Review of key features of rocket propulsion and fuels

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Abstract:

The use of spatial technologies is developing at a rapid pace, leading to an increase reliance on satellites particularly for Earth observations and telecommunications. Rocket engines, with their various propellant options, are essential for reducing the cost of launching satellites into orbit and thus of the services they are meant to provide. This pursuit of maximum efficiency also raises the question of whether it is appropriate to repair rather than replace end-of-life satellites, as well as the best way to produce and store the selected propellants before use. Rocket engines are sophisticated energy systems operating over a wide range of temperatures, from deep cryogenic temperatures (storage) to several thousand degrees (combustion chamber). While the choice of engine technology and their associated propellants is primarily based on thrust performance and propellant storage requirements, including safety, and potentially on the cost of the latter, it would be interesting to extend the considerations to overall energy efficiency and, to a lesser extent, to environmental considerations.

Keywords:

Rocket engines; Thermodynamics; Energy; Rocket Fuels; Combustion.

1. Introduction

Space propulsion relies on the use of rocket engines. A rocket engine is a type of reaction engine (Newton's third law), that is to say, a device that ejects a substance, usually a gas, towards the rear of the vehicle to which it is attached, producing by reaction a thrust, of equal magnitude and opposite direction, on the latter.

Generally, a rocket engine works by expelling gases produced in a combustion chamber by an exothermic chemical reaction between propellants (fuel and oxidizer). These gases are then accelerated to high speed as they pass through a nozzle (convergent-divergent nozzle or *Laval nozzle*). Since it must be able to operate in vacuum, that is, in an environment devoid of atmosphere, this type of engine requires that both the fuel and the oxidizer be stored, usually separately, inside the vehicle. This distinguishes it from jet engines used in aviation, which use atmospheric oxygen to burn the onboard fuel.

Propellant pairs used in astronautics can involve different combinations of *fuel* (reducing agent) and *oxidizer* (oxidizing agent). The desired properties of propellants are essentially:

- high combustion (or decomposition) energy;
- highest possible combustion temperature[1];
- high density to reduce tank volume;
- stable combustion products (low dissociation);
- combustion products with low molar mass[1].

The optimum thrust is achieved when the pressure P_o is equal to the ambient pressure, which is very low above an altitude of 20 km.

The performance of rocket engines, more specifically that of their propellants, is often compared in the literature based on what is called their *specific impulse* I_{sp}. This is defined as the impulse, or force, provided to the rocket by the consumption per unit time of one kilogram-weight (i.e., on Earth, 9.80665 N) of propellant:

[1] This results from the fact that the ejection velocity is given by [2] :

$$v_e = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{\tilde{\boldsymbol{r}} \cdot \boldsymbol{T_c}}{\tilde{\boldsymbol{m}}}\left[1 - \left(\frac{P_o}{P_i}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

$$I_{sp} = \frac{F}{\dot{m} \cdot g} = \frac{v_e}{g} \quad \text{in } [s] \tag{1}$$

The adjective "specific" refers to the fact that the specific impulse I_{sp} is linked to the nature of the propellant (just as a specific mass is linked to the nature of a substance). The specific impulse is expressed in units of time, usually seconds. Therefore, this quantity can also be interpreted as the duration for which one kilogram-weight of propellant can provide a thrust equivalent to its weight in the Earth's gravitational field.

In fact, the concept of specific impulse was proposed long before physicists standardized the definition of units, notably by distinguishing between kilogram-mass and kilogram-weight. Before this standardization, the impulse-to-mass ratio was naturally expressed in seconds. This physical quantity should nevertheless be abandoned today (it is difficult to give it a coherent meaning when spacecraft now travel in gravitational fields other than Earth's), in favor of the single ejection velocity, from which it differs only by a constant: the acceleration of Earth's gravity, g, which is specifically linked to our terrestrial environment. Specific impulse therefore provides no additional information compared to ejection velocity, which does not suffer from the same lack of "universality.".

2. Liquid propellants

The most common propellants are *dipropellants*[2], consisting of two liquids, one acting as fuel and the other as oxidizer. A distinction is made between propellants that can be stored in liquid form for several days at room temperature, such as nitrogen tetroxide or UDMH (*asymmetric dimethylhydrazine*), and cryogenic propellants, which require storage at very low temperatures, for example, oxygen, which has a boiling point of -183 °C, or hydrogen, which has a boiling point of -252.9 °C. The latter cannot be stored for long periods in a tank and must be quickly drained in case of a launch delay. Propellant pairs are also called hypergolic if they react spontaneously when brought in contact.

Liquid propellants have the major drawback of being difficult to store, but have the advantage of allowing control, and even interruption (followed by a possible restart), of combustion by modifying their injection rate into the combustion chamber. The characteristics of some of the main liquid propellants used in astronautics are given in (Table 1).

The ejection velocity v_e at pressures below 6895 kPa, can be estimated by applying the conversion factors given in (Table 2).

Liquid hydrogen, generally denoted by the acronym LH2, is by far the lightest and most efficient element. Its ejection velocity is nearly 30% greater than that of RP-1. Furthermore, it reacts with many oxidants, releasing a significant amount of energy. With liquid oxygen (LOX) as oxidant, the energy released per mole is 242.8 kJ. This value assumes, however, that all the reactants are at the same temperature. Yet, as we have seen, the reacting substances are stored in liquid form at very low, albeit different, temperatures. The water produced by their reaction (18 g/mol) is in gaseous state, heated to a high temperature by the released heat.

The temperature produced by the reaction is theoretically close to 3000 °C. The ideal ratio between the reducing agent and the oxidizing agent (stoichiometric mixture) is (see the corresponding reaction in Table 1): (2 g/mol H_2)/(16 g/mol 1/2 O_2). In practice, since the mixture of the two propellants cannot be perfect, it is advantageous to increase (enrich) the proportion of fuel in the mixture by 10 to 20%.

From the above consideration, we can deduce that hydrogen is currently the best fuel (meaning: the most energy-efficient one). However, it has the significant drawback of requiring storage at extremely low temperatures, even lower than that required by LOX. This limits the possibility of storing it in a tank at the desired temperature to just a few hours. Furthermore, the tank has to be large, given that the apparent density of an LH2/LOX propellant is nearly 30% lower than that of an RP-1/LOX propellant. Moreover, liquid hydrogen is a particularly volatile substance, requiring very careful handling due to the high risk of explosion. Fuel systems are therefore subject to considerable constraints, and robust cryogenic technology is necessary to safely handle a fluid whose temperature never exceeds 20.25 K (i.e., very close to absolute zero). Finally, hydrogen liquefaction is a relatively complex and energy-intensive process.

Given the difficulties encountered with using hydrogen as a rocket propellant, attention is now turning towards another option: methane. Despite an exhaust velocity approximately 18% lower, the methane (CH_4)/LOX mixture offers several operational advantages that make it competitive with the LH2/LOX mixture. Methane is

[2] There are also *monopropellants* that operate based on the exothermic decomposition of a single chemical compound, generally under the action of a catalyst. Hydrazine (N_2H_4) is the most commonly used monopropellant, particularly during the descent phases of space probes to their target or during orbital insertion (e.g., Viking, Phoenix, and Mars Reconnaissance Orbiter missions).

significantly less expensive to produce and easier to handle than hydrogen. It is liquid at -161.5 °C compared to -252.9 °C for hydrogen; this temperature, close to that of liquid oxygen, simplifies the design of the tanks and fuel systems for both propellants. In particular, it can be used to pressurize propellant tanks (autogenous pressurization), thus avoiding the need for complex and expensive pressurization systems based on helium, for example. Methane is also six times denser than hydrogen, therefore requiring six times less storage volume. Rocket stages can thus be more compact and potentially easier to recover for reuse (for launches from Earth, of course). But above all, with future crewed missions to Mars in mind, methane can be produced locally and then stored until needed, by implementing the Sabatier reaction ($CO_2 + 4\ H_2 \rightarrow CH_4 + 2\ H_2O$) and the reverse water gas reaction (RWGS: $CO + H_2O \rightarrow CO_2 + H_2$) within the framework of in situ resource utilization (ISRU) technologies, extracting CO_2 and H_2O from the atmosphere and the planet's ice reserves respectively.

Table 1: Characteristics of some of the main liquid propellants used in astronautics [1]

I) At sea level; II) in vacuum

Oxidizer	***Reducing agent (fuel)***	***Reaction***	$\widetilde{m}$ [g/mol]	E_c [kJ/mol]	T_c [°C]	v_e [m/s]	
Cryogenic propellants							
O_2	H_2	$H_2 + 1/2\ O_2 \rightarrow H_2O$	18	242,8	2740 2978	3816 4462	I) II)
	CH_4	$CH_4 + 2\ O_2 \rightarrow CO_2 + 2\ H_2O$	44+2x18	891,0	3260 3290	3034 3615	I) II)
	RP-1 *		relatively high		3403 3428	2941 3510	I) II)
F_2	H_2	$H_2 + F_2 \rightarrow 2\ HF$	20	547	3689 3985	4036 4697	I) II)
	CH_4	$CH_4 + F_2 \rightarrow CH_3F + HF$	34+20	484	3918 3933	3414 4075	I) II)
OF_2	RP-1 *		relatively high		4436 4432	3424 4021	I) II)
Storable propellant							
N_2O_4	UDMH (50%) +N_2H_4 (50%)	$H_2N\text{-}N(CH_3)_2 + 2\ N_2O_4 \rightarrow$ $3\ N_2 + 4\ H_2O + 2\ CO_2$	2x28+4x1 8+2*44	83,3	3095 3096	2831 3349	I) II)

(grey)	Non-hypergolic
(green)	Hypergolic

* "Rocket Propellant 1", a form of kerosene (a mixture of hydrocarbons, ranging from $C_{10}H_{22}$ to $C_{14}H_{30}$) specially refined for use as a storable liquid fuel for space launchers.

Table 2: Conversion factors

[kPa]	6895	6205	5516	4826	4137	3447	2758	2068
[PSI]	1000	900	800	700	600	500	400	300
Coeff.	1,00	0,99	0,98	0,97	0,95	0,93	0,91	0,88

The future flagship engine for European launchers, *Prometheus* (Precursor Reusable Oxygen METHane Cost Effective Propulsion System) will use, as its acronym indicates, methane and oxygen as propellants and is meant to be reusable. *Prometheus* (with up to 50% of its parts produced using 3D printing, which should significantly reduce its cost) is designed to provide variable thrust, up to a maximum of 100 tonnes; the future Ariane 6 Next launcher (scheduled for launch in 2030) will take off with seven engines of this type. This choice is part of other past Ariane 6 options being considered in Europe, some based solely on the use of methane and others with upper stages possibly using hydrogen [3]. Europe is not the only country interested in using methane as a propellant; the *Raptor* engine (v.2 or v.3) from *SpaceX* (founded by Elon Musk), with 33 units planned to power the first stage of the *Starship* spacecraft, as well as the BE-4 engine from *Blue Origin* (founded by Jeff Bezos, CEO of *Amazon*), which propels the *New Glenn* heavy-lift launcher, also rely on this technology. For its part, *Rocketdyne* also conducted preliminary studies in the 2000s on a similar engine the RS-18, designed for NASA's *Constellation* program, which was canceled in 2010 and has since been replaced by the *Artemis* program. Several other projects are also under development in other major spacefaring nations such as Russia and China.

RP-1 is a rocket fuel developed during the Cold War in the United States and the USSR to achieve high-performance, storable liquid propellants for their intercontinental ballistic missiles. Used with liquid oxygen as an oxidizer, it forms a propellant that, while less powerful than LH2/LOX, is much simpler and safer to handle, and also has a significantly higher energy density. *RP-1* manufacturing process begins with conventional refining to produce kerosene, followed by a desulfurization step to remove unsaturated fractions (thus preventing problems related to potential solid deposits) and then overly linear fractions (avoiding fragmentation of molecules into light byproducts). The resulting molecular species are highly branched or polycyclic C_{12}-saturated hydrocarbons. Because it is not very volatile, *RP-1* must be pressurized in its tank by a dedicated system, usually using nitrogen or helium. A fuel similar to *RP-1*, developed in the former USSR under the name *T-1*, is still in use in Russia; it notably powers the engines of the *Soyuz* launch vehicle.

Dimethylhydrazine, or *UDMH* — with the chemical formula $H_2N\text{-}N(CH_3)_2$ — is a storable, non-cryogenic liquid propellant used as fuel in the lower stages of many launch vehicles, as well as for propelling space probes, thanks to its long-term storage capacity. It is a clear, volatile, and hygroscopic liquid with an ammoniacal odor that decomposes spontaneously at around 400 °C under atmospheric pressure, producing methane and nitrogen. It is miscible in all proportions with water, ethanol, and kerosene. When mixed with nitrogen tetroxide (N_2O_4), it forms a hypergolic propellant. *UDMH* is a hazardous and potentially carcinogenic product; although it does not explode on impact, it releases flammable vapors at concentrations ranging from 2.5% to 95% in air. *UDMH* was preferred to pure hydrazine (N_2H_4) as a reducing propellant because it performs better at high temperatures, reducing the risk of explosion; it also remains liquid at lower temperatures (it freezes at -57 °C, compared to 1 °C for hydrazine). Its density is also relatively high (793 kg/m³), though less than that of hydrazine (1004.5 kg/m³), which gives it good performance for space propulsion. When blended with hydrazine at a 50% mass ratio, *UDMH* produces a propellant called *Aerozine 50*, which combines the high density of hydrazine with the stability of *UDMH*. Many launchers are propelled by pure *UDMH* or *Aerozine 50*, such as for example: the Russian *Cosmos* and *Proton*, the American *Delta*, or the Chinese *Long March* (as well as the second stage of the former European launcher *Ariane 5* (*G* and *ES* versions).

3. Solid propellants

A solid propellant rocket engine uses a material for propulsion, in the form of a powder bound and stabilized by an organic binder, which is combustible but stable at room temperatures. The propellant can be homogeneous, such as *nitrocellulose*, or heterogeneous (a composite) where the fuel is intimately mixed with the oxidizer.

The first category includes so-called dual-base propellants, mainly composed of *nitrocellulose* and *nitroglycerin*; they also contain additives to control their ballistic and mechanical properties and maintain their chemical stability. Their combustion results in relatively low ejection velocities ($v_e = 2300$ m/s).

Composite propellants are primarily composed of a solid oxidizer, a fuel (generally a mineral salt) finely dispersed throughout the oxidizer mass, an organic binder, and other additives. The powder blocks can be molded or extruded. The most common composite used in modern launchers is a mixture of ammonium perchlorate (NH_4ClO_4, oxidizer), aluminum powder (the reducing agent), and polybutadiene (BR, the fuel which also acts as a binder; a synthetic rubber with the structural formula $H_2C{=}CH\text{-}CH{=}CH_2$)[3]. The resulting ejection velocities are of the order of 2450 m/s. High-performance composites obtained by incorporating a small amount of high-power explosive such as *HMX* (with the chemical formula $C_4H_8N_8O_8$; 1.7 times more powerful than *TNT*) and *RDX* (with the chemical formula $C_3H_6N_6O_6$, one of the most powerful military explosives) into the

[3] As an example, each of the two space shuttle boosters contained 450 kg of a mixture of: 16% powdered aluminum (fuel), 69.6% ammonium perchlorate (oxidizer),0.4% iron oxide powder (catalyst), 12% polybutadiene acrylonitrile (binder), 2% polyepoxides.

mixture allow ejection velocities of nearly 2700 m/s, but are little used due to the higher risk of accidents and their lower stability and resistance to humidity.

There are also advanced energetic propellants based on new molecules – *DNA*, *FOX*, *GAP* (source: *Roxel Propulsion Systems*).

Solid propellants have the advantage of being stable and easy to store, which explains why they are often preferred for military applications. However, they have the disadvantage that once ignited, their combustion cannot be easily regulated or stopped before the entire propellant block is burned, except by opening valves (vents) on the side opposite the nozzle. The gas ejection through these valves is calculated so that the thrust generated is equal, but in the opposite direction, to the propulsive force obtained at the nozzle. This procedure is easy to implement in solid propellant rockets because the combustion chamber, formed by the central channel left empty in the center of the propellant block, extends from one end of the propellant to the other. This channel most often has a star-shaped cross-section (with 4 or 5 points) to obtain a more or less constant thrust throughout the duration of combustion.

The thrust depends on the surface area of solid propellant subject to combustion (the gas flow rate being proportional to it). This combustion is often initiated along the entire length of the propellant block using a flammable rod inserted into the central channel[4]. This surface area can vary as the propellant is consumed from the center of the propellant outwards. The initial shape of the central channel determines how this surface area evolves during combustion. With an initial circular cross-section, this surface area would tend to increase (a circular cross-section with an increasing radius). To maintain a more or less constant surface area, the central channel must initially be given a star-shaped cross-section such that its perimeter is, in principle, always equal to the inner circumference of the outer casing. However, combustion is no longer as regular at the end of operation, and significant vibrations and even explosive phenomena can occur due to the fragmentation of the remaining solid propellant block. This can compromise the integrity of the booster if its outer casing does not remain airtight. This airtightness is of course absolutely crucial in the case of solid rocket used as "boosters" for crewed spacecraft. It was the failure of a seal on the right solid rocket booster of the *Challenger* Space Shuttle that caused a perforation in its casing (on January 28, 1986). The resulting blowtorch effect then reached the adjacent main tank, filled with hydrogen, causing its explosion and the destruction of the shuttle, resulting in the death of its seven astronauts.

The propellant can be shaped before being placed in the rocket body or poured and then solidified by baking within the body itself. Large modern rockets generally use the second option. Once the casing is ready, with its thermal protection applied to the inner wall and coated with a binder to ensure adhesion between the propellant and the structure, the propellant ingredients, previously thoroughly mixed like bread dough, are poured into it. The central channel is created using a solid core temporarily inserted into the center of the cylindrical space defined by the casing; this core gives the channel the desired shape to achieve the required thrust over time. The quality of the assembly is verified using non-destructive testing methods (X-rays, ultrasound, thermal imaging). Such large modern rockets may be equipped with a steerable nozzle for guidance, thrusters to control the rocket's orientation, self-destruct mechanisms in case of malfunction, and an auxiliary power generator.

Table 3 Example of a high-power solid propellant booster, the Ariane 5 EAP [1]

Propellant	Ammonium perchlorate, aluminum powder, polybutadiene
Dimension, height x diameter	31 x 3 meters
Nozzle	metallic and composite
Central canal shape	star (upper segment), cylindrical (other segments)
Total mass (empty mass)	269 tons (31 tons)
Mean thrust	4’885 kN ou 498 tons (6’580 kN ou 671 tons, void)
Thrust orientation	6° by hydraulic cylinder
Maximum pressure	61 bars
Ejection velocity	2’698 m/s
Combustion time	129 seconds

[4] In some applications, frontal combustion, in parallel layers, like a cigarette, is possible rather than radial combustion. In this case, the combustion surface area is small but constant.

This type of propulsion system, simpler in design than a liquid-propellant launcher, can provide significant thrust (each Space Shuttle booster delivered 12,000 kN of thrust, or nearly 1,250 tons) at a relatively low cost. This is why solid-propellant propulsion is more specifically used for the first-stage boosters of rockets (Table 3).

Liquid propellant engines, which offer higher exhaust velocities (especially those using hydrogen as fuel), are preferred for upper stages. On the other hand, solid propellant thrusters are widely used to place satellites into their final orbit (apogee thruster) due to their simplicity and reliability, as well as their compact size and relatively high mass energy density.

Improving the performance of solid-propellant rocket motors relies primarily on developing new chemical mixtures and mastering the combustion process of these propellants, as well as the manufacturing process of the powder blocks. This field of chemistry is constantly progressing. Nevertheless, only a few countries master today the use of the highest-performing propellants.

Another critical aspect that needs to be dealt with is the environmental damages that can be caused by some metal components [4].

4. Litherpropellants (hybrid propellants)

A hybrid rocket engine can combine a solid propellant, usually the fuel, with a liquid propellant, in this case the oxidizer. The idea is of course to take advantage of the strengths of both technologies, but, as it is the case in any hybrid system, it is not possible to avoid also combining the disadvantages of both (at least in part).

Compared to a fully liquid-propellant rocket engine, the litherpropellant engine is simpler in design (no turbopump, reduced piping, less sophisticated injection system), and consequently, in principle, more reliable, but also safer since it uses non-explosive propellants As for the comparison with solid-propellant rocket engines, it is primarily the ability to modulate thrust by injecting more or less oxidizer, or even to interrupt it (and possibly reactivate it afterward), that gives hybrid rocket engines a significant advantage.

Hybrid propulsion, however, has several drawbacks which, until now, have limited its application to a few exceptional projects. The most publicized example is that of the spaceplanes developed by Mojave Aerospace Ventures/Virgin Galactic – *SpaceShipOne* and *SpaceShipTwo* – which use such hybrid propulsion systems for their suborbital flights; a choice dictated here by considerations of cost and safety.

The main drawback of litherpropellant rocket engines is that the fuel and oxidizer do not mix well because the vaporization of the fuel block surface is too slow (a so-called regression rate that is too slow), due to insufficient heat transfer. To accelerate combustion, the propellant block can be perforated with several channels, but this reduces the fuel volume and leads to instability at the end of combustion. Furthermore, the combustion efficiency is degraded by the progressive enlargement of the combustion chamber, leading to a less efficient oxidizer-to-fuel ratio. The oxidizer must also be injected at a high pressure (higher than that in the combustion chamber), which requires a particularly robust, and therefore heavy, tank, or alternatively, the use of a turbopump, thus reintroducing the complexity and mechanical fragility that hybrid propulsion was specifically designed to avoid. The fact that traditional fuel block molding techniques to form solid grains are laborious, expensive and prone to grain defects presents another challenge in meeting the high demands of commercial spaceflight.

The main components of a hybrid rocket engine are a pressurized tank containing the oxidizer in liquid (possibly gaseous) form, a combustion chamber integrated into a solid block of fuel (reducer), and a valve separating these two elements. The engine is started by opening the valve, allowing the pressurized liquid or gaseous oxidizer from the tank to enter the combustion chamber (via the injector) where it vaporizes and reacts with the solid propellant, igniting thanks to a dedicated device (ignition system). In its simplest case, the combustion chamber is just a longitudinal channel drilled through the center of the propellant block (which can take various shapes; see the case of solid propellant rocket engines). At the end of this channel is the nozzle through which the produced gases are accelerated and then ejected. To obtain greater thrust, several channels can be drilled, thus increasing the surface area of the propellant block exposed to combustion. Although this type of propulsion is still far from being widespread, recent research into new, more efficient propellants could open up new perspectives.

5. Non-chemical space propulsion systems

In the field of space propulsion, the application of Newton's third law, "action = reaction," is not limited, in principle, to the ejection of hot gases resulting from a chemical reaction even though this is currently the most common, and even practically the only, way to propel rockets and other spacecrafts. In reality, the high-speed ejection of any substance (assuming, of course, that it possesses a certain mass), regardless of how the necessary energy is supplied, can achieve this objective.

5.1 Nuclear thermal space propulsion (fission energy)

The principle of nuclear thermal propulsion in space is to replace the energy produced by the combustion of fuel in the presence of an oxidizer in a conventional rocket engine with the energy resulting from fission reactions, and potentially even fusion reactions. Apart from this, a nuclear rocket engine is very similar to a combustion rocket engine (gases heated to high temperature, accelerated, and finally ejected through a Laval nozzle). The main advantage of nuclear energy lies in its energy density (specific energy), which is several orders of magnitude greater than that of chemical energy. Indeed, the fission of one kilogram of uranium-235 releases 83 million megajoules of energy, while the combustion of one kilogram of hydrogen provides only 142 megajoules. The fission of plutonium-239 (obtained in a breeder reactor by neutron reaction with the isotope uranium-238, a so-called *fertile* material) or of uranium-233 (obtained in the same way from thorium-232), "artificial" fissile isotopes, uranium-235 being the only fissile isotope available as a natural resource on Earth, and even in the whole solar system, leads to energy densities of the same order: 83.6 million MJ/kg and 82 million MJ/kg respectively. It is therefore possible, in principle to obtain a very significant improvement in ejection velocities by using nuclear energy rather than chemical combustion energy; the ejection velocity being proportional to the square root of the energy density e[5].

This does not mean, however, that the performance gains achievable with nuclear propulsion compared to chemical propulsion are proportional to the square root of the energy densities mentioned earlier. In reality, on the one hand the efficiency with which the energy produced by nuclear reactions can be transferred to the propellant fluid is lower; for example, a chemical rocket engine using an LH2/LOX mixture as propellant recovers more than 80% of the combustion energy of these propellants as kinetic energy, a performance that no nuclear propulsion concept can achieve due to the temperature limits of the structural materials, on the order of 3000 K, heat transfer losses, and the fact that only a (relatively small) portion of the fuel is consumed. On the other hand, the additional mass to be propelled due to the essential shielding designed to protect personnel and equipment, especially electronic instruments, from the dangerous radiation emitted by the fission products must be taken into account.

The advantage offered by nuclear propulsion remains nevertheless significant. This is especially true given another factor in its favor: the possibility of choosing the propellant with the lowest possible molar mass, namely hydrogen (H_2, with M = 2 g), which is ideal, as previously mentioned. In contrast, the most efficient chemical propellants combine hydrogen with oxygen in a combustion reaction that produces water vapor (H_2O), whose molar mass is already 18 g. Ultimately, a nuclear rocket engine should allow for an ejection velocity (and therefore a specific impulse) approximately twice that of the best chemical rocket engine.

It is obviously out of question to use a nuclear engine to launch a rocket directly from Earth (or even from another planet). A ground explosion would permanently contaminate the launch pad and surrounding areas, and potentially an even larger territory if the explosion were to occur in flight. The risk is greatly reduced, however, if the reactor is transported to high orbit by a chemically propelled rocket before being used. Indeed, a reactor that has not yet become operational does not contain, by definition, any fission products or transuranic elements, the main sources of radioactive radiation (nuclear fuels themselves are only slightly radioactive; however, it must be taken into account that plutonium, if used as nuclear fuel, exhibits very high biotoxicity).

Since the beginning of the space age, various types of nuclear reactors for space propulsion have been studied. Some have a design similar to those used in Earth-based nuclear power plants, while others, more complex but more efficient in this context, such as gas-core reactors, are specifically designed for this purpose. A nuclear propulsion reactor must meet constraints very different than those faced by terrestrial power reactors. The fuel and certain structural materials must withstand high temperatures (> 2000 K) that must moreover be reached in very short periods to achieve the desired velocity increments. However, the cumulative operating times over the lifetime of such reactors remain extremely short compared to those of nuclear power plant reactors. In space propulsion applications, the nuclear fuel is therefore subjected to high thermal stresses but at a lower burn-up rates. Furthermore, it is obviously crucial to maintain the thrust-to-weight ratio as low as possible for engines designed to propel spacecraft, a constraint that does not exist for ground-based installations. It is essential to consider that physics dictates a minimum fuel mass, known as the critical mass, for chain reactions to be maintained in the core of a nuclear reactor. If this condition is not met, the number of

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[5] This relationship between v_e and e can be found from the equation given in the footnote 1. Neglecting P_o, which is very small for a nozzle adapted to the pressures prevailing at high altitudes or in the vacuum of space, we have (energy and mass expressed per mole):

$$v_e \approx \sqrt{\frac{2\gamma}{\gamma-1}\frac{\tilde{\boldsymbol{r}}\cdot T_0}{\tilde{\boldsymbol{m}}}}\text{, with } \gamma = 1{,}4 \text{ and } e = \frac{(7/2)\cdot\tilde{\boldsymbol{r}}\cdot T_0}{\tilde{\boldsymbol{m}}} \text{ for a diatomic gas like } \mathrm{H_2}$$

Thus: $v_e \approx 1{,}414\sqrt{e}$, with v_e in [m/s] and e in [J/kg].

neutrons absorbed or escaping from the reactor core is insufficient to guarantee that at least one of the two to three neutrons emitted by fission is available to initiate a new fission reaction.

5.2 Nuclear electric propulsion (ionic propulsion)

The use of a nuclear reactor[6] can also be considered, as is usually the case on Earth, to primarily produce electricity, which will then be used to power the type of rocket engine called ionic because it relies on the ejection as a propulsive mass of ionized molecules or atoms, accelerated to very high speed by an electric or electromagnetic field.

To obtain the desired ions (in this case, cations, i.e., positively charged ions), atoms can, for example, be heated to high temperatures, which partially or completely deprives them of their electrons. For example, heating a tungsten plate to approximately 1500 K and spraying an alkali metal — cesium or potassium — onto it as vapor, results in near-total ionization (over 99%). The resulting gas is then electrically charged and can be accelerated by electrostatic or electromagnetic forces. For a potential difference of 1000 volts between two electrodes, ejection velocities of around 40000 m/s are achieved, significantly higher than the speed that can be imparted to hydrogen in a conventional nuclear thermal engine. However, cesium and potassium corrode the engine walls, which is why a neutral gas, xenon, is today preferred.

The concept of ion propulsion dates back to the 1910s when it was proposed by the astronautics pioneers Robert Goddard (USA) and Konstantin Tsiokovski (Russia). Despite the time that has passed since then, many practical difficulties remain to be overcome to obtain a truly efficient engine. One of these problems concerns the free electrons produced by ionization. Indeed, if these electrons are not expelled at the same rate as the ions, they will attract the ions (two opposite electrical charges attract each other) to the point of eventually hindering their ejection. It is therefore necessary, in one way or another, to allow the recombination, just before the engine exits, of the two types of charged particles in the ejected mass, which is then, under these ideal conditions, electrically neutral again. Despite considerable efforts, this problem has not yet found a fully satisfactory solution. Around the thruster, areas with a predominance of ions or electrons (space charges) are created, which slow down the ejection of gases.

An ion thruster therefore essentially consists of the following three elements:

- an ion source created by a device that separates ions from electrons, which can be achieved through contact ionization, electric arc, or high-frequency induction;
- an ion acceleration and focusing system to shape the ejected beam;
- a mechanism for recombination of ions with electrons before they exit the engine.

The advantage of electric propulsion lies in its very low fuel consumption compared to chemical propulsion. However, time is a crucial factor, because while fuel-efficient and capable of long-duration operation, the ion engine produces only very low thrust (of the order of a few hundreds millinewtons, approximately 100 times less than the chemical engines typically used on space probes). Conventional rocket engines provide significant acceleration in a short period of time, but consume a large quantity of propellant to do so; ion engines, on the other hand, produce a low but constant propulsive force. They generate significantly more work per kilogram of inert gas onboard than conventional rocket engines. Thus, even after a much longer time, they can give the spacecraft the same speed, or even a significantly higher speed, than that obtained with a chemical propellant, but at the cost of a much lower thrust mass consumption.

At the same power output, this type of engine is also significantly lighter, resulting in further savings in propulsion costs. Besides the problem of their low thrust, by their very principle — the ionization of a propellant gas — ion engines can only operate in the vacuum of space; it is therefore completely out of question to use them for launches from a planet, especially one with an atmosphere. However, as soon as the vehicle reaches space, the ion engine can, in principle, take over (particularly in the case of probes).

In conclusion, the nucleo-ionic (or possibly photovoltaic-ionic) engine is interesting, at least for certain applications, but it is still far from achieving the performances required to revolutionize astronautics. For an engine type to bring about a real change, it would need to allow very high ejection velocities without resulting in excessively low ejected mass flow rates and therefore extremely low thrust.

The VAriable Specific Impulse Magnetoplasma Rocket engine, *VASIMR* is a promising candidate for reconciling, at least in part, these seemingly contradictory requirements. NASA academic institutions such as MIT, the University of Texas, the University of Houston, Rice University, and the University of Michigan, major US national laboratories like Los Alamos and Oak Ridge, as well as private companies, particularly Ad Astra Rocket, based in Houston, Texas, and Guanacaste, Costa Rica, are actively working on this type of rocket

[6] For low to medium power levels, the necessary electrical energy can also be supplied by an isotopic generator instead of a nuclear reactor. This is a simple thermoelectric generator that produces electricity from the heat resulting from the spontaneous decay of radioactive isotopes. Another possibility is to use photovoltaic panels, but this is only feasible for relatively low power levels and for missions limited to the inner planets of the solar system (beyond this range, the solar intensity becomes too low, which would require too large PV capture surfaces).

engine that could theoretically allow us to approach relativistic speeds. However, while the thrust provided by such a system is far greater than that of more conventional ion and plasma engines, it remains significantly lower than that of chemical or nuclear-thermal engines.

Technically, the operating principle of a VASIMR engine is as follows:

1. A neutral gas, hydrogen or helium, is injected into the device's inlet as fuel.

2. A quartz tube, resistant to high temperatures and completely transparent to the ionizer's electromagnetic waves, collects and confines this gas before ionization.

3. A helicon antenna, similar to a microwave oven, preheats and ionizes the gas using radio frequencies. The ionized gas is thus raised to a temperature of approximately 30,000 K (five times higher than the temperature at the surface of the Sun).

4. The ionized gas is kept away from the walls (more to prevent its instant cooling than to protect the walls) by solenoids surrounding the cylindrical chamber, which create an axial magnetic field within it.

5. A second "furnace," the ICRH (Ion Cyclotron Resonant Heating) antenna, fully ionizes the gas by raising its temperature to 10 million degrees and induced an electric field that accelerates the ions toward the nozzle along a helical trajectory.

6. A magnetic nozzle (necessary to prevent any physical contact between the plasma and the nozzle walls) regulates and directs the plasma flow at the outlet. This "variable magnetic geometry" nozzle allows the specific impulse and thrust to be varied at constant power by modulating the intensity of the magnetic field and the geometry of its field lines. This magnetic nozzle thus plays a role somewhat analogous to that of the gearbox in a vehicle whose engine is supplied with fuel at a constant speed.

According to Franklin Chang-Diaz, a former NASA astronaut and current CEO of *Ad Astra Rocket*, *VASIMR* is designed to develop various power outputs in the megawatt range. He claims that a spacecraft powered by a 200 MW *VASIMR* engine could reach Mars in 39 days, roughly five times faster than with the best current chemical propulsion systems, while consuming only 10% of the fuel used by those systems. Between April and September 2009, a prototype of this engine, the VX-200, equipped with two Tesla superconducting magnets, operated in a stationary position for 25 seconds at a power output of 200 kW, thus validating the concept.

Further test results presented in January 2011 confirmed that at its optimal operating point, the VX-200 could achieve an ejection velocity of approximately 50000 m/s, corresponding to a specific impulse exceeding 5000 s, for a thrust of 5 newtons. In 2013, this prototype underwent more than 10000 full-power tests using argon as propellant, demonstrating an efficiency over 70% relative to the RF energy input. In August 2019, *Ad Astra Rocket* announced the successful completion of tests of a new generation 120 kW radio frequency (RF) transmitter power unit (PPU), ten times lighter (52 kg) than its competitors and achieving 97% RF-to-electricity conversion efficiency. A semi-empirical model predicts that the improved version of this engine, the VX-200SS, will achieve an overall propulsion efficiency of 64% at a specific impulse of 6,000 seconds and a power output of 200 kW. The first industrial production prototypes are expected to produce specific impulses reaching 30000 seconds with thrusts up to 500 newtons.

The major problem with the VASIMR is the very high power level it requires. This is all the more true given that this type of engine is not very efficient in this respect. With 5 N of thrust for a power output of 200 kW, the VX-200 is nearly half as efficient as the more conventional *NEXT* ion engine developed by NASA, which produces 0.327 N with only 7.7 kW of power. The 200 MW engine envisioned by Chang-Diaz to significantly shorten the journey time to Mars would therefore require a nuclear fission reactor, the only technology currently considered viable for providing the required power level in space, with performance levels far beyond what is achievable with the current state of technology. According to Chang-Diaz, the power-to-mass ratio should be 1000 watts per kilogram, whereas the most powerful nuclear reactor for space use currently in operation, *Topaz-II* (USSR), offered only 10 kW of power with a power-to-mass ratio of 10 watts per kilogram for a weight of about 1060 kg.

The designer of *VASIMR* believes, however, that technological advances will make it possible in a not-too-distant future to achieve performance levels of 500 W/kg (making a journey to Mars possible in 60 days), or even the hoped-for 1000 W/kg. A NASA study published in 2011, "Multi-MW Close Cycle MHD Nuclear Space Power Via Nonequilibrium He/Xe Working Plasma," seems to confirm that such objectives are indeed achievable in the long term. However, another difficulty to note is that the relatively poor energy efficiency of the VASIMR engine means that heat sinks the size of a football field would be required to dissipate the heat generated by a 200 MW engine. All of these elements (ionization and confinement systems, nuclear reactor, heat sink) add to the overall weight of the spacecraft and make it unsuitable for applications requiring a high power-to-weight ratio, such as launches from a planet's surface, even though very high thrust could potentially be achieved.

In conclusion, variable specific impulse magneto-plasma propulsion offers the possibility of long-distance journeys for astronauts in the solar system, but only on the express condition that substantial progress is made in the development of high-power nuclear reactors for space use. Unfortunately, progress in this field is currently almost at a standstill, even though a renewed interest in this technology can be observed in the last years in the United-States (for example: the NASA Kilopower Reactor Using Stirling technology, or *KRUSTY*, that should provide 10 kilowatts of electric energy in a stationary way during at least 10 years).

5.3 Nuclear fusion propulsion

Rather than using nuclear fission energy, some advocate turning to nuclear fusion energy to ensure the propulsion of future spacecraft, an energy source that releases more energy per unit of mass invested (light nuclei in this case) than fission (using heavy nuclei)[7]. This should maximize propulsion efficiency while reducing the amount of fuel required and potentially the engine size. However, it should be noted that despite the considerable resources invested in this field since the 1950s, mastery of this type of energy has still not been achieved for its terrestrial applications (electricity production). More specifically, we have not yet managed to achieve what is known as "*breakeven*," that is, obtaining at least as much energy from a fusion reactor as is used to create the conditions — a temperature of approximately 150 million degrees and a sufficiently long confinement time — to initiate fusion reactions. Proponents of fusion propulsion in space emphasize that these would be small reactors, typically on the order of tens of megawatts (compared to thousands of megawatts for a nuclear fusion power plant), making them much smaller, lighter, and easier to build and launch into space. This would allow them to power robots and probes, as well as future crewed missions (in which case, several such reactors could be coupled if necessary, offering a reliability advantage: if one reactor fails, the others can still allow to complete the mission). Such a reactor would weigh around ten tons and occupy a cylindrical volume approximately 4 to 8 meters high and 1.5 meters in diameter; it could produce 1000 W/kg, precisely the power density required for the *WASIMR* concept (see above).

The surge in nuclear fusion research in the 1970s, particularly inertial confinement fusion (as opposed to magnetic confinement[8], used in tokamaks, for example), sparked renewed interest in the older concept of pulsed nuclear propulsion, this time based on the use of micro-fusion "bombs." The advantage of fusion over fission is that the former does not require reaching a critical mass; propulsion can therefore be based on more precisely calibrated detonations, making it possible to envision more efficient and compact vehicles. The *Daedalus* project, for example, a study conducted from 1973 to 1978 by a dozen scientists and engineers from the British Interplanetary Society, aimed at nothing less than the creation of an automated interstellar probe capable of reaching a nearby star in about fifty years. *Daedalus* was to be propelled by the fusion of microspheres made of a mixture of deuterium and helium-3, bombarded by electron beams in a magnetic nozzle. The projected specific impulse was approximately one million seconds; the vehicle would have reached 12% of the speed of light after about four years of operation. However, given its size and the technologies involved, *Daedalus* remains far from being feasible.

6. Conclusion

The most energetic propellant combination is the liquid dipropellant LH2/LOX, which is also the less environmentally harmful. However, its production requires a complex infrastructure due to the extremely low temperature of liquid hydrogen and the difficulty in keeping it contained. Therefore, the current trend is toward the use of LCH_4/LOX that is less efficient because of combustion products having a higher molar mass but have the advantage of requiring the same order of magnitude of cryogenic temperatures for both propellants. Liquid propellants allow for better control of the combustion including the ability to stop combustion, which is not possible with solid propellants. However, solid propellants offer the advantage of being stable and easy to store, but are often more damaging to the environment.

Non-chemical space propulsion systems, including nuclear thermal or ionic propulsion systems, are under development but could only be considered for applications outside of planetary gravity wells.

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[7] In particular, one could consider that the *VASIMR* concept could one day be based on a fusion plasma which would then itself constitute the propulsion fluid.

[8] In the "inertial confinement" approach to fusion… there is actually no confinement; it would therefore be more accurate to simply speak of "*inertial fusion*". It is simply inertial forces that briefly confine the plasma long enough for fusion reactions to ignite, after it has been created, then highly compressed and heated to a high temperature by the irradiation of fuel microspheres with very high-power lasers for example, or ion/electron beams.

Nomenclature

$\tilde{\boldsymbol{m}}$ molar mass (g/mol) of the reaction products (gas)

E_c reaction energy produced per mole of reducing agent (partially calculated)

T_c equilibrium temperature in the combustion chamber

F thrust, N

g gravitational acceleration, m/s^2

v_e ejection velocity of the produced gases, for optimal expansion at 6895 kPa (combustion chamber pressure)

$r = \tilde{\boldsymbol{r}}/\tilde{\boldsymbol{m}}$, mass constant of an ideal gas with $\tilde{\boldsymbol{r}}$ = *8.3145 [J/(mol·K)], universal constant)*

T_c = gas temperature in the combustion chamber

P_i, P_o , pressures at the input (i) and the output (o) of the nozzle respectively

$\dot{m}$ mass flow rate, kg/s

t temperature, °C

v_e ejection velocity, m/s

I_{sp} specific impulse, s

$\gamma = c_p/c_v \cong$ *1.4* for diatomic gases, the ratio of specific heat capacities (respectively at constant pressure and volume)

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