HIGH POWER NUCLEAR ELECTRIC PROPULSION

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The research leading to these results has received funding from the European Union Seventh Framework Programme (FP7/2007-2013) under grant agreements n°s 218859 and 284081.

Abstract:

The 'High Power Electric Propulsion; a Roadmap for the Future (HiPER)' project has investigated the next generation of high power European electric propulsion. However sample return missions to the outer planets, or even orbital capture, only look feasible with substantial nuclear power generators. Once developed the technology has wider applications in shuttling large infrastructure to the inner planets or Lagrange Points or even the commercial mining of near earth objects (NEOs). A preliminary report of the scope of the study for nuclear electric power generation was presented at propulsion 2010. It set ambitious targets within the constraints of current or foreseen technology, launch capabilities and operating and safety constraints. This paper describes the extent to which the targets are considered feasible and sets out a roadmap to overcome the technical challenges identified. The paper summarises the final study findings, roadmap recommendations and subsequent developments.

Introduction

The EC FP7 High Power Electric Propulsion: a Roadmap for the Future (HiPER) investigated nuclear fission electrical power generation for the exploration of the outer solar system. Mission analysis[1,2] identified a range of applications from one way journeys to Uranus, return missions to the Jovian and Saturn planetary systems and multiple, shorter infrastructure, or manned, delivery. The nuclear 'space tug' concept was seen as the best fit to the range of applications.

The 'space tug' had to be capable of launch on an Ariane 5 ECA. The main constraints are the fairing dimensions and the lift capability. From previous studies a specific mass of 25kg/kWe was thought the best that could be achieved with current and emerging technology. In principle this constrained the power generating capability to 200 kWe which became the target for a concept design.

The technical starting point was a Rolls Royce Nuclear Technologies for Space Applications survey[3] and an Acta[4] Shield Design Study which provided the baselines for modelling and simulation studies and the consequent concept design development. The three main pillars of the investigation were the core reactor, the shield and the power conversion system. However the potential to achieve the individual capabilities had also to be consistent with a viable, overall system architecture as well as each other. Initial investigation demonstrated the specific mass benefit of Brayton cycle power conversion compared to thermo-electric or thermionic systems for the power range under consideration. However the relative merits of direct cycle, gas cooled, epithermal and indirect cycle liquid metal cooled reactor based systems are highly dependent upon technical development required to realise the concept design, operating and safety

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1 Nuclear Electric Propulsion (NEP) Rationale and Strategy (SEP/HiPER/WP3.2/SD 1st January 2009).
2 HiPER Mission & Transportation Scenarios (HiPER-ALT-D-2.3-i1r1 30th June 2009).
3 Nuclear Technologies for Space Applications - Technology Survey (HiP- R-R- TN – 001- i1R130th June 2009)
4 Radiation Shielding Design for NEP spacecraft

considerations. It was not possible to determine a clear advantage of either technology and concept designs were developed for both, considerably extending the planned level of effort.

Design Considerations

The technology survey concluded that direct or indirect Brayton cycle thermal to electrical power conversion was the preferred option for medium (100-500 kWe) power generation and that the technology is scalable. High temperature materials are essential to facilitate high temperature operation to minimise system (and in particular radiator) mass. Radial conversion machinery was preferred for compactness together with a mixture of xenon and helium operating gas optimised for molecular mass and thermal properties.

The more compact liquid metal indirect Brayton cycle reactor was found to be 25% of the mass of the Direct Brayton cycle (476 compared to 2017 kg) before consideration of the heat exchanger mass. Consequently reductions in the direct cycle design mass, without affecting performance, control and safety, were investigated.

Circulation of the operating gas through a fixed, tubular radiator was preferred to a deployable heat pipe design for several reasons. Deployable structures are larger and require flexible connections; there is a risk of heat pipe freezing during shut down periods and they are more prone to radiation scattering effects. However deployable radiators could overcome the Ariane 5ECA fairing size constraints for lower temperature systems and they may still be appropriate for the separate cooling of electronics. They therefore remain an option in the Roadmap.

Reactor Core Physics

The findings of the reactor modelling and simulation are reported in Rolls Royce HiPER Nuclear Power Generator Modelling and Simulation Details[5]. Both indirect liquid metal cooled and direct gas cooled reactors could generate 200 kWe over the 10 year lifetime or longer if required. Fuel composition and reactivity control are compatible with water immersion safety requirements.

Neutron and gamma escape fluxes from the core were calculated at all exterior surfaces. In the indirect concept, little variation in the escape fluxes was identified between the different surfaces. In the direct concept, the neutron and gamma escape fluxes were approximately an order of magnitude higher from the radial surface than from either axial surface. In the indirect concept design, the neutron escape flux peaked towards higher energies, which was in keeping with its fast neutron spectrum. In the direct concept design, the neutron escape flux was relatively constant over the majority of neutron energies. Coolant gas activation calculation for the direct design demonstrated no problems.

The direct cycle core physics is dominated by the inlet/exit gas flow paths, which occupy about 35% of the fuelled region volume in the design which was analysed. Higher gas pressure and acceptance of increased core resistance permit the gas flow areas to be halved, perhaps with the number of fuel bed annuli reduced from 12 to 8. The core diameter can then be reduced to 91% of the reference value without reduction of fuel loading. (Reduction in optimum reflector diameter might also result from the reduced neutron transparency of the core.) The core length can also be reduced in proportion to the diameter (with a proportional reduction in fuel load) and the ratio of fuel mass to surface area of fuelled volume envelope returns to the original value when the core length becomes 0.5 m. (The ratio used is an indicator of neutron leakage.)

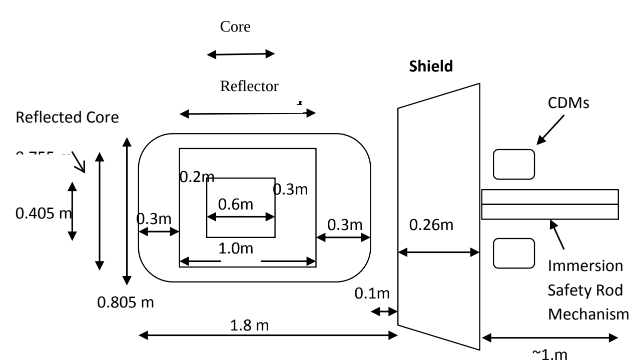

Figure 1: Direct Cycle reactor core arrangement.

[5] HiPER: Space Reactor Core Physics (HiP- R-R- TN – 003- i1R1 April 2010)

Both direct and indirect reactor designs exhibited large excess reactivities with the control media 'withdrawn' which were left as a margin to allow for losses arising from the implementation of engineering features and realistic materials. The margins could be 'trimmed' in future iterations of the designs, if required, to reduce the fuel loading and/or fuel enrichment.

Radiation Shielding

The shield modelling and simulation used US Government MCNP-MCNPX code. For the direct cycle the neutron and gamma flux radiated from the reactor is seen from Figure 2 to decrease as one moves from the centreline thus permitting a corresponding reduction in shield thickness toward the outer edges. (A similar distribution with higher radiation intensity was derived for the Indirect Cycle.)

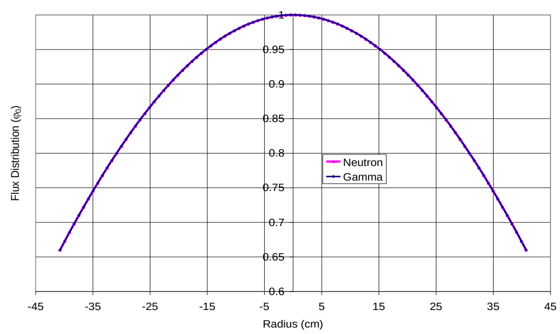

Figure 2: Direct Cycle Neutron and Gamma Ray Flux Distribution.

The general arrangement for the shielding and the internal structures is shown in Figure 3.

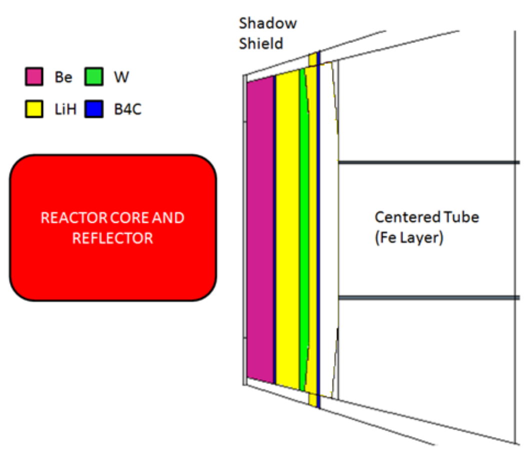

Figure 3: Direct Cycle Shield Arrangement.

Account was also taken of the shielding benefits from the turbo-alternator pods and heavy electrical equipment in the PMAD and EP power processing units (PPUs). As the location and size of propellant tanks is unknown no account was taken of further shielding from xenon.

The final modelling results[6] gave an end of life gamma dose or neutron flux generally lower for a factor of two to four than the stated requirements apart from the gamma dose at payload entrance for the direct cycle reactor, where the margin is only of about 20%. (The requirements are 1.6 mRad/s and 31700 n/cm^{2}s based on a 22.5 metre boom separation to the payload and derived from the SP100 requirements criteria.)

This was considered a reasonable margin for more detailed design once the spacecraft architecture is fully established taking account of location of internal structures, routing of coolant pipes and control mechanisms and shaping 'fins' to shield protuberances. (This analysis was based on the assumption of radial control rods only requiring a 0.1m gap between the reactor and shield). The most significant result was to reduce the shield diameter for the Direct Cycle (at the face closer to the reactor) from about 1.6m to about 0.56m, and depth from 0.51 to 0.26m, so reducing shield mass by a factor of two

Similarly for the Indirect Cycle, it was possible to reduce the minimum shield diameter to 0.52m and the thickness to 0.41m. The greater thickness reflects the higher radiation flux from the indirect reactor. It also assumes that the control rods are partly housed within the radial reflector possibly requiring some extra shielding to counter 'streaming' through the control rod mechanism.

Power Conversion

Rolls Royce investigation of the power conversion options[7] showed Brayton Cycle efficiencies of between 17% and 19%.

[6] Space Reactor Shielding -The Radiation Shield Design - ACTA/HiPER-WP 3.2-RSD/003/2009 May 2011.

[7] Design of a Power Conversion Unit for a Space-Based High-Power Electric Propulsion System (HiP- R-R- TN – 002- i1R1 May 2010)

Specifically the indirect cycle was the more efficient as a result of the reheat loop but it is a mechanically more complex arrangement.

Although higher power conversion efficiency for the Brayton cycle is achievable in principle this is not compatible with the main design objective of minimising specific mass in this particular case. Modelling and simulation lead to optimisation based on a number of factors including realistic turbine and compressor speeds, pressure and temperature differentials. The mass of the power conversion system is mainly a function of the operating temperature which determines the size of the heat sink radiator. The upper temperature is constrained by the creep life of the first turbine stage. The power conversion for a recuperated direct cycle system is illustrated in Figure 4. Gas coolant, direct from the reactor core, feeds an HP turbine driving the compressors before a free turbine drives the turbo-alternator (which needs its own cooling radiator).

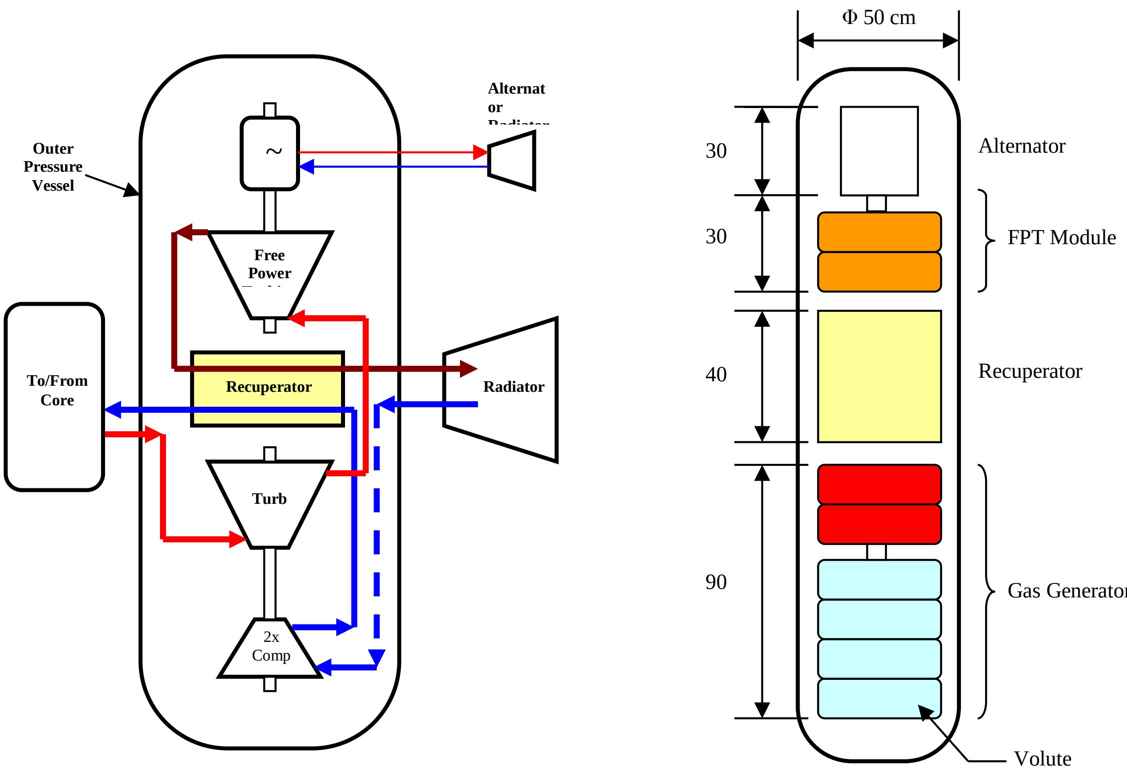

Figure 4: Layout of the direct cycle Power Conversion Unit and associated modules

Because the indirect cycle was found to be more efficient than the direct cycle it achieved an 11% lower radiator mass calculated through scaling the idealised area. The optimum turbo-machinery configuration was achieved by employing a two-shaft compressor with a reheat loop driving a free power turbine. The reheat loop increases the temperature of the fluid entering the free power turbine and consequently the temperature of the fluid at the radiator inlet. It is thus a direct product of the reheat loop that is responsible for the reduction in mass of the radiator when compared to the direct cycle.

Higher operating temperatures significantly reduce the radiator mass and size through a quartic relationship. Below a turbine inlet temperature of 1300°K a fixed radiator would be too large to fit within the Ariane 5 fairing for launch. Also unless made from low mass materials once micro-meteoroid protection is added the mass is likely to be excessive. The creep life for 10 years for today's most advanced single-crystal super-alloys limits turbine inlet temperature to 1100°K before considering the challenges of high temperature radiator design.

Even these alloys are in themselves not currently suitable for such an application due to their anisotropic strength properties. They would require significant development in replacement polycrystalline materials and associated manufacturing techniques by the year 2020.

In principle the direct cycle can deliver a turbine inlet temperature of ~1500°K although the indirect cycle is likely to be limited by heat exchanger (between the liquid metal coolant and operating gas) to ~ 1200°K. The following technical developments were therefore identified in the Roadmap.

Turbine Inlet Design

The turbine rotor rather than the guide vanes is the vulnerable component. When it is rotating the rotor experiences a lower stagnation temperature because of the relative motion of the turbine blades to the coolant gas. Indicative calculations suggest the drop in stagnation temperature could be ~80ºK, or higher, thus permitting the turbine inlet gas temperature to rise to ~1180 to 1300ºK.

Turbine Blade Cooling

Another possibility is to bleed cooler coolant gas from the compressor onto the turbine rotor blades. It is understood that there has been research in Russia for applying this technique to high temperature terrestrial gas turbines as an alternative to fabrication from very expensive alloys containing Rhenium. The technique may enable the turbine blades to remain at up to 200ºK below the temperature of the inlet gas.

Refractory Metal Alloys

There is understood to be a great deal of research into high temperature refractory metal alloys (niobium, molybdenum, tantalum, tungsten and rhenium) in Europe, Russia and the US. It is also understood that longer creep life to reduce maintenance in, for example jet engines, is the subject of new research. Extrapolation of creep test results from 10,000 hour tests to a 100,000 hour design life would probably be necessary to evaluate the potential of these alloys to meet the high temperature turbine requirements. Niobium, for example, has lower density and should not be susceptible to oxidation when operating in a xenon and helium environment provided there was adequate protection from any out-gassing of oxygen in the system. Tungsten has a higher melting point (and onset of creep) but the higher density makes it more prone to stress. In terrestrial applications the alloys are often coated with materials to prevent oxidation. This is an expensive process. Although oxidation may not be an issue, investigation of suitable coatings to provide thermal insulation was recommended.

Ceramic Materials.

Ceramic materials in theory have the thermal and creep properties to offer a very high temperature solution but tend to be prone to stress fracture. Another possible technique is a thin ceramic layer on the turbine to give a highly efficient thermal barrier. It is used successfully in terrestrial applications and investigation of its adaptation to radial machinery with the necessary creep life was also recommended.

Whereas turbine inlet design and blade cooling may make inlet temperatures up to 1300ºK feasible higher temperatures may well require refractory metal alloys or ceramics. Refractory metal alloys may be necessary anyway to achieve the necessary anisotropic properties.

Radiators

Two types of radiator were considered: fixed and deployable. Fixed radiators are of more simple construction and more compact but can be high mass and more difficult to protect against micro-meteoroids. A ≥1300ºK turbine inlet temperature for a 200 kWe generator is required to keep the size compatible with the Ariane 5 fairing dimensions. A deployable radiator can operate at lower temperatures but it will have a much a larger radiating surface. It is easier to protect against micro-meteoroids but more complex.

A Rolls Royce initial fixed radiator heat sink design was based on a stainless steel or Inconel 600 or 718 (nickel alloy) simple tube arrangement taking up part or the whole of the cylindrical length of the Ariane 5 fairing (10m). The material has a density of 8250 kgm^{-3} and a conductivity of $19wm^{-1}K^{-1}$ giving a significant mass penalty. The original assumption of a 1100ºK turbine inlet temperature required a radiator inlet temperature of 640 ºK, outlet of 420 ºK and mean of 530 ºK for Direct Cycle (inlet 694 ºK, outlet 405 ºK and mean 555 ºK for Indirect Cycle).

The radiator calculations assumed a total cycle pressure drop of 5% of the upper working pressure equivalent to 1 bar total. The radiator tubes are allowed 20% of this budget (0.20 bar) and are limited to using a minimum tube diameter of between 7.5 and 10mm outside diameter. An assumed 20MPa hoop stress limit corresponds to a wall thickness of between 0.18mm and 0.24mm. Under these circumstances the tube weight alone amounts to between 1.1 and 1.5 tonnes without any allowance for headers, connecting pipework or supporting structure.

It has not been possible to find evidence of space radiator design to date with these features but the design concept is considered realistic. Further research for compatibility with higher operating temperatures is recommended in the Roadmap: ie. For Indirect cycle 1200 ºK turbine inlet, radiator inlet 800ºK, outlet 505ºK mean 655ºK; for Direct Cycle 1300 ºK turbine inlet radiator inlet 840ºK, outlet 620ºK mean 730ºK; for Direct Cycle 1500ºK turbine inlet, radiator inlet 1040ºK, outlet 820ºK mean of 930ºK.

High temperature carbon-fibre tubing can offer comparable thermal and structural properties for specific mass ~ 1750 kgm^{-3}. Although not helping to reduce the required area the lower mass benefits could be significant. (Rough estimates suggest > 1000 kg mass saving on a 1300 ºK, 200 kWe Direct Cycle system.) On its own, carbon-fibre is porous to the helium coolant gas so it would be necessary to find some method of sealing against this. If a thin metal liner was used, for example, the composite density could rise to ~ 2500 kgm^{-3} but still give significant mass savings. A 500 kg radiator mass saving reduces a 200 kWe generator specific mass by 2.5 kg/kWe and further research into the use of these materials is recommended.

Micro-meteoroid protection for coolant tubes creates a gap between an outer skin and the surface to be protected. The very high velocity of impact will vaporise the micro-meteoroid and the thermal energy generated is dissipated in the gap. The gap does not have to be very wide provided there is sufficient space for the dissipation without further damage. Coolant carrying tubes have to be protected because a leak would be catastrophic but other structures such as radiating fins can tolerate most likely impacts. A small vacuum filled gap between the radiator tubes and the space environment will reduce the radiating efficiency and fins require a larger area to radiate the same quantity of heat. Either barrier tubes or protective fins may be considered as shown in Figure 5.

Figure 5: Tube Radiator with Barrier Tubes or Shaped Fin Design.

A fixed radiator for a direct cycle 200 kWe generator with 1300 ºK inlet temperature requires an area of 110 m^2 which will fit within cylindrical section of the Ariane 5 ECA fairing (140 m^2) without micro-meteoroid protection. Rolls Royce proposed a system of micro-meteoroid protection based on a ‘bumper’ shield of smaller empty tubes to protect the coolant containing. The appropriate number of pressure tubes for acceptable gas pressure drop appears to be in the range from 500-700 for tubes of about 15mm diameter up to 1000-1400 for tubes of about 10mm diameter. Temperature loss across the barrier is dependent on the ratio of barrier tube thickness to the square of barrier tube diameter (as well as on metal thermal conductivity). There is an incentive to minimise barrier tube diameter, so far as the meteorite defence requirement permits while retaining good thermal conductivity.

It was estimated that this arrangement would increase the radiator area by ~ 30%, to account for the lower thermal efficiency of the design, increasing the required area to ~143 m^2 (For a 1300ºK Direct Cycle system). The 140 m^2 constraint would restrict the generator output to about 195 kWe unless other cooling surfaces could be used. It was also estimated that there would be a 70% increase in mass (from 1523 to 2589 kg) both for the larger area and the additional ‘bumper’ tubes. Investigation into lower density, high radiative materials for both coolant and barrier tubes is recommended together with self-sealing mechanisms in the radiator design and shaped fins rather than barrier tubes.

NASA has researched a deployable radiator design for a 200 kWe Brayton cycle nuclear generator[8] (see Figure 6).

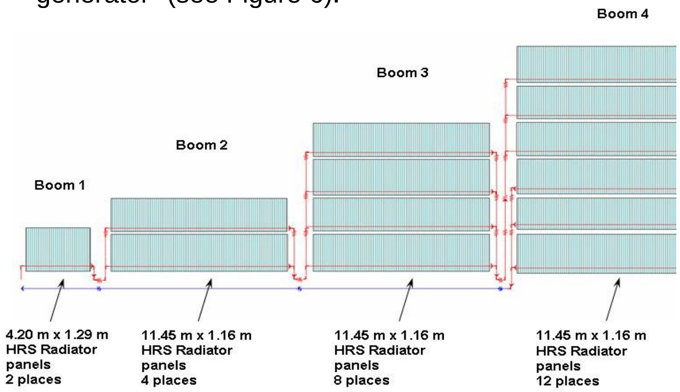

Figure 6: NASA Deployable Radiator Design.

The design is based on heat pipes separated by cooling fins. The heat pipes are 'fed' from a circulating coolant which might be water or potassium. Heat is transferred from the xenon and helium turbo-machinery operating gas in a heat exchanger. This isolation from the primary (or secondary for the Indirect Cycle) coolant means that rupture of the radiator circulating coolant main could probably be limited to a wing or even a panel and would not necessarily be catastrophic. The resulting design requires a much larger area (two wings of ~165 m^2), as shown in Figure 6, than a fixed radiator but it can be made from lower density materials (~3 kg/m^2).

There is the added vulnerability from all the flexible connections which must survive the radiator deployment and the complexity of a secondary or tertiary (Indirect Cycle) coolant circuit. More challenging is how to fit the radiator within the Ariane 5 ECA fairing once space has been allocated to the reactor, shield, conversion machinery, PMAD, EP Systems, Propellant Storage and structure. The NASA panels in the example are longer than will fit within the cylindrical section even if it is empty and the overall ~330m^2 required for a 200kWe 1200ºK turbine inlet temperature design represents a significant packaging challenge.

[8]NASA/TM—2006-214121 A Comparison of Coolant Options for Brayton Power Conversion Heat Rejection Systems by John Siamidis (Analex Corporation) and Lee S. Mason (Glenn Research Center) June 2006

The NASA design appears to be consistent with a shadow angle of less than 10º which will lower shield mass but implies a longer boom length and associated harness mass. With a 14º shadow angle a comparable boom length to the HiPER concept design should be possible if the radiator can start immediately behind the shield. The area and mass in the NASA design are appropriate for a 1100-1200ºK turbine inlet temperature and some modest reductions may be expected if the inlet temperature is raised to 1300ºK.

Micro-meteoroid protection is achieved by surrounding the coolant circulating pipes with an outer protection layer and using the heat pipes for heat rejection as illustrated in Figure 7. The low mass carbon fibre fins between the heat pipes may be punctured by micro-meteoroids with minimal effect on performance. The heat pipes themselves are protected by a foam surround and the circulating coolant is protected by foam and an air gap.

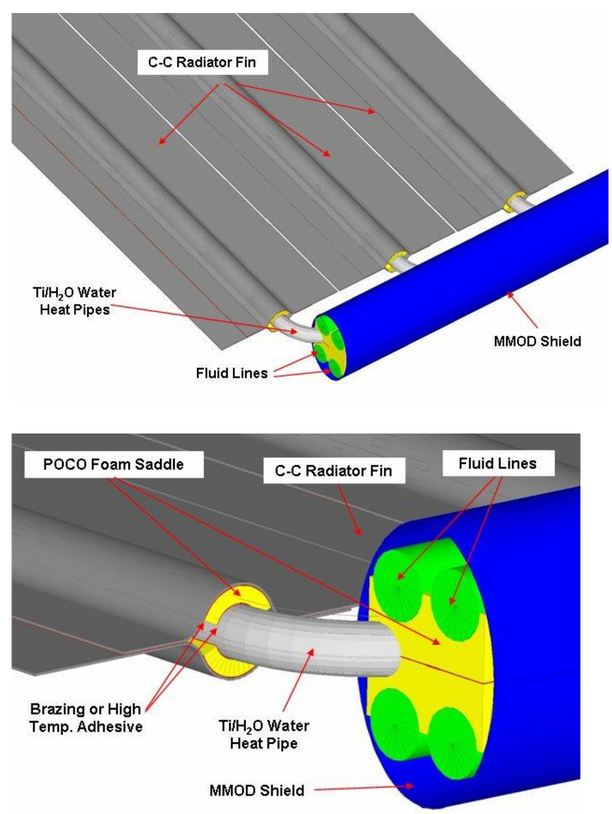

Figure7: Deployable Radiator Design Detail.

Operating Constraints

In addition to the constraints imposed by the Ariane 5 ECA fairing dimensions and lift capability, core reactivity control and high temperature materials, the following were also taken into account in the concept design:

- Rotating machinery to be in balanced contra-rotating pairs to avoid inducing unwanted torques,
- The battery (or alternative) power required for initial commissioning and a 'cold start' in space; the battery may also have a useful role in helping to protect the system from large transients,
- Resilience to sudden, large load induced transients and emergency shut down,
- Commissioning, starting and shut down,
- Standby and low power operation,
- Reliability, damage control and disposal.

System Design Constraints

The Roadmap has recommendations to manage the following design constraints:

- To avoid radiation damage from the reactor and thruster exhaust plume impingement conventional design puts the reactor and propulsion systems at opposite ends of the spacecraft. Thrusters with very narrow exhaust plumes may be able to be located relatively close to the reactor with shaped shielding. Otherwise novel methods of power distribution are required to avoid a substantial mass penalty.
- The structure must be sufficiently strong to support the high mass reactor and shield high up in the launcher fairing during the launch. Although the proposed structural layout has this requirement in mind, structural design was outside the scope of the study.
- Electrical power systems and components in the 200kWe range still have to be developed for space applications. Components, especially in the turbo-alternator pods, must also be able to sustain high temperature (~470°K). A 500V or possibly 1kV DC power bus may best match the EP system requirements. A rectified supply of this nature is therefore assumed for each generating pod.
- The EP thrusters are most efficient if the thrust vector is through the centre of mass in the direction of the required motion.

Safety

The US SP100 safety considerations were used.

Concept Design

Initial concept design was based on 1200ºK Indirect cycle and 1300ºK Direct cycle turbine inlet temperatures. Fixed nickel alloy radiators without micro-meteoroid protection gave the mass and radiator area figures shown in

SYSTEM & BASELINE EXAMPLE		Recuperated Direct Brayton (Epi)		Indirect Brayton (Fast)	
T hot, ºK		**1300**		**1200**	
Power					
MWth		1.18		1.12	
MWe		0.200		0.200	
η		0.169		0.175	
Reactor Mass	kg	**1627**	*8.14*	**528**	*2.64*
Shield	kg	**800**	*4.0*	**600**	*3.0*
Reactor Control	kg	**42.6**	*0.21*	**31.1**	*0.16*
IHX	kg			**366**	*1.83*
Generation	kg	**1656**	*8.28*	**1612**	*8.06*
Radiator					
Area	m^2	**110**		**128**	
Mass	kg	**1523**	*7.62*	**1770**	*8.85*
Total Mass		**5648**		**4907**	
Sp. Mass, kg/kWe		**28.2**		**24.5**	

Figure 8. Initial concept design Direct and Indirect Brayton Cycle Systems *(Italics are specific mass)*.

Although the Indirect system design appears to meet the target specific mass of 25kg/kWe at 200kWe there are several factors which must be taken into consideration. Armouring the radiator with a bumper tube design will increase the area to 166m^2 and the mass to 3009kg. Not only will this not fit within the Ariane 5 fairing; the specific mass increases to 30.73kg/kWe. If one also adds ~400kg of battery for cold start and commissioning this rises to 32.73kg/kWe. A deployable radiator design would have an area of ~ 330 m^2 and approximately 1800kg mass which with the additional battery suggests specific mass ~26.5kg/kWe might be achievable.

For the Direct system a barrier armoured radiator would have an area of 143m^2, which only marginally exceeds the cylindrical volume available in the Ariane 5 fairing but would have a mass of ~ 2589kg, bringing the specific mass up to 33.57kg/kWe. A deployable radiator could possibly reduce this to 29.15Kg/kWe and a low mass fixed radiator (specific mass 3.8 kg/kWe) could in theory bring the system specific mass down to at 200kWe ~27kg/kWe.

This assumes that the PMAD, electric propulsion system, structure and propellant specific mass contributions are common to all the design options. However it should be noted that significantly more energy is required to raise the temperature of a liquid metal (Indirect cycle) reactor to an operating point than for the (Direct cycle) gas cooled.

The concept designs were scaled between 100kWe and 2 MWe as illustrated in Figure 9 for Direct Cycle 1500°K turbine inlet temperature with ceramic turbines and barrier tube nickel alloy coolant tubes and micro-meteoroid protection.

T hot, K		**1500**			
Power **MWth** **MWe**		 0.592 0.100	 0.888 0.150	 1.183 0.200	 11.83 2.0
η		0.169	0.169	0.169	0.169
Reactor Mass	kg	**1226**	**1436**	**1627**	**6468**
Shield	kg	**598**	**708**	**800**	**2306**
Reactor Control	kg	**33**	**38.3**	**42.6**	**113**
IHX	kg				
Generation	kg	**1229**	**1449**	**1656**	**7693**
Radiator Area Mass	 m^2 kg	 **40.5** **750**	 **60.6** **1098**	 **80.8** **1441**	 **808** **12768**
Total Mass	kg	**3856**	**4729**	**5566**	**29349**
Sp. Mass, kg/kWe		**38.6**	**31.5**	**27.8**	**14.7**

Figure 9. Initial concept design Direct 1500°K Brayton Cycle Scaled from 100kWe to 2MWe.

This emphasises the benefits of very high temperature operation. The heat exchangers for a deployable radiator would be technically very challenging. However, a low mass fixed radiator (3.6kg/kWe) might reduce the overall specific mass at 200 kWe to ~ 24.3kg/kWe.

Roadmap

The system design specific mass target analyses illustrate clearly the rationale for the roadmap focus on high temperature turbine and low mass radiator development. However validation of the technical development required to achieve power conversion efficiency and compact, high temperature reactor control systems and coolant pipe routing are all necessary. The advanced shielding design is a very significant contributor to lowering the overall specific mass also requiring validation through bread boarding and prototyping. Recommendations to improve the Concept Design, to optimise specific mass are:

Common to both Direct and Indirect Cycles

- ➢ Higher turbine inlet temperature (Direct to 1300ºK or even 1500ºK; Indirect to 1200ºK),
- ➢ Increase turbine efficiency from 85% to 88%,
- ➢ A higher temperature radiator,
- ➢ Reducing the mass and volume penalties of micro-meteoroid protection,
- ➢ Efficient routing of coolant pipes around, and design of Control Drive mechanisms (CRDM) through, the shield.

Direct Cycle

- ➢ Reduce reactor core size through a combination of higher operating pressure, increased TRISO particle density, and replacing control drums with rods to reduce the overall area required for the passage of coolant gas,
- ➢ A narrower shield shadow angle if radiator dimensions and internal pod location permit.

Indirect Cycle

- ➢ Increasing core exit temperature to 1200ºK (with high temperature heat exchanger (IHX)),
- ➢ Investigating 'hot launch' strategies to reduce commissioning power requirements.

Commissioning and Operation

Rolls Royce estimated a 40kWhr battery would be needed to commission a Direct Cycle reactor and up to 100 kWhr for the Indirect cycle. With the most advanced space battery technology this will have a mass of ~250kg (Direct) – 650Kg (Indirect). However the battery may have a useful function for in-orbit cold re-start, as a partial shunt for sudden load loss and for managing large power steps in start-up and shut down.

Very little information exists on PPU characteristics because space qualified components to operate at these high powers do not currently exist. Although this subject lay outside the scope of this particular study the interface must be included in future research and development because it is a critical consideration in the reactor emergency shut down design.

There is, for example, a significant trade-off in the mass and capacity of a coolant by-pass in the turbo-alternator to that of a shunt to absorb a sudden fall in electrical power demand.

Other Design Issues

Xenon propellant stored in supercritical condition at about 300ºK has a density of ~1100 kg/m3 and can provide useful shielding until the propellant is used. It could probably be maintained at between 30 and 90 Bar for much of the mission. The tanks are light and have no shielding value but need to be designed to minimise radiation streaming through the gaps between. An arrangement of 4 relatively wide diameter, say 1.5 metres, would need to be less than 0.5 metres long to hold 4 tons of propellant. Located at the separation plane they can help counterbalance the reactor and shield when establishing the centre of mass.

The Ariane 5 ECA constraint of a payload centre of mass 2.5 metres above the separation plane may also prove to be significant. Two of the heavier elements, the reactor and shield, are at the top end of the fairing. Care is required to balance these masses for launch. Relaxations to this constraint should be investigated.

The overall structural and thermal integrity of the spacecraft fell outside the scope of this study. Together with thruster and reactor location and electrical system requirements, the issues have been highlighted in this study and in many cases suggestions made for the way forward. It is considered that there is sufficient information now to conduct an analysis of the architecture and structural and thermal integrity.

The Way Forward

The Roadmap proposed a development schedule of preliminary technical research, feasibility, project definition, development, qualification, launch and operation. The nominal schedule was 30 years including a 10 year mission. The technical issues identified in the HiPER project were consistent with the findings of previous studies and the next step was seen to be a review of the European appetite to embark on a space nuclear fission generator programme.

In the subsequent FP7 Disruptive Technologies for Space Power and Propulsion (DiPOP) project we are investigating which missions might be seen to justify such a programme and to assess the resources required. This includes existing expertise and facilities and that which would have to be developed. A review of this nature was recommended by the European Working Group on Nuclear Power Source for Space report of 30th March 2005. We are also attempting to understand the potential commitment to a European fission nuclear project.

The scale of development may prove very challenging for current budgetary pressures in Europe. To help make best use of existing nuclear fission power experience and to explore any possibilities for collaboration we have the assistance of an Advisory Board of European, Russian and US nuclear experts. This is proving most helpful in developing a realistic perspective of the many issues.

The main DiPOP space fission nuclear power study deliverable is a Roadmap contributing to a recommendation of the European Working Group for Space Nuclear Power[9]. “A Roadmap for the development and use of nuclear power sources for space should be elaborated, differentiating in terms of the typology and the timescale. It should include a comprehensive inventory and assessment of all potentially relevant existing facilities and capabilities in Europe.”

We started with a draft Roadmap based on space fission nuclear power generation to date, the applications for which the technology is required or could bring significant benefit and an initial assessment of relevant European capabilities. An important issue is also the probability of public acceptance of a European space fission programme and the associated safety and regulatory requirements.

The draft Roadmap was reviewed by the Advisory Board in February and with their guidance we are now focussing on the assessment of European capability and interest with a view to reporting in October this year.

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[9] Re p o r t European Working Group on the Nuclear Power Sources for Space *30 March 2005*