NUCLEAR ELECTRIC PROPULSION/ POWER PROCESSING UNIT REPORT

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ABSTRACT

This document constitutes the deliverable D23.2 and is in two parts. The first investigates processes for evaluating the relative merits of high power electric propulsion systems couples to fission nuclear power sources. The second considers the relative merits and design challenges of fission nuclear power thermal to electrical conversion and subsequent distribution and application. It takes account of electric propulsion system input requirements, efficient power conversion and transmission and system robustness. A very high level set of criteria for electric propulsion system comparison and an associated evaluation matrix based on the earlier considerations are then included.

KEY WORDS DiPoP, Nuclear, Electric Propulsion, Power Processing, Evaluation Matrix.

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[R 2]	www.DiPoP.eu
[R 3]	http://www.its.caltech.edu/~jsnyder/thermoelectrics/engineering.html
[R 4]	NASA / CR--1999-209164/VOL1 Stirling Space Engine Program Volume lm Final Report 1999
[R 5]	HiPER Nuclear Power Generation Model And Simulation Details October 2010.
[R 6]	HiPER Nuclear Power generator Roadmap 6th May 2011.
[R 7]	HiPER Consolidated mission analysis document 8th December 2011

1. Introduction

This Technical Note is in two parts: an assessment of mission and nuclear electric propulsion system options and power management and distribution and electric propulsion power processing issues. There is a tendency to compare EP systems on their stand-alone merits which unfortunately can be of limited value and often misleading. In this Technical Note we therefore start by identifying the constraints on design options from external factors such as mission requirements and launch capability. Even these can be seen to vary widely, some favouring one electric propulsion technology and others another. Although most electric propulsion (EP) technologies can probably be adapted for the full range of missions it is not possible given the extent of current knowledge to identify one which achieves this better than any other. Consequently the preferred way ahead at this stage is for a nuclear electric propulsion generator (NEP) to be compatible with all the available EP technologies.

The key to this compatibility is interface between the NEP generator and the EP systems known as the power management and distribution system (PMAD). This is a function of the method of thermal to electrical power generation and the characteristics of the 'electrical load' in the form of the EP system (seen as the PPU). It is also a function of the spacecraft architecture and environment. The features which must be considered are:

- The thermal to electric power conversion which may be thermo-electric, thermionic, Stirling, Brayton or Rankine cycle,
- The characteristics of the electrical load in terms of electric propulsion systems (GIE, HET, HEMP, MPD) and other spacecraft functions, especially the reactor control and coolant functions,
- The degree of current and voltage regulation to match the supply to the load,
- Stored energy for commissioning (assumed to be in-orbit to meet safety requirements) and re-starting,
- Protection against the effects of sudden unplanned changes in load or power supplied,
- Thermal, radiation and vacuum environmental constraints,
- Mass efficient power distribution.

2. Mission and Nuclear Electric Propulsion System Options

2.1. Mission Design Constraints

Two important mission deign constraints are the available launch lift capability and the time to execute the mission. Manned missions need fast transit times and therefore high thrust to power ratios but at the cost of a large launch lift budget. Robotic missions can tolerate longer mission times which are normally constrained by the maintenance free life of the spacecraft systems. High specific impulse and low thrust to power may allow the initial lift to be constrained to one or two launches. These factors are likely to be the dominant features in any comparison between nuclear electric propulsion systems for particular missions.

Nuclear propulsion can only be justified where a mission is not achievable with solar electric or chemical propulsion. In principle missions to the inner solar system and Mars can be achieved by conventional means. In practice if nuclear propulsion is developed for missions to the outer solar system it can also be used for missions to Mars and closer NEOs assuming that the recurring costs of a nuclear 'space tug' are competitive with conventional power generation. Possible benefits are the high thrust enabling fast transits offered by nuclear thermal propulsion (NTP) and the multi-journey capability of a nuclear electric (NEP) 'space-tug' once in orbit. Another option is to deliver the infrastructure for a manned mission to Mars in advance of the human arrival using slower but less expensive NEP to enable the most efficient, fast transit of the human cargo once everything for descent to and ascent from the planetary surface is safely in place.

2.2. Missions to Jupiter and Saturn

In the EC FP7 High Power Electric Propulsion; a roadmap for the future (HiPER) project NEP mission analysis (Reference 7) was based on a double Ariane 5 ECA launch: one for the NEP 'space tug' and one for the payload. It was assumed that the NEP 'space tug' and the payload would be attached at Earth Moon L1 and the objectives were to deliver a reasonable payload to orbit around Jupiter and Saturn in a time compatible with a 'sample return' mission. The study of nuclear power generation in HiPER had shown that a Brayton cycle NEP of up to 200kWe was the maximum compatible with the Ariane 5 ECA lift and fairing volume capabilities.

Assuming up to 10 years operation in a 15 year lifetime required a transit time of less than 5 years for the outward journey on the basis that the reactor would be in a very low power or standby state for the period in orbit around the destination planet. Optimal transit and payload delivery was achieved with Isp of 5000-10000s to Jupiter and 10000-15000s to Saturn.

A single launch with a smaller NEP generator may also be considered but specific mass tends to increase significantly with lower power with consequent reduction in the useful payload or increase in trip time or both. Russian plans to develop larger single launch lift (70 or 120 tons to LEO compared to Ariane 5 ECA 20 tons) may offer the prospect of increasing NEP power levels to 1-2 MWe in a single launch as long as there is compatibility with the fairing size and shape.

2.3. Missions to Mars, NEOs and the Sun/Earth Second Lagrange Point (SEL2)

Mission analysis for robotic (sample return) missions to Mars and the closer NEOs indicated that optimal trip time and payload delivery was achieved with Isp ~ 5000s for a one or two Ariane 5 ECA launch strategy. The analysis was based on SEP rather than NEP but it is reasonable to assume that the benefit of NEP constant thrust would be countered by the higher specific mass. Similar results were achieved for round trips to SEL2.

2.4. Costs

Another dominant feature is the non-recurring cost of developing an NEP fission generator. The US Prometheus project cost estimates were in the region of B$10 (B€7.5). The closest comparison for a European project is the EC Allegro (terrestrial) advanced gas cooled research reactor project which is expected to cost several billion Euros over a 10 year development cycle. For a space project one would also have to add all the space qualification aspects which would almost certainly double the cost. From the current costs of ESA Bepi Colombo and Exomars missions it seems unlikely that any recurring mission costs would be less than B€1-2 if only because there would have to be a continuing contribution to maintain the nuclear 'design, build and operate' infrastructure.

It is probably best to compare costs on the 'recurring cost basis' because the development of NEP for a 'one-off' mission looks very difficult to justify. Even then these are very rough because little hard information currently exists. If we consider the double launch scenario for a Jupiter sample return costing B€1.5, for example, the launch costs are likely to amount to 20-

25% based on current prices. Experience with the ESA Rosetta mission suggests that operations costs during these long transits are relatively modest and an overall operations cost of 20% seems reasonable. (Bear in mind that this has to take into account in-orbit commissioning of the NEP generator and operations at the destination planet.) Qualification of the NEP generator is likely to take up a large part of remaining 65-70% because its robustness and resilience must be very thoroughly demonstrated as it has to be commissioned in orbit for safety reasons.

Extrapolation from today's prices suggests that the recurring cost of a 25 kW EP system could be ~ 5 M€ (say M€40 for a 200 kWe NEP system). Then a price variation of ± 20% between EP technologies, for example, at ±M€ 8 represents approximately ±1% of the total mission cost. This is not to be ignored but given the potential error bars on all of these cost estimates it is suggested that any comparison between technologies at this stage must be treated with extreme caution.

In principle each additional launch will add ~15% to the mission costs should these be necessary for additional propellant for higher thrust to power systems. The costs of obtaining propellant from in-situ sources is not estimated but might require levels of energy for extraction which contribute indirectly to the overall mission cost. If operations costs are relatively low during long transits then the extra launches are likely to be the most dominant cost variable.

2.5. Electric Propulsion Technology Options

2.5.1. General Principles

It can be seen that no single electric propulsion technology is likely to be compatible with all possible NEP generator based missions. Higher Isp tends to be needed for the missions to the outer solar system for which NEP may be seen as a unique enabling capability. However lower Isp is probably more suitable for 'spin-off' applications closer to earth. And once the technology is available these could become the more predominant applications. Also whereas Ariane 5 lift compatible power levels in the 200 kWe region may suit 25kW GIEs, HETs and HEMPs a 2MWe NEP generator may operate for efficiently with 100 (or higher) kW MPDs. Equally a 30 kWe NEP generator, for example, could operate with existing GIT and HET or even arcjet systems.

In principle it is possible to design the NEP generator to be compatible with a range of EP technologies and the key to this is the interface which is described as the power management and distribution system (PMAD). This depends on many factors. It must deliver power of the required quality to the EP systems and protect the NEP generator from sudden un-programmed load changes. It must have access to the energy for in-orbit commissioning and cold re-starts as well as controlled power up, power down and safe standby or low power operations. The harness mass efficiency is likely to be a significant factor. It is influenced by the selection of AC or DC power distribution, NEP turbo-alternator and EP PPU design and the architecture determining the distance between the NEP generator and the EP systems.

2.5.2. Comparison Matrix

It would be easier to quantify the relative merits of different technologies and many of the key features compensate in one way or another with a design activity. A principle criteria is mass efficiency but comparisons on dry mass alone tend to ignore the full picture and are only really valid when considered with the wet mass and implications for delivered payload, trip time, etc. If one takes the case of the GIE and the HET, for the same electrical power, the GIE thruster is lower mass than the HET and the PPU normally higher mass; the narrower GIE exhaust plume beam width gives greater flexibility in location on the spacecraft and therefore shorter harness length but that has to be set against the lower voltage HET requirements and simpler electrical configuration. The structural and thermal challenges of integrating the EP systems to the spacecraft are broadly comparable.

The complexity and range of considerations to be taken into account is illustrated by a basic Comparison Matrix in Section 6.

3. Thermal to Electrical Energy Conversion.

3.1. Thermo-Electric.

Metallic thermo-electric generators achieve energy conversion efficiencies in the range of 1-5% but claims have been made more recently for high temperature semiconductor devices to give efficiencies up to 15% (Ref 3). The temperatures required are up to 975°K and a temperature range of 675°K which tend not to be compatible with terrestrial commercial applications but could be developed for a space application. There are significant technical challenges in creating the temperature difference.

3.2. Thermionic.

Thermionic energy conversion occurs when a very hot ‘emitter’ (~1600-2500°K) is close (a few microns) to a ‘collector’ at a significantly lower temperature. Then the electrons will flow to the collector and create an electric current. In principle operation should be in a vacuum but usually a gas (eg caesium) is used to "encourage“ the electron flow (space charge elimination). The entire device, known as a thermionic diode, can achieve energy conversion efficiencies in the range 5-7% but leakage and loss of caesium has tended to a significant life-limiting factor (the design life was 3 years but only one year was demonstrated life in Space, while diodes tested at CEA in the 60’s achieved more than 50 000 hours operation with electrical heating).

3.3. Stirling Cycle.

The Stirling cycle introduces moving parts to increase energy conversion efficiency to >15% with a target of up to 25% and lifetime in excess of 6 years continuous operation (Ref 4). NASA research is based on an input temperature to the engine ~ 950°K and a system mean operating temperature ~ 525 °K which was at the very upper limit for the alternator operating in helium and the fixed magnets. The temperatures are also too high for conventional insulation and highly dependent on ceramic materials. A significant part of the research was to establish the resilience (including creep life) of the high temperature materials, highly efficient thermal design and long life bearings for moving parts. The NASA research was a part of the US SP100 programme aimed at a 100 kWe output although the reference system built delivered 12.5 kWe from 60 kW_{th} in the laboratory, To date Stirling systems are mainly considered in the 0.5 (radio-isotope) – 15 kWe (fission) power ranges.

3.4. Brayton Cycle.

At higher power levels Brayton cycle power conversion offers lower specific mass than thermo-electric, thermionic and Stirling cycle power conversion. The cost of launch and the lift capability of launch vehicles tend to make mass the main design driver for fission nuclear power generators. Counter-intuitively high efficiency is not directly associated with low specific mass.

There is a complex trade-off particularly for Brayton cycle systems with multiple compressor and turbine stages (Ref 5). Specific mass for thermo-electric and thermionic systems tend to exceed 100 kg/kWe. A comparison of specific mass scaling for different types of Brayton system are shown in Figure 1 showing the improvement at higher operating temperatures and higher power output. (In the scaling 'SP100' is based on Stirling; 'Brayton' is reheated direct and 'R Ind' reheated indirect cycle; 'Rec 1200' and '1500' are recuperated Brayton direct cycle at 1200 and 1500 °K turbine inlet temperature.)

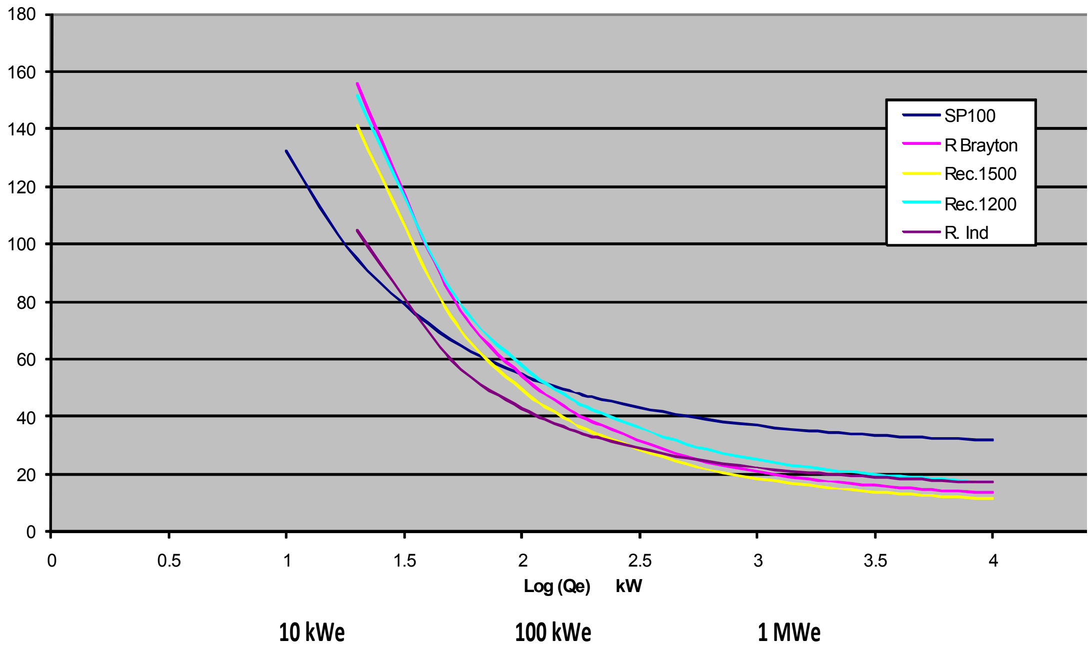

Figure 1 Brayton Cycle specific mass scaling.

The indirect cycle Brayton has the advantages of a smaller and much lower mass liquid metal cooled fast reactor and shield but the additional mass of heat exchangers between the liquid metal and the turbine operating gas. The direct cycle has the simplicity of the reactor coolant being the turbine operating gas but requires a larger reactor and shield for adequate gas flow. Radiator size and mass reduces significantly at higher operating temperatures but turbine inlet temperature above 1200-1300°K is considered very challenging (and impractical for indirect cycle heat exchanger technology). Most recent uncooled helicopter turbines operate at 1300 K but the operating life and therefore all important creep life (less than 10 000 h) is shorter than required for NEP operation. However, the inert atmosphere of the cycle is compatible with materials (niobium alloys, silicides) not allowed for oxidizing atmosphere and operating at higher temperature ~1500 K. For megawatt class applications, a cooled turbine with ceramic coating could be used as in the case of turbojets, enabling temperature around 1800K.

3.5. Rankine Cycle.

The potassium Rankine cycle offers high efficiency in a low mass power conversion system to which heat would be supplied by a liquid-lithium-cooled fast reactor. System complexity is a major disadvantage at low power, but this is potentially the lightest high power system. However, there are major technical issues, including management of two-phase fluid in micro-gravity and design of a potassium condensate feed pump for high-pressure, low-flow performance. The high temperatures that are associated with published descriptions of this concept may not be feasible with existing materials. Lower temperatures would correspond to lower power density and increased radiator size. On the other hand, the recognised advantage of Rankine cycle is the almost constant radiator temperature leading to a much smaller radiator size than in the case of Brayton.

The initial melting of potassium (or other alkali metal) is also challenging. It requires a very large energy. If electrical heating is selected, the energy may be stored in a lithium ion battery or delivered by a hydrazine APU. The mass penalty may be quite high.

A last challenge is the erosion of turbine blades by liquid metal droplets. This effect could be alleviated by using dry steam and a centripetal condensing turbine.

The MHD conversion can be connected to Rankine cycle. It uses liquid metal and metal steam (e. g. caesium steam / liquid sodium). The vapour/ liquid two phase flow is expanded in a supersonic nozzle and the liquid metal fed to a MHD channel with a transverse magnetic field; the mixture is condensed in the radiator. Extensive tests have been performed at CEA with total efficiency around 20%.

3.6. Applications.

Thermo-electric power conversion has the attraction of no moving parts for smaller systems but has high specific mass. Thermionic conversion is considered to have little benefit over thermo-electric and the life limiting by caesium loss to be a significant disadvantage.

Stirling cycle is attractive for power levels in the 10-25 kWe range if all the technical issues of high temperature moving part operation can be demonstrated to be resolved, including any implications for integration into the spacecraft and effects on spacecraft performance. Multiple systems are likely to be a sensible redundancy measure although this will increase specific mass.

Recent studies (Ref 6) suggest that compatibility with current launch lift to a safe operating orbit constrains the Brayton nuclear generator size to ~ 1 MW_{th}/200 kWe. As for Stirling cycle there are a range of technical issues to be resolved for the use of rotating machinery such as torque cancellation indicating a need for multiple machines (the torque cancellation could be effected by a roll control RCS as for an upper stage and is only needed during alternator speed transients). Slow speed variations can be treated by a momentum wheel. A 100 kW turbo-alternator is lighter and has a better efficiency than the two counter-rotating 50 kW units. This also assumes that the advantages of very high operating temperatures can also be achieved. Some research has indicated that better performance may be achievable with Brayton or Rankine cycle but with greater system complexity.

On this basis, the most promising forms of thermal to electrical power conversion are considered to be thermo-electric and Stirling cycle up to 100 kWe and Brayton at higher powers (some US studies suggest Brayton is better than Stirling above 10 - 15 kW). Thermo-electric has a natural DC output but Stirling and Brayton have alternators which can generate AC or DC. Provided thermoelectric (or thermionic diodes or MHD) can provide a voltage in excess of 70 V, a "direct" coupling with MPD thruster can be contemplated. But the compatibility of the generator V-I characteristic with the thruster V-I characteristic shall be verified. For other thrusters (HET and ion) a fairly heavy PPU is required (insulation transformer), On the other hand, Stirling and Brayton voltage output can be tailored to the input voltage of the HET or ion thruster, a simple diode bridge rectifier and filter is sufficient to provide the thruster input voltage. A transistor bridge allows for voltage regulation. No insulation transformer is required as the alternator output is floating. Thermo-electric is relatively unaffected by a sudden load loss. Brayton rotating machinery will not tolerate an over speed of 10% or greater or an overload larger than 100%. Stirling is also vulnerable to sudden load loss.

The role of PMAD will be to avoid these detrimental effects. For example, the PMAD can include a set of resistors, transistor controlled, able to provide a load the alternator in case of load loss. The same circuit could be used for the starting and shut-down transient (the alternator shall be set to it nominal speed before switching thrusters on).

The PMAD design will be different if the alternator is a permanent magnet one or if the rotor includes excitation coils. In this last case (similar to power plant alternators) the regulation is much easier.

4. Electric Propulsion Input Requirements.

4.1. Background.

The requirements quoted here are based on published data for existing electric propulsion systems and those identified for future high power electric propulsion in the HiPER project (ref 7).

4.2. Gridded Ion Engines (GIE)

For nuclear electric applications GIEs may be considered in the 5 kW 3000-5000s Isp and the 20-25 kW 10000-15000s Isp range. For the former the PPU generates high voltage in the 1.6-1.8 kV and low voltage in the 30A/35V range (Kaufman) or an RF generator of similar power level (RIT). For the higher power, higher Isp the High voltage increases to 6-8 kV. Normally power supply input stability of ~ ± 3% will be specified but the processing to generate high and low voltages will normally take care of power supply variation. In principle power input can be AC or DC and be anywhere in a range between hundreds of volts and several kV.

Since the beam power is by far the largest part of the power, it is advisable to use a direct rectified source for this function (elimination of heavy transformers). On the other hand, the weight penalty is acceptable for discharge voltage and auxiliary functions (cathodes, accelerating grid).

4.3. High Energy Magneto Plasma (HEMP)

HEMP discharge characteristic is very similar to the HET one, but the discharge voltage is higher (typically 1 kV instead of 400 V). Demonstrated power is up to now 5 kW. For the purposes of comparison in general terms the HEMP will be considered part of the HET family.

4.4. Hall Effect Thruster (HET)

For nuclear electric applications HETS may be in the range of 5kW 2000-3000s Isp and 20-25 kW 2500-3500s Isp (for xenon). Works are under way to allow operation with krypton and argon leading to higher specific impulse for a given discharge voltage. The PPU (or input circuitry) generates 200V to 400 V for the lower range and 400 to 800V for the higher power. The engine is tolerant of operation with an unregulated power supply but in practice a limit of ~ ± 3% is advisable to limit thrust fluctuation. The discharge characteristic being "vertical", a voltage source is naturally stable with the discharge. Ideally the input power voltage will be DC and match that of the main power supplied to the thruster (ie 200 to 800 V, depending on the point selected for the thruster operation).

As in the case of ion propulsion, it is advisable to set the alternator voltage to suit the discharge voltage and use insulation transformers for the other functions (magnet supply if any, cathode, valves)

4.5. Magneto Plasma Dynamic (MPD)

For nuclear electric applications MPDs may be in the range of 100kW or 1000kW. The main power supply to the thruster is at 50 or 150V (higher power) to the cathode. The need for power input stability is not fully defined at this stage but an assumption common to that for GIEs and HETs is probably prudent. In principle the PPU will be simpler if the input voltage is DC and matches the main 50 or 150V supplies to the thruster.
A PWM regulator may be needed in this case as the MPD discharge characteristic may not be compatible with the source characteristic. This PWM unit may for example perform current limitation. Its efficiency may be as high as 97%. Bearing in mind that the discharge current may reach 1500 – 2000 A for the 100 kW case, it could be necessary to split the PWM into 20 units in parallel, each handling 100 A current.
The DC power source may be either:

- Thermo electric,
- MHD,
- Homopolar generator (liquid metal electrical contacts).

100 kW MPD are generally of the applied field type, so a different power supply may be needed for the electromagnet.

4.6. Other Considerations.

One of the objectives in considering the electrical power systems is to avoid duplicating functions in the electrical power generator, the management and distribution or in the electric propulsion 'load' and the PPU. For a thermo-electric generator regulation can be largely achieved with conventional electrical principles and a large battery and a DC supply at 100-200V would be a reasonable expectation. Also at lower power levels harness mass is less likely to be a significant issue.

It is necessary to find a reliable mechanical switch to control the 1000 to 2000 A DC current with an acceptable mass penalty.

In the case of thermoelectric power, the switching problem is simplified by the fact that many thermo elements strings are provided in parallel (each element may provide less than 10A) but the problem is more difficult at PMAD level (each thermo element string delivers a voltage slightly different compared to the other ones, so a regulation unit is needed for each string (current regulation).

Regulation for Stirling and Brayton cycle is more complicated because some regulation is achieved by controlling the torque (and speed) of the moving machinery.

In the case of Brayton, the short term (a few seconds) regulation could be performed by a by-pass valve acting on the turbine. The speed variation is expected to be small between power idle and full power. The alternator load defines the turbine torque demand, itself set by the by-pass valve. The longer term power demand could be modified by acting on the gas temperature (reactor power).

In the case of Hirn-Rankine, the short term regulation is generally provided by a valve upstream of turbine and the longer term regulation by setting the reactor temperature (vapour pressure). These systems are more vulnerable to sudden load changes and require power to start them from cold (e.g. an inverter could provide 3 phase AC power to “launch” the turbo-alternator to its minimum self-operating speed).

The PMAD shall provide fast protection against over-speed (electrical ballast brake), this system could be used also for starting transient (the alternator is launched without EP load and the alternator power delivered to the ballast is regulated according to speed limitation.

5. Power Management and Distribution (PMAD)

5.1. Range of Services.

The PMAD supplies high power to the main payload which may be propulsion systems, a ground penetrating radar, a powerful laser, high data rate inter-planetary communications, mining equipment or general power supplies to a planetary outpost. At the same time a range of low power supplies are also normally required. These can range from normal spacecraft services for communication, operating ancillary equipment and the control of spacecraft functions. One would expect these services to be less than 10% of the generated power.

Stirling, Brayton and Rankine power conversion is vulnerable to sudden load changes if not protected by a ballast or a suitable regulation. In the case of helicopter turbines, the engine shall survive the over-speed induced by the loss of main gearbox torque (this is affected by the regulation). Regulation response time shall be less than 10 ms. A Brayton turbo-generator for example will not tolerate > 10% over-speed. Heavy load devices with large fluctuations in load are therefore best suited to thermo-electric power conversion (or MHD) unless integration with a large energy sink or battery is possible to compensate in delays in thermal decay.

Stirling and Brayton power conversion systems require emergency shut-down systems to cope with sudden, unplanned large load loss or overload (e. g. electrodes shorting). In which case ancillary systems could probably rely on a large battery for some time.

5.2. Regulation

Thermo-electric power conversion is relatively insensitive to load fluctuations and power regulation requirements will be mainly determined by the input requirements of the services being powered. The battery size is most likely to be determined by the energy required for in-orbit commissioning and re-start.

Stirling and Brayton turbo-alternators offer the choice of AC or, if rectified, DC power supply. The lower mass of AC harnesses has to be traded against the mass of high power rectification in the supplied service. A degree of regulation is provided by control of the torque (and to a lesser extent of the speed) of the rotating machinery (ie ~5% or better to give a good margin against over-speed). The level of further regulation required depends on the input requirements of the services. The size of battery required for in-orbit commissioning and re-start may well avoid the need for additional regulation devices.

5.3. Harness and Power Transmission

High power distribution can incur very significant mass penalties particularly if there is some distance between the generator and the main load. Conventional designs have put the reactor at one end of the spacecraft and the electric propulsion, for example, at the other. In the example in HiPER with 10m from the power generator to the start of a 22.5m boom, attaching the payload module, and a further 10m to the rear of the payload capsule some 50m of harness would be required allowing for routing. For a 2000 A maximum current (100 kW AF MPD), a 2 cm2 copper section is required, leading to a copper mass of 90 kg without insulation and without cooling (active cooling will be necessary in space to reduce the copper losses). Flexible couplings would be required for boom deployment and a conservative estimate suggested a 200 kWe DC harness mass ~ 400kg.

The mitigating options are:
- Investigate mass savings from using AC rather than DC power distribution,
- Investigate novel distribution technologies such as high temperature super-conductors,
- Investigate ways to put electric propulsion and the reactor at the same end of the spacecraft (requires thrusters with very narrow exhaust plumes).

5.4. Environmental Constraints

Up to 80% (90% for thermo-electric) of the reactor power must be dissipated as waste heat although some of that waste heat can also be used for habitability purposes such as maintaining adequate temperature in manned planetary outposts. Some is also required to keep the spacecraft temperature at a reasonable level in deep space. However most of it must be dissipated with a large radiator and the spacecraft will tend to be at a high temperature compatible with equipment such as electronics, control mechanisms and propellant tanks. Electrical equipment forming part of Stirling or Brayton turbo-alternators will probably need additional cooling to keep the temperature below 470K°.

A third radiator is required for the PMAD and PPU cooling. The preserve the electronic components reliability, a 300 K radiator temperature is advisable. Taking a fairly optimistic total efficiency of 95% for PMAD and PPU, the dissipated power will reach 10 kW for the 200 kW case. The corresponding radiator surface will reach 20 -30 m^2.

Electrical PMAD equipment may therefore be expected to need qualification for both high power and high temperature with an increasing reliance on ceramic insulation.

Conventional designs tend to be based on a combination of shielding and separation (normally by a boom) to ensure tolerable radiation levels at the payload. The power generation and much of the PMAD is likely to be much closer to the reactor and may even form part of the shielding strategy.

However, existing “rad-hard” civil use electronic components are limited to 100 krad, i. e. incompatible with a location close to reactor.
Although large electrical passive devices may be relatively unaffected by radiation smaller control devices can be quite vulnerable. Even then the resilience of the larger devices to radiation

effects will need to be investigated and almost certainly some form of additional local shielding provided for more vulnerable devices.

Another feature of rotating machinery is the creation of unwanted torques. Former Brayton cycle conversion projects (e. g. ERATO) involved contra-rotating arrangements to avoid this normally in the form of even numbers of turbo alternators. More recent projects, like OPUS, used a single turbo-machine associated to a RCS and a momentum wheel. RCS is anyway absolutely necessary for initial deployment and pointing.

5.5. Commissioning and Cold Starts

The launch a fission reactor in a critical state is very unlikely to be acceptable. In-orbit commissioning in Earth orbit below 800km is also thought to be unacceptable. A lot of energy is required to start a nuclear reactor from cold in orbit (this energy is an order of magnitude higher for liquid metal cases). This may be mitigated to some extent if the structure is preheated until just before launch. If not calculations for a 200kWe gas cooled reactor indicated a requirement for up to 40kWhr to bring the reactor to 10% of power and achieve self-sustaining operation (~ 3.3 kWhr). An LM reactor is likely to require at least twice as much because the whole surroundings structure has to be raised to a temperature at which the LM will liquefy. This may be slightly easier with heat pipes.

There may be alternatives to a very large battery such as gas generators. However if the battery forms other useful functions such as regulation and compensating for sudden load shifts it may be the best option. Also a one shot gas generator would not cope with a cold re-start later in the mission (a monopropellant or bipropellant gas generator is preferred). Combinations of batteries and other energy sources can look superficially attractive but introduce unwanted complexity and risk.

5.6. Controlled and Emergency Shut Down

Another use for a large battery is to manage large changes in load for Brayton cycle power conversion. In the case of alternators with electrical rotor excitation, the problem can be simply solved by a constant speed alternator and adapt the excitation to the load variation. However, this is not feasible with permanent magnets alternators, the sudden load variation shall be smoothed by the ballast. The current delivered to PPU will be monitored and the ballast current set to cancel fast variation of alternator current. This will be effected by a PWM current controller to ballast (response time: a few ms). Power up : the ballast load will be progressively increased before starting a thruster. Unexpected power down (thruster switch-off) : the ballast power will replace the sudden power loss and be progressively decreased. Individual thrusters can well exceed 10% of the overall load and even if shut down individually could cause the risk of over-speed. To some extent the EP systems can be throttled but the role of the battery (or ballast) in managing step changes is almost essential. With an AC bus, the optimum battery location is between each rectification bridge and thruster i. e. inside each PPU. There is also the complication of keeping turbo-alternators operating at similar speeds to avoid unwanted torques (hence the interest of torque cancellation by RCS momentum wheel combination).

Both the LM and gas cooled reactor designs in the HiPER study could be operated for reasonable periods at about 10% full power. The design envisaged a by-pass valve on the turbo-alternator to prevent over-speed in the event of sudden load loss. Emergency shut-down arrangements for total sudden unplanned load loss were based on spring loaded insertion of the control rods. The initial thermal decay is quite rapid and it was envisaged that the battery could restore 10% of full power within several hours of the emergency shut-down. The turbo-alternator can act as a freewheeling fan to eliminate the radioactive decay heat even if no electrical power is produced.

6. Conclusion and Evaluation Matrix

The main conclusion is that there are many factors to take into consideration, many of which are mission dependant. Consequently, an awareness of the principle advantages and disadvantages of the various technical options is the best that can be achieved at this stage. Also many features of the technology options are to some extent overlapping and have significant variations in potential performance.

Another aspect to consider is that performance measurement error for more advanced technical options can be significant. Laboratory thruster test results have been known to be very different to subsequent in-orbit performance. Although techniques for improving in-orbit thruster performance measurement have improved many system aspects, particularly high power electrical processing and conditioning, are at an early stage of research.

It is also important not to underestimate the cost and time to develop a prototype technology to a mission ready capability. For the purposes of this comparison it is assumed that these factors will apply equally to all technical options.

The Basic Evaluation Matrix is in Tables 1 and 2. Table 1 gives the evaluation criteria taking account of general principles, EP system characteristics and propellant considerations. A column is allowed for weighting but at this time all the weights are unitary. It is not considered possible to ascribe meaningful weighting to the general overview. Weighting may be introduced at a later stage when considering a specific mission and where the design drivers are specified to sufficiently detailed criteria. Even then this is a risky technique where there is a danger that results may be manipulated to achieve a desired result rather than a true comparison.

Table 2 offers a very top level of the characteristics of the HET, GIE and MPD families for Medium, High and Very High Isp classes of missions. At this stage one can only draw the following rather general conclusions:

- Missions to the outer planets, or longer duration return journeys, either need higher Isp, to achieve acceptable propellant budgets, or require new propellants with greater abundance and lower cost. (No account is taken of the additional cost of multiple launches for high wet mass missions.)
- New propellants may not have as beneficial characteristics as those in current use (eg xenon) and therefore in turn affect performance and cost.
- At high level there is considerable similarity between the application of technologies for medium and high Isp operation and there only tends to be a difference at very high Isp.
- Lifetimes for longer duration missions may be met with EP system redundancy or longer operating life. New qualification methods will almost certainly be needed for the latter and are already being pioneered by ESA for GOCE and Bepi Colombo missions. The target for a nuclear power operating life is currently targeted at 10 years (within a 15 year mission timeline allowing for periods when full power is not needed).

It is recommended that Table 2 is viewed as illustrative rather than definitive and therefore a starting point for more detailed analysis for specific mission scenarios.

COMPARISON CRITERIA FOR NUCLEAR ELECTRIC PROPULSION				
CRITERIA	**WEIGHT**	**CHALLENGING OR UNSATISFACTORY**	**ACHIEVABLE OR ACCEPTABLE**	**OPTIMAL OR BEST**
GENERAL				
TRL	1	1-3	4-6	6-9
Complexity	1	Many sub-systems and regulation	Some sub-systems and regulation	Few sub-systems and regulation
Lifetime (years)	1	>10	5-10	05-Oct
Redundancy	1	Fully Redundant	Critical Functions only	No single point failure
Safety:	1	Multiple high hazard levels	Fewer hazards	Minimum hazards
- Toxicity,	1	√		
- Explosive,	1	√	√	
- Electrical	1	√	√	√
- Radioactive	1	√	√	√
Dry EP System Cost (k€/kW)	1	>20	5-20	<5
PERFORMANCE				
Dry Specific Mass (kg/kW)	1	>20	10-20	<10
Thrust/Area (N/m2)	1	<1	1-25	>25
Thrust/Power (kW/N)	1	>100	100-30	<30
Exit Velocity (km/s)	1	<1	1-2.5	>2.5
Efficiency %	1	<30%	30-60%	>60%
Clustering ability	1	Serious problems	Manageable problems	No problems
Thrust range %	1	<10%	10-50%	>50%
Isp range %	1	<10%	10-30%	>30%
PROPELLANT				
Availability	1	Limited supplies	Good supplies	Abundant
Cost (€/kg)	1	>100	100-10	<10
Tank Mass (% of wet mass)	1	>15	5-15	<5
Evaporation Temperature (°K)	1	>1000	500-1000	500-250
Contamination	1	Major problems	Manageable problems	No problems

Table 1 **Comparison Criteria.**

CRITERIA	LOW ΔV			MEDIUM ΔV			HIGH ΔV			COMMENTS
	Near NEO Sample return			Mars Sample return			Jupiter of Saturn sample return			From C^3 Low 1-5 km/s, Medium 5-15 km/s, High >25km/s.
	Mars one way			Jupiter or Saturn one way			Kuiper belt one way			
SPECIFIC IMPULSE (I_{sp})	Medium	High	Very High	Medium	High	Very High	Medium	High	Very High	Medium 1500-2500s, High 2500-8000s, Very High 8000-15000s
TECHNOLOGY	HET[1], MPD	HET[2], GIE[1]	GIE[2]	HET[1], MPD	HET[2], GIE[1]	GIE[2]	HET[1], MPD	HET[2], GIE[1]	GIE[2]	HET[1],GIE[1] First generation; HET[2], GIE[2] second generation.
GENERAL										
TRL	*** *	** ***	*	*** *	** ***	*	*** *	** ***	*	
Complexity	** **	** **	**	** **	** **	**	** **	** **	**	Inludes thruster, pointing mechanism, PPU and porpellant system
Lifetime (years)	** **	** ***	***	** **	** ***	***	** **	** ***	***	Length of achievable design performance
Redundancy	** **	** **	**	** **	** **	**	** **	** **	**	
Safety:	** **	** **	**	** **	** **	**	** **	** **	**	High risk *, Medium Risk **, Low risk ***
- Toxicity,	*** ***	*** ***	***	*** ***	*** ***	***	*** ***	*** ***	***	Possibly in very high power insulation.
- Explosive,	*** ***	*** ***	***	*** ***	*** ***	***	*** ***	*** ***	***	Only in pointing release mechanisms if inert gas propellant
- Electrical	** **	** *	*	** **	** *	*	** **	** *	*	Very high power levels
- Radioactive	* *	* *	*	* *	* *	*	* *	* *	*	
Dry EP System Cost (k€/kW)	** **	** **	**	** **	** **	**	** **	** **	**	Broad estimates from current and past programmes
PERFORMANCE										
Dry Specific Mass (kg/kW)	** ***	** **	**	** **	** **	**	** **	** **	**	MPD waste heat needs large cooling system
Thrust/Area (N/m2)	** ***	** **	***	** ***	** **	***	** ***	** **	***	
Thrust/Power (kW/N)	*** **	** **	*	*** **	** **	*	*** **	** **	*	
Exit Velocity (km/s)	* *	** **	***	* *	** **	***	* *	** **	***	Along the thrust axis
Efficiency %	* *	** **	***	* *	** **	***	* *	** **	***	Ratio of the energy in thrust on thrust axis to EP system energy input.
Clustering ability	* **	** ***	***	** **	** ***	***	** **	** ***	***	Highly dependent upon clusterint arrangements
Thrust range %	*** **	** **	**	** **	** **	**	** **	** **	**	Wider thrust range may have a lifetime penalty
Isp range %	** **	** **	**	** **	** **	**	** **	** **	**	Wider Isp range may have alifetime penalty.
PROPELLANT										
Xenon use (% world stock)	* *	** ***	***	* *	** **	***	* *	* *	**	Exceeds supply *, Exhausts supply **, Supply adequate *** .
Xenon mission cost (€/kg)	* *	** **	***	* *	* **	***	* *	* *	**	Product of unit cost and amount used
Xenon Tank Mass (% wet mass)	** **	** **	**	** **	** **	**	** **	** **	**	
Argon use (% world stock)	*** ***	*** ***	***	*** ***	*** ***	***	** **	*** ***	***	Exceeds supply *, Exhausts supply **, Supply adequate *** .
Argon mission cost (€/kg)	** **	** **	**	** **	** **	**	** **	** ***	***	Product of unit cost and amount used
Argon Tank Mass (% wet mass)	** **	** **	**	** **	** **	**	** **	** **	**	
Evaporation Temperature (°K)	** **	** **	**	** **	** **	**	** **	** **	**	Argon and xenonevaporation temperatures below 700°K.
Contamination	*** ***	*** ***	***	*** ***	*** ***	***	*** ***	*** ***	***	Neither argon or xenon give contamination

Notes: Optimal or Best is indicated by ***, Achievable or acceptable is indicated by **, Challenging or Unsatisfactory is indicated by * .

Table 2 **Basic Evaluation Matrix.**