excessive weight penalty associated with chemical or solar propulsion.

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SYSTEM ANALYSIS

In order to rendezvous with a KBO at a nominal distance of 40 AU in a minimum time, one must accelerate to a substantial velocity, perhaps coast for awhile, and then decelerate to rendezvous speed. The total velocity capability of the spacecraft (ΔV) is the sum of the acceleration, deceleration, and rendezvous adjustment velocities. Typically, the total ΔV for a rendezvous in 12 years is about 50 km/s.

The rocket equation (Sutton, 1986) dictates that the total mass of fuel needed increases exponentially with the ratio of the desired velocity increase (ΔV) over the specific impulse of the fuel (I_{sp}) times the gravitational acceleration on earth (g):

$$m_{fuel} = m_{nonfuel} (\exp(\Delta V / (g I_{sp}) - 1), \quad (1)$$

where m_{fuel} is the mass of the propellant, $m_{nonfuel}$ is the mass of everything that is not propellant (scientific payload, communications, guidance, rocket engines, power supply, etc.). One of the best current chemical propellants is liquid oxygen/liquid hydrogen which has a specific impulse of 460 seconds. To achieve a total ΔV of 50 km/s with a payload, power system, engines, and other inert masses ("non-fuel mass") of 5000 kg would require 324,000 tonnes of fuel. Clearly, chemical propellants cannot do the job. But NASA has recently developed very reliable electric propulsion units with a demonstrated specific impulse of 3300 s (Polk, 1998). An even higher I_{sp} can be achieved either by using a larger voltage on the grids or by using a lighter gas. NASA is also developing a VASIMR electric thruster which is projected to achieve up to 20,000 s I_{sp} (Chang Diaz, 1998). To reach a total ΔV of 50 km/s with a specific impulse of 10,000 seconds and 5000 kg of non-fuel mass requires only 3.3 tonnes of propellant. This is much more reasonable.

Electric propulsion requires very large amounts of electrical energy. Nuclear fission, in the form of a small reactor, is the only credible near-term means for obtaining this energy, especially in deep space. Solar energy is not practical because the spacecraft needs to decelerate by about 25 km/s at a distance of about 35 au where the solar intensity is 1000 times dimmer than at earth. A fission reactor and power conversion system for space (excluding the radiator) would be about the size of an automobile and similar in power to research reactors found at many universities.

A system analysis was performed based on the following assumptions: (1) the design point goal is a rendezvous with a KBO at 40 AU, (2) the reactor and propulsion system operates at steady power and accelerates at a fixed thrust for a specified duration (typically many years), (3) the spacecraft then coasts for awhile, (4) the reactor switches to low-power mode during the coast period for spacecraft maintenance, (5) the spacecraft then decelerates at full power and fixed thrust to achieve rendezvous velocity with the KBO, and (6) some propellant is reserved for maneuvering and rendezvousing with additional KBOs. In deference to historical rockets, the propulsion duration will be called a "burn," even though the propellant is not actually burned.

There are two general approaches to the mission scenario. The first approach is to launch the spacecraft into earth orbit and use the electric propulsion system to propel the spacecraft out of earth orbit and then on to the Kuiper belt. The second approach is to launch the spacecraft into an earth-escape trajectory (called a $C3=0$ trajectory) and then use the electric propulsion system. The advantage of an earth-orbit launch is that it does not require as large a launcher for a given spacecraft weight because the highly-efficient electric propulsion system will be used to get the spacecraft out of earth's gravity well. A disadvantage is that it requires a longer total burn time for the reactor and electric thruster. It also requires the spacecraft to survive a relatively slow passage through the Van Allen radiation belt. Overcoming these difficulties is one reason to use a $C3=0$ launch. In addition, the safety analysis for a $C3=0$ launch might be easier since reactor operation would not begin until an earth-escape trajectory is achieved. But the penalty for a $C3=0$ launch is that typically the mass that a given launcher can deliver to earth escape is about three times less mass than it can deliver to low-earth orbit.

There are many approaches that can be used in comparing options. One could assume fixed spacecraft mass, fixed spacecraft power, fixed launcher type or launcher cost, fixed travel time, etc. One could minimize travel time, launch mass, total mission cost, etc. by varying specific impulse, burn duration, coast duration, etc. We will not

present a comprehensive span of possibilities in this paper. Rather, we will present some representative cases that show the potential feasibility of this mission.

A simple orbital mechanics code was developed using the fundamental laws of gravity and acceleration along with the algebraic and Runge-Kutta solution packets in the Mathcad (MathSoft, 1999) commercial software. Three cases were considered. For case 1, the spacecraft was assumed to be launched into a C3=0 trajectory; for case 2, the spacecraft was identical to that in case 1 except that it was launched into low earth orbit (LEO) and additional propellant and tankage to get the spacecraft to C3=0. The added tankage (small) was jettisoned after earth escape. This allowed a simple comparison of the C3=0 and LEO options. Case 3 is like Case 2 except it assumes a lower Isp during earth escape, such as might be available from a VASIMR thruster. This reduces the time needed for earth escape, but increases the mass required in LEO. For both LEO cases, the initial orbit is still high enough that the orbital decay time would be long enough to allow reactor fission products to decay back down to background levels before reentry (about 600 years).

Table 1 shows the key parameters. The "science payload" is assumed to be 1000 kg. Everything else (i.e. everything except the science payload and propellant) is called the "tug" and has a dry mass of 4000 kg. The tug consists of the reactor, the reactor shield, the power conversion system, the thermal radiator, the electric thrusters, the guidance and control system, the communication system, propellant tankage, structures, etc. The tug electric power is 100 kW. The total efficiency of the thrusters (propellant power over electrical power) is assumed to be 75%. Acceleration, coast time, and deceleration duration were adjusted so as to end up at 40 AU with the proper rendezvous velocity (4.7 km/s around the sun). It was determined that the trip time was not a strong function of coast time if the burn time could be adjusted accordingly. A coast time of 7 years yields an attractive configuration.

TABLE 1. Reference mission configurations.

Parameter	Case 1 (C3=0)	Case 2 (LEO)	Case 3 (LEO)
Power in the tug (kWe)	100	100	100
Mass of the tug (kg)	4000	4000	4000
Mass of the science payload (kg)	1000	1000	1000
Specific impulse (s)	10000	10000	10000
Efficiency of converting electrical to exhaust power (%)	75	75	75
Acceleration duration (yr)	3.79	3.79	3.79
Coast duration (yr)	7.00	7.00	7.00
Deceleration duration (yr)	2.24	2.24	2.24
Propellant mass used from C3=0 to KBO rendezvous (kg)	2984	2984	2984
Altitude of initial LEO (km)	N.A.	700	700
Isp for trip from LEO to C3=0 (s)	N.A.	10000	3000
Propellant mass from LEO to C3=0 (kg)	N.A.	634	2315
Duration from LEO to C3=0 (yr)	N.A.	1.29 yr	0.42 yr
Total spacecraft wet mass at launch (kg)	7984	8618	10299
Total mission ΔV to rendezvous (km/s)	45.9	53.4	53.4
Total burn time to arrival (yr)	6.03	7.32	6.45
Total flight time to KBO rendezvous (yr)	13.03	14.32	13.45

The total trip duration is about 13 years, which is not unreasonable for a rendezvous trip as far out as Pluto. The total amount of high-power burn time for both the reactor and thrusters is 6 years for case 1, which is a reasonable lifetime for these systems. There will need to be some low-power maintenance during the 7-year coast and the several-year science mission after arrival, so component longevity needs to be considered for this mode also.

The total mass for the C3=0 option is 7984 kg, which would require a launcher such as a Delta IV-M (under development). However, if the launch is only to LEO and the tug uses its own power to escape earth orbit, smaller launchers can be used. The mass launched to LEO for Cases 2 and 3 are 8618 kg and 10299 kg, respectively. However, the trip time and the burn time increases by 1.3 yr and 0.4 yr, respectively, so the savings might not be worth it, especially since this increases the time in the Van Allen radiation belt.

Figure 1 shows the trajectory for Case 1. The trajectory is plotted on a polar plot with the radius on a logarithmic scale. The dots in the trajectory each represent about 7 days. The trajectory from about 5 AU (Jupiter) to 40 AU is essentially straight. The code does not explicitly calculate the final maneuver to match orbits with the KBO (which consists of turning a residual 4.6 km/s velocity vector 90 degrees), so instead we allow an additional 4.6 km/s of ΔV at the end for this maneuver. Thus the velocity in the figure ends up at zero, rather than 4.6 km/s.

Figure 2 shows the total velocity (relative to the sun) vs. time. The total velocity plot shows the three thrust regimes (accelerate, coast, and decelerate). After earth escape, the spacecraft has a total velocity of about 30 km/s, which is the velocity of the earth around the sun. Then, as the spacecraft slowly thrusts along its velocity vector, it puts itself into an elliptical orbit in which the sun's gravity slows it down as it travels outward, even though it is still thrusting. The spacecraft is gaining energy at this time, but it is gaining potential energy rather than kinetic energy (like climbing a mountain). Eventually the continuous thrust starts to overcome the sun's pull and the spacecraft begins to pick up speed. After 4 years, it still does not have as much total velocity as it started out with, but that velocity vector is now pointed almost directly away from the sun. At this point the spacecraft enters the coast phase for a period of 5 years. Finally, 9 years after launch, it begins deceleration so that it can match velocity with the KBO.

Figure 2 also shows the spacecraft distance from the sun vs. time and along with the orbital distances of the outer planets. (Pluto orbits between 30 and 50 AU.) This spacecraft, if directed in the right direction, could fly by Jupiter in 3 years, Saturn in 4 years, Uranus in 7 years, Neptune in 9 years, or Pluto in 9 to 10 years after launch (but not all of them in one trip). This would be a large improvement over present capabilities.

FIGURE 1. Trajectory in log-radius coordinates.

FIGURE 2. Distance from sun and total velocity vs. time.

TUG DESIGNS

To achieve the performance described, the total tug dry mass must be about 4000 kg with an electric power of 100 kW. The thrusters and power conditioners are estimated to weigh about 500 kg, and the tankage for 3000 kg of propellant is estimated to weigh about 150 kg. Communications, guidance, structures, etc. might be an additional 1000 kg. That leaves a total of about 2350 kg for the power source which consists of the reactor, radiation shield, conversion system, and thermal radiator. The specific mass of the power system is thus about 23 kg/kWe. This is higher in power and lower in specific mass (kg/kWe) than any space nuclear power system that has been fielded in the past, but reasonable for estimated masses of near-term space-reactor systems.

The choice of technologies for the reactor system is fairly wide, although there is no off-the-shelf space-reactor system presently available. The SNAP program flew one space reactor in 1965 and built six other working reactors

during the program. All of these used UZrH fuel with a NaK coolant (Anderson, 1983). The subsequent SP-100 program designed a reactor which used UN fuel with a Li coolant (Mondt, 1994; Mondt, 1995; El-Genk, 1994). The Russian Rorsat reactors (about 30 flown in space) used UMo fuel (Angelo, 1985). The Russian Topaz II reactor used UO₂ fuel and UZrH moderator. There are numerous proposed designs in the literature (Angelo, 1985; El-Genk, 1994).

Systems for conversion to electricity (for the electric thrusters) include thermoelectric (TE), thermionic, Rankine, Stirling, and recuperated Brayton cycles. Thermoelectrics have been the most used in space and typically result in only 5% conversion efficiency. Thermionics are a bit more efficient but require a higher temperature. Brayton, Rankine, and Stirling achieve typically 20-30% efficiency or more but require heavier parts and moving parts. Higher efficiencies can be achieved if a large radiator is available to provide a low-temperature heat sink for the cycle. There has been extensive development of all of these conversion systems on earth, but no dynamic conversion system has been used in space to date.

Given the need for a small reactor and shield mass, and a high conversion efficiency, we propose consideration of the following three options for the KBO mission: (1) a UZrH-fueled, NaK-cooled reactor with a steam Rankine conversion system, (2) a UN-fueled gas-cooled reactor with a recuperated Brayton conversion system, and (3) a UN-fueled heatpipe-cooled reactor with a recuperated Brayton conversion system. All three systems use the same assumed low-mass deployable thermal radiator to maximize the conversion efficiency. Table 2 compares the three systems. Subsequent sections describe the options in more detail. There has not been sufficient detailing of the designs to perform an accurate weight estimate for each system, but rough estimates indicate that all of these systems have the potential to be below the weight limit of 2350 kg.

TABLE 2. Comparison of three electric power systems for the KBO mission.

Component	UZrH/NaK-cooled/Rankine	UN/gas-cooled/Brayton	UN/Heatpipe-cooled/Brayton
Electric power (kW)	100	100	100
Thermal power (kW)	345	220	333
Thermal efficiency (%)	29	46	30
Nuclear fuel	UZrH	UN	UN
Moderator	Be	BeO	N/A
Reflector	Be	Be	Be
Primary coolant	NaK	He/Xe	Na
Number of fuel pins	211	361	138
Enrichment (%)	93	93	93
Total mass of ²³⁵ U (kg)	8	70	75
Fuel clad material	Hastelloy	Nb1%Zr/Re	Nb1%Zr/Re
Vessel material	316 SS	Super Alloy	Mo
Core vessel outer diameter (m)	0.235	0.40	0.20
Core + reflector outer diam. (m)	0.40	0.70	0.42
Conversion cycle	Rankine	Brayton	Brayton
Working fluid	water	He/Xe	He/Xe
Thermal radiator type	heatpipe/fin	heatpipe/fin	heatpipe/fin
Working fluid	ammonia	ammonia	ammonia
Two-sided radiating area (m ²)	360	252	252
Reactor coolant exit temp (K)	723	1200	1200
Average radiator temp (K)	330	306	306

UZrH-Fueled NaK-Cooled Reactor with Rankine Conversion System

The SNAP series of reactors used UZrH fuel and a "thermal" neutron spectrum. That is, the neutrons released by fission were slowed down by collisions with the hydrogen in the fuel so that they could interact with the uranium more easily. This results in a minimum amount of fuel needed to achieve a self-sustaining reaction, which allows the reactor, and also the radiation shield, to be near minimum mass. The SNAP program produced six complete

operating reactor systems at various power levels in the 1960s (Angelo, 1985). One system, SNAP-10A, was flown in space.

The SNAP-8 reactors were derivatives of SNAP-10A, and the candidate reactor design would be similar to SNAP-8 and SNAP-10A. We would operate the reactor with a peak fuel temperature of only 800 K to extend the lifetime of the fuel. An enrichment of 93% is preferred, but a lower enrichment is also possible. The SNAP-10A spaceflight reactor and shield weighed about 268 and 98 kg respectively, summing to nearly 366 kg. To allow for a larger power and total burnup capability, we estimate the reactor mass would be 500 kg. To allow for shielding a large radiator, we estimate the shield would be 400 kg.

UZrH fuel cannot be operated at as high a temperature as UN fuel because of hydrogen dissociation. This necessitates the use of a low-temperature conversion system such as a steam Rankine conversion. Steam Rankine systems are a highly mature technology on earth with an extremely large industrial and extensive experience with reactor systems. However, they have never been tested or used in space, and this represents a major technical risk for this option. Figure 3 shows an overview of the conversion cycle. Heat is extracted from the reactor coolant (NaK) and converted to steam. The steam drives a turbine as it expands, and then condenses at a heat exchanger connected to the thermal radiator. A pump recirculates the condensed water back to the boiler. The turbine, pump, and alternator are all on a single shaft floating on a liquid bearing. The system has one moving element. The radiator consists of many parallel and separate capillary pumped loops with ammonia as the coolant. The temperature increase in the steam generator and superheater is about 383 K, but the temperature increase in the reactor coolant can be considerably less by designing the steam generator appropriately.

The Department of Energy and Jet Propulsion Laboratory developed operational hardware for a 25-kW toluene Rankine system driven by solar thermal energy (Nesmith 1985). The program also generated several conceptual designs for 100 kWe systems. We estimate the weight of a TAP with a multi-stage turbine for the KBO mission would be about 90 kg. The additional components for conversion system components (excluding the radiator) are estimated to be about 160 kg, summing to a total of 250 kg.

FIGURE 3. Overview of Rankine conversion system.

A major key to success for this system is development of a large lightweight (deployable) radiator. This allows heat rejection to space at a low temperature, which can result in a high conversion efficiency overall. NASA and commercial firms have been working on such radiators (some of the designs and specific diagrams are proprietary). The most advanced designs use an array of parallel capillary pumped loops with thermal fins between them. Ammonia is typically used as the coolant. The use of many separate loops minimizes the need for shielding from micrometeor impact. The total mass of these radiators is about 2 kg/m² (which would be 1 kg/m² of radiating surface if used in a flat configuration with two-sided radiation). Capillary pumped loops, loop heat pipes, and heat pipes

have all been demonstrated in space, but these new lightweight radiator configurations have not. The heat load required for each individual heat pipe has been demonstrated on earth, but not at the required lengths yet.

The total radiating area needed for the baseline design is about 360 m^2 at 330 K. Traditionally space reactor radiators have been designed as conical with only the outside surface radiating. This was done to maximize the radiator area behind the radiation shield. However, to allow easier deployment of the radiator, to minimize the radiator mass, and to help reduce the shield mass, we presently envision the radiator as being flat with both sides radiating. Thus the physical radiator would be 180 m^2 with a total radiating surface of 360 m^2 . The radiator would deploy from a manifold extending along the boom. With this potential design and the mass numbers quoted for small advanced systems as background, the radiator system is estimated to weigh 360 kg.

The sum of all these estimated weights (reactor, shield, conversion, radiator) is 1510 kg. Structure, additional controls, other components, and contingency will add to this total estimate, but staying below the 2350-kg limit seems achievable.

UN-Fueled Gas-Cooled Reactor with Brayton Conversion System

The most recent U.S. space reactor power program, SP-100, developed detailed designs, advanced reactor fuel, a "zero-power" reactor critical assembly, radiation-hardened control drives, and various other hardware components in the 1980s. The baseline SP-100 was designed to produce 2400 kW thermal and 100 kW electric with a lifetime of 7 to 10 years (Mondt, 1994, Mondt 1995, El-Genk 1994). The specific power at program termination was about 42 kg/kWe. It used a high-temperature advanced fuel (UN) which was developed and proven with nuclear burn-up tests during the program. The fuel was not designed to slow down the fission neutrons, so the neutron spectrum was "fast" and the resulting core size and U-235 mass for the reactor to achieve criticality was thus larger than for the SNAP series.

The proposed design consists of a gas-cooled fission reactor using the SP-100 developed UN fuel and a closed Brayton cycle for power conversion at 100 kWe. There is a very extensive industrial data base and fabrication experience for open-cycle Brayton units: they form the basis for commercial and military jet engines as well as helicopter engines. The Brayton conversion system could be designed to have only one moving part: a single shaft connected to the turbine, the electrical generator rotor, and the compressor. In closed systems, this single shaft floats on a gas bearing bled off from the main gas flow and returned to it. Space Brayton units would be weightless and at constant power. They should easily last for many years without any maintenance. A 52,000-hr ground test of a 10.7-kWe closed Brayton unit at NASA/LeRC in 1965 supports these expectations.

The reactor is gas cooled (30/70 mole-% He/Xe) which directly drives the Brayton system. The active core is 0.40 m in diameter and 0.5 m long. The radial reflector is 0.15-m thick beryllium, and the axial reflector is 0.10-m thick BeO. The fuel rods are held in a lattice of BeO with a 2.0-mm thick flow channel around each rod for the gas coolant. The BeO provides a small amount of moderation. There is a strong negative thermal feedback which allows the reactor power to naturally follow variations in load without needing adjustment of the control elements. There is a thick radiation shield on top of the reactor to shield the payload from the reactor neutron and gamma radiation.

Figure 4 shows a schematic of the Brayton cycle and the associated state points. The reference design produces 100 kWe with 46% total thermal efficiency and has a specific mass of about 26 kg/kWe. A key feature is the heat exchanger which recuperates some heat from the turbine exit and uses it to preheat the gas returning from the heat sink before it re-enters the reactor. This recuperation step gives the cycle a greater conversion efficiency than a "non-recuperated" Brayton cycle. The radiator is opened out flat to radiate from two sides and is 126 m^2 per side. This allows a low heat-sink temperature and enables the high electrical conversion efficiency. Detailed weight estimates are not available for this configuration, but the larger reactor size will likely make it heavier than the UZrH Rankine system.

FIGURE 4. Brayton cycle schematic and associated state points.

UN-Fueled Heatpipe-Cooled Reactor with Brayton Conversion System

Houts, Poston, and Emrich have reported on various designs for heatpipe cooled reactors (Houts, 1998; Poston, 1996). In the heatpipe power system (HPS) heatpipes are inserted into the reactor core at regular locations to remove heat by boiling and wicking of the coolant. This heat is then transferred via a heat exchanger to the working fluid of a conversion system. For our application, we would use UN fuel in the reactor, sodium in the heat pipe, and He/Xe in the Brayton conversion system. Thus the system state points would be very similar to the gas-cooled Brayton system. Figure 5 shows a fuel/heatpipe module and a cross section of the reactor.

There are numerous advantages and features of the HPS that make it an attractive near-term system:

1. **Safety.** The HPS is designed to be subcritical for all credible launch accidents.
2. **Reliability.** The HPS has no single-point failures.
3. **Long life.** The design lifetime is in excess of 10 years.
4. **Modularity.** The HPS consists of independent fuel/heatpipe modules which can be tested individually.
5. **Testability.** The HPS system launch hardware can be tested at full power using electrical heaters in place of fuel rods. Unirradiated fuel rods inserted before launch. Full-power nuclear tests might not be required.
6. **Versatility.** The HPS can use a variety of fuel forms, structural materials, coolants, and conversion systems.
7. **Scalability.** The HPS design scales well to beyond 1000 kW thermal power.
8. **Simplicity.** There are few system integration issues since there are no in-core shutdown rods, no hermetically sealed refractory metal vessel or flowing loops, no electromagnetic pumps, no coolant thaw systems, no gas separators, and no auxiliary coolant loop for decay heat removal.
9. **Fabricability.** Most of the fabricated parts are small modules with similar metals; there is no pressure vessel.
10. **Near Term.** The system needs no development of advanced materials or components. It can be developed quickly and inexpensively with few nuclear tests.

11. **Low Mass.** The HPS system has a high fuel fraction in the core since it uses no in-core shutdown rods. The potential for in-space fueling (because of no pressure vessel) allows a more compact form while still meeting launch safety requirements.

FIGURE 5. Module and reactor for heatpipe power system.

LAUNCH APPROVAL

There is a precedent for operating small research-sized reactors in space. There are presently over 30 shut-down nuclear reactors orbiting earth at about 600 km altitude. All but one of these are Russian reactors from Rorsat high-power radar satellites. The one U.S. reactor is SNAP-10A, launched in 1965. Every U.S. launch of a payload involving nuclear material must be reviewed by an Interagency Nuclear Safety Review Panel (INSRP) (Sholtis, 1994). The INSRP reviews the sponsors assessment of the risk and reports to the Office of Science and Technology Policy under the Executive Branch. The President or his designee (usually the Science Advisor) then decides whether to grant launch approval. This process has been followed for 25 launches of nuclear materials over the past 40 years and approval has always been granted. All but one of these launches have involved radioisotopic power sources, but a space reactor would follow the same process.

One key safety feature of a fission reactor is that it is barely radioactive before it is used. The radiological inventory in the fresh fuel is about the same as a 100 cubic yards of Florida dirt or beach sand. The space reactor can be tested prior to launch at essentially zero power to prevent any inventory buildup, so the primary concern for launch safety will be to assure that the reactor will not turn on for any conceivable launch accident scenario. This requires solid engineering, but is not difficult. In addition, once the reactor begins operation in space, it should be assured that it cannot reenter the earth's atmosphere. With high circular orbits and low-thrust ion-propulsion, or with a launch to an earth-escape trajectory before the reactor is turned on, this should be fairly straightforward. Once it is on its way away from earth, the safety issues vanish, and waste disposal is automatic.

SUMMARY

A mission to rendezvous and land on one a Kuiper Belt Object is stressing and requires a large amount of total ΔV . A fission-powered system with an electric thruster could rendezvous at 40 AU in about 13.0 years with a total DV of 46 km/s. It would deliver a 1000-kg science payload while providing ample onboard power for relaying data back to earth. The launch mass of the entire system (power, thrusters, propellant, navigation, communication, structure,

science payload, etc.) would be 7984 kg if it were placed into an earth-escape trajectory ($C=0$). Alternatively, the system could be placed into a 700-km earth orbit with more propellant, yielding a total mass in LEO of 8618 kg, and then spiral out of earth orbit to arrive at the KBO in 14.3 years. To achieve this performance, a fission power system with 100 kW of electrical power and a total mass (reactor, shield, conversion, and radiator) of about 2350 kg. Three possible configurations are proposed: (1) a UZrH-fueled, NaK-cooled reactor with a steam Rankine conversion system, (2) a UN-fueled gas-cooled reactor with a recuperated Brayton conversion system, and (3) a UN-fueled heatpipe-cooled reactor with a recuperated Brayton conversion system. All three of these systems have the potential to meet the weight requirement for the trip and be built in the near term.

ACKNOWLEDGMENTS

This activity supported by Sandia National Laboratories. Sandia is a multiprogram laboratory operated by Sandia Corporation, a Lockheed Martin Company, for the United States Department of Energy under Contract DE-AC04-94AL85000.

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