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Nuclear Propulsion

Claudio Bruno
DIMA, University of Rome (La Sapienza), Roma
Italy

1. Introduction

Nuclear propulsion (NP) concepts go back to the very end of WW II. Scientists informed about the effects of the US atomic bomb thought of exploiting its energy release for applications like commercial electric power generation, but also rockets and space flight [Shepherd and Cleaver, 1948, 1949; Bussard and DeLauer, 1958]. However, space flight was still considered science fiction, and the military had to deal with more concrete things, like the Cold War. Thus, besides power generation, second stages of ICBM, submarine propulsion, long range and long duration airplanes and missiles became the focus of nuclear energy applications.

It was the second-stage and airplane application that drove R&D in nuclear propulsion. With the advent of reliable ICBM (the Atlas missile) and lighter fission and thermonuclear warheads, a nuclear-powered second stage became no longer necessary. Airplane applications were found impractical: the Convair NB-36 required such a heavy lead shield for the crew that testing and operation were much restricted. Nuclear-powered missiles were easier to design, e.g., project PLUTO, but still far more complicated compared to conventional. The Soviets investigated airplanes and rockets powered by nuclear power as well, and discarded them too.

The history of NP can be found in [Czysz and Bruno, 2009, Chapter 7; Lawrence, 2008; Lawrence et al, 1995; Gunn and Ehresman, 2003; Dewar, 2004] and will not be reported here. Basic technology is also discussed in the references above, in particular reactor design is in [Lawrence et al, 1995]. In the US investing in NP technology by NASA and LASL stopped in 1972, but was kept alive by the USAF, and to some extent also at Los Alamos Science Laboratories (LASL, now LANL) who focused on unconventional concepts. In the former Soviet Union, now Russia, research was slowed down, but never completely stopped, because it was thought indispensable for future space missions.

The present work focuses rather synthetically on NP based on fission. (Fusion has been demonstrated, but the net power gain budget is still negative, and power generation is still far in the future, so NP based on this technology will not be discussed.) Note that only reluctantly space agencies or international study groups like ISECG [ISECG] are beginning to include *fission-based NP* in their exploration scenarios. The cost of R&D and experimenting with NP technology is in fact much higher than with conventional rocket propulsion, and until now MIR first, and then ISS, have absorbed most of the human exploration R&D moneys. After completing the ISS, space agencies are now discussing in-house and among themselves what should be the next goal of *human* space exploration. This question is critical to the survival of space industry in the US, EU and Japan; less so in Russia, and definitely not in emerging space-faring nations such as India, China and S.Korea. Note the *satellite* industry is in much better financial shape and has no such problem.

2. Human exploration: Constraints

In a long-term vision, the question becomes, what will be future manned interplanetary missions, and what do they need? NASA and Roskosmos are looking at 'far' destinations, i.e., not the Moon, although the Moon is still seen by most space planners an indispensable stepping stone to Mars and near Earth objects or asteroids (NEO or NEA). NASA favors, or favored, NEO missions as less expensive than going to the Moon, since no lander is needed (currently, no space agency is planning to build a man-rated Moon lander), but the life support system for a NEO mission does not exist yet, and, in fact, NASA is in the process of reassessing NEO vs. Moon goals. Such issues are more and more influenced by the growing consciousness of the health risks posed to crews by solar and cosmic-galactic radiation (SR and GCR). These risks are still difficult to quantify, but caution suggests to shorten interplanetary missions, e.g., to Mars as much as feasible. In turn, this means discarding conventional Hohmann trajectories (minimum energy trajectories). A Hohmann-type, chemical propulsion (*CP*)-powered mission to Mars lasting 9-mo or more each way would result in a total dose to the crew > 1 Sv, while the average dose per person on Earth, due to all sources (solar and background) is about 0.6 *mSv* per year. Although much research is necessary to quantify *the damage* due to space radiation, estimates of excess cancer probability due to 1 Sv range from a few percent to 30% [Durante and Bruno, 2010]. Even the lower bound is considered unacceptably high by most life scientists.

Note that the Earth magnetic field protect us from radiation provided we are within the van Allen belt. Belt-crossing itself exposes humans to much increased doses; outside the belt, in space, radiation dose depend on SR + GCR, including the SR contribution due to [still unpredictable] solar flares.

The SR + GCR dose to astronauts is directly proportional to the duration of exposure, i.e., dose = (radiation flux) x (exposure time). To reduce dose there are two logical solutions: either to reduce flux to crew by some form of shielding, or to shorten exposure time (faster missions).

2.1 Shielding

Concerning the first solution, shielding depends mainly on the areal mass density (usually given in g/cm^2) of the structure surrounding the astronauts. Typical ISS values, depending on location and thickness of scientific equipment (that contributes to areal mass), vary between 5 and 20 g/cm^2. This would be insufficient during a solar flare, where proton flux intensity may increase by a factor 10 to 50, or on the Moon. Dividing areal density by material density gives a good idea of the material thickness of the shield , so increasing areal density is effective in reducing dose, but translates also into increasing shield mass and thus orbiting cost; secondary radiation, due to interaction between high energy GCR and shield, remains a problem. *Active* shielding (electrostatic or electro-magnetic, EM) has been considered, e.g., by the Nobel Prize in physics Sam Ting, who worked on the original superconducting (SC) magnet of the Alpha Magnetic Spectrometer recently installed outside the ISS to detect dark matter. After discarding the superconducting magnet (B ~ 1 - 2 Tesla ≡ T; 1 T = 10,000 gauss) in favor of simpler permanent magnets [Covault, 2011], Ting thought of testing SC magnets in space to deviate high energy GCR particles and thus protect future astronauts. However, the B field required is of order 10-12 T or higher for a sufficiently large crew habitat. Besides feasibility, the high-B health effects on humans are still unknown; and even 10 T cannot deviate or stop the highest energy band of the GCR spectrum.

While research in this area continues, most experts have concluded that active shielding is for the foreseeable future still unfeasible [Parker, 2006], and that the cost of orbiting to LEO the massive shield required to reduce crew dose during a CP-powered ('long') Mars mission is much too high.

2.2 Interplanetary orbits

Thus, for the time being, the second solution (shortening travel time) should be sought. Any way to speed up interplanetary travel pays off in terms of astronauts safety. We have *some* understanding of the effects of microgravity on bone mass and metabolism, but we are just now beginning to explore the *psychological* consequences of crew confinement inside cramped quarters for many months (the recently completed Mars 500 ground experiment is the first example).

The question at this point becomes of if and how interplanetary trips may be shortened. Here this question is discussed by using basic physics, leaving details (but no show stoppers) to future engineering. In this context the basic physics of interplanetary travel is presented below, starting from what is current practice.

In fact, all present or planned mission are dominated by the key constraint to save mass to be orbited. Anything to be used for space missions must first be orbited, at a cost, for LEO, roughly between 10 and 20 k$/kg, depending on launcher provider. With CP, most of the mass is propellants, see Eqn (1). Physics tells us that once in LEO, the minimum energy (or the minimum ΔV) required to change orbit, for instance to go to Mars, is that of Hohmann trajectories. These are *ellipses* osculating the starting orbit (for instance, that of the Earth around the Sun) and the final one (for instance that of Mars), having assumed the two are on the same ecliptic plane. Hohmann trajectories require the least expenditure of energy, thus the least expenditure of propellants, and are realized by applying thrust (**F**) for a time $\Delta t <<$ of the total interplanetary travel time T. In the limit $\Delta t / T \rightarrow 0$ thrust becomes impulsive, and it is this feature that minimizes energy losses due to non-parallel spacecraft **F** and instantaneous velocity **V**. If **F** is impulsive, **F** and **V** are parallel, and all thrust is used to increase **V** by an assigned **ΔV**. It is always possible for a spacecraft to change its orbit by using slow acceleration ($\Delta t / T$ finite $\rightarrow$ low F); however, in significant gravitational fields such as Earth's, the trajectory becomes initially a *spiral* where **F** and **V** form an angle and thrust work is partly wasted. Low thrust ('spiraling') losses may be relatively large, e.g., in a low thrust trajectory from LEO to L1 (theoretical ΔV about 4 km/s) the actual 'expenditure' of ΔV may become about 6 km/s; see [Martinez-Sanchez and Pollard, 1998] for a LEO to GEO quantitative example.

2.3 Mass consumption

Using CP consumes much propellants mass. Tsiolkovski's equation states

$$\Delta V = \mathrm{Isp} \ln(\mathrm{Min} / \mathrm{Mfin}), \text{ or } \mathrm{Min} / \mathrm{Mfin} = \exp(\Delta V / \mathrm{Isp}) \quad (1)$$

and, with CP the specific impulse Isp, defined by Isp = F/(dm/dt), *happens to be of the same order of the ΔV needed* to change orbit in our solar system, see Figure 1 showing the ideal ΔV

with Hohmann orbits associated, for instance, to a Mars mission. The numbers tell the mass fraction will be >> 1. In Eqn (1) the (dm/dt) is the rate of mass consumption of propellants ejected from the spacecraft.

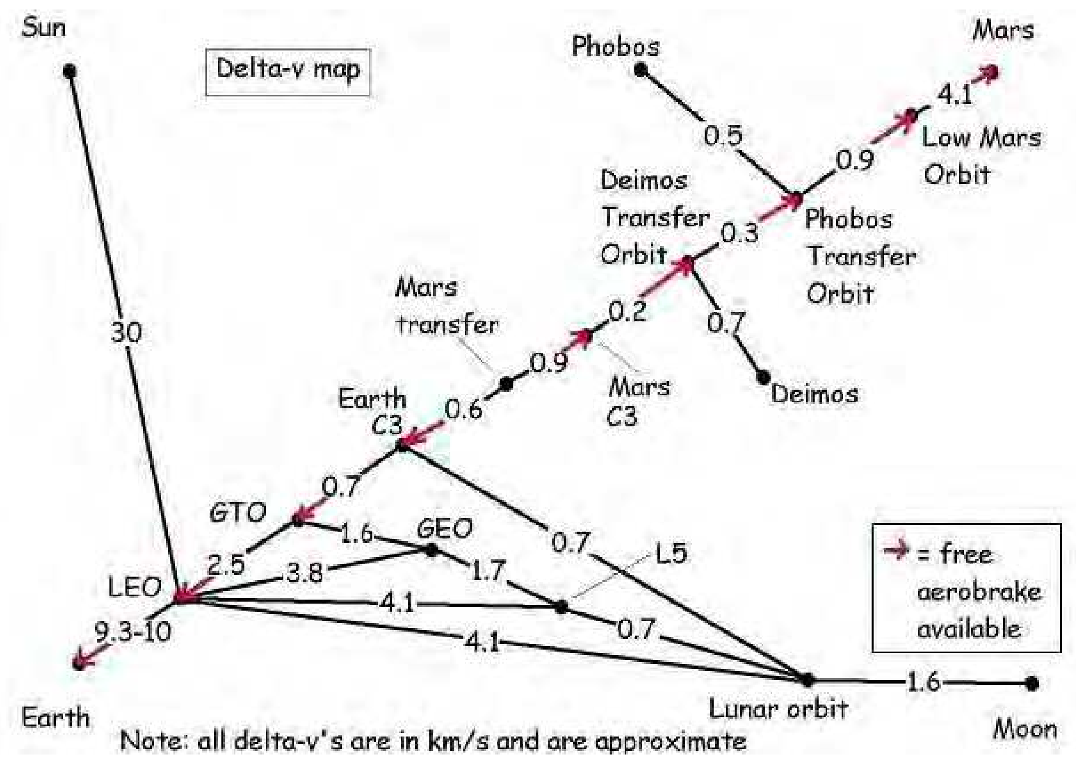

Fig. 1. Minimum ΔV using Hohmann's orbits associated to a Mars mission

Note that, according to Newton's second principle, F is the rate of change of momentum of the mass ejected. Assuming for simplicity that this mass has zero velocity prior to being accelerated, the instantaneous momentum change is Ve times (dm/dt). Provided the acceleration process is ideal, i.e., 100% efficient, the *Isp just defined coincides with Ve*, the ejection velocity. In real chemical rockets Ve and Isp differ, because the gasdynamic expansion of combustion products is incomplete, that is, never reaches zero pressure and temperature.

Actual Isp values depend on fluid and on energy conversion process. In CP this means: maximum and minimum cycle temperatures; in practice: combustion temperature, average molecular weight of combustion products, and expansion ratio. The highest practical Isp is that of the LOX/LH2 combination: in space, about 4500 m/s (or 4500/g seconds, g being the standard gravitational acceleration, about 9.81 m/s^2, when using old engineering units). Isp may be higher with 'exotic' combinations, e.g. , LF2 + LO2 and others, but their logistics and safety implications are forbidding. In practice, CP limits Isp to about 4.5 km/s. The Tsiolkowski equation shows that to achieve a certain ΔV the mass to eject depends *exponentially* on ΔV/Isp, this ratio being of order unity.

Thus ways to raise Isp are the key to economically feasible fast interplanetary travel.

3. Propulsion

The Tsiolkowski equation (1) is the form taken by the second Newton's principle when the mass of an object being accelerated varies with time. In fact, the physics we can bring to bear to solve the problem of faster travel is based solely on Newton's three principles. The third,

the most important in all kinds of propulsion, states that in space velocity of an object can be increased only *by ejecting mass,* that is, accelerating it from 0 to Ve (in the spacecraft reference system): so far, no massless 'space drive' has been invented.
Mass may be ejected either 1. via *internal* molecular collisions (i.e., through thermodynamic expansion), or 2. by applying an *external* force *directly* to atoms/molecules. These two different strategies are common to all space propulsion systems.
1. CP is based on the first strategy: in special relativity, the relativistic potential energy PE = $(\Delta m)\ c^2$ of a mass Δm becomes kinetic energy (KE) of the remaining mass m, that is ½ m $(Ve)^2$. The percentage $\Delta m/(m + \Delta m)$ of mass converted into KE depends on the *fundamental force* used. CP uses the potential of the *second force* (electro-weak), the potential energy released when chemical bonds are broken and rearranged.
In fact, the Standard Model of physics includes *only three* fundamental forces: gravity, electro-weak (the result of unifying electro-dynamic and weak forces in the '80s), and the strong, or nuclear, force. The electroweak force binds molecules and atoms together, and is responsible for the existence of matter as we know it. The strong force binds sub-atomic particles (nucleons) together, preventing them from disintegrating due to Coulomb repulsion.
The $\Delta m/m$ fraction varies from 10^{-27} of gravity to about 10^{-3} of the nuclear force. In classic mechanics, the simple 0-D relationship between PE change and KE gain is

$$\Delta(\mathrm{PE}) = ½\ \mathrm{m}(\mathrm{Ve})^2 \tag{2}$$

predicting that the Ve (the Isp) scales with the *square root* of $\Delta(E)/m$, a weak dependence. (Note the equation above does not refer necessarily to a fluid.)
Table 1 shows conversion ratios and energy/unit mass, J, of the three forces. (metastable nuclear isomers are still a controversial subject, see [Czysz and Bruno, 2009].)

Type of force	*Potential*	*alpha*	*Energy density, J (J/kg)*
Gravity	gravitational	10^{-27}	10^{-11}(*)
Electro-weak	chemical (H_2/O_2 combustion)	1.5×10^{-10}	1.35×10^{7}
Strong Force	Nuclear: Fission (^{235}U)	9.1×10^{-4}	8.2×10^{13}
	Fusion (D-T)	3.75×10^{-3}	3.4×10^{14}
	Metastable (^{180m}Ta)	2×10^{-7}	1.8×10^{10}
	Annihilation (p^+-p^-)	1.0	9×10^{16}

Table 1. Force, potential, conversion factor alpha = $\Delta m/m$ and energy density J = PE/m of the three fundamental forces. The J of gravity refers to two unit masses at 1 m distance [Czysz and Bruno, 2009].

Comparing these three forces shows that the J of CP, based on the second, is eighteen orders of magnitude larger than gravitation, but six smaller than the third. Dividing J of the second force by the specific heat of chemical reactions products yields the maximum temperature of a CP cycle, not much higher than 3500 K, and anyway limited by the melting point of materials (ruled by the same force). When using realistic numbers, Eqn. (2) shows that the Ve and Isp are correspondingly limited to 4-5 km/s. Hence, as said, Isp is of order of the ΔV needed to change orbit, and Eqn (1) predicts large mass consumption: for instance, the ΔV from ground to LEO (in practice, 9-9.5 km/s) results in a (Min – Mfin)/Min ratio, the payload fraction, of order of a few percent. Thus, during a satellite launch, almost all propellants mass and energy release is used to lift and accelerate the propellants themselves, not the payload.

2. When adopting the *second* strategy, [ionized] molecules/atoms are still accelerated by electroweak forces, but the Coulomb or Lorentz force is external to the plasma and requires an *external electric power source*. The Ve possible are much larger than in a thermodynamic expansion, since they are neither limited by the upper and lower temperatures of the cycle, nor by the melting point of materials. Accelerating ionized matter by Coulomb or Lorentz is, in principle, limited only by relativity, although in practice voltages and magnetic fields cannot be made large at will by arcing, magnet mass, critical current density and Meissner effect.

Using *classical* (Newton's) mechanics, the conclusions for space propulsion are as follows:

- Isp scales with Ve
- Thrust, F, scales with (flowrate) times (Ve), thus with $(Ve)^2$
- Power of the ejected flowrate is the kinetic energy (of the mass ejected) per unit time, or ½ (flowrate) times $(Ve)^2$, and thus scales with $(Ve)^3$.

(*Relativistic* relationship are formally different, but predictions differ only when Ve is > 0.99c, and will be ignored here).

Based on these simple scaling laws, to reduce mission time F must be increased. This implies Ve must be as large as possible; when doing so also mass consumption decreases. There is a price, however, and that is the power required, scaling with $(Isp)^3$. As said, CP is *fundamentally limited in Isp* (in Ve) by J = Δ(E)/m that can be extracted from rearranging chemical bonds.

Thus the next logical step is to seek *other potentials* providing *larger energy density* J. Of the three fundamental forces we know, only the *nuclear force* remains, since gravity is so weak that only assist (flybys) manoeuvres, exploiting the mass of entire planets, can supply enough energy to a spacecraft, at the expense of lengthening, not shortening, mission time. The *nuclear force*, see Table 1, converts about 0.09% of the 'fuel' mass into KE in the case of ^{235}U fuel, and 0.3% in the case of D-T (deuterium-tritium) fusion, a factor ~10^6 larger than possible in chemistry. It is logical to look at nuclear reactions as the *only solution* to the need to increase J, thus Isp, thus thrust, finally enabling faster missions with drastically lower mass consumption. Thrust, and travel time, scaling with $(Ve)^2$, will depend on the NP strategy outlined already. The first, nuclear thermal propulsion, or NTP, has produced in time the family of NP devices called nuclear thermal rockets (NTR), the second, nuclear electric rockets, or thrusters, or (generally) nuclear thermal propulsion (NTP). Strategy 1 began to be explored in the '50s. Strategy 2 was explored by Stuhlinger during WW II, but started being investigated only much later.

4. Nuclear thermal propulsion (NTP)

4.1 Principles and performance

In both US and SU NTP took the form of a compact nuclear reactor (NR) transferring heat to a coolant (e.g., LH2) brought to a temperature T limited only by materials. This temperature was about 2300 K in US reactors, close to 3000 K in SU. Heated hydrogen was then expanded in a conventional nozzle, similarly to what done in CP rockets. The Isp of such NTR is higher than in CP, because the molecular weight of GH2 (about 2) is a factor ~ 5 lower than that of a LOX/LH2 rocket (about 10). See [Lawrence, 2008] for details.

The heat produced by a NR is due to nuclear reactions 'burning' (that is, fissioning) nuclear 'fuel', e.g., ^{235}U-enriched uranium, for instance, encapsulated in bars, rods or pins. Since U has a melting point about 900°C, it is alloyed with O, N or C to form U-ceramics, with much higher melting point. Fissioning splits U nuclei into smaller fragments (FF), including neutrons fissioning other U nuclei. This chain reaction produces therefore FF consisting of unstable isotopes (most belonging to one or another of two families, with mass centered around 90 and 130, respectively); neutrons; and gamma photons. FF possess kinetic energy in the O(1) to O(100) MeV range. By colliding with nuclei of the rod material and of the fuel itself, they deposit their kinetic energy as heat ('thermalize'). In a solid-core NR this heat is transferred to the coolant/propellant flowing inside the many channels present in each rod, see Figure 2.

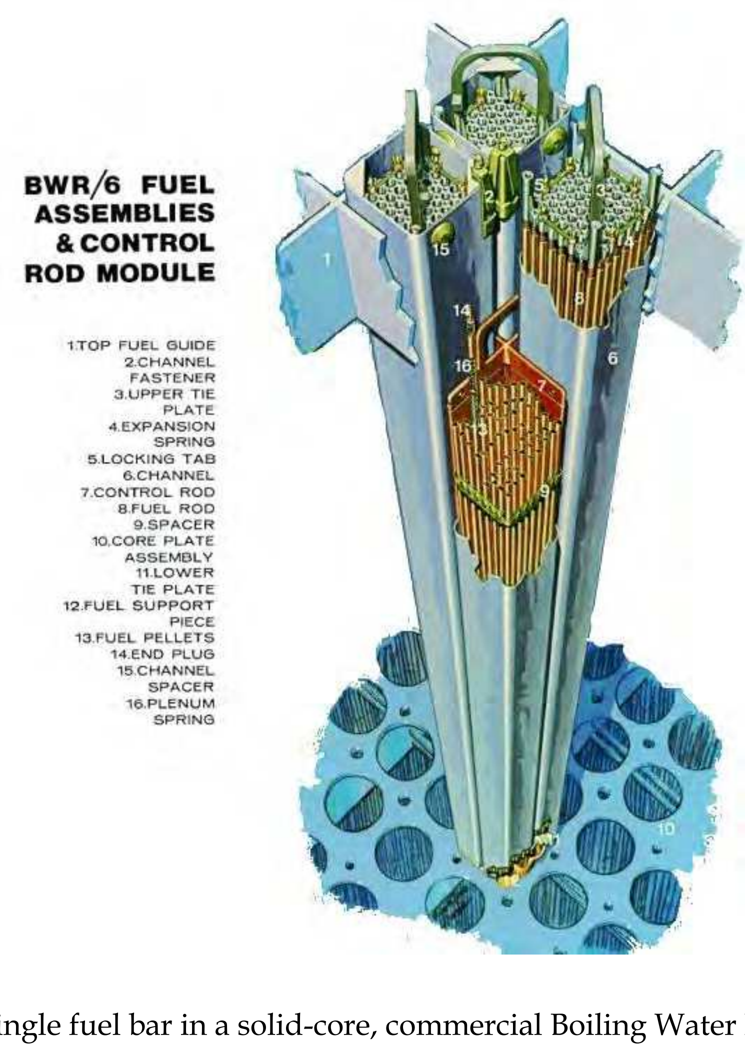

Fig. 2. Sketch of a single fuel bar in a solid-core, commercial Boiling Water NR

This strategy is the same of LRE, except the heat source is external to the propellant, not due to its chemistry. It started in the US, where space NR research (under the ROVER project) became the responsibility of Los Alamos Scientific Labs (LASL, now LANL). Westinghouse was responsible for engineering the LASL *reactors* into *rocket engine*s. After ROVER was terminated in 1972 by the Nixon administration, the USAF took over and built NR using ceramic fuel (CERMET) capable of withstanding hundreds of on-off thermal cycles without cracking, and more compact NR using fuel *pebbles* instead of bars (pebble bed reactors, PBR). In terms of sheer power the high point of ROVER was the Phoebus IIA reactor, tested at 4200 MW for 12 minutes. Less powerful (10-100 MW) but more versatile NR were designed and tested; see [Turner, 2005; Gunn and Ehresman, 2003] for details.

Not as well documented is NR work by the US Navy. The Navy wanted torpedoes faster than conventional (60-70 knots). The Brookhaven National Labs (BNL) designed NR for a standard-size torpedo (about 25 in dia.). Power was in the tens of MW range (note that life expectancy was only few minutes). Thus thermal power/unit volume was roughly of order MW/liter. In time, this work was applied conceptually to a family of space NTR concepts with the MITEE acronym by the US Plus Ultra Technologies company. Plus Ultra Technologies looked at even more compact NR using fuels with low critical mass, e,g., Am 242 and others [Powell et al, 1998, 1999]. Lower critical mass means lower fuel volume and thus lighter shielding, since the thickness of a neutron + gamma shield is a constant for a given material. (with graphite, a few decimetres), and thus shield mass scales with NR area. Am-based fuel was also proposed in Israel by Prof. Y. Ronen at Ben Gurion University, and by C. Rubbia in Italy for their independently developed FF NTR powering manned Mars missions (M3), see [Accettura et al, 2008; Ronen et al, 2000].

Still not translated is most SU/Russian work. Contrary to what happened in the US, SU work in NTP never stopped, and All Union conferences have been held until recently. In 2010 General A. Perminov (then head of Roskosmos) claimed to have the NP technology for a M3, and his statement was echoed by a statement by NASA Administrator Bolden, who went in print wishing to collaborate.

Russian capability is apparent from articles, e.g., [Koroteev et al, 2007] emphasizing the importance of bi-modal approach also proposed at NASA-Glenn by S. Borowski. In this approach part of the thermal energy of the NR is used for thrust, and part to generate electricity to power all spacecraft systems and also high Isp electric thrusters (ET). This strategy can produce large thermal thrust (many kN or even tens of kN, depending on NR power), at moderate Isp (880 s were demonstrated by Westinghouse NTR in the '60s; Russian engines were capable of more than 900 s; 1000 s are claimed feasible with MITEE technology). The thermal thrust would allow a spacecraft to quickly reach escape speed from LEO; then the NTR would be switched off and the much lower thrust ET, with Isp in the 4,000 to 15,000 s range, would be turned on, saving propellant mass during most of the interplanetary trajectory.

Dual mode operation avoids in principle the spiral trajectories inevitable with the low thrust typical of all ET, and thus avoids crossing the deadly van Allen belt tens or hundreds of times. Plus Ultra Technologies has also proposed a dual-mode concept based on MITEE [Powell et al, 2003].

In sum, NTP is a well explored and mature technology in Russia, and still viable in the US, although experts are retiring, retired or passed away. It is capable of thrust in the tens of kN, with Isp in the 800 to 1000 s range. Uranium nitrides and carbides (and more advanced) fuels are available. The large thrust of NTP enables interplanetary missions much faster than with CP, but Isp is only about a factor 2 better than with CP, so LH2 consumption implies substantial LEO orbiting cost. This said, in combination with ET could be a viable and very promising propulsion technology, indeed suggesting itself as a candidate system for M3 in the framework of future global collaboration. Although much less mature, the Ronen/Rubbia engine is another strong NTP candidate, with F in the O(1) kN, and a very interesting Isp ~ 2000 s.

4.2 Unconventional NTR

4.2.1 Gas-core NR

Almost all NTR tested in the US and the former SU have a solid core (rods, bars or pins) containing the fissionable fuel. Liquid- and gas-core NR have also been investigated, but (to this author's knowledge) never built or tested. US and Russian work [Howe et al, 1998; Koroteev et al, 2002] has focused on gas reactors because their Isp (and thrust) may be much higher than in solid-core NR, e.g., in the many 1000s of s. Their main difficulty is how to transfer energy from fissioning hot gas to propellant. The only heat transfer mechanism found to reduce the inevitable mixing losses is radiation, and that requires a heat exchanger made of a radiation-transparent material. This concept goes under the name of "nuclear light bulb", see [Czysz and Bruno, 2009, Section 7.12].

4.2.2 Rubbia and Ronen FF engine

Another unconventional NTR concept briefly mentioned was that originally proposed in [Shepherd and Cleaver, 1947] and rediscovered independently by C. Rubbia and Y. Ronen in the late '90s [Accettura et al, 2008; Ronen, 2000]. It is based on thermalizing the fission fragments generated by fissioning an unconventional fuel (e.g., ^{242m}Am) directly inside the propellant, without first heating solid rods. In this concept GH2 flows inside a chamber, or channel, where walls have been coated with a fissionable thin layer of fuel (Y. Ronen has theoretically proved the feasibility of fission inside a thin layer). The isotropically emitted FF thermalize partly inside the flow of GH2, that heats and then expands in a nozzle, and partly inside the coating, that must be cooled. The coolant may be the propellant itself, f.i., LH2, that is injected and gasifies in the channels. This strategy is similar to regenerative LRE cooling.
The advantage of the Rubbia/Ronen Engine is its Isp, predicted in the 2000 - 2500 s range, coupled to the significant thrust typical of all NTR. Assembling a cluster of such engines can supply many hundreds, perhaps thousands, of N in a compact volume. Extant problems are cooling of the reactor walls, of the nozzle, and materials.

4.2.3 Propellant-less NTR concepts

This form of NTR avoids all external propellants. That is, the NTR uses as propellant the FF themselves [Czysz and Bruno, 2009, Section 7.13]. Neutrons and gamma photons do not contribute to thrust, being too light, but isotopes created by fissioning U do. These have energies of order O(100) MeV, with theoretical Isp in the 10^6 to 10^7 s range, but their thrust is very small, and collimation hard to achieve. The concept was pushed to its limit by Eugen Saenger, the inventor of the photonic rocket [Saenger, 1958]. This rocket uses a NR to heat a paraboloid surface to high T. The surface emits and collimates photons, so the *ideal* Isp available is the speed of light c = 3 x 10^8 m/s). In fact, when accounting for consumption of fissile fuel mass the effective Isp = F/(dm/dt) is less than c, and Saenger calculated actual Ve of about O(100) km/s. The same concept was proposed in the '90s by C. Rubbia, who was not aware of the Saenger work. In Rubbia's example, a 1-Gw NR perfectly converting all its energy to light would produce 3 N, with Isp = c (Rubbia did not include in Isp the mass consumed by the Einstein relationship). Although fascinating, the S&T problems posed by the photonic rocket are immense: to achieve the performance predicted by Rubbia the temperature and emissivity of the paraboloid surface would have to be that of a perfectly black body at about 4000 K. This said, the concept is interesting and (to 0-th order) physically sound, although hardly realizable with foreseeable technology.

4.2.4 The Orion project

This concept offers an alternative to the dichotomy "either high Isp and low F, *or* viceversa". The Orion is an idea (1947) of Stan Ulam (the father of the US thermonuclear bomb). Ulam thought of exploding atomic 'bombs' inside a rocket chamber to produce thrust, but quickly realized 1. that atomic explosions created pulses of energy and momentum many orders of magnitude larger than chemical, and therefore 2. that there was no way of confining an atomic explosion inside a chamber. Thus the solution was to have the atomic explosion(s) take place *outside* a vehicle, unfortunately with possibly destructive effects on its structure.
Thus his remained a pure idea until the 1954 US thermonuclear bomb test on Bikini: heavy steel spheres coated with graphite and suspended before the test, were found intact but at some distance away. The data analysis that followed the test showed that the ablating graphite protected the spheres, and that their motion was the effect of gas-dynamic (and radiation) momentum transfer from the expanding fireball gas to the spheres. These facts fueled interest in a proposal to the AEC, and the "Orion Project" took off in 1958. It lasted until the 1963 Partial Test Ban Treaty banned all atmospheric tests that killed it.
The Orion concept consisted of accelerating a spacecraft by periodically exploding astern atomic 'bombs' of yield from 0.03 kton to 0.35 kton each. The yield depended on spaceship mass, while the number of explosions was fixed at 800 to reach LEO. A massive thrust plate fitted with shock absorbers received the momentum pulse from each explosion and accelerated the spaceship at 2-4 g.
In fact, under Orion a family of spaceships was developed capable of orbiting to LEO payload from 300 to 6100 ton. The word 'spaceship' is appropriate, as all were single-stage (taking off from the ground) see [Ewig and Andrews, 2003] for details. The 'bombs' were in fact sophisticated atomic shaped charges. A sketch of an Orion spaceship is in Figure 3.

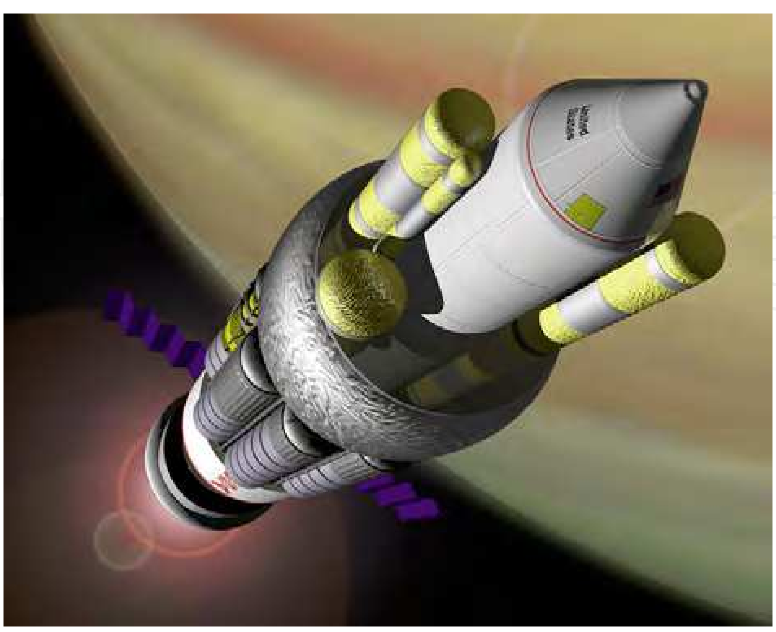

Fig. 3. Notional sketch of an Orion spaceship near Jupiter. Thrust plate and spacing between plate and shock absorbers assembly are shown.

All mechanical, thermal and control issues associated with this method of propulsion were analyzed, in particular the thrust plate, and all eventually were solved with engineering know-how of the time. A test with conventional HE confirmed performance and scaling expected. That requires discussion of Orion peculiar gas dynamics.

In Orion spaceships the energy release occurs *outside* the vehicle, not internally. The fundamental point, as Ulam realized, is the fact that the velocity of the shock wave created by each fireball is *at least* an order of magnitude larger than the Ve of CP-rockets, and actually depends on distance between fireball and thrust plate. This distance is a compromise between capturing as much as possible the momentum pulse and avoiding plate destruction by radiative heat transfer and by the expanding gas momentum. A typical calculated distance was 200 m, where the debris velocity, here still named Ve, was (2 - 3) x 10^4 m/s.

In fact, fundamental thermodynamics, i.e., Eqn (2), shows that a 10 kton (4.18 x 10^7 MJ) explosion produces after a few ms a 100-m fireball at temperature ~ 50 to 70 k°C. The fireball expands at speed Ve scaling of course with (temperature)$^{1/2}$. For the 10 kton yield, initial Ve is ~ 100 km/s, dropping with the inverse of distance squared. With higher yield the Ve scales according to Eqn (2).

It is the hot gas at velocity Ve that transfers most of its momentum (some is also imparted by radiation pressure) to a surface; if the energy release is isotropic, the fraction transferred depends roughly on the solid angle of the cone with apex in the center of the fireball and base equal to the thruster plate surface. In fact, each 'bomb' was conceived as a *shaped charge*, that is, containing a 'reaction' mass designed on purpose to impinge on the plate, so the momentum calculated was higher than that predicted solely on the basis of the angle.

In practice, the *empirical* (but physically consistent) relationship for the 'equivalent Isp' of each explosion pulse is Isp = (C_0) x (Ve), where C_0 is a collimation factor of order 0.5 when fireball diameter and thrust plate diameter are well matched.

The calculations indicated this NP method ensures equivalent *Isp about 2-3 times larger* than any NTR, with a thrust F scaling with (Isp) x (reaction mass of each bomb) x (explosion frequency). In most of the Orion family spaceships the pulse frequency was 1 Hz. F can also be estimated from ship mass and its impulsive acceleration data. The F reported for the smallest Orion spaceship (880 ton at take-off) was about *10^6 N.*

If restricted to space, the Orion concept is still valid. In his "Cosmos" TV series Carl Sagan proposed, in fact, to use the charge of nuclear warheads dismantled because of nuclear ban treaties, as fuel for future Orion-type spacecraft. It was estimated that a single Orion spaceship could lift a complete human outpost to the Moon.

Based on physics, the Orion concept offers the largest possible Isp and F combination for a given energy pulse (yield). Engineering it into a spacecraft has its problems, but the work done indicated they are solvable. Rather, the issues are political (exploding bombs in space), and the availability and consumption of nuclear fuel to build the shaped charges. Given the political will (a rare commodity these years) Orion could be made into a true, *reusable* spaceship. In fact, Orion showed how to exploit the third force to create the ultimate (high Isp, high F) NP system, the only capable of exploring our planetary system in reasonable times.

The Orion concept was resurrected in recent years and molded into an ET called MagOrion, then scaled down to the MiniMagOrion (MMO) described in Section 5.1.

5. Nuclear Electric Propulsion (NEP)

The second NP strategy consists in using a NR to generate electric power fed to an ET. ET may use either the Coulomb force, e.g., Gridded Ion Thrusters, GIE, or the Lorentz force, as in magneto-plasma-dynamic (MPD) thrusters. Hall thrusters, VASIMR and others are of this type. Their description is documented, for instance, in [Goebel and Katz, 2008; Rossetti et al, 2008]. A review of NEP has been recently published in [Cassady et al, 2008]. Here the focus will be on near term to long term ET powered by NR rather than on current ET technology.

GIE are mature and are commercialized, e.g., as the Boeing XIPS family, with Isp 4000 to 4500 s. Xe is the propellant of choice for GIE. Ionizing Xe absorbs 5-6% of the total electric power. More importantly, in all types of ET ionization means low pressure, because recombining ions and e^- need a third body, and thus recombination rate grows with $(\text{density})^3$. Because of low pressure operation, the actual thrust of GIE, and ET in general, is very small. The thrust of a state of the art 'high thrust' satellite GIE (powered by solar power) may be a fraction of N. F is basically limited by solar power available, in turn limited by solar panel area and efficiency. If significant thrust is desired, GIE must be assembled in clusters. These would be orders of magnitude bulkier than CP thrusters in the same thrust class. However, their mass consumption, ruled by Isp, is orders of magnitude less.

The low F of ET means that significant spacecraft ΔV requires either engine-on time comparable to the entire mission duration, $\Delta t/T \sim 1$, or that an ET consisting of tens or hundreds of individual modules, with a corresponding increase in electric power. A key issue with high power (high voltage) GIE is grid life, since some of the ion beam current impacts the grid system. With most MPD thrusters the key issue is anode/cathode erosion. In both cases these issues are due to ion impingement, and determine ET life. Because ET thrust is 'low' (at least using solar power), mission time with ET propulsion may be years, as in ESA's SMART-1 or JAXA's Hayabusa spacecraft.

The VASIMR engine being developed by the Ad Astra Company [AdAstra, 2011] by F. Chang-Diaz, and to be tested on the ISS in the near future, is as promising a MPD thruster as it is ambitious. It is electrode-less, with ionization and energy feed realized by plasma radio-frequency. Its technology is borrowed from tokamak fusion technology. As reportedly planned, it should deliver *any* combination of F and Isp while keeping their product (~ power) fixed. The Isp of VASIMR tested so far is still about 5000 s rather than the 10,000 s planned, seemingly due to power losses ~ 30 to 40 %. These are under investigation but are probably due to radiative heat transfer and incomplete conversion inside the magnetic nozzle of the ionic circumferential velocity component to axial. A 200 kWe VASIMR will be tested on the ISS in the near future, but the power needed by tests will be from batteries, not a NR. VASIMR systems in the 10s and 100s of MW have been proposed; if built, they will need nuclear power: solar power (and thermo-ionic power) is inadequate above a few 100s of kWe.

The disadvantages of NEP noted, its advantages are remarkable in terms of Isp and ease of coupling between power generator and ET. Note that EU and the US have extensive GIE know-how, while Russia leads in Hall (MPD) thrusters. In addition, Russians have designed many conceptual dual-mode engines, and advocate NP as the best solution for a M3 [Karash, 2011]. VASIMR is the most promising concept in ET, but is still not mature. In any case, low F remains the Achille's heel of all ET.

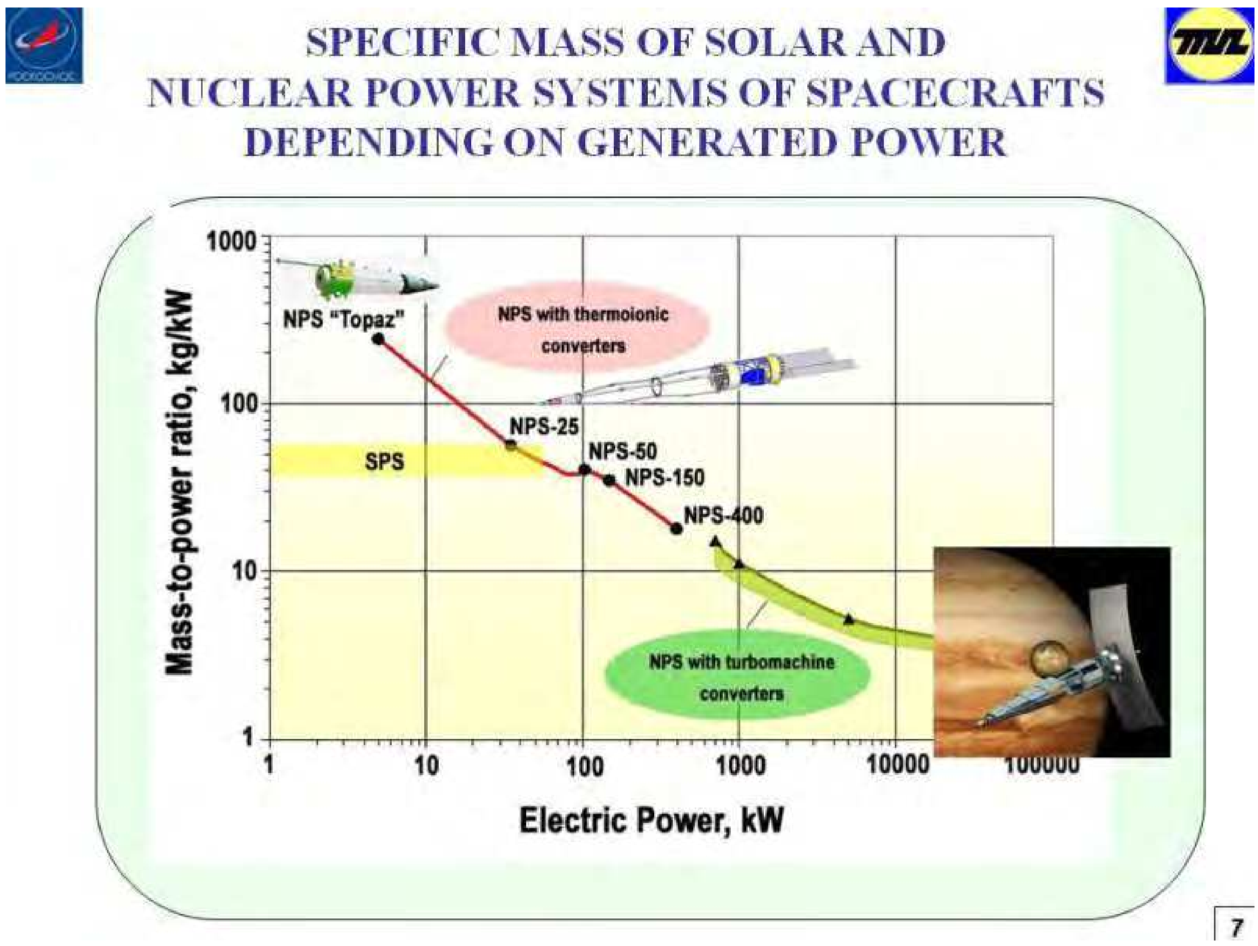

Fig. 4. The alpha parameter (kg/kW of propulsion system) of thermo-ionic and NR-powered electric propulsion systems. Courtesy of A. Koroteev, Keldysh Research Center (KERC), Russia.

Clustering nuclear ET may solve these problems, but if the NR must function reliably and autonomously for most of the mission, the issue becomes that of NR refuelling. The NP group led by A. Koroteev at KERC has presented NR concepts where refuelling is done mechanically. To date, this concept has not been tested.

NEP has other drawbacks besides low F. The thermal to electric conversion step is one. Standard conversion is thermodynamic (e.g., a Brayton closed cycle with heat exchanger). The mechanical power drives alternators/dynamos. In the MWth range this technology is considered mature by many. Unofficially, Soviet NR using liquid Na or NaK eutectic are known to have powered some of their reconnaissance satellites requiring O(100) kWe. Conversion efficiency (thermal to electric) $\eta \sim 25$ % is feasible with total mass is ~ 1 to 2 ton, including shielding. Heat extraction may recommend a gas cycle, a liquid metal cycle, and may use heat pipes, see [Lenard, 2008].

Because of thermodynamics, the third drawback of NEP is the 'bottom' T at which the (1-η) fraction of the thermal power not converted to mechanical work must be dumped. Dumping must be done by a space radiator, and it is a fact that such radiator may be more massive than the NR itself, and certainly bulkier. Note that the bottom temperature of a standard NEP cycle is critical: too low, it cannot radiate away enough waste power; too high, the η drops.

Thus NEP not only may waste up to ¾ of the NR power, but adds much weight and bulk to the spacecraft. This explains why the parameter *alpha = mass/power of electric propulsion systems* (measured in *kg/kWe*) is at the moment about 10 or a little less, see Figure 4. For comparison, NTR systems have alpha about 100 times lower. Alpha is very important in determining the performance of EP-, or NEP-powered missions, see [Stuhlinger, 1964] for a must-read reference.

Work in NEP is at the moment being carried on in Russia, the US and France, with Russia favoring dual mode NTR/NEP operation. Within the next 1-2 years a decision about an international Mars mission is likely and will indicate also the preferred propulsion system for future M3.

5.1 Unconventional NEP

Like NTR, all NEP thrusters mentioned (GIE, Hall, VASIMR) use steady operation. The Orion history suggested to Andrews Space & Technology (AS&T) Corporation, in the US, to investigate NEP whereby pulsed nuclear explosions become a Lorentz force periodically pushing the spacecraft. Thus energy pulses are converted to momentum pulses electro-magnetically rather than through gas dynamics. Andrews Space's concept (called MagOrion) is therefore a pulsed MPD thruster. The frequency of atomic detonations was assumed 1 Hz. This concept was called MagOrion. Each explosion produces ionized debris, that interacts electro-magnetically with a 2-km superconducting ring. This accelerates the debris and produces thrust, see [Ewig and Andrews, 2003; Lenard and Andrews, 2006] for details. The thrust/weight ratio was predicted to be in the 0.2 to 10 range (0.2 to 10 m/s^2 acceleration).

Because of size (especially that of the superconducting ring, impossible to achieve in practice) MagOrion was eventually scaled down, taking the name MiniMagOrion (MMO). In MMO the superconducting ring was replaced by a magnetic nozzle analyzed in collaboration with the University of Washington. MMO was tested piecewise, meaning that some ersatz components were tested using the unique facilities and energy sources at the Sandia National Laboratories.

This partial testing confirmed part of MMO physics. The future of this concept envisaged in 2006 was a manned spacecraft, for instance shaped as in Fig. 5. Calculations indicated Isp of order 10,000 s is potentially feasible, with thrust depending on vehicle size and fuel yield, but, typically, in the 10^2 ton. An interesting feature of MMO is that each nuclear fuel pellet is [periodically] compressed to the critical target density (the fuel density times pellet radius, in g/cm^2) by the plasma effect called Z-pinch, driven by fast rising current density. (This method of compression is similar to that proposed by the late R. Bussard to adiabatically compress and fuse D-T fuel in his "QED" inertial confinement fusion concept.)

MMO research at AS&T investigated and compared several fuels, ^{235}U, ^{233}U, and ^{239}Pu among others, and settled on ^{245}Cm, a fuel with the theoretical lowest critical target density. ^{245}Cm is a component of commercial NR waste, with convenient neutronics properties and good 'burn' fraction. In the MMO is in the form of hollow pellets coated with a Be neutron reflector layer.

Estimated performance of the baseline MMO vehicle in [Ewig and Andrews, 2003; Andrews and Lenard, 2006] shows Isp = 10,000 s, thrust 1100 kN, and that a ΔV = 100 km/s is feasible with a 100 ton payload. The parameter alpha is of order 2.5 x 10^{-4} kg/kWe, to be compared with alpha = 10 or so for solar-powered ET. Although appealing, the MMO physics has not been investigated in the same depth as Orion's. The efficiency of converting energy pulses to thrust still awaits confirmation; but to this author's knowledge, all research in MMO has stopped.

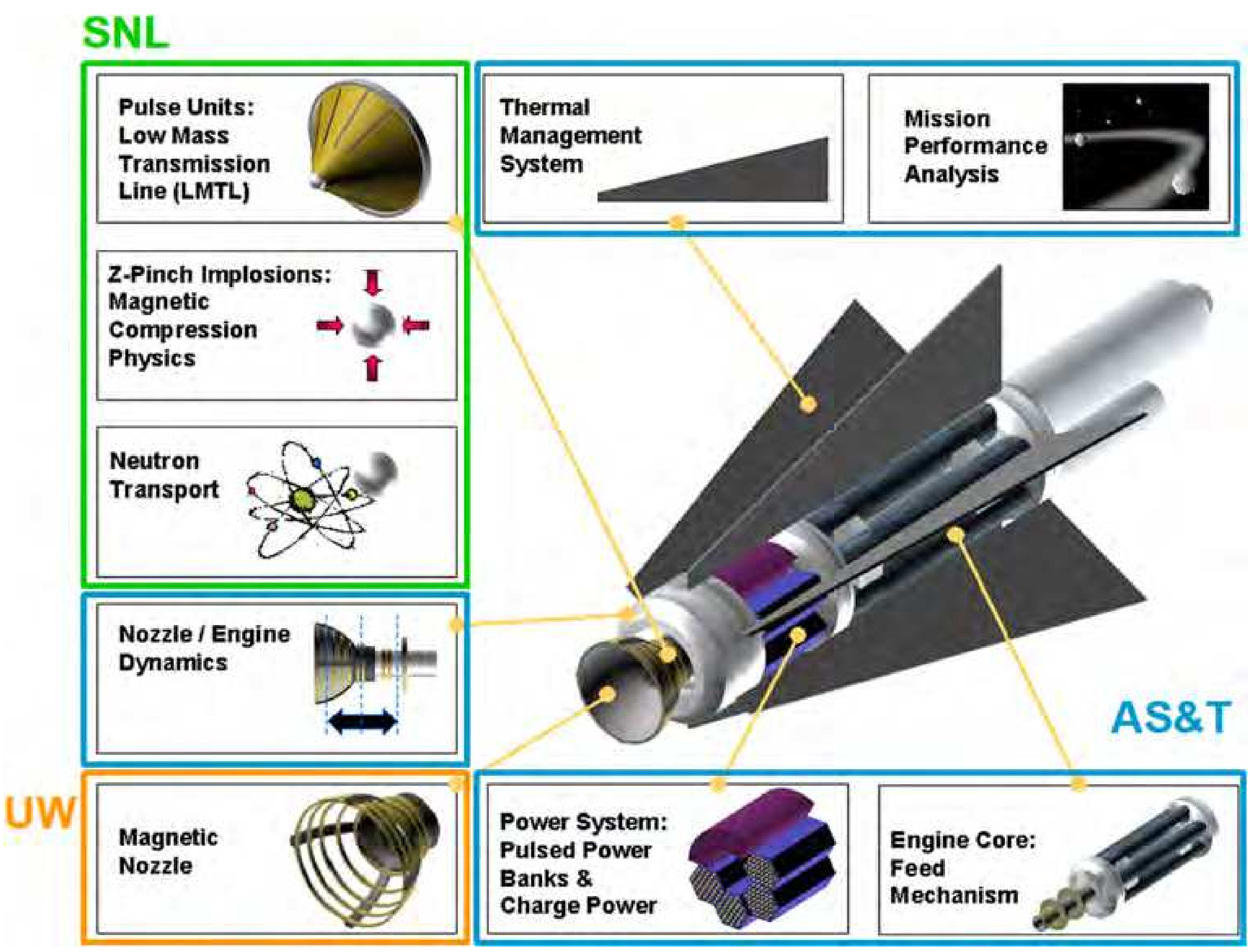

Fig. 5. Sketch of a notional MiniMagOrion spacecraft and of its main components (Ewig and Andrews, 2003).

5.2 Pulsed operation

Orion concepts beg the question of whether releasing energy in pulses is more convenient, or performing, than steady-state operation. Time-averaging constant frequency pulses shows thrust is lower than steady state operation of the same amplitude. In fact, there might be some advantages. It is known, see Section 2.2, that impulsive F results in Hohmann minimum energy transfer trajectories. Besides, ideal propulsion efficiency depends on the ratio between spacecraft speed and exhaust jet speed, and peaks when this ratio is unity. Thus impulsive thrust may be more efficient, as the ratio between [time-averaged] ΔV and exhaust speed may be made close to one 'instantaneously', rather than slowly. In rocket launches the lift-off, when Ve is of order 3 km/s and V is a few m/s, is notoriously the most inefficient fraction of the launcher trajectory.
A third consideration is that fluid dynamics of a suddenly accelerated fluid (f.i., expanding in vacuo, as is the case of Orion), predicts its "limit velocity" higher by the factor $[2/(\gamma - 1)]^{1/2}$ than the steady state limit velocity for the same reservoir conditions [Thompson, 1972].

This said, the case for impulsive Orion and MMO concepts is made simply because the effects of detonating a nuclear charge (Ve and thrust) are per se very large, and because a steady, or quasi-steady nuclear explosion is unlikely (possible exception: X-ray laser technology). In this light the three effects mentioned are probably insignificant.
The limit to this strategy is posed by maximum acceleration tolerable by structure, payload and especially crew. Humans can tolerate steady acceleration ~ 3 g, much higher for much shorter times even if periodically repeated, e.g., 10g can be tolerated for times < 0.1 s. A spacecraft shock absorbing system is a must in this context. In the case of MMO, the EMP may damage payload and humans: these have evolved in a B = 0.6 gauss (6 x 10^{-5} T) environment, and do not take easily to EMP of order 1 T or higher. For instance, anecdotal reports of the effects of B ~ 1 T for periods ~ 1 s include 'seeing' colored lights.

5.3 The future of NEP

NEP is being considered by many space agencies and experts as potentially more suited than NTP to manned interplanetary missions. The reasons are partly scientific and partly based on engineering.
With NEP, electric thruster (ET) and NR are effectively uncoupled, and lend themselves to preliminary separate optimization (as the aeroengines for commercial airliners, for instance). Mathematical criteria to pick the best Isp for given mission speed and payload have been developed since the '60s [Stuhlinger, 1964], and reliable ET have been space-qualified and flown for many tens of thousands of hours. The coupling between power source and ET is more flexible and less sensitive to feedback than in NTR. In principle, at constant NR power, that is at constant (Thrust) x (Isp), thrust and Isp may be chosen to optimize a mission, for instance by using high F to reach quickly escape speed from LEO, and then switching to lower F and high Isp, thus consuming less propellant, during the interplanetary leg.
The specific type of ET may influence the propulsion architecture considerably, and may ultimately determine its future. The thrust of GIE depends on the *square root* of voltage, and 70 kV is cited as the future limit in [Fearn, 2008]. With this voltage the future thrust of a single GIE may eventually reach 10 N. Thus 100 N, a thrust significant for interplanetary missions, will need a (3x3)-module GIE. With high current 60 cm dia. modules, the total assembly bulk is not significant. An advantage of clustering is that the **F** vector can be modulated step-wise. At the same time, heat dissipation and arcing may be problematic.
On the MPD side, high power thrusters (e.g., Hall, VASIMR) absorb many 1000s A. Ohmic losses may become limiting, suggesting superconductor wiring and magnets. Using LH2 propellant would enable the switch to this technology. Ground testing of MPD ET becomes another problem at the power level enabled by NEP, dictating individual or of downscaled module testing, when feasible. One positive side of MPD thrusters is their higher thrust per unit power, even though that means inevitably lower Isp at fixed NR power.
In the long term, the jury is still out on which of the two ET families will become the standard, and probably the choice will depend on mission optimization; however, it seems that future nuclear-powered MPD lend themselves to better integration of propellant, thrust and Isp management by using superconductivity. In both families ionization of the propellant will eventually do away with electrodes, that either because of massive current or of very high voltage tend to erode, degrade and break down. Electrode-less ionization by RF, as in fusion tokamak, has been implemented successfully

in the VASIMR thruster, and there is no reason why it should not work also in future ET absorbing tens or even hundreds MW.

As for performance, future nuclear-powered GIE built with tested tokamak refueling technology may accelerate hydrogen to 100 km/s (Isp = 10^4 s) producing F = 10 N. Ideal power is then 1 MWe [Fearn, 2008]. This number, times a factor 3-4 gives the thermal power of the NR.

Future GIE and MPD with even higher Isp (~ 10^5 s) are potentially feasible with the same technology and will need 10 MWe, and hundreds of MWe for thrust of order 100 N required by fast missions (missions lasting only a few months).

These figures give pause to any ET expert, used to solar-powered ET in the 10 kW range. But power of order 100 MWe is nothing to commercial NR engineers. Thus, IF space agencies worldwide will collaborate, for instance in a M3, such figures may well become commonplace.

5.4 When is NP useful?

The crucial advantage of NP is the power the NR can supply, thus thrust, and Isp. The energy density of NR is orders of magnitude larger than any other power source, in particular of solar panels (SP), see Table 1. A rule of thumb is that above 250 – 300 kWe a NR is lighter and less bulky than SP, but that depends also on mission and solar panel technology. Certainly it is true for missions toward the outer planets, since solar power decreases with $1/r^2$.

The issue of power becomes the key issue when looking at the main objectives of human exploration under current consideration by space agencies. According to the ISECG international organization coordinating future human exploration, they are the Moon, NEO/NEA, and Mars, see [ISECG].

Going to the Moon does not require NP, but building and maintaining an outpost there will definitely need a NR during the long lunar nights and especially to carry on power intensive surface activities, e.g., ISRU, or to dig regoliths to build radiation shelters. In fact, NR in the 1-MWth range are included in most lunar scenarios. Mining will of course need far more power.

For human mission to [still unspecified] NEO/NEA the question of power becomes acute. Distance from the sun will likely rule out *present* technology SP. Restricting this discussion to propulsion, NP power will depend on planned mission time. Due to SR+ GCR dose, crew exposure time must be less than 5-6 months to limit cancer risks, although these numbers must be taken with a grain of salt, waiting for more research on the effects of radiation.

Roughly speaking, these times dictate thrust in the many 100s of N at the very least, to avoid crossing the van Allen belt too many times while escaping. This thrust would be too large for *SP*-powered clusters of GIE or [future] MPD ET. Note that NTP systems in this thrust range were never built, so NEP is the obvious choice for the trans-NEA portion of a trajectory. To avoid spiraling around the Earth and crossing several times the van Allen belt while accelerating toward escape speed, pure NEP must be complemented by either a chemical rocket, or be replaced by a hybrid NEP-NTR combination. This conclusion is actually tentative: VASIMR progress may indeed result in a ET capable of varying F and Isp inversely at fixed NR power. One may roughly estimate that for F of order O(100) N and a reasonably high Isp, say, O(10,000) s [Coletti, 2011], electric power needs be O(10) MWe. While significant, this power is modest in terms of NR capability. Thrust in this range enables a roundtrip mission to some asteroids lasting only a few months, with a significant reduction in NR and ET life requirements.

This said, the future of SP includes [initial] efficiencies up to 42% (current high performance cells are at 28%). When these are mature, SEP may, or might, become competitive for certain NEO missions, even those requiring beyond 300 kWe. This tentative conclusion is important in view of the 2025-2030 timeline envisaged by ISECG for future deep space missions.

A M3 is a good deal more challenging. A conceptual pure-NEP-driven M3 mission has been already studied [Ferraro et al, 2009], admittedly using GIE with Isp equal to the highest tested in the lab, ~ 37,500 s (this Isp is nine times higher than commercially available now, but the tokamak fusion technology already mentioned will in due time result in Isp in the many 100s of ks [Fearn, 2008]. With these assumptions a NEP-powered Mars transfer was predicted to take between 200 and 100 d with GIE power between 100 and 300 MWe. Total spacecraft mass was between 150 and 100 ton, respectively. A significant fraction of mission time was due to spiraling to reach Earth escape speed.

The crew dose for such fast trip would be quite tolerable with structural areal density (Al, or Al + equipment), 5 to 20 g/cm^2 as on the ISS [Durante and Bruno, 2010]. The latter showed that NEP is a key enabling technology to lower radiation risk. Presumably, in combination with NTR, or based on VASIMR-type ET, NEP may be even more convenient.

A second example of the capability of NP is associated to the future of ISS. The ISS life has been extended beyond 2020 and perhaps even beyond, to 2028. In this context a concept has been proposed [Lorenzoni and Bruno, 2011] fitting the ISTAR vision illustrated at the recent ISS Mars Conference in Washington by NASA's C. Bolden and supported by US industry. This vision has also been endorsed by Russia [Savinyk, 2011]. In the ISTAR scenario the now-complete ISS is used not only for scientific purposes, but becomes incrementally a testbed of technologies indispensable to deep space exploration. ISTAR includes eventually installing propulsion on the ISS, that (whole or in part) would be moved from LEO to some location making it a deep space habitat (DSH). A possible location is L1. The reasoning behind is that a DSH is needed as a stepping stone toward a NEA and/or future Mars mission, but, in the current financial scenario, there is no money to build one. The space community will make do with whatever space assets already exist, and that is the ISS. The testing of the 200 kWe VX-200 VASIMR thruster [AdAstra, 2011], under discussion between Ad Astra Company and NASA may be coincidental, or may be consistent with the ISTAR vision.

At the ISS Mars Conference mentioned [Lorenzoni and Bruno, 2011] compared the ISS transfer from LEO to L1 using SEP to that using NEP. The ISS mass was assumed 190 ton, and a ΔV = 6 km/s accounted for the spiral transfer using ET. SP power was assumed limited to 110 KWe. A GIE and a Hall thruster, with Isp = 10,000s and 3200 s, respectively, were the ET chosen for this comparison. (The Isp of the GIE is that predicted feasible in the '20s by the U. of Southampton, UK.) Note that the ISS inertia moments are huge, limiting maximum transfer acceleration and thrust. Thus expected transfer times from LEO to L1 with ET are significant. A simplified 0-th order solution indicated ~10.6 yr the transfer and 9.9 ton of Xe propellant with GIE, and ~6.4 yr and 33.7 ton with Hall thruster. These numbers may improve with future (3-junction) solar cells, that at Spectralab have already shown 42% initial efficiency.

For comparison, with a modest 1 MWe NEP system using the same GIE, but powered by a 1 to 2 ton NR, the transfer time becomes 620 d, and the propellant consumed is about 9.8 ton, a factor 4 better in terms of transfer time, and 3.3 in propellant consumption in the worst case. Although the fraction of time spent within the van Allen belt is non-negligible, the comparison is noteworthy, even not accounting for the difference in orbiting cost due to the much bulkier solar panels when compared to a compact NR system.

Finally, the future may see a new class of interplanetary missions combining robotics with human activity. This is asteroid capturing. Far from being science fiction, this concept has been proposed originally in [Tantardini, 2010] and is now being preliminarily explored at JPL by John Brophy and others. The purpose of such mission is manifold, i.e., scientific, but also to train astronauts near/on very low gravity bodies, for instance to dock, and to fasten objects by drilling or by other means. (this is a crucial task to deviate asteroids from their orbit, as in planetary defence).

There is a dearth of small asteroids passing close to Earth, , e.g., 2011 MD went by on June 27, 2011 ~ 7600 mi. above the Atlantic Ocean [Chang, 2011]. Moving asteroids in the 20 to 30 m size (meteoroids) from their orbit to L1, or even close to the ISS, is technically feasible in the near future. For instance, using current SEP technology (Isp = 3000 s, 150 kWe panels feeding thrusters capable of 50 mN/kWe), the thrust available, F = 7.5 N, could be applied to the 2000 LG6 meteoroid. Its size (5m) and mass (200 ton) can be handled relatively well: its acceleration would be 3.75×10^{-5} m/s^2, its ΔV, after a year of tugging, about 1.2 km/s. With solar power this mission requires 8 ton/yr of propellant and about *500 m^2* of SP. With the same [modest] NEP system of the previous example the ΔV time could be halved.

Note that for this class of missions a NTR would be much more convenient (thrust and acceleration would be many orders of magnitude larger), and given the modest ΔV required, the larger propellant consumption would probably not matter much. However, no known NTR was ever designed with power less than many *tens* of MW.

6. Conclusions

NP is an outstanding, if not *the* outstanding propulsion candidate for deep space human missions to Mars and possibly also to NEA, although future high efficiency solar panels might become a good competitor. Because of its inherent power density NP is capable of reducing drastically travel time and thus crew radiation dose, with its known and newly emerging health risks. While there are still engineering issues to be solved to apply NEP, NTP, or their combination to future human missions, its capability and potential outclass any CP performance, enabling faster travel with drastically reduced mass consumption. This last is a very important factor in planning financially affordable human exploration.

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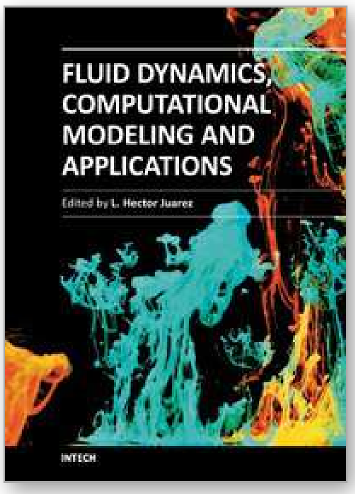

Fluid Dynamics, Computational Modeling and Applications
Edited by Dr. L. Hector Juarez

ISBN 978-953-51-0052-2
Hard cover, 660 pages
Publisher InTech
Published online 24, February, 2012
Published in print edition February, 2012

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University Campus STeP Ri
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InTech China
Unit 405, Office Block, Hotel Equatorial Shanghai
No.65, Yan An Road (West), Shanghai, 200040, China
中国上海市延安西路65号上海国际贵都大饭店办公楼405单元
Phone: +86-21-62489820
Fax: +86-21-62489821