NUCLEAR SPACE PROPULSION WITH A PURE ELECTRO-MAGNETIC THRUST.

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Abstract.

The exploration of the planetary system, one of the goals of our civilisation, implies journeys over billions of km. The performance of an engine for deep space travel is fundamentally different of the one of today’s rockets. Rather than a very large thrust, as the one needed for lift-off, one must produce a relatively small force, but persistent during most of the necessarily very long mission.

Nuclear power is the most suited energy source, since the energy produced for unit fuel mass about $10^6$ times larger than the chemical one. In the fission process, a fraction $\Delta m = \chi m$ of the fuel mass ($\chi \approx 10^{-3}$) is transformed directly into thermal energy, $E = \Delta m c^2$. If this energy is in turn emitted in the form of directionally focussed, Stefan-Boltzmann spontaneous electromagnetic radiation, the resulting, recoiling kinetic impulse is equal to the one of a matter propelled rocket, in which the mass $\Delta m = \chi m$ is ejected with the speed of light, $c = 300000$ km/s. The spacecraft is expelling pure energy in the form of massless photons, instead of particles of finite mass, without the need of a propellant material. The total nuclear fuel mass being the equivalent of the classic, propellant mass, the effective specific impulse is $I_{sp} = 30600$ s, about 68 times the one of the best chemical rocket.

In analogy with "solar sails", the method is based on radiation reaction: however solar radiation is now replaced by a "lamp", made of a very hot plate heated by nuclear power. The engine is very simple, since it consists of a naked critical structure cooled by its own radiation.

A number of exemplar missions have been considered, starting from a "coasting" earth's orbit. Evidently the use of nuclear power is limited for environmental reasons to the journeys in interplanetary space and spiralling to/from target orbits. It is concluded that the potentialities of this propulsion method, once fully developed, may be such as to achieve capture in a suitable orbit of almost any celestial body of the solar system, including the larger planets (Jupiter, Saturn and Uranus) and their satellites. For a great deal of cases, and specifically in the case of Mars, also round trips without refuelling are possible. With refuelling, the spacecraft can be made reusable indefinitely.

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1.— Need for new propulsion methods.

The propulsion in empty space so far has been based almost exclusively on

i) the ejection of a high speed propellant, which according to the third’s Newton’s law produces a reaction force on the spacecraft.

ii) In addition, energy is directly provided by the chemical reaction involving the propellant itself.

Such a method has been very successful in the conquest of space. But substantial limitations begin to emerge in the cases of long distance travel in outside the earth’s attraction (deep space), as soon as one considers the exploration of other planets of the solar system and of their satellites.

Present chemical propulsion methods — although with a substantial thrust force — generate a exhaust speed of the propellant, \( v_{exh} \) which is too small, when compared to the velocity change \( \Delta v \) required by the mission\(^1\). The mass of propellant \( m_{prop} \) needed to produce a velocity change \( \Delta v \) for a final mass \( m_f \) is given by the rocket equation:

\[
m_{prop} = m_f \left[ \exp(\Delta v/v_{exh}) - 1 \right].
\]

When the mission requirements become such that \( \Delta v > v_{exh} \), the need for propellant grows exponentially, becoming quickly prohibitive. In order to reduce the amount of propellant initially on board, one must increase \( v_{exh} \).

In a given mission, what is generally specified is the time profile of the engine’s thrust force, \( F(t) = v_{exh}(dm/dt) \): from the formula, as already pointed out, a higher \( v_{exh} \) implies a smaller propellant mass rate \( dm/dt \) for a given thrust force. However the power to be generated by the engine, \( P = v_{exh}F(t) \), for a given force, grows proportionally to \( v_{exh} \). Therefore smaller propellant mass rate is not without demands, since it requires a correspondent increase in the total propulsion energy for a given mission. The specific energy to be provided to unit propellant weight is therefore growing like the square of the specific impulse, \( dE/dm = v_{exh}^2 \). Chemical power can only provide a limited specific impulse with the specific energy \( dE/dm \) is generated by the chemistry of the propellant itself and instead less propellant means less energy. Typically, for optimal H\(_2\)-O\(_2\) burning, \( v_{exh} \approx 4.4 \) km/s (\( I_{sp} \approx 450 \) s). In order to exceed such a limit the option ii) above should be abandoned.

The expected performance of an engine for deep space travel is fundamentally different of the one of today’s rockets. Rather than a very large thrust, as

\(^1\) Instead of the exhaust speed, it is usually quoted also the so called specific impulse, with the dimension of a time, defined as \( I_{sp} = v_{exh}/g \), where \( g \) is the gravitational constant (it represents the time duration over which one can generate a thrust equal to the propellant’s used weight).
the one needed for lift-off, one must produce a relatively small thrust, but persistent during most of the necessarily very long mission. Here the typical accelerations to be impressed are to be compared with the one produced by the Sun attraction at a few astronomical units and accelerations of the order of $a \approx 10^{-3}$ m/s$^2$ are sufficient, rather than of one at the earth surface attraction ($a = g = 9.81$ m/s$^2$) which is determinant at the lift-off. For instance an acceleration as small as $a = 10^{-3}$ m/s$^2$ during a time of $t = 10^7$ seconds (115.7 days), produces the considerable final speed change of $\Delta v = at = 10$ km/s. Such an acceleration will be produced by a tiny force of 1 Newton exercised on a mass of 1 ton. This has profound implications on the way in which the engine is designed and operated for long journeys in the interplanetary space.

We have shown that the primary energy density to be transferred to the propellant in the form of $dE/dm$ is rising as the square of $v_{eh}$: for $v_{eh} \gg 4.4$ km/s the most viable, primary energy source is Nuclear Energy. If on earth, nuclear energy is presently competing with other methods, this has some unique features which make it practically indispensable for any major conceptual advance in deep space travel. Provided an almost complete nuclear burning is achieved, nuclear fuels may provide an energy per unit mass in excess of $10^6$ times the chemical ones. For instance only a few kilograms of nuclear fuel generate an amount of energy which is many times the one produced by the largest existing chemical rockets. With Fusion or Fission, a fraction $\chi = 10^{-3}$ of the fuel mass is transformed directly into energy according to the classic relation $E = mc^2$.

The perception of its “risk” — one of the major drawbacks of nuclear power — has a completely different value for instance in a manned exploration of Mars. This added “nuclear” risk has to be compared to the other risks of the mission, for instance the long exposure to an intense and often unpredictable (solar flares) ionising radiation of solar origin. In order to ensure that a fall-out is on Earth is impossible, nuclear devices should be exploited exclusively when in vacuum and at a safe distance. We believe that the use nuclear power will have positive effects in reducing the over-all risk, making the over all mission safer, cheaper and faster.

However the straightforward transformation of the thermal energy produced by a nuclear reactor into thrust of an appropriate (non reacting) propellant — if $v_{eh}$ has to be in excess of the one corresponding to the highest temperature of a solid fuel$^2$ — will require the two step conversion of heat into electricity,

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$^2$ The NERVA and similar devices exploit directly the nuclear heat in order to heat up the (hydrogen) propellant. The maximum temperature is limited to about 3000 °C because one has to maintain the mechanical solidity of the fuel elements. The corresponding specific impulse, mainly because of the light atomic mass of hydrogen, is $I_{sp} \approx 900$ s.
followed by some form of electric propulsion. It is very hard to produce in space vast amounts of electric power with conventional turbine-generator technology, since the only way to eliminate the excess heat of the thermal cycle is by radiation. Such a nuclear powered rocket will be most likely very heavy, costly and complicated, involving for instance an unprecedented number of fast moving components. In addition, ion propulsion, which can provide a high specific impulse is best suited for very small thrusts, insufficient to propel a spacecraft of many tons to distant planets.

2.— Space propulsion with a pure electro-magnetic thrust.

The almost unlimited energy availability of nuclear reactions suggests also abandoning also point ii) above. We consider here another method based on radiation emission recoil in the spacecraft is expelling photons instead of particles of finite mass. This method of propulsion by radiation has no need of carrying along the propellant material, with the added advantage of an outmost simplicity.

A photon of frequency $\nu$ has an energy $E_\nu = h\nu$ and carries a momentum $p_\nu = h\nu/c$, where $c$ is the speed of light and $h$ the Planck constant. Therefore $E_\nu = c p_\nu$, which signifies that the photon rest mass is zero.

Momentum conservation permits to state some simple rules. If the photon is absorbed at a surface, it transmits its momentum $p_\nu$ to the surface. If a photon is reflected by a mirror after an impingement with angle $\alpha$ with respect to the normal, it transfers a momentum $2 p_\nu \sin(\alpha)$ to the mirror. If a light source emits a photon, a momentum $p_\nu$ is transferred to the source.

The power emitted by the source must however be very large, because of the small coefficient between impulse and energy, $p_\nu = E_\nu/c$. For instance if a normal flash light were switched on in space, it would reduce its mass by a fraction $\chi = 10^{-11}$ and reach a terminal velocity of $10^{-4}$ m/s. However we shall show that direct nuclear heat used to radiate energy at the place of the filament, in view of the high temperature and of the released power, is largely sufficient in order to propel a multi-ton spacecraft in a number of ambitious missions in the solar system.

Sun light recoiling off a large mirror surface has been proposed in order to “sail” on solar “wind” in some long interplanetary missions. The principle diagram is shown in Figure 1. At the optimal incidence angle ($\alpha = 0$) a thrust force points in the direction away to the Sun. Different incidence angles are possible,
however at expense of thrust, since \( F(\alpha) = 2\Phi S \cos^2(\alpha)/c \), where a \( \Phi \cos(\alpha) \) term comes from the reduced solar flux on the (inclined) reflecting surface and another \( \cos(\alpha) \) from the vector composition of Figure 1. The thrust force must have at all times a radial component always pointing away from the Sun. These constraints, in addition to the weakness of the thrust, seriously complicate the planning of the missions, limiting them in practice to outbound missions from earth.

Instead, radiation generated by a source on board of the spacecraft, as discussed in the present note, can evidently be pointed to any direction.

A parallel beam of photons of total energy \( E = \sum E_{\nu} \) is emitted by a source, the generated thrust force is given by

\[
F = \frac{d}{dt} \sum p_{\nu} = \frac{1}{c} \frac{dE}{dt} = \frac{1}{c} W
\]

[1]

where \( W \) is the power of the beam. The thrust force to power ratio of propulsion by radiation is therefore simply

\[
\frac{F}{W} = \frac{1}{c}
\]

[2]

At the earth’s radius the solar flux is \( \Phi = 1.35 \text{ kW/m}^2 \). A solar “sail” of surface \( S \), optimally oriented, will give twice the thrust force of Equation [1], namely \( F = 2\Phi S/c = 9 \times 10^{-6} S \text{[m}^2\text{]} \), pointing off the Sun’s direction. In order to produce \( F = 1 \text{ N} \), we need \( S = 0.1 \text{ km}^2 \). Although very nice, such a method of propulsion is not without difficulties!
In order to overcome the insufficient amount of solar radiation, we consider here a much stronger radiation source made of a very hot nuclear reactor block. The emitted by the block light must be redirected in a roughly parallel beam of light in order to produce a finite thrust force.

In the process of a complete nuclear fission of an appropriate fuel, a fraction $\Delta m_{nf} = \chi m_{nf}$ of the fuel mass ($\chi \approx 10^{-3}$) is transformed directly into energy, $E = \Delta m_{nf} c^2$. The produced power is therefore $W = \dot{m} c^2$, where $\dot{m}$ is the fissioned mass for unit time. If such power is transformed in parallel radiation without losses, according to Equation [2], the trust force will be $F = \dot{m} c$. In the case of a chemical rocket with a combustion rate of propellant $\dot{m}_p$ and exhaust speed $v_{exh}$, the corresponding expression is $F = \dot{m}_p v_{exh}$. Therefore the performance of propulsion by radiation is equivalent to the one of a matter propelled rocket in which the fissioned fractional mass $\Delta m_{nf} = \chi m_{nf}$ is ejected with the speed of light, $c = 300000 \text{ km/s}$.

Realisation of a significant thrust requires a huge power. For instance in order to produce 10 Newton of thrust force, according to Equation [1], the radiative power must be $W = 3 \text{ GWatt}$, an ordinary industrial nuclear reactor. By itself, producing such an amount of nuclear thermal power does not seem too extravagant, in the sense that it is well within of the potentialities of nuclear energy. Nuclear technology permits to conceive very compact critical structures which can produce extremely high power densities with a relatively simple fuel-moderator configuration. Large powers are on the other hand customary in space programme. For instance, in order to generate a thrust $F \approx 10^7 \text{ N}$, (1020 ton weight), the engine must develop an instantaneous power of $W \approx 30 \text{ GWatt}$.

Both the nuclear fuel and the propellant represent elements which enable the transport of the engine and of the payload. In order to evaluate more realistically the mass budgets, it would be appropriate to compare the nuclear fuel mass $m_{nf}$ with the propellant mass $m_p$. The performance of propulsion by radiation is then similar to matter propulsion in which the full mass of the nuclear fuel $m_{nf}$ is used instead of the propellant mass $m_p$ and with the effective exhaust speed $v_{exh} = \chi c$ (a specific impulse $I_{sp} = \chi c/g$). For $\chi = 10^{-3}$ we find $v_{exh} = 300 \text{ km/s}$ ($I_{sp} = 30600 \text{ s}$), about 68 times larger than the one of the best chemical rocket based on H$_2$-O$_2$ burning.

The mass of a chemical rocket is decreasing during the journey due to the emission of propellant. In these conditions, the well known “rocket equation” gives the surviving rocket mass as a function of the attained speed in units of the exhaust speed $v_{exh}$. If the spent nuclear fuel is disposed in space after frequent

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3 This is an optimistic assumption, since it assumes that the whole nuclear fuel is actually fissioned.
refuelling, the rocket equation will remain valid, with the mass of the nuclear fuel used and gradually left in space replacing the mass of the ejected propellant. Notwithstanding, in view of the very high effective exhaust speed \( v_{\text{exh}} = \chi c \approx 300 \text{ km/s} \), compared with the much smaller velocity changes required for interplanetary missions, the effect of releasing the spent fuel during the mission has small consequences. It is a good approximation to assume that the mass of the radiation propelled rocket will remain constant during the journey. Note that the relativistic mass reduction due the fuel mass transformed directly into energy is very small, only \( \chi \) times the nuclear fuel mass.

3.— Conversion of nuclear energy into radiation.

The most immediate conversion of on board produced nuclear thermal energy into radiation in space is achieved automatically by radiation emitted by an opaque surface of a high temperature body, heated in its inside by nuclear fission. Since is no other outlet to the produced energy, the temperature of the body will rise to the equilibrium condition between radiation emission and energy production. Therefore, in stationary conditions, the conversion will be absolutely complete.

The radiation emitted at a wavelength \( \lambda \) by a “grey” surface of absolute temperature \( T \) and emissivity \( \varepsilon < 1 \), at an angle \( \alpha \) from the normal direction, is given by

\[
dW = \varepsilon(\lambda)L_\text{e}(\lambda,T) \cos \alpha \ d\lambda dS d\Omega,
\]

where

\[
L_\text{e}(\lambda,T) = \frac{dL}{d\lambda} = \frac{2hc^2\lambda^{-5}}{\exp(hc/kT\lambda)-1}
\]

in which \( h \) and \( k \) are respectively the Planck and Boltzmann constants. The function \( L_\text{e}(\lambda,T) \) has a maximum for \( \lambda_m = (2897 \text{ K})/T \mu \text{m} \). The solar radiation is well approximated to the thermal radiation of a body at \( T = 5900 \text{ K} \), corresponding to a peak wavelength \( \lambda_m = 0.49 \mu \text{m} \). As shown in Figure 2, the light spectrum emitted by a body at \( T = 3000 ^\circ \text{C} \) will therefore have the same general shape of the solar spectrum but at wavelengths which are about twice as long, peaked at \( \lambda_m = 0.88 \mu \text{m} \).

The radiated power for unit surface for a black body \( (\varepsilon = 1) \), integrating \( d^3W \) and Equation [3] over emission angles and wavelength, is given by the Stefan-Boltzmann formula

\[
\Phi_{\text{rad}} = \sigma T^4
\]

where \( \sigma = 5.67 \times 10^{-8} \text{ W m}^{-2} \text{ K}^{-4} \). For a temperature \( T = 3000 ^\circ \text{C} \) and for a black body emission, we find \( \Phi_{\text{rad}} = 6.5 \text{ W m}^{-2} \).

\[\text{This is a few times the surface power density of an ordinary fuel element in a fast reactor.}\]
MW/m² (650 W/cm²) i.e. about 5000 times the solar constant. The corresponding thrust force (in Newton) for a parallel light beam, built starting from such a source, is

\[ F = \Phi \cdot c \cdot S \cdot \text{mrad} = \Phi \left( \frac{S}{46} \right), \]

where \( S \) represents the radiating surface.

A thin hot plate emits light to both sides. Therefore in order to generate a net thrust, the back face must not radiate appreciably. This can in principle be achieved with coatings such \( \varepsilon_{\text{back}} \ll \varepsilon_{\text{front}} \). It is not known if it would be possible to realise at such a high temperature a fully reflecting coating. Alternatively (see Figure 3A) one could locate to the back side a reflecting surface physically separate but very near to the hot plate, such as to send back to the plate its emitted radiation. In turn, in order to reduce the temperature of the reflector \( T_{\text{refl}} \), even for a hot plate “black” at both sides it must have the configuration \( \varepsilon_{\text{back}} \gg \varepsilon_{\text{front}} \), since \( T_{\text{refl}} = T_{\text{hot}} \left( \varepsilon_{\text{front}} / \varepsilon_{\text{back}} \right) \) i.e. a more reasonable temperature.

However a hot plate does not emit parallel light. The radiating power emitted by an elementary surface \( S \) in a direction with an angle \( \theta \) with respect to the normal \( \hat{n} \) of the plane, on a solid angle \( d\Omega = d(\cos(\theta))d\phi \) by a body of absorption coefficient \( \varepsilon \) is given by

\[ d\Phi = \varepsilon SL \cos(\theta)d\Omega \]

where \( L = \sigma T^4 / \pi \) is the black body specific radiance, usually given by the Stefan Boltzmann formula. For symmetry reasons, the components of the force, normal
to $\vec{n}$, integrated over $\theta$ and $\varphi$ are zero. The surviving force, parallel to $\vec{n}$ is given by

$$F_{\parallel} = \frac{1}{c} \int d\varphi \int S d\omega \cos^2(\theta) d\cos(\theta) = \frac{2}{3} \frac{W}{c}$$  \[5\]

where $W = \pi \epsilon L_\omega$ is the radiated power. Therefore a naked, flat surface produces a thrust force, normal to the surface equals to $2/3$ the one of a parallel beam of the same power.

This inefficiency can be improved with a reflecting conical structure (non focusing optics, Winston’s cone), which can funnel the off direction radiation toward the direction $\vec{n}$, as shown in Figure 3B. The specific discussion of such a structure, well known in the solar concentration technology, is beyond the purpose of this paper. However for such devices, if well designed, the product of the solid angle of the light cone times the cross section of the structure is conserved. Therefore in a reflecting cone of gradually increasing cross section, light rays gradually straighten up. A larger output cross section hence generates a more collimated light beam, hence with a more efficient thrust. We can assume that an adequate cone could recover the missing factor $1/3$ and a heated plate

Figure 3. Conceptual layout for the radiating structure. In (A) we show the general arrangement for a hot radiating plate. In order to generate thrust, radiation must be directed only in one direction. The unwanted radiation is reflected back onto the plate by a reflector. In (B) a concentrating cone is added to geometry (A) in order increase the directionality of the emitted light. In order to keep the temperature of the reflectors in (A) and (B) to a reasonable value, while the surface facing the hot plate is made highly reflective, the outer surface is blackened to enhance heat dissipation. All reflecting structure must be extremely light in order to minimize the mass of the propulsion system, profiting from the technology developed for the “sails”.

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with a simple non-focusing conical optics (the “light-nozzle”) approaches the performance of a parallel light beam.

The relation [2] between radiated power and thrust force is of course independent of the temperature of the radiating body. The parametric choice $T = 3000 \, ^\circ\text{C}$ is therefore only indicative. Evidently a higher temperature reduces the dimension of the radiator, though it corresponds to a higher power density to be radiated. We display in Figure 4 the temperature dependence of the surface and surface power. One can deduce that the actual operating temperature will be the result of the engineering choices, rather than being an intrinsic feature of the method. For instance, reducing the temperature to $T = 2500 \, ^\circ\text{C}$ will increase the radiating surface by a factor 1.94, and the radiating power density to $\Phi_{\text{rad}} = 334 \, \text{W/cm}^2$.

Figure 4. Temperature dependence of (A) the required radiating surface for perfect focusing (parallel light beam) and a flat radiating half plate; (B) emitted power density for unit surface $\Phi_{\text{rad}}$. All curves are for a thrust force of 1 Newton, corresponding to a thermal power of $W=300\,\text{MW}$ for perfect focusing and $450 \, \text{MW}$ for a flat radiating plate.
4.— Specific Space missions.

In order to evaluate the potentialities of the method, we have considered a number of idealised space missions, selected on the basis of existing transport capabilities in low earth orbit and of the most economical transfer trajectory (Homann). The performance data should be taken as approximate value only. In all cases, a reduction in flight time could be traded against payload, and vice versa. The most desirable concept of the spacecraft will be the result of a mission oriented realistic engineering analysis.

A realistic upper limit for the propulsion time may be 2 to 3 years. This limit, which is determined by factors like corrosion, radiation and meteoroid damage, vaporisation losses of hot surfaces, poisoning of the reactor elements, etc. may expand further as technology advances.

There are a number of space missions for which this type of propulsion is suited. In our considerations we shall limit to the case of a intra-planetary transfer towards the planets of our solar system, namely the transfer between a low (≥ 540 km) initial earth orbit to a circular low altitude orbit in the field of the target planet. In the cases of planets with satellites, in order to reach them as well, the capture to an orbit with the satellite’s orbital parameters may be considered.

The general scenario will therefore consist in three nuclear phases, with intermediate coasting phases, preceded and eventually followed by more conventional phases eventually with chemical fuel, in particular the transfer of the spacecraft from ground to low earth orbit and all the manoeuvring phases in the field of the target planet or satellite. Evidently the use of nuclear power should be limited for environmental reasons to the journey in interplanetary space. The nuclear phases are as follows:
   
i) Escape from the earth’s gravitational field, spiralling from the initial orbit, with the thrust oriented tangentially to the orbit. The ensuing trajectory will then be a spiral with increasing pitch, until the escape will occur. Let \( a_o \) be the constant, small acceleration impressed to the spacecraft. It can be shown that escape will occur for arbitrarily small values of \( a_o \) (in m s\(^{-2}\)), with a escape time in days which can be parameterised as

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\(^5\) The masses of the satellites of the planets of the solar system are generally very small, the largest being of the order of the mass of the moon. Therefore the subsequent capture in an orbit around them and eventually the landing can be performed with on board chemical fuels.

\(^6\) In the following we have assumed an initial mass for the spacecraft of 20 ton, since this is a realistic payload for present launchers (Space Shuttle and Ariane 5).
$t_{esc} = \frac{6.42}{a_o \sqrt{r_o}}$ \hspace{1cm} [6]

where $r_o = 6370 + h_o$ is the radius of the departure orbit in km with orbital altitude $h_o$. In Figure 5 we display the radiative power in GW as a function of the escape time in days for a spacecraft of 20 ton for different departure orbital altitudes.

We notice the weak dependence on $h_o$. A radiative power $W$, proportional to the spacecraft mass $M_o$ of a few GW for $M_o = 20$ t, is adequate for a reasonable escape time ($t_{esc} = 140$ days, for $h_o = 2000$ km and $W = 3$ GW). The nuclear fuel burn-up, defined as the product $B_{esc} = Wt_{esc}$ (expressed in GW day/ton) is, according to Equation[4], independent of $W$ and proportional to $M_o$, and for $h_o = 2000$ km it amounts to $B_{esc} = 21.0 M_o [Ton]$. Typical fuels should permit $B = 500$ GW day/ton and hence the fuel mass burnt in the escape procedure is a mere 4.2% of the spacecraft mass $M_o$.

ii) Transfer to the Hohmann trajectory, i.e. an elongated elliptic solar orbit tangent both to the earth and the destination planet. This requires a velocity change $\Delta v_{\text{earth} \rightarrow \text{H}}$, which depends on the target planet. In principle such a velocity change should be localised; but numeric calculations show that it can be performed without major perturbations also over a finite time. Since the power of the engine is determined by the requirement i) above of a reasonable escape time,
we assume that the engine will continue with the same power output and adjust correspondingly the running of the engine. Results of the calculations for the five principal planets is given in Table 1.

**Table 1.** Injection into the Hohmann trajectory for a spacecraft mass of 20 ton and a peak engine power of 3 GW.

<table>
<thead>
<tr>
<th></th>
<th>Venus</th>
<th>Mars</th>
<th>Jupiter</th>
<th>Saturn</th>
<th>Uranus</th>
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<tbody>
<tr>
<td>Planet orbit radius</td>
<td>A.U.</td>
<td>0.72</td>
<td>1.52</td>
<td>5.20</td>
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<tr>
<td>Duration of trip</td>
<td>146.0 d</td>
<td>258.9 d</td>
<td>2.80 y</td>
<td>6.06 y</td>
<td>16.0 y</td>
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<td>Speed var. ( \Delta v_{\text{earth-H}} )</td>
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<td>2.942</td>
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<td>Total veh. Mass, ( M_o )</td>
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<td>20.0</td>
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<td>20.0</td>
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<tr>
<td>Peak power, ( W )</td>
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<td>Engine on time</td>
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<td>68.1</td>
<td>203.4</td>
<td>238.2</td>
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<tr>
<td>Burn-up</td>
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<td>173.6</td>
<td>204.3</td>
<td>610.3</td>
<td>714.6</td>
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<td>2.04</td>
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<tr>
<td>Fuel mass/ ( M_o, \text{sofar} )</td>
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<td>6.24</td>
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</tbody>
</table>

We remark that, as previously, the burn-up is independent of choice of the engine power \( W \) and proportional to the spacecraft mass \( M_o \). The mass of the fissile burnt fuel has been calculated under the assumption that \( B = 500 \text{ GW day/ton} \). This mass is reasonably small in all cases. It is of the order of a mere 2 % of the spacecraft mass \( M_o \) for the nearer planets Venus and Mars with an engine on time of the order of some 60 days. In the case of the larger planets Jupiter and Saturn, which are much further out, as reflected by the longer duration of the trip, the fuel consumption remains quite modest, respectively 6.1 % and 7.1 % of \( M_o \). Finally in the case of Uranus, although the fuel consumption remains moderate, the trip time is probably too long (16 years). A faster trajectory, though demanding more energy should be studied. Once injected into the Hohmann solar orbit, the engine is switched off and the spacecraft is freely coasting, until injection into the orbit of the target planet. The fraction of the time with engine off is very substantial for the planets which are further out in space. In these cases, as mentioned, one could trade transit time with a further activation of the engine.
iii) On approaching the target planet, the spacecraft must be slowed down in order to be captured on the (solar) orbit of the planet, at the edge of its attraction potential. The orientation of the engine is therefore reversed and it is operated according to the values of Table 2.

**Table 2.** Exit from the Hohmann trajectory for a spacecraft mass of 20 ton and a peak engine power of 3 GW.

<table>
<thead>
<tr>
<th></th>
<th>Venus</th>
<th>Mars</th>
<th>Jupiter</th>
<th>Saturn</th>
<th>Uranus</th>
</tr>
</thead>
<tbody>
<tr>
<td>Speed var. ( -\Delta v_{\text{H-plan}} ) km/s</td>
<td>2.712</td>
<td>2.237</td>
<td>5.641</td>
<td>5.440</td>
<td>4.657</td>
</tr>
<tr>
<td>Total veh. Mass, ( M_o ) ton</td>
<td>20.0</td>
<td>20.0</td>
<td>20.0</td>
<td>20.0</td>
<td>20.0</td>
</tr>
<tr>
<td>Peak power, ( W ) GW</td>
<td>3</td>
<td>3</td>
<td>3</td>
<td>3</td>
<td>3</td>
</tr>
<tr>
<td>Thrust N</td>
<td>10</td>
<td>10</td>
<td>10</td>
<td>10</td>
<td>10</td>
</tr>
<tr>
<td>Acceleration ( \text{m/s}^2 \times 10^3 )</td>
<td>0.5</td>
<td>0.5</td>
<td>0.5</td>
<td>0.5</td>
<td>0.5</td>
</tr>
<tr>
<td>Engine on time ( \text{days} )</td>
<td>62.8</td>
<td>51.8</td>
<td>130.6</td>
<td>125.9</td>
<td>107.8</td>
</tr>
<tr>
<td>Burn-up ( \text{GW day/t} )</td>
<td>188.3</td>
<td>155.3</td>
<td>391.7</td>
<td>377.8</td>
<td>323.4</td>
</tr>
<tr>
<td>Fissile used mass ton</td>
<td>0.377</td>
<td>0.311</td>
<td>0.783</td>
<td>0.756</td>
<td>0.647</td>
</tr>
<tr>
<td>Fuel mass/( M_o ) %</td>
<td>1.88</td>
<td>1.55</td>
<td>3.92</td>
<td>3.78</td>
<td>3.23</td>
</tr>
<tr>
<td>Fuel mass/( M_o, \text{sofar} ) %</td>
<td>7.82</td>
<td>7.79</td>
<td>14.22</td>
<td>15.13</td>
<td>15.26</td>
</tr>
</tbody>
</table>

5.— The descent nearer to the celestial body.

At this point one has to further specify the mission. All planets of Table 2, with the exception of Venus, have satellites which may be worth targeting. Therefore we have taken into consideration both (a) the transfer to a low altitude orbit of the main planet and (b) the matching to the speed of a specified satellite. In all cases, the objectives are a closer exploration and/or landing. These specific manoeuvres strongly depend on local environmental conditions, like the presence or absence of atmosphere, the purpose of the mission and so on. They will be generally performed with conventional, non-nuclear techniques, like chemical fuel, air braking or parachute, whose mass allowance is included within the “nuclear” payload of the mission.

A list of possible missions is listed in Table 3, where the main parameters of the capture process from the conditions of Table 2 trajectory are given. Numbers given in Table 3 refer as before to a 20 ton full load and a thrust power of 3 GW.
Table 3. Perspective missions to planets orbits and to their satellites. The figures to the left refer to the transfer to the final orbit from exiting the Hohmann trajectory (Table 2). The last three columns refer to the full mission from initial earth’s orbit.

<table>
<thead>
<tr>
<th>Satellite or orbit</th>
<th>Fraction of earth mass</th>
<th>Orbit r (x10^3 km)</th>
<th>Capture (days)</th>
<th>Orbital (km/s)</th>
<th>Burnup (GWd)</th>
<th>Fuel (ton)</th>
<th>Fuel %</th>
<th>Time(y)</th>
<th>EON (y)</th>
</tr>
</thead>
<tbody>
<tr>
<td>VENUS</td>
<td>0.817</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3 planet radii</td>
<td>—</td>
<td>18.30</td>
<td>97.6</td>
<td>4.22</td>
<td>292.9</td>
<td>0.59</td>
<td>2.93</td>
<td>10.75</td>
<td>0.98</td>
</tr>
<tr>
<td>5 planet radii</td>
<td>—</td>
<td>30.50</td>
<td>75.6</td>
<td>3.27</td>
<td>226.9</td>
<td>0.45</td>
<td>2.27</td>
<td>10.09</td>
<td>0.92</td>
</tr>
<tr>
<td>MARS</td>
<td>0.108</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3 planet radii</td>
<td>—</td>
<td>10.25</td>
<td>47.4</td>
<td>2.05</td>
<td>142.1</td>
<td>0.28</td>
<td>1.42</td>
<td>9.21</td>
<td>1.22</td>
</tr>
<tr>
<td>5 planet radii</td>
<td>—</td>
<td>17.08</td>
<td>36.7</td>
<td>1.58</td>
<td>110.1</td>
<td>0.22</td>
<td>1.10</td>
<td>8.89</td>
<td>1.19</td>
</tr>
<tr>
<td>Phobos</td>
<td>4.52E-9</td>
<td>9.35</td>
<td>49.6</td>
<td>2.14</td>
<td>148.7</td>
<td>0.30</td>
<td>1.49</td>
<td>9.28</td>
<td>1.23</td>
</tr>
<tr>
<td>Deimos</td>
<td>3.06E-10</td>
<td>23.40</td>
<td>31.3</td>
<td>1.35</td>
<td>94.0</td>
<td>0.19</td>
<td>0.94</td>
<td>8.73</td>
<td>1.18</td>
</tr>
<tr>
<td>JUPITER</td>
<td>305.6</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3 planet radii</td>
<td>—</td>
<td>214.12</td>
<td>552.1</td>
<td>23.85</td>
<td>1656.2</td>
<td>3.31</td>
<td>16.56</td>
<td>30.78</td>
<td>2.81</td>
</tr>
<tr>
<td>5 planet radii</td>
<td>—</td>
<td>356.88</td>
<td>427.6</td>
<td>18.47</td>
<td>1282.9</td>
<td>2.57</td>
<td>12.83</td>
<td>27.05</td>
<td>2.47</td>
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<tr>
<td>Io</td>
<td>1.32E-2</td>
<td>422.00</td>
<td>393.3</td>
<td>16.99</td>
<td>1179.8</td>
<td>2.36</td>
<td>11.80</td>
<td>26.02</td>
<td>2.38</td>
</tr>
<tr>
<td>Europa</td>
<td>7.99E-3</td>
<td>671.00</td>
<td>311.9</td>
<td>13.47</td>
<td>935.6</td>
<td>1.87</td>
<td>9.36</td>
<td>23.58</td>
<td>2.15</td>
</tr>
<tr>
<td>Ganymede</td>
<td>2.58E-2</td>
<td>1071.00</td>
<td>246.9</td>
<td>10.66</td>
<td>740.6</td>
<td>1.48</td>
<td>7.41</td>
<td>21.63</td>
<td>1.97</td>
</tr>
<tr>
<td>Callisto</td>
<td>1.24E-2</td>
<td>1883.00</td>
<td>186.2</td>
<td>8.04</td>
<td>558.5</td>
<td>1.12</td>
<td>5.59</td>
<td>19.81</td>
<td>1.81</td>
</tr>
<tr>
<td>SATURN</td>
<td>95.1</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3 planet radii</td>
<td>—</td>
<td>181.50</td>
<td>334.4</td>
<td>14.45</td>
<td>1003.3</td>
<td>2.01</td>
<td>10.03</td>
<td>25.16</td>
<td>2.30</td>
</tr>
<tr>
<td>5 planet radii</td>
<td>—</td>
<td>302.50</td>
<td>259.1</td>
<td>11.19</td>
<td>777.2</td>
<td>1.55</td>
<td>7.77</td>
<td>22.90</td>
<td>2.09</td>
</tr>
<tr>
<td>Mimas</td>
<td>6.19E-6</td>
<td>185.60</td>
<td>330.7</td>
<td>14.29</td>
<td>992.2</td>
<td>1.98</td>
<td>9.92</td>
<td>25.05</td>
<td>2.29</td>
</tr>
<tr>
<td>Enceladus</td>
<td>1.24E-5</td>
<td>238.30</td>
<td>291.9</td>
<td>12.61</td>
<td>875.6</td>
<td>1.75</td>
<td>8.76</td>
<td>23.89</td>
<td>2.18</td>
</tr>
<tr>
<td>Tethis</td>
<td>8.24E-5</td>
<td>295.10</td>
<td>262.3</td>
<td>11.33</td>
<td>786.9</td>
<td>1.57</td>
<td>7.87</td>
<td>23.00</td>
<td>2.10</td>
</tr>
<tr>
<td>Dione</td>
<td>8.99E-5</td>
<td>377.40</td>
<td>231.9</td>
<td>10.02</td>
<td>695.8</td>
<td>1.39</td>
<td>6.96</td>
<td>22.09</td>
<td>2.02</td>
</tr>
<tr>
<td>Rhea</td>
<td>2.96E-4</td>
<td>527.40</td>
<td>196.2</td>
<td>8.48</td>
<td>588.6</td>
<td>1.18</td>
<td>5.89</td>
<td>21.02</td>
<td>1.92</td>
</tr>
<tr>
<td>Titan</td>
<td>1.99E-2</td>
<td>1223.00</td>
<td>128.8</td>
<td>5.57</td>
<td>386.5</td>
<td>0.77</td>
<td>3.87</td>
<td>19.00</td>
<td>1.73</td>
</tr>
<tr>
<td>Hyperion</td>
<td>1.14E-5</td>
<td>1481.00</td>
<td>117.1</td>
<td>5.06</td>
<td>351.2</td>
<td>0.70</td>
<td>3.50</td>
<td>18.64</td>
<td>1.70</td>
</tr>
<tr>
<td>Iapetus</td>
<td>3.80E-4</td>
<td>3563.00</td>
<td>75.5</td>
<td>3.26</td>
<td>226.5</td>
<td>0.45</td>
<td>2.26</td>
<td>17.39</td>
<td>1.59</td>
</tr>
<tr>
<td>Phoebe</td>
<td>3.22E-6</td>
<td>12950.00</td>
<td>39.6</td>
<td>1.71</td>
<td>118.8</td>
<td>0.24</td>
<td>1.20</td>
<td>16.32</td>
<td>1.49</td>
</tr>
<tr>
<td>URANUS</td>
<td>14.56</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3 planet radii</td>
<td>—</td>
<td>76.20</td>
<td>202.0</td>
<td>8.73</td>
<td>606.0</td>
<td>1.21</td>
<td>6.06</td>
<td>21.32</td>
<td>1.95</td>
</tr>
<tr>
<td>5 planet radii</td>
<td>—</td>
<td>127.00</td>
<td>156.5</td>
<td>6.76</td>
<td>469.4</td>
<td>0.94</td>
<td>4.69</td>
<td>19.95</td>
<td>1.82</td>
</tr>
<tr>
<td>Ariel</td>
<td>8.34E-5</td>
<td>191.00</td>
<td>127.6</td>
<td>5.51</td>
<td>382.8</td>
<td>0.77</td>
<td>3.83</td>
<td>19.09</td>
<td>1.74</td>
</tr>
<tr>
<td>Titania</td>
<td>2.28E-5</td>
<td>438.00</td>
<td>84.3</td>
<td>3.64</td>
<td>252.8</td>
<td>0.51</td>
<td>2.53</td>
<td>17.79</td>
<td>1.63</td>
</tr>
<tr>
<td>Oberon</td>
<td>1.81E-4</td>
<td>586.60</td>
<td>72.8</td>
<td>3.15</td>
<td>218.4</td>
<td>0.44</td>
<td>2.18</td>
<td>17.44</td>
<td>1.59</td>
</tr>
</tbody>
</table>
Both power and fuel mass are proportional to the vehicle mass, $M_o$. In the case in which the main target is the planet, we have considered the capture into two, somewhat arbitrarily chosen, circular, equatorial orbits of 2 and 4 planet radius’s altitude. In the case in which the satellite is being targeted, we capture the vehicle to an orbit identical to the one of the chosen satellite. Eventual landing on the satellite should be an operation to be performed with conventional methods and of a relatively low energetic cost, even in absence of atmosphere. In Table 3 we list satellite masses in units of earth mass; their additional gravitational strength is very modest, in view of their relatively small masses\(^7\). Individual planets will briefly examined, with the help of Table 3:

i) Venus, a planet which has no satellites and a very thick and hot atmosphere, can be approached with nuclear propulsion up to a circular orbit of medium altitude ($\approx 12000$ Km, e.g. 2 planet radii). Capture from the interplanetary journey to a medium altitude orbit should require some $75 \div 100$ days. The consumption for the whole mission, i.e. from low earth orbit to low Venus orbit requires in fuel mass a bit more than $0.1\, M_o$ and the outbound trip time is about 1 year. Return trip, even with no mass change for the inbound phase, will be possible since an initial fuel load of $(0.2 \div 0.25)\, M_o$ is perfectly reasonable.

ii) Mars is the easiest target planet of Table 3. With the help of nuclear propulsion, a circular Martian orbit of medium altitude ($\approx 6800$ Km, e.g. 2 planet radii) can be reached, starting from the Earth’s orbit, with a over-all fuel allowance of a bit more than $0.09\, M_o$. Round trips between the Earth’s and Mars’ coasting orbits are therefore possible with a fuel consumption of less than $0.2\, M_o$. Mars has two small satellites, Phobos and Demos, which turn in the equatorial plane with almost circular orbits of 9350 km and 23400 km respectively (revolution time: 7 h and 39 m and 30 h 18 m). They are irregular rocks with meteorite craters. Phobos, of a diameter of about 20 km, could be an excellent “staging post” for further exploration of Mars. Landing on Phobos requires the Mars’ capture time of about 50 days, adding to a total journey time from an earth orbit of 1.22 years, of which the engine will be on (EON) for 0.85 years.

The larger planets of the solar system, Jupiter, Saturn and Uranus are fundamentally different from the much smaller objects Mercury, Venus Earth and

\(^7\) The heaviest amongst satellites, e.g. the four main satellites of Jupiter and Titan (Saturn) have masses of only few percent of the earth’s mass. The Moon’s mass, in comparison, is 1.23 % of the earth’s mass.
Mars. They have a much larger diameter due to a lower density, presumably due to highly compressed gas, mostly hydrogen and helium. Their mass (see table 3) being respectively 306, 95 and 14.5 times the mass of the earth, the gravitational strength is much larger than the one of earth. However due to their lower density (for instance the density of Saturn is only 0.68 the density of water) and hence radius, gravity at surface is relatively modest, being at the equator respectively 2.3, 0.88 and 0.99 making it possible in principle for a human to descend on them. However, their surface temperature is in the harsh range -150 °C ÷ -200 °C, though slightly higher than the one due to solar radiation, because of an (unknown) source of internal heat. Therefore instrumental exploration is likely to be highly preferred, including the descent of probes on the planets themselves and eventually return of samples on Earth.

The length of the nuclear journey outbound (see Table 3), is typically of 4 years for Jupiter, 7 years for Saturn and 17 years for Uranus, on no major concern for instrumental exploration. Return journeys seem of rather specific importance, primarily in order to bring back some samples on Earth. Because of the large gravitational field produced by these large planets, the capture to a reasonable planet’s orbit is relatively long, adding generally of the order of 1 year to the journey, and costly in terms of energy, since the reactor must operate continuously for that length of time (see Table 3).

However the very long response time for signals from and to earth (0.58 ÷ 0.86, 1.18 ÷ 1.45 h and 2.51 ÷ 2.79 hours respectively, depending on the location on the orbit), does not permit to have a reasonable feed-back time from earth. Any sophisticated operation implies either an extraordinary development of artificial intelligence or the local presence of man in a space station around the planets, eventually with an instrumented probe descending on the surface.

Looking in more detail to each of these planet, one may add:

iii) Jupiter, the heaviest of the solar planets (306 times the earth) is shown a considerable amount of activity on the surface, including the famous red spot, as large as three times the earth. Internal heat contributes to about + 40 °C to the temperature, which is presently an absolute mistery. Equally interesting are the four Galilean satellites, Io, Europa, Ganymede and Callisto which are at growing distance from the planet with masses of the order of a few percent of the one of the earth, about the one of the moon. Europa is the most interesting one, since it is covered by an icy cap, presumably molten into a liquid water ocean in the interior, in which, in analogy to the under ice cap Vostok lake in Antarctica, life may be able to survive. Reaching out for these satellites requires, in the capture process, a considerable amount of kinetic slowdown (see Table 3). For instance, in the case of
Europa, this manoeuvre requires, by itself, a bit less than one year and $0.0936 M_o$ of fuel, more than what is needed for the full journey Earth to Mars.

The descent to Jupiter itself to an orbital altitude equal twice the planet radius is even more costly (552 days and $0.1656 M_o$ of fuel). Because of its fast rotation, the iso-synchronous satellite orbit is even lower, at an altitude of 0.8 planet radii. Its atmosphere is rich of methane, hydrogen, helium and ammonia. The atmospheric pressure is of several atmospheres.

iv) Saturn has a surface gravity very close to the one on the earth, and a surface temperature of $-160 \, ^\circ C$. Its atmosphere similar to the one of Jupiter, except with less ammonia since it may liquefy (incidentally on Uranus ammonia is almost all liquefied). The most impressive property are his ring, which extend to about 136000 km. It is unlikely that any orbit could descend below this orbital altitude, without colliding with the material of the ring. We have considered (see Table 3) the lowest orbit to be at twice the planet radius, which can be reached by spiralisation in about 330 days with an amount of fuel equals to $0.10 M_o$.

Amongst satellites, of which 10 are known (see Table 3), the heaviest is Titan, of a mass close to the one of the moon reachable in about 130 days and with $0.0387 M_o$ of fuel.

v) Finally, at even greater distance one may aim at Uranus, which is certainly one of most inhospitably cold amongst the planets mentioned sofar. The time of the journey is very long (17 years), of which more of 15 years of idle coasting on the Hohmann trajectory. The amount of fuel for the total engine on (EON) time is $0.2 M_o$. Because it has a lower mass (about a factor ten more then the earth) capture is more easily performed, in times which are typically of the order of 1/2 year.

It is certainly premature and outside the purpose of this paper to speculate further on what one may search for in detail in each mission. However from the technical point of view, our propulsion method is capable of ensure realistically all the missions described in Table 3.

Other missions, like reaching the Asteroids, flying out of the plane of the solar system, encounters to a comet etc. although not reported in Table 3, give similar figures and therefore similar conclusions.

One may therefore conclude that the present propulsion method is a general purpose, realistic method suitable to reach and to descend on any celestial
body up to the outer bound of Uranus. For a great deal of cases, and specifically in the case of Mars, also round trips without refuelling are made possible.

In these examples, the EON time is generally less than 3 years, which is a realistic time for the operation of the reactor.

Extending the targets further to Neptune and Pluto is not prohibitive from the point of view of engine performance, in the sense that the Hohmann trajectory requires only some percent of additional energy. However the time of the journey approaches 30 years for Neptune and 45.6 years for Pluto. It is unlikely that one is prepared to wait that long.

However, since most of the time is spent coasting, an extended EON time may lead to more effective trajectories, which however have still to be examined in detail.

6.—— Very preliminary engine design considerations.

The actual design of a nuclear reactor with the required characteristics is outside the purpose of this paper. However some general considerations will be given, in order to explore if such a performance is within the realm of the reasonable extrapolations of the present state of the art.

Nuclear power reactors have operated at the high power, high power density and high temperature as required by the present application, but not simultaneously on all fronts.

For instance the thermal power of a modern commercial PWR for electricity production exceeds the reference value of 3 GW used in the previous section. But this power is primarily generated by the fission of $\approx 3.3$ tons of U-235 contained in $\approx 100$ ton of enriched Uranium oxide fuel. In the case of a space propulsion dedicated reactor, the fuel should be as light as possible and an almost pure U-235 fuel should be used. The reduction of a factor 33 of the total mass of the fuel for the same power represents no problem for the nuclear processes, but now the thermal power density is $\approx 33$ times larger. In order to ensure a good heat extraction, the fuel must have both a higher thermal conductivity and be made of smaller size elements.

Most of these problems have been explored with a number of test reactors, with many possible solutions. In particular, and in reference to space applications, it is worth mentioning the NERVA project, in which a (fast) reactor about 0.6 ton of almost pure U-235 in the form of a composite carbide U-Zr-Nb is used.
to heat up hydrogen gas to a (stagnation) temperature of 3000 °C. This gas cooled reactor has operated for extended periods with a power $W \approx 0.3$ GW. The reactor for this application may therefore largely profit from the work on NERVA.

The simplest solution would be the one of letting the nuclear fuel radiate directly the electromagnetic energy. However there is contradiction between the "open" geometry needed to enhance the radiative surface and the requirement to achieve criticality with a minimum mass of reflector. For instance, as already pointed out, the surface needed to radiate 3 GW at 3000 °C is $S = W/\Phi_{rad} = 461.5$ m². The corresponding radiator thickness is extremely thin, $t = M_{rad}/S = 0.22 M_{rad}[\text{ton}] \ g/cm^2$, where $M_{rad}$ is the radiator mass. It is therefore impossible to reconcile a low $M_{rad}$ with a criticality condition. The alternative arrangement would then consist of a nuclear reactor thermally coupled with the radiator, with the help of an appropriate heat transfer arrangement based on a specific coolant. Basically the reactor is a naked fast neutron reactor, whose main weight is the nuclear fuel. Using enriched uranium there should be no difficulty in achieving criticality even with a modest neutron reflector. The sensitive equipment, like in the case of NERVA may require local shielding.

However gas cooled reactors generally require considerable gas volumes, pressures and fuel contact surface. In order to achieve good thermal coupling and high heat extraction rate, we consider more appropriate a liquid metal coolant. In the present application there are many reasons which suggest to use a boiling liquid as a coolant, where advantage would be taken of the large latent heat of vaporisation. The reactor must be designed in such a way as that bubble formation causes a decrease, rather than an increase of reactivity. Because of the self regulating effect of steam-bubble formation, a reactor can be operated continuously in a stable fashion under boiling conditions. If the boiling occurs more rapidly than what required by the normal operating power, the additional bubble formation will decrease the reactivity and less heat will be produced by fission. This will tend to bring the boiling rate back to the desired value.

Another advantage of the method is that both the boiling and subsequent condensation processes are essentially iso-thermic. The heat transfer between the reactor and the radiating fins is performed with modest pressures and temperature differences and it requires a greatly reduced pumping power.

The choice of the appropriate boiling liquid depends on many technical considerations, like corrosion, neutron behaviour, radiation damage and so forth and cannot be made at this level. An interesting candidate as a pure metal is Beryllium ($^9\text{Be}_4$), which has a melting point of 1278 °C and a boiling point at 2970 °C. This metal has a number of desirable properties. It is one of the lightest of all
metals, it has an appropriately high boiling point, a high specific boiling heat \((297 \times 10^3 \text{ J/mol, e.g. } 32.9 \times 10^6 \text{ J/kg})\), a small neutron absorption cross section, a substantial \((n,2n)\) cross section at high energies which leads to a significant neutron multiplication.

Transfer of 3 GW from the reactor to the radiating panels demands a coolant’s mass flow of \((3 \times 10^9 \text{ W})/(32.96 \times 10^6 \text{ J/kg}) = 91.2 \text{ kg/s}\) i.e. in total about 50 liter/s of boiling liquid metal. The vapour volume is of course much larger, \((\text{Be}_2\text{, absolute pressure }\approx 5 \text{ bar, } T = 3000 \degree \text{C})\) namely of order 220 m³/s.

The total radiant (black) surface required to emit the chosen power into space is given by \(S = W/\Phi_{\text{rad}}\). The temperature dependence of \(\Phi_{\text{rad}}\) is shown in Figure 4. For instance for \(T = 3000 \degree \text{C, } W = 3 \text{ GW, } \Phi_{\text{rad}} = 6.5 \text{ MW/m}^2\), the total surface, eventually composed of several segments, is \(S = 461.5 \text{ m}^2\). (Reducing the temperature to \(T = 2500 \degree \text{C}\) will roughly double the surface \(S\)). In order to maintain the temperature of 3000 °C, one must condense — on the inner wall of the radiator plate — gas into liquid at the rate \(21.7 \text{ mol m}^{-2}\text{s}^{-1}\). For the above conditions, \((\text{Be}_2\text{, absolute pressure }\approx 5 \text{ bar, } T = 3000 \degree \text{C})\) the local gas speed toward the condensing plate is 0.5 m/s, a reasonable value. The resulting, liquid volume is of course much smaller, namely \(\approx 0.11 \text{ litre/s}\) for each square meter.

The radiator should then consist of a very light structure of an appropriate material capable to withstand the high temperature, like for instance carbon fibres, with a distributed tubing system in which the coolant is transmitted to the radiating surface. The inner geometry should be such as to permit an efficient condensation of the metal vapour and the collection of the resulting liquid, to be sent back to the reactor. In order to collect sufficiently rapidly such condensed liquid, an artificial gravity force may be generated by putting the radiator assembly into axial rotation. The movement of the liquid should be sufficiently fast in order to minimise the amount of coolant required. For instance if the cooling cycle is performed in 10 second, the coolant mass is of the order of 1 ton.

The feasibility of such a massive heat transfer from the reactor to the radiating panels is a critical item in the feasibility of the proposed method.
Travel in deep space to planets and their satellites in the solar system requires engine performances which are fundamentally different than the one offered by chemical fuels. The amount of energy required to attain targets at billion of kilometres from Earth in a reasonable time, is a compelling argument in favour of nuclear (fission) energy. However the nuclear resource may only be used for the main part of the journey, classical methods being maintained for all local manoeuvres.

The main nuclear engine must provide a relatively small thrust (several Newton), but for a very long time, up to years of operation. It must be conceptually simple, with a minimum number of fast moving parts and extremely reliable. The large amount of propellant in a classical propulsion method, determined by the rocket equation, reduces dramatically the payload for a given initial mass, requiring a very high exhaust speed, i.e. a very large specific impulse, like the one offered today by ion propulsion.

Therefore the present method must be directly compared with advanced nuclear/electric methods based on ion propulsion. Assume for instance that we wish to produce the thrust of 10 N with a specific impulse of 30000 s — which are the parameters of the 3 GW radiative engine on which the perspective missions have been evaluated — with singly ionised Xe atoms, corresponding to a ion current of 23.3 A and a beam power of 1.5 MW. If the overall combined efficiency of the conversion into electricity and of the subsequent acceleration in the ion gun is, say, 1%, the thermal power of the associated reactor, mostly (99%) dissipated as radiation into space, is of the order of 150 MW. The electromagnetic thrust of this radiated energy is then $F_{em} = W/c = 0.5$ N, namely a factor $\approx 20$ lower than the one provided by ion propulsion. The key question is then if increasing the thermal power of a naked reactor by such a factor $\approx 20$ is preferable to the complexities of a full fledged conversion into electricity and subsequent ion acceleration in space.

Compared with "solar sails", from which this method has been inspired, the availability of an "on board" light source of appropriate size represents operationally an immense advantage, since, unlike the Sun, it can provide thrust in an arbitrary direction and it is not attenuated by a growing distance from the Sun during the mission. In addition the local source of light being much more powerful than solar radiation, the driving surface is reduced by a factor of the order 5000 at the earth’s position and even more for any elaborate mission involving targets more peripheral from the sun than the earth.