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Electric Thruster Selection Criteria

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Abstract

The paper presents a set of very well-known criteria to be taken into account when selecting an electric thruster concept for a propulsion system aboard mainly nano or small spacecrafts. Unfortunately, those criteria are sometime forgotten by some thruster designers or "people who do differently" who propose hardware that can be simply highly non-competitive with respect to other concepts. A list of the known possible concepts of Electric Propulsion will be presented.

The criteria to be taken into account are described in the paper. Among them one can cite: the number of thrusters used in the system, the unified propulsion concept, the worst-case design, the redundancy philosophy, the system mass impact and the possibility of sharing (partly and/or timely) the electrical power with the EP thrusters. In order to allow an objective selection among the thruster concepts, it is proposed in the paper to rely on the definition of Issp (the System specific impulse) introduced by Peter Erichsen in the 90s. Issp number takes into account all parameters influencing the impulse-dependent part of the propulsion system mass, such as the mass of the propellant storage and electric power supply systems, and therefore it characterises the performance of propulsion systems particularly for the size of the system considered for nano or small spacecrafts much better than the Isp "Thruster-specific Impulse" alone which only includes the propellant mass.

Here, Issp has been adapted for covering the cases of unified or non-unified propulsion system as well as the worst-case design concepts with or with no redundancy. Examples of output of selection will be presented for several relevant cases in order to allow thruster designers or "business angels" to make their own best selections before taking any investment decision into a particular concept.

1 Introduction

The propulsion aboard any small, nano, micro-satellite (like the CubeSat standard) represents certainly a major disruptive step in the sustainability potential of such kind of satellite families.

However, the interfaces between the satellite designer and the propulsion thruster designer are sometimes contradictory.

For example, the satellite designer may only want a "plug and play" brick of propulsion system that does not interfere greatly with the other parts of the satellite: no power or a minimum power used by the propulsion, with a minimum added mass impact (only the propellant mass, the thruster mass should be negligible with respect to the propellant mass), without any thermal interferences, with no EMI, without major constraints on the attitude control, etc.

On the side of the propulsion thruster designer, mimicking with what happens for large satellites, the mass of the propulsion is limited to the one of the hardware delivered (the propulsion package mass). The impact of the required power in terms of mass increase is not really a concern. Neither are the thermal loads coming from the propulsion thermal dissipation. The attitude control needs for performing long thrusting periods of months are forgotten. Hence the majority of the thruster proposal comprise a unique thruster with the hope that attitude control will be obtained within the devices of the satellite.

Hence it is clear that the two quite opposite approaches may lead to severe disappointment between parties. In order to resolve this situation, only the satellite designer is able to take into account properly the different aspects needed for making a relevant design and to decide if it is possible. But the main problem that the satellite designer has to solve is to make the right choice of the propulsion system among the so numerous potential proposed thrusters. This task is very difficult, moreover most of arguments coming from propulsion thruster designers are not really relevant for an implementation aboard a satellite. On can mention thruster designers claiming impossible cases with current technologies (see further a thruster designer claiming in its advertisements a capability with its thrusters to make more than 2 km/s of delta-V on a 3 kg satellite mass), thruster designers speaking mainly of their thrusters specific impulse forgetting the dry mass and other induced mass (like thermal rejections), or speaking mainly in terms of efficiency while forgetting the added operation's complexity and dry mass needed and induced.

In order to clarify thruster's selection, the paper discusses on how to set a suitable "Selection Criterion" in the case of "Electric Thrusters". Such criterion can also be used by any "business angels" to make their own best selection before taking any investment decision into a particular thruster concept whatever the designer may say.

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2 Conventional approach to thruster's performance

The thrust vector can be seen as directly related to the mean exhaust velocity opposite vector $-\vec{v_e}$ and to the ejected propellant mass flow $\dot{m}_p$ (i.e. $\frac{dm_p}{dt}$) by Newton's law of motion:

$$\vec{F} = -\dot{m}_p \vec{v_e} \text{ [N]} \tag{1}$$

The thrust level F reflects the duration of a manoeuvre and its precision. The higher the thrust is, the shorter the manoeuvre duration is and the less precise it is. Another essential characteristic of reaction jet is the ΔV budget called ideal velocity or velocity increment that relates to the quantity of manoeuvre that can be performed. The Tsiolkovski equation expresses the ideal velocity as a function of the exhaust velocity modulus (supposed here as a characteristic constant of the thruster concept) and the spacecraft mass ratio $\frac{m_0}{m_f}$:

$$\Delta V = v_e \ln\left(\frac{m_0}{m_f}\right) \quad \text{[m/s]} \tag{2}$$

with m_0 the mass of the spacecraft at the beginning of life, $m_f = m_0 - m_p$ the mass of the spacecraft at the end of the manoeuvre, m_p the total propellant masse used.
It is of interest to recall that eq.(2) can come from the following derivations, with $m(t)$ the time-varying satellite mass and V its velocity, once again according to Newton's law of motion applied to the satellite:

$$F = m(t)\frac{dV}{dt} \qquad \text{hence:} \qquad \frac{dm_p}{dt} v_e = m(t)\frac{dV}{dt} \quad \text{[N]} \tag{3}$$

$$\text{Having obviously} \quad dm_p = -dm \qquad \frac{-v_e dm}{m(t)} = dV \qquad \int_{m_0}^{m_f} \frac{-v_e dm}{m(t)} = \int dV = \Delta V \quad \text{[m/s]} \tag{4}$$

$$\text{and we can write using the structural index } k = \frac{m_f}{m_p} \qquad v_e \ln\left(1 + {}^{1}/_{k}\right) = \Delta V \quad \text{[m/s]} \tag{5}$$

$$\text{or if} \quad \frac{m_p}{m_0} \ll 1 \qquad \frac{v_e m_p}{m_0} \approx \Delta V \quad \text{[m/s]} \tag{6}$$

The above development shows two cases depending on the structural index value: for launch vehicles, $\boldsymbol{k}$ is quite low and eq. (5) shall be used, while for high structural index (case of CubeSat propulsion in particular), eq. (6) can be used.

To compare thrusters' performance without considering the satellite, the total impulse I_{tot} is introduced. The total impulse is defined as the impulse that a thruster can produce during its operational lifetime τ. For such impulse, it is needed to feed the thruster with an operational total mass of propellant $\boldsymbol{m_{pMax}}$ according to the following eq.:

$$I_{tot} = \int_0^{\tau} F\, dt = v_e \int_0^{m_{pMax}} dm_p \qquad \text{so that} \qquad I_{tot} = v_e\, m_{pMax} \quad \text{[N.s]} \tag{7}$$

For all thrusters, this parameter $\boldsymbol{I_{tot}}$ is important because it characterises the **operational lifetime** capabilities of the thruster. This $\boldsymbol{I_{tot}}$ can be a constraint for the satellite, because when the needed total impulse aboard a satellite must be higher that the single thruster capability, then a second thruster has to be implemented aboard…
Also, when thrusters have different operating points validated in the course of a qualification program, the total impulse capability for one thruster can depends on the operating point and on the corresponding operational lifetimes (for example, for one point: higher thrust with lower performance and higher lifetime and for a second point: lower thrust but with higher performance and probably lower lifetime). Hence in such case, $\boldsymbol{I_{tot}}$ of the thruster depends on the corresponding operational point.

For the case with high structural index (case of CubeSat propulsion $\frac{m_p}{m_0} \ll 1$) and $\boldsymbol{m_p} \leq \boldsymbol{m_{pMax}}$, i.e. $\boldsymbol{m_p}$ is small, one gets the ideal velocity simply from eq. (6), (7):

$$\Delta V \approx \frac{I_{tot}}{m_0} \qquad \text{[m/s]} \tag{8}$$

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Using the total impulse I_{tot} as a thruster characteristic is obviously important in terms of operations and its constraints, but it is irrelevant as propulsion criterion index because it depends linearly on the used mass of propellant up to its maximum acceptable at thruster end of life m_{pMax}. Using the ΔV as propulsion criterion index is also irrelevant because it depends on the satellite mass ratio.

Therefore, a more intrinsic parameter commonly accepted in the case of large rocket engine or large thrusters is to evaluate the performance of a thruster thanks to the specific impulse $I_{sp\,thr}$:

$$\boldsymbol{I_{sp\,thr}} = \frac{I_{tot}}{m_{pMax}\,g_0} \qquad \boldsymbol{I_{sp\,thr}} = \frac{F}{\dot{m}_p g_0} \quad [\mathrm{s}] \tag{9}$$

with $g_0 = 9.80665$ [m.s^{-2}] the gravitational constant.
Note the suffix "$_{thr}$" stand for thruster in order to avoid any confusion in the course of this paper.

Specific impulse corresponds to the impulse delivered per unit weight of propellant[1]. A high $I_{sp\,thr}$ means that the propulsion system has a good efficiency is terms of propellant consumption: at similar fuel mass, a system with a higher $I_{sp\,thr}$ will provide the satellite with more ΔV capabilities.

It shall be highlighted that the mechanical energy provided by the thruster is linked to the total impulse

$$\boldsymbol{E_{meca}} = \frac{1}{2}\boldsymbol{v_e I_{tot}} \quad \text{or} \quad \boldsymbol{E_{meca}} = \frac{1}{2}\boldsymbol{g_0 I_{sp\,thr} I_{tot}} \quad [\mathrm{J}] \tag{10}$$

which means that for the same total impulse, the energy needed to produce it is proportional to the exhaust velocity. Hence selection of high velocities $\boldsymbol{v_e}$ i.e. high $\boldsymbol{I_{sp\,thr}}$ for a thruster concept is paid by more energy consumed. In the case of electric propulsion, with a power source $\boldsymbol{P_{el}}$, one gets the very well-known relationship of the power-to-thrust ratio proportional to the $\boldsymbol{I_{sp\,thr}}$ taking into account $\boldsymbol{\eta}$ the efficiency of the overall process of conversion from the power source to the mechanical power:

$$\frac{\boldsymbol{P_{el}}}{\boldsymbol{F}} = \frac{\boldsymbol{v_e}}{\boldsymbol{2\eta}} \quad \text{or} \quad \frac{\boldsymbol{P_{el}}}{\boldsymbol{F}} = \frac{g_0 I_{sp\,thr}}{\boldsymbol{2\eta}} \quad [\mathrm{W/N}] \tag{11}$$

Eq. (11) highlight the fact that high $\boldsymbol{I_{sp\,thr}}$ is not free of charge, but paid by more power demand for the same thrust.
Hence, for electric propulsion aboard CubeSat, $\boldsymbol{I_{sp\,thr}}$ can be seen more like **a criterion of information** of the thrusters with respect to their **power demand**. At similar thrust force**,** systems with **higher $\boldsymbol{I_{sp\,thr}}$** will need **higher power capabilities** from the satellite.

3 Limitations of the conventional approach for small satellites

At thruster level, the thrust along with the mass flow rate are the main data for evaluating the thrusters' performance, because most of the time they can both be measured experimentally. In the traditional approach, the specific impulse $\boldsymbol{I_{sp\,thr}}$ (ratio of the two previous measured data) is the most commonly used criterion. We have explained in the previous section why it is only relevant when the foreseen mass of propellant is much larger than the dry mass of the propulsion system.
The Figure 1 is an application of this traditional approach of displaying thrusters' performance, applied to CubeSat constraints, i.e. focused on nanosatellites, that is to say 1-10 kg CubeSats (up to about 6U-CubeSats), which represents the majority of small satellites [R 1]. In terms of power, it is common to assume (as peak value) few watts per CubeSat unit, which means 3U-CubeSats require propulsion systems that consume much less than 10 W for quasi continuous propulsion.
Deployable solar panels mounted on the structure can increase the power received (up to doubling it or more), but they will also increase the mass and complexity of the satellite operations. Hence, a state-of-the-art of propulsion systems weighing less than 3 kg and consuming less than 40 W peak power is represented.
By means of the classical parameters that we have defined above and with the hypothesis of low structural index -- propellant mass much higher than the propulsion system dry mass -- it is possible to compare the performance

[1] Another unit is available in the literature for $I_{sp\,thr}$ or $g_0 I_{sp\,thr} = \frac{F}{\dot{m}_p} = \frac{I_{tot}}{m_p}$ [N.s.kg^{-1}].

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of different CubeSat propulsion types. A general introduction to the propulsion system options is available in the annex, highlighting the differences in thruster concepts with respect to their compatibility with the unified propulsion system or not [R 4] which is detailed in further chapters. From Figure 1 we can observe two main categories of propulsion systems: the chemical and electrothermal propulsion systems, delivering usually from 1 mN to 1 N of thrust and the classical criterion $I_{sp\ thr}$ of less than 200 s. On the contrary, electromagnetic and electrostatic propulsion systems deliver from 10 µN to 10 mN of thrust and the classical criterion $I_{sp\ thr}$ from 300 s up to 3000 s or even more. The latter is therefore "recommended" by thruster designers for high ΔV mission profiles, whereas the former will perform faster manoeuvres.

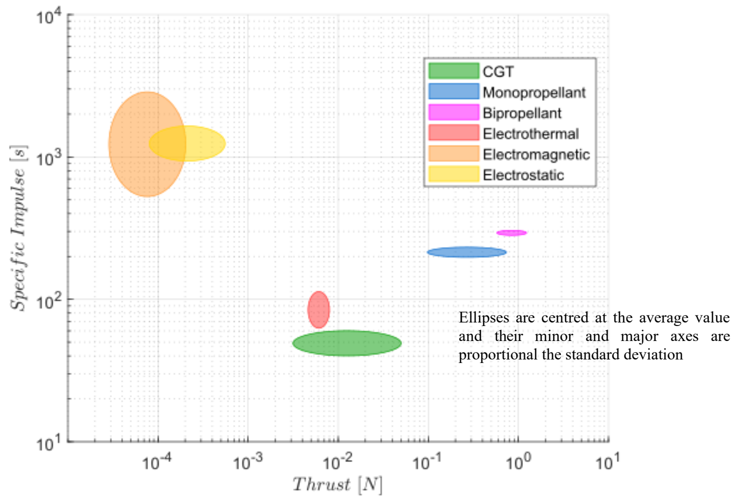

Figure 1: Traditional approach thrusters' performance: Specific impulse versus thrust, prop. systems < 3 kg ; power < 40 W.

Although the classic approach using $I_{sp\ thr}$ allows a first comparison of available thrusters, the use of the thruster specific impulse fails short in capturing many essential system aspects, such as the hardware mass, the electric power supply, the thermal and electromagnetic shielding when needed or the required attitude control capabilities in particular during thrust periods. These insufficiencies appear obvious when these criteria are applied to nano/micro-satellites class such as CubeSats for which it is clear that the underlined hypothesis stating that they have a low structural index --propellant mass to dry mass -- is false. As a matter of fact, such systems have high indexes with dry masses 5 to 100 times heavier than the propellant mass. In addition, the impact of Electrical propulsion on the power system of the CubeSat may become very significant either when thrusters are used by intermittence or quasi continuously.

3.1 Case of Electric propulsion: power impacts

When the electric propulsion is used by intermittence, more instantaneous power can be used with the help of suited batteries. Such intermittence use of propulsion is correct for short duration tests, but it cannot allow long duration propulsion for hours. Indeed, the high-power battery size has to increase proportionally to the thrust duration, and so does the time to recharge such batteries from low power solar arrays (greatly limiting the propulsion duration). The advantage of an intermittence use of propulsion is that the attitude of the satellite can be easily switched between the thrust mode attitude and the Solar cells Sun pointing mode attitude when charging the batteries, which are two independent modes.

When the propulsion is used quasi continuously, the electrical power shall come primarily from the solar arrays. There are two cases to be assessed for the solar array: rotating solar array or fixed solar arrays.

With rotating solar array (as for much larger satellite, with thrust axis non colinear to the rotation's axis of Solar array), it is always possible to combine the thrust mode attitude with the Solar cells Sun pointing attitude because both modes allow one free degree of freedom (DoF) different from the other thanks to their non-collinearity. However, using rotating solar arrays implies higher mass of hardware and more reliability concerns. In addition, batteries shall be used during eclipses and charged again from solar array when in the sun, hence increasing the nominal power and eventually increasing the dry mass.

With fixed non-rotating solar array (as it is the case for all small satellites considered), giving that a certain attitude is requested during the orbit transfer (for example thrust vector tangent to the orbit), it is not always possible

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to get a Solar cells Sun pointing attitude, even with the DoF available around the thrust axis. To perform assessments anyway, a common practice is to rely on the Orbit Average Power (OAP) which is the power really available in average for one orbit when the attitude of the satellite has to follow a prescribed attitude pointing law.
For instance, let us assume a spacecraft in LEO. We take into account eclipses, the possibility of rotating the spacecraft around the thrust axis to maximize the area exposed to the Sun and we consider that the line of sight of the Sun is in the orbital plane. At mid-life, one gets OAP ≈ 34% of the peak Power at Beginning of life (BOL). *For higher orbit altitudes, OAP/Power$_{peak}$ increases a bit, but never higher than $2/\pi$ i.e. 64% at BOL for a line of sight of the Sun in the orbital plane. Considering helio-synchronous orbit 6-18h (case of line of sight of the Sun orthogonal to the orbital plane) could increase the power for the first orbits, but as the orbit transfer progress, the synchronism is lost and eclipses will occur anyway.*

3.2 Electric propulsion: Typical example

Let us consider an electric propulsion system [R 2] that consumes 40 W of power and claims 3770 s of $I_{sp\ thr}$. For a 3 kg satellite at BOL (typical mass for a 3U CubeSat), the designer translates the typical performance of its thruster into ΔV= 2879 m.s^{-1} of ideal velocity.

Even high compared to other thruster concepts, such a ΔV capability is actually penalised by the low thrust arcs losses which lead to an efficiency of about 70% compared to an impulsive manoeuvre. The fact that out of the low Earth orbit an orbit transfer mission with a unique thruster or actuator is simply unforeseeable (because of the lack of attitude control to correct the parasitic torque disturbances from the thruster alignment or from the Sun radiation pressure) being for now disregarded, such ΔV capability would allow a CubeSat launched in GTO to escape the attraction of the Earth and perform some interplanetary mission , although such an orbit transfer would take many years due to the very low acceleration. And from LEO, the ΔV of 2879 m/s would allow a spiral up orbit transfer from 500 km to 11 400 km (continuous thrusting, with thrust tangent to the velocity).

The same designer mentions that the propulsion system has a dry mass m_{dry}= 0.75 kg and carries 0.25 kg of propellant (with a usable efficiency taken at 90%). Thus, the "propulsion package" wet mass is 1 kg and the rest of the considered CubeSat mass is 2 kg. The corresponding total impulse is $I_{tot} = \Delta V.m_{\boldsymbol{pMax}}/\ln\left(\frac{m_0}{m_f}\right)$=8310 Ns.
For taking advantages of the huge ΔV promised by the designer, the only obvious choice is to be able to use quasi continuously the propulsion and thus, using the electrical power from the solar arrays (or during eclipses, the power stored into suited batteries).
But the required power of 40 W to be at least continuously produced by the solar arrays is not very common for 3 kg CubeSat. We need to assess the mass impact on the CubeSat of the added solar panels. One uses the so called α_{pow} mass-to-power ratio (units of kg/kW), with the mass in numerator and the power kept fixed, because it allows to get the total mass by simply adding the α_{pow} numbers of different components such as solar arrays, power processing units, thrusters etc.

On the horizon, it is foreseen that solar cells efficiency will increase, thin film will decrease the mass giving α_{pow} ≈1 kg/kW or less. This is of course not currently available but maybe within the next decades or never. And it is important to notice that such ratio concerns mainly the solar cells (at BOL), while what is needed for a spacecraft is solar panels and arrays with deployable capabilities (for example spring-loaded hinges and hold-down/release or roll-out assembly of the fragile thin film making the packaging quite challenging) including the potential solar cells degradations at mid-life and the impact on the power system H/W needed, wiring mass...

Currently, the α_{pow} values for full deployable solar panels and arrays having good TRL are unfortunately much higher, with α_{pow_BOL} in the range [7, 74] kg/kW, see [R 11] to [R 15], while most of the time the currently market available plug and play devices are around α_{pow_BOL}=22 kg/kW. With an OAP/Power$_{\text{peak BOL}}$ = 34% as described above for a thrusting pointing tangent to the orbit (along the orbital velocity), the α_{pow_OAP} goes up to 64 kg/kW.
This simple ratio allows to assess the mass of the solar arrays needed for being able to feed continuously a thruster of 40W. *As an average power it includes of course the extra power needed for re-charging the batteries used during eclipses --however considering an efficiency of charge/recharge of 100%--.*
We obtain a required mass of solar arrays of 2.56 kg.
This mass can be considered as a hidden mass linked to the thruster concept.
One shall mention that this mass may be rather optimistic because the mass of the batteries needed has been neglected as well as the impact of such high power on the H/W Power Management And Distribution (PMAD) of the CubeSat and also the thermal losses impact from the power system and from the thruster itself, and on top of that the mass of

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the needed attitude control system (with actuators like other thrusters or by solar sailing with sufficient control authority with respect to the thruster parasitic torques) for a continuous spiral up to near MEO...

As a result, the total mass required for the propulsion system to operate is > 1+2.56 kg. The drawback is that such mass 3.56kg becomes already higher than the initial total mass of the satellite at BOL set by the thruster designer at 3 kg (=2 kg +1 kg of propulsion). This shows that the case claimed by the thruster designer leads to an impossible operational option. This result makes it unrealistic to consider such a propulsion system for the considered 3kg CubeSat. On can see that this case is characterised by the fact that the propulsion dry mass (including its hidden required mass) is in total 13 times higher than the propellant mass.
What is wrong in such designer claims? This is the subject of the next chapter.

4 First possible "selection index" proposed for small satellites

A first answer is proposed with the introduction by P. Erichsen [R 8] of I_{ssp} the so-called "system-specific impulse[2]":

$$\boldsymbol{I_{ssp}} = \frac{\boldsymbol{I_{tot}}}{\boldsymbol{m_{PS}\, g_0}} \text{ [s]} \tag{12}$$

with m_{PS} the mass of the propulsion system. Thus defined, the system-specific impulse requires specifying propulsion system mass for the two main concepts of thrusters.

For chemical thrusters, m_{PS} is

$$\boldsymbol{m_{PS}} = \boldsymbol{m_{H/W}} + \boldsymbol{m_{PSS}} \text{ [kg]} \tag{13}$$

where $m_{H/W}$ is the propulsion hardware, including the thrusters, valves and piping, and m_{PSS} is the mass of propellant with its corresponding tankage.

When it comes to electric propulsion, the mass of the propulsion system is adapted to take into account the added part of power supply and control system m_{El}:

$$\boldsymbol{m_{PS}} = \boldsymbol{m_{H/W}} + \boldsymbol{m_{PSS}} + \boldsymbol{m_{El}} \text{ [kg]} \tag{14}$$

Going back to the example proposed previously, the I_{ssp} computed is 8310 / (3.56 * 9.80665)= 238 s, which is only 6% of the thruster $I_{sp\,thr}$ of 3770 s.

With a such value of I_{ssp} the satellite designer can already see that if applied as is the I_{ssp} instead of the $I_{sp\,thr}$ in the Tsiolkovski equation, then **the final mass becomes ridiculously small** for performing the huge ΔV claimed by the thruster designer. Hence it is clear that the I_{ssp} **is a kind of selection criterion** suited for a rough comparison between several propulsion concepts. But one shall add that the I_{ssp} definition shows that its value depends on the total impulse considered (8310 Ns in the case above). If smaller total impulses are needed, the I_{ssp} to be considered is roughly proportional: for delivering 1000 Ns with the same thruster, I_{ssp} becomes roughly 1000 / (3.56 * 9.80665) = 28 s. *Of course, if the propellant mass (set to 0.25 kg) can be adjusted, then the real needed propellant mass has also to be reduced proportionally, then* I_{ssp} *=1000 / ((3.56-0.25+0.25*1000/8310) * 9.80665) = 30 s.*
Hence it is important to mention that for being applied into the Tsiolkovski equation, the suited I_{ssp} is only one value in the range starting from zero: $I_{ssp} =]0, 238]$ s.

This kind of I_{ssp} range]0, 238] s is however not so bad, it is already a quite good performance compared to some other thruster concepts, even if, for the considered, the warm-up time constraint for getting operational in hot stand-by state as well as the thermal rejections can be considered as important drawbacks. The use of I_{ssp} may prevent the satellite designer from building **impossible cases**.

5 A new definition of the thruster "selection criterion" for a full system performance

5.1 New definition of the I_{ssp} as a "selection criterion"

Despite the promising handling of the power supply, the above form of system-specific impulse coming from Erichsen [R 8] does not cover all the aspects that we consider essential when it comes to characterize propulsion systems and to

[2] Same as for the I_{sp}, the I_{ssp} can be defined without the gravitational standard.

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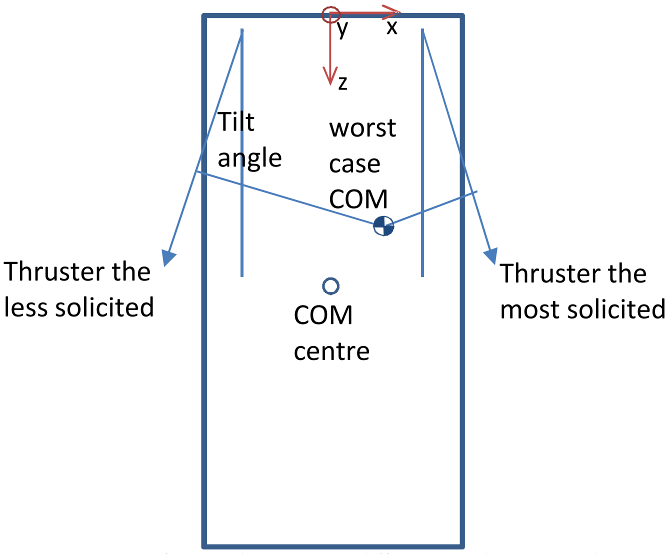
enable comparisons between thruster concepts. The system and thruster characteristic criteria considered are listed below:

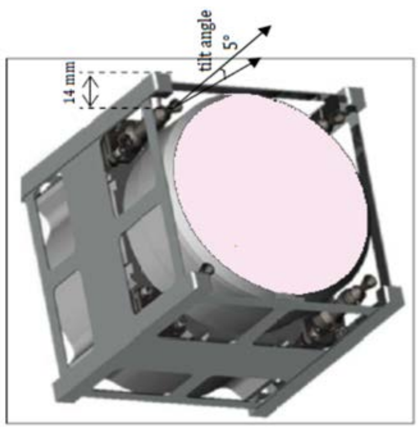
1. System redundancy philosophy,
2. Number of thrusters,
3. Continuous use of propulsion or propulsion by intermittence,
4. Thruster concept enabling unified propulsion system or not,
5. Thruster concept availability (for example: system warm up time & power needed before each thruster use),
6. Thruster system concept hot stand-by power,
7. Thruster concept thermal rejections,
8. Thruster concept “qualified” lifetime or operational total impulse,
9. System worst case total impulse,
10. Attitude control capability and consistency with thrust vector misalignment’s wrt the COM,
11. Resultant magnetic dipole of the propulsion (On and Off),
12. Swirl of the propulsion plume (torque generated by some thrusters around their axis),
13. Power needs and management of simultaneous or sequential use of thrusters,
14. Thruster system mass (dry including its electronic and wet)
15. Thruster costs and system cost
16. Thruster development level (TRL)
17. Etc.

The above criteria are dealing with the system philosophy chosen as well as with the thruster concept assessed. They comprise some of their explicit or hidden advantages and draw-backs.

Moreover, some generally hidden interconnections between system and thruster are highlighted. For example, the impact of a redundancy philosophy with the use of a thruster concept not compatible of the unified propulsion system concept defined as the impossibility for the thrusters to be fed by a common propellant tank (i.e. all thrusters concept relying on solid or electrically conductive propellant, see Annex Figure 5, this fact being sometimes advertised by some thruster designers as a great added value for their concept) may reduce the I_{ssp} by a factor up to 2.

This kind of drawback can also be easily seen on the total impulse at system level in a worst-case design with several thrusters not compatible of the unified propulsion system. Of course, when a CubeSat uses only a single thruster, the fact of being compatible or not compatible of the unified propulsion system makes no difference, but with a single thruster or actuator, no orbit transfer mission out of the low Earth orbit can be foreseen (magneto-torquers are non-efficient, reaction wheels have to be off-loaded with actuators), that is only short thrust pulses can be performed most of the time. Hence, using several thrusters in a system, thruster concepts not compatible with the unified propulsion concept have a great penalty with a worst-case design because this comes from the differences in lever arms as shown in Figure 2.





Example of propulsion system with 4 tilted thrusters for primary propulsion and 3-axis control (*ref. MPS-120 from Aerojet Rocketdyne*).

Figure 2: Design in a worst-case of COM dispersions: difference in lever arms between thrusters.

When the centre of mass (COM) is located at the centre of geometry of the satellite, there are no differences between thrusters: all shall be used about the same time for the compensation of the parasitic torques. If four thrusters are

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considered, the total impulse at system level is the sum of each thruster unit, that is 4 units. But when the worst-case variations of the COM are taken into account it is obvious that one thruster will be more solicited than the others in order to compensate for the parasitic torques. For a tilt angle of 5°, and with the CubeSat specification of COM dispersions [R 16], one of the thrusters shall be turned on proportionally 100% of the time while it has been determined that the others have to be run only 38% of this time. And a similar effect occurs once again when considering the torques around the orthogonal axis (x) and a bit around the swirl axis (z). That means that when the first thruster runs out of propellant, it has been determined that it will still remain 73% of propellant in average into the other 3 thrusters. Because a first thruster running out of its own propellant implies that the mission is necessarily ending at this point, this reduces the operational total impulse at system level to 1.9 units instead of a potential of 4 units (in other words the worst-case total impulse represents only 1.9/4 of the best-case capability). Hence, the worst-case design shall be made with a reduced performance: the operational total impulse is reduced to about 50% of the loaded total impulse. This highlights the high penalty for systems non-compliant with the unified propulsion system concept.

In order to reduce this high penalty, constraints on the CubeSat designer could be imposed by the thruster designer to reduce the COM dispersions, but such approach is highly questionable regarding the failure in flight of the D-Sat CubeSat that asked a very small dispersion of only 1.5 mm [R 5].

Also, the attitude control capability may interact with the number of thrusters. For orbit transfer out of the low Earth orbit missions a minimum of 4 thrusters or solar sailing actuators is needed for enabling 3 axis attitude control in addition to the main thrust (that is a "3.5 DoF" which is still compliant with the Wiktor and Chen theorem [R 3] stating a need of ≥"m+1" thruster for having "m" DoF).

Last but not least, for enabling 3 axis attitude control out of propulsion phase (in the case of propulsion by intermittence) for allowing a Solar cells Sun pointing mode, the thruster concept availability shall be always quasi-immediate within seconds (without hours of warm-up time before being able to use such thrusters) unless some solar sailing torques actuators can be used for allowing the Sun pointing mode.

Let us consider the following evolution of I_{ssp} definition for a propulsion system using one or several thrusters

$$\boldsymbol{I_{ssp}} = \frac{\mathbf{1}}{\boldsymbol{g_0}} \frac{\boldsymbol{I_{tot}.k_2}}{\boldsymbol{(m_p+m_{tank}).k_1+m_{thruster}k_3+(m_{El}+m_{The})k_4+m_{ACS}k_5}} \quad [\mathrm{s}] \tag{15}$$

where

- I_{tot} is the total impulse capability per thruster; g_0=9.80665 m/s^2.
- m_p is the mass of usable propellant per thruster.
- m_{tank} is the mass of the tank needed, for large system it is a percentage of the propellant mass. It may include the systems involved by the tank (insulation, heater control, etc.). For non-exotic propellants and large systems, $\frac{m_{tank}}{m_p}$ is similar to the structural index presented above and can be rather a constant, $\frac{m_{tank}}{m_p} = k$.
- k_1 is a parameter generally set to 1, but may be set to 2 in a mission requiring reliability's redundancy when the thruster concept is incompatible with the unified propulsion system.
- k_2 is a parameter generally set to 1, but may be set to ≈ 0.5 in a mission worst-case, with several thrusters thrusting in about the same direction, when the thruster concept is incompatible with the unified propulsion system.
- $m_{thruster}$ is the dry mass of one thruster (the definition is given for one thruster total impulse, the use of several thrusters is taken into account in the factor k_4 defined below).
- k_3 is a parameter generally set to 1, but may be set to 2 in a mission requiring reliability's redundancy.
- m_{El} is the mass of the electric system dedicated to one thruster: this includes the thruster power supply as well as, if any, the dedicated mass of solar array and batteries for providing the power which is related to the power by α_{pow} kg/kW. For missions with continuous propulsion, the value of $\alpha_{pow\ OAP}$ shall be used instead of the value mentioned before.
- m_{The} is the mass of the thermal system management dedicated to one thruster thermal rejections, the ones of its power system as well the ones coming from the hot stand-by power if any.
- k_4 is a parameter generally set to 1/N, but may be set to 1 in a mission requiring a simultaneous use of the N thrusters.
- m_{ACS} is the mass of the attitude control system to be added in case of insufficiencies.
- k_5 is a parameter generally set to 0 in the case of N=4 thrusters suited to perform the 3 DoF for attitude control, but may be set to 1 in the case of insufficient thrusters number

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Because $\boldsymbol{I_{tot} = m_p\, g_0 I_{sp\,thr}}$ one also gets the following direct equation between the specific impulse at thruster level $\boldsymbol{I_{\mathrm{sp\ thr}}}$ and the system specific impluse $\boldsymbol{I_{ssp}}$:

$$\boldsymbol{I_{ssp}} = \frac{\boldsymbol{I_{\mathrm{sp\ thr}}.k_2}}{\boldsymbol{(1+k)k_1+\frac{1}{m_p}[m_{thruster}k_3+(m_{El}+m_{The})k_4+m_{ACS}k_5]}} \quad [\mathrm{s}] \tag{16}$$

where $\boldsymbol{k}$ is a structural index (tank dry mass per propellants mass ratio), defined by $\boldsymbol{k} = \frac{m_{tank}}{m_p}$.

5.2 Application of the I_{ssp} as a "selection criterion"

In order to simplify the analyses, the systems considered are composed of:

A. one single thruster system with short thrust pulses: I_{ssp} can be assessed with the next eq. (17) for N=1
B. a multiple thruster system with N=4 thrusters having their thrust about along the main thrust axis, and working sequentially (in order to minimize the mass, cost and complexity impact on the power subsystem): I_{ssp} can be assessed with eq. (17) when the thruster concept is compatible of Unified propulsion system or with eq. (18) when the thruster concept is not compatible of Unified propulsion system (i.e. with individual tank for each thrusters).
Note: Some thrusters concepts need a heating for becoming in hot stand-by status i.e. being ready for operations (case of Iodine sublimation or Indium liquefaction, ...): within a system of 4 thruster, the power to be considered is the nominal power of one thruster plus the hot stand by power of the 3 other thrusters.

It is clear that in the case A with "one single thruster system" only missions crossing or in the Low Earth Orbit are allowed. Even in LEO mission, the impact of the thrust function on the CubeSat ACS can be significant. But if the thrust pulses are short with respect to the orbital period, then the impact can be negligible, and then the coefficient k_5 in the term $m_{ACS}k_5$ can be set to 0. As a consequence, in the case A, large orbit transfers cannot be performed and high ΔV missions may take huge duration probably incompatible with the CubeSat lifetime (unless dedicated other propulsion systems/actuators are implemented aboard the CubeSat, complex case not considered in the present paper).

On the contrary, the case B "with 4 thrusters" allows missions into LEO but also toward MEO, GEO or the Moon, etc. In such cases, there are no needs for any additional attitude control if the thrusters are able to be thrust modulated or duty cycled for performing the 3 DOF attitude control, and then the coefficient k_5 in the term $m_{ACS}k_5$ can be set to 0 (even the CubeSat can be designed without any other ACS than the propulsion system). Recalling that k_4 is a parameter generally set to 1/N, but may be set to 1 in a mission requiring a simultaneous use of the N thrusters, one has for a sequential use of the thrusters to consider $\boldsymbol{m_{El}}$ and $\boldsymbol{m_{The}}$ the mass of the electric system and thermal system dedicated to one thruster only (except the hot stand-by power needed for the other thrusters).

For a given propulsion system with N thrusters, compatible with the Unified propulsion system concept, also valid in the case of N=1:

$$\boldsymbol{I_{ssp}} = \frac{\boldsymbol{1}}{\boldsymbol{g_0}} \frac{\boldsymbol{I_{tot\ all\ N\ thrusters}}}{\boldsymbol{m_{pTotal}+m_{tankTotal}+N.m_{thruster}+m_{El}+m_{The}}} \quad [\mathrm{s}] \tag{17}$$

For a given propulsion system with N thrusters, NOT compatible with Unified propulsion system, in a worst-case mission where the mission ends when the first thruster runs out of propellant, about half of the total impulse is available in operation worst case:

$$\boldsymbol{I_{ssp}} = \frac{\boldsymbol{1}}{\boldsymbol{g_0}} \frac{\boldsymbol{0.5 * I_{tot\ all\ N\ thrusters}}}{\boldsymbol{m_{pTotal}+m_{tankTotal}+N.m_{thruster}+m_{El}+m_{The}}} \quad [\mathrm{s}] \tag{18}$$

A first application of the process described above has been performed taking into account a power subsystem mass needed to be dedicated to the propulsion system with α= 64 kg/kW for single thrusters requiring more than 5 W. By lack of relevant data from thrusters' designers, thermal rejection impacts of the propulsion system on the CubeSat are

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for now not possible to assess, hence the mass in the term $\boldsymbol{m_{The}}$ could not be taken into account in the current analyses. However, it is clear that the power needs are more or less linked to amount of thermal rejection, making that higher power are more difficult to manage and implement than lower power.

The Table 1 and Table 2 come mainly from a recent reference [R 10] with additional data found in the open literature (note: some thruster designers, prudent to publish any relevant data, are not included of course in this review). The references cited in the column "Ref." are the ones of the cited reference. In this preliminary paper, the System Specific Impulse $\boldsymbol{I_{ssp}}$ has been computed on a selection of thrusters and systems described with sufficient data available. Sometimes it is difficult to distinguish between sustainable "qualified" data and expected capabilities (if all goes well for low TRL concepts), hence unfortunately for this preliminary work, this important aspect could not be taken into account. The Figure 3 plots the results from the Tables (> 100 items) of $\boldsymbol{I_{ssp}}$ versus $\boldsymbol{I_{\mathrm{sp\,thr}}}$ and Power versus $\boldsymbol{I_{ssp}}$. One *shall mention that some thrusters are not enough detailed for their hot stand-by power consumption: in particular the RF Ion thrusters using Iodine, hence for those thrusters the* $\boldsymbol{I_{ssp}}$ *are maybe very optimistic and their representative point is crossed in the figure.*

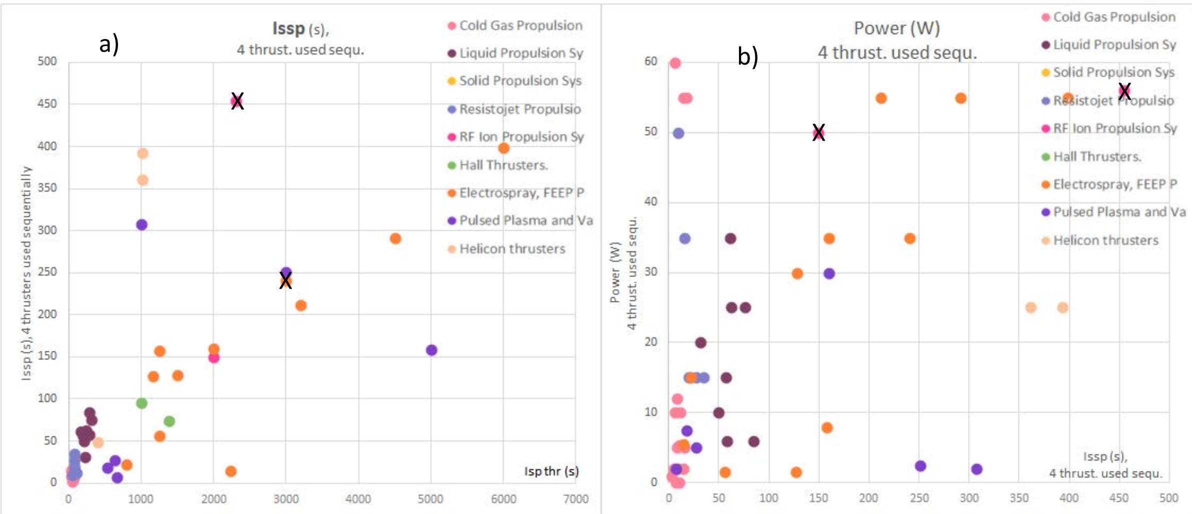

Figure 3: a) System specific impulse versus thruster Isp b) power versus System specific impulse. The comparison shows the irrelevance of the thruster specific impulse when used aboard CubeSat within a system of 4 thrusters used sequentially

The Figure 3 allows a rough comparison of systems of 4 thrusters used sequentially (to minimize the power consumption, but allowing the thrust orientation attitude mode without other actuators): the figures of $\boldsymbol{I_{ssp}}$ and $\boldsymbol{I_{\mathrm{sp\,thr}}}$ are shown. As already mentioned above, the high structural index of CubeSat makes that the $\boldsymbol{I_{ssp}}$ is more than 10 times lower than the $\boldsymbol{I_{\mathrm{sp\,thr}}}$. This shows the poor value of the $\boldsymbol{I_{\mathrm{sp\,thr}}}$ as selection criterion for their real use aboard CubeSat, as mentioned above, it can be seen as an indication of the specific power demand of thruster's concepts according to eq. (11**)**. On the contrary, $\boldsymbol{I_{ssp}}$ allowing an objective comparison between thruster concepts at system level can be considered as a valuable selection criterion. And some concepts have quite good potential regarding their use aboard CubeSat while they would not be considered on the base of their $\boldsymbol{I_{\mathrm{sp\,thr}}}$. In this preliminary paper, one shall highlight that the Figure 3 shows only a partial picture of thruster concepts comparison: many useful data could not be found in the documentation available. The power needs plotted in Figure 3 versus the $\boldsymbol{I_{ssp}}$ can provide an indication on the difficulty to implement the thruster system along with its thermal rejections and the impact in terms of added mass on the CubeSat. Thruster designers should be encouraged to use the above definition of System specific impulse and to publish their results, or to provide the data needed to compute it unambiguously.

6 Conclusions

The new definition of the System Specific Impulse I_{ssp} coming from an adaptation of the one of P. Erichsen [R 8] for covering the cases of unified and non-unified propulsion system as well as the worst-case design concepts with or without redundancy has a great impact on the thruster concepts to be chosen as best suited for dealing with particular missions. This is coming from common sense.

It can be considered as a new selection criterion of thruster concepts in order to allow thruster designers or "business angels" to make their own best selections before taking any investment decision on a particular concept.

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The current definition of the System Specific Impulse I_{ssp} presented in the paper is the following:

$$I_{ssp} = \frac{1}{g_0} \frac{I_{tot}.k_2}{(m_p+m_{tank}).k_1+m_{thruster}k_3+(m_{El}+m_{The})k_4+m_{ACS}k_5} \quad \textbf{[s]}$$

The formulation is suited for a single thruster as well as for a system of several thrusters used simultaneously or sequentially.

This System Specific Impulse I_{ssp} value can be used into the Tsiolkovsky equation, but it must be highlighted that if the total impulse needed for a mission is lower than the operational capability of a thruster concept, the value to be used into the Tsiolkovsky equation is only a value in the range $]0, \boldsymbol{I_{ssp}}[$, roughly proportional to it.

In this first improvement of the System Specific Impulse I_{ssp} many of the thruster advantages or disadvantages could not be taken into account. This shows that further improvements analyses are needed for a complete "selection criterion" picture.

The most important aspect to be pointed out is the fact that the data relative to some thruster concepts are sometimes not explicit, are sometime too optimistic, or sometime only based on the thruster specific impulse $I_{\mathrm{sp\,thr}}$ which is as shown in the paper not a relevant selection criterion.

Also, to be highlighted is that the System Specific Impulse I_{ssp} as used here takes mainly into account the power needs as well as the attitude control needed for the propulsion. Regarding the management of the thermal rejections and their impact in terms of added mass on the CubeSat along with the complexity of radiators implementation if any, those aspects are for now quite difficult to assess in the general formula by lack of data from thrusters' designers. However, those aspects have to be taken into account as general drawback linked (or proportional) to the power consumption (in thrusting mode of course but also in hot stand-by mode).

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Table 1: A selection of thruster ready or foreseen for CubeSats after [R 10]

Company/Institution with Location	Engine	Propellant	Remarks // Heritage	Thrust (mN)	Power (W)	Dry mass (kg)	Propel-lant (kg)	I_{tot} (Ns)	Isp thr (s)	I_{ssp} (s) single thruster, short pulses	Unified Prop. System	I_{ssp} (s), 4 thrust. used sequ.	I_{ssp} (s), 4 thrust. redundant	Ref. of [R 10]
Table 1. Cold Gas Propulsion Systems.														
SSTL, Guildford, United Kingdom	SNAP 1	Liq. Butane	flown on Giove-A (600 kg)	50					43		Yes			[12,15,16]
UTIAS-SFL, Toronto, ON, Canada	CNAPS	SF6	// CanX-4 (6 kg), CanX-5 (6 kg)	10		0.50	0.18	175	35	26	Yes			[17]
	...CNAPS other op.			40		0.50	0.18	175	35	26	Yes			
UTIAS	CNAPS	SF6		60	5.4	0.91	0.18	115	65	8	Yes	10	5	[AA]
Microspace Rapid, Singapore	POPSAT-HIP1	Argon	// POPSAT-HIP1 (3U/3.3 kg)	1					43		Yes			[18]
Microspace Rapid, Singapore	POPSAT-HIP1	Argon		1	2		0.02	10	43					[AB]
GOMSpace, Denmark	MEMS Cold Gas	Methane	// TW-1 (one 3U and two 2U) also flown on PRISMA (180 kg)	1					50		Yes			[9,19,20]
	...MEMS other op.			1					75		Yes			
Nanospace	MEMS	C4H10		1	2	0.22	0.06	40	67.9	15	Yes	15	7	[AC]
VACCO	MEPSI MiPS	C4H10		53	1	0.46	0.53	23	44	2	Yes	2	1	[AD]
VACCO	Palomar	C4H10		35	5	0.89	0.17	85	50	8	Yes	8	4	[AE]
GomSpace/NanoSpace	NanoProp 3U	C4H10		1	2	0.30	0.50	40	60	5	Yes	5	3	[AF]
GomSpace/NanoSpace	NanoProp 6U	C4H10		5.5	2	0.77	0.13	80	60	9	Yes	9	5	
VACCO Industries, El Monte, CA, USA	CPOD	R134a	8 thrusters // CPOD (3U)	25		1.00	0.35	138	40	#N/A	Yes	10	5	[21–23]
VACCO	C-POD	R134a		25	5	0.77	0.47	186	40	15	Yes	15	8	[AG]
VACCO	MiPS	standard	R134a	10	10	0.43	0.11	44	40	4	Yes	6	4	[AH]
		end-mounted	R134a	10	10	0.44	0.24	93	40	7	Yes	11	6	[AI]
VACCO Industries, El Monte, CA, USA	MIPS		5 thrusers /system	55		0.46	0.05	34	65	#N/A	Yes	7	3	
University of Texas	ACS	R236-fa	designed for the Interplanetary NanoSpacecraft Pathfinder In Relevant Environment	21	60	0.91	0.18	115	65	2	Yes	6	4	[AJ]
VACCO	JPL MarCO MPS	R236-fa		25	55	1.57	1.92	755	40	11	Yes	18	10	[AK]
VACCO	NEA Scout MiPS	R236-fa		23	55	1.26	1.28	500	39.8	8	Yes	15	9	[AL]
VACCO	CuSP	R236-fa		25	12	0.51	0.18	69	39.7	5	Yes	8	4	[AM]
Table 2. Liquid Propulsion Systems.														
Aerojet Rocketdyne, Sacramento, CA, USA	GPIM Propulsion System	AF-M315E		400					235		Yes			[29]
Aerojet Rocketdyne	GR-1	AF-M315E	will fly on GPIM mission in 2018-2019	1420	22				231					[AN]
	...GPIM other op.			1100					235		Yes			
Aerojet Rocketdyne, Sacramento, CA, USA	MPS-120 CHAMPS	Hydrazine	-	260					215		Yes			[30–34]
Aerojet Rocketdyne	MPS-120	Hydrazine		250	10	1.06	0.38	775	210	38	Yes	49	26	[AO]
Aerojet Rocketdyne, Sacramento, CA, USA	MPS-130 CHAMPS	AF-M315E	-	1.5					240		Yes			[30,32]
Aerojet Rocketdyne	MPS-130	AF-M315E		750	25	1.06	0.50	1200	244.4	39	Yes	62	35	[AP]
ECAPS, Solna, Sweden	HPGP	ADN based LMP-103S	flown on PRISMA (180 kg)and SkySat-3 (10.5 kg tank)	1000					232		Yes			[19,35]
ECAPS	HPGP	LMP-103S		1000	10	0.34					Yes			[AQ]
VACCO / ECAPS	ADN MiPS	LMP-103S	One thruster for orbit control and four for attitude with different thrust	100	15	1.09	0.53	1036	199.81	41	Yes	57	30	[AR]
Busek, Natick, MA, USA	BGT-X1	AF-M315E		100					214		Yes			[9,36]
Busek, Natick, MA, USA	BGT-X5	AF-M315E	-	500					225		Yes			[36,37]
Busek	BGT-X5	AF-M315E		500	20	1.24	0.26	565	220	21	Yes	32	17	[AS]
VACCO	Lunar Flashlight PS	LMP-103S	for Lunar Flashlight Mission	150	35	3.00	2.00	3320	169.0	47	Yes	61	32	[AT]
Tethers Unlimited, Bothell, WA, USA	HYDROS	Liquid water		250					256		Yes			[9,38]
	...HYDROS other			600					256		Yes			
TUI	HYDROS-C	H2O		1200	25	1.90	0.74	2252	310	54	Yes	76	40	[AU]
Hyperion Technologies	PM400	N2O(ox) C3H8(fuel)		1000	6	1.40	0.63	1750	285	74	Yes	84	43	[AV]
Hyperion Technologies	PM200	N2O(ox) C3H8(fuel)		500	6	1.10	0.31	850	285	48	Yes	58	30	[AW]
Table 3. Solid Propulsion Systems.														
Aerospace Corporation, El Segundo, CA, US	Is p 30 s	-	-	37					187		No			[39]
Orbital ATK, Dulles, VA, USA	STAR 4G	Al and Ammonium perchlorate	-	13					269.4		No			[46]
DSSP,Reno, NV, USA	CAPS-3	HIPEP-501A	flown on SPINSAT (57 kg)						245		No			[44,47]
	...CAPS-3 other op.								260		No			
DSSP, Reno, NV, USA	CDM-1	AP/HTPB	-	76					226		No			[9]
			flown on D-SAT (fail)											
Table 4. Resistojet Propulsion Systems.														
SSTL, Guildford, United Kingdom	LPR	Xe	flown on NovaSAR-S (100 kg)	18	30				48		Yes			[59,60]
CU Aerospace, Champaign, IL, USA and VACCO Industries Inc., Huntsville, AL, USA	PUC	SO	-	5.4	15				65		Yes			[30,61]
CU Aerospace	PUC 0.25U	SO		5	15	0.45	0.27	184	70	11	Yes	20	11	[AX]
CU Aerospace	PUC 0.5U	SO		5	15	0.51	0.47	320	70	17	Yes	27	15	[AY]
CU Aerospace	PUC 1U	SO		5	15	0.64	0.86	593	70	25	Yes	35	19	[AY]
CU Aerospace, Champaign, IL, USA and VACCO Industries Inc., Huntsville, AL, USA	CHIPS	R134a, R236fa	-	30	30				82		Yes			[58,62]
CU Aerospace	CHIPS	R134a		31	25		0.64	478	76		Yes			[AZ]
CU Aerospace	CHIPS	R236fa		23	25		0.74	433	60		Yes			[AZ]
Busek, Natick, MA, USA	AMR	R134a, R236fa	-	10	15				150		Yes			[9,30,49]
University of South. Cal., Los Angeles, CA, USA	FMMR	Water	-	0.129					79.2		Yes			[56]
Sitael	XR-50-050	Xe, Kr		150	50	0.22	0.25	120	49	3	Yes	10	7	[BA]
Mars Space Ltd.	VHTR			100	100	0.25	0.25	245	100	4	Yes	12	10	[BB]
CU Aerospace	MVP	Plastic	inspired by 3D printing technology	6.7	35	0.54	0.66	540	83	16	No	16	9	[BC]

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Table 2: A selection of thruster ready or foreseen for CubeSats after [R 10], cont.

Company/Institution with Location	Engine	Propellant	Remarks // Heritage	Thrust (mN)	Power (W)	Dry mass (kg)	Propel-lant (kg)	I_{tot} (Ns)	Isp thr (s)	I_{ssp} (s) single thruster, short pulses	Unified Prop. System	I_{ssp} (s), 4 thrust. used sequ.	I_{ssp} (s), 4 thrust. redundant	Ref. of [R 10]
Table 5. RF Ion Propulsion Systems														
Busek, Natick, MA, USA	BIT-1	Xe	-	0.18	28				2150		Yes			[9,71]
	...BIT-1 other op.	Xe		0.1	28				3200		Yes			
	...BIT-1 other op.	Iodine	hot stand-by power TBD/thruster	0.1	28				3200		No			
Busek, Natick, MA, USA	BIT-3	Iodine	will fly on Lunar IceCube (6U) hot stand-by power TBD/thruster	1.15	75				2500		No			[71,72]
	BIT-3	Iodine	hot stand-by power TBD/thruster	1.2	56	1.40	1.50	33833	2300	532	No	454	258	[BD]
Airbus, Lampoldshausen, Germany	RIT-µX	Xe	-	0.5					300		Yes			[73]
	RIT-µX	Xe		0.275	50	0.44	0.10	1961	2000	53	Yes	149	106	[BE]
				0.05					3000		Yes			
Airbus, Lampoldshausen, Germany	RIT 10 EVO	Xe	available in 3 designs	5					3200		Yes			[73]
	...RIT other op.	Xe		15					3000		Yes			
	...RIT 1 other op.	Xe		25	145				1900		Yes			
Table 6. Hall Thrusters.														
Busek, Natick, MA, USA	BHT-200	Xe, Kr	flown TacSat-2 (370 kg) and FalconSat-5 (180 kg)	12.8	200				1390		Yes			[46,85,89]
Busek, Natick, MA, USA	BHT-200	Xe, I2	hot stand-by power TBD/thruster	13	200	1.20	0.25	3371	1375	24	Yes	74	56	[BF]
	...BHT-200 other propellant	I	hot stand-by power TBD/thruster	12.8	200				1390		No			
Busek, Natick, MA, USA	BHT-600	Xe, Kr	-	39.1	600				1530		Yes			[85]
	...BHT-600 other propellant	I	hot stand-by power TBD/thruster	39.1	600				1530		No			
Sitael Aerospace, Mola di Bari, Italy	HT 100	Xe, Kr	-	10	100				1100		Yes			[90,91]
	HT 100	Xe		6	120	0.45	0.25	2452	1000	30	Yes	95	75	[BG]
Sitael Aerospace, Mola di Bari, Italy	HT 400	Xe	-	50	100				1750		Yes			[92]
MIT, Cambridge, MA, USA	MHT-9	Xe	-	50	200				1500		Yes			[46]
	...MHT-9 other op.	Xe		20	30				300		Yes			
UTIAS-SFL, Toronto, Ontario, Canada	CHT	Xe, Ar	-	1	200				1139		Yes			[79]
	...CHT other op.	Xe, Ar		10	200				1139		Yes			
Table 7. Electrospray, FEEP Propulsion Systems.														
MIT, Cambridge, MA, USA	S-iEPS	ionic liquid	also called IMPACT // Aero-Cube-8 (1.5U)	0.1	1.5	0.10	0.03		1200		No			[9,94,106]
MIT, Cambridge, MA, USA	S-iEPS	ionic liquid		0.074	1.5	0.10	0.03	319	1160	254	No	127	63	[BH]
Accion Systems, Boston, MA, USA	TILE 5000	ionic liquid	-	1.5	30				1800		No			[30]
	TILE 50	ionic liquid		0.05	1.5	0.05	0.00	60	1250	111	No	56	28	[BI]
	TILE 500	ionic liquid		0.4	8	0.60	0.24	3000	1250	226	No	157	84	[BI]
	TILE 5000	ionic liquid		1.5	30	1.10	0.33	4800	1500	146	No	128	73	[BI]
Busek, Natick, MA, USA	BET-1mN	ionic liquid	flown on LISA Pathfinder (476.3 kg)	0.7	9				800		No			[107–109]
	BET-1mN	ionic liquid		0.7	15	1.07	0.08	605	800	29	No	22	12	[BJ]
Busek, Natick, MA, USA	BET-100	ionic liquid	-	0.005					1800		No			[9]
	BET-100	ionic liquid		0.1	5.5	0.54	0.01	175	2230	20	No	14	8	[BK]
	...BET-100 other op.	ionic liquid		0.1					1800		No			
ENPULSION GmbH, www.enpulsion.com	IFM Nano Thruster FEEP	Indium	hot stand-by power =5W/thruster	0.42	40	0.75	0.25	7800	3200	223	No	212	138	[R7]
	...IFM other op.	Indium	hot stand-by power =5W/thruster	0.3	40	0.75	0.25	14700	6000	421	No	399	260	[R7]
	...IFM other op.	Indium	hot stand-by power =5W/thruster	0.25	20	0.75	0.25	4900	2000	219	No	160	98	[R7]
	...IFM other op.	Indium	hot stand-by power =5W/thruster	0.2	20	0.75	0.25	7350	3000	329	No	240	146	[R7]
	IFM Nano	Indium	hot stand-by power =5W/thruster	0.35	40	0.67	0.23	10150	4500	299	No	291	193	[BL]
Table 8. Pulsed Plasma and Vacuum Arc Thrusters.														
Mars Space Ltd., Southampton, United Kingdom	µPPT	PTFE	PPT		2	0.15	0.01	57	578	36	No			[122]
Mars Space Ltd., Southampton, United Kingdom	PPTCUP	PTFE	PPT	0.04	2	0.28	0.01	39	670	14	No	7	4	[BM]
	...µPPT other op.	PTFE			10	0.15	0.01	71	727	9	No			
Mars Space Ltd., Southampton, United Kingdom	NanoPPT	PTFE		0.09	5	0.32	0.03	190	640	55	No	28	14	[BN]
Primex Aerospace Company Redmond, WA, USA	EO-1 PPT	Teflon	PPT; flown on Dawgstar (13 kg)	0.14	12.5				1150		No			[123,124]
Busek, Natick, MA, USA	MPACS	PTFE (Teflon)	PPT; flown on FalconSat-3 (54.3 kg)	0.144	10				830		No			[46,125]
Busek, Natick, MA, USA	BmP-220	Teflon	PPT	0.14	7.5				536		No			[9,30]
	BmP-220	Teflon	PPT	0.14	7.5	0.46	0.04	220	536	23	No	18	10	[BO]
GWU, Washington, D.C., USA	µCAT	Nickel	VAT // BRICSat-P (1.5U)	0.001					3000		No			[126]
	µCAT	Nickel		0.05	2.5	0.20	0.04	1180	3000	501	No	251	125	[BP]
	...µCAT other op.	Nickel		0.02					3000		No			
Comat	Plasma Jet Pack	Cathode	VAT	0.045	30	0.72	0.08	4000	5000	150	No	159	98	[BQ]
University of Illinois, Champaign, IL, USA	µBLT	Aluminum	VAT	0.054	4						No			[9]
Wurzburg University, Wurzburg, Germany	UWE4 Arc Thr.	Ti, W	VAT	0.002	0.5				900		No			[127]
	...UWE4 other op.	Ti, W		0.01	2				1100		No			
Project LPPT		Liquid PFPE	non conductive liquid 4 thrusters	0.05	2	0.45	0.20	1960	1000	#N/A	Yes	307	#N/A	[BR]
Table 9. Helicon thrusters														
Study Spangelo with CAT iodine Original	RFTk	Iodine	Ambipolar (CAT) TRL 3-4	1.86	25	0.85	3.00	30123	1010	749	No	393	198	[BS]
Study Spangelo with CAT iodine	RFTk	Iodine	Ambipolar (CAT) TRL 3-4	1.86	25	0.85	3.00	30123	1010	564	No	361	190	[BS]
Phase Four	RFTk	Xe	derived from Ambipolar (CAT)	2.7	160	1.00	0.50	1942	396	17	Yes	49	36	

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Annex: propulsion system types

This work is concerned with reaction jets which produce a control force by the expenditure of mass. As a consequence, propelantless systems such as solar sails are not considered here.

Chemical propulsion uses gases under pressure (an ambient or hot temperature) that are accelerated through a nozzle.

- A cold gas thruster (CGT) is a propulsion system in which the propellant does not undergo combustion or electromagnetic acceleration. Although cold gas thrusters are among the simplest propulsion systems, they have low specific impulse.
- Hot gas propulsion systems on the other hand comprise liquid and solid propellants. An exothermal combustion reaction of the propellant is needed to produce high temperature reaction products that are expelled. Hot gases in general require one more step compared to cold gases (except the vaporizing solid), and liquid propellants must be stored in pressurized tanks.

Electric propulsion by-passes the fundamental limitation of chemical propulsion, which is by definition the output velocity from the chemical enthalpy of a propellant. It uses electric or electromagnetic energy to eject matter flow rate at higher velocities. To be more specific, an external electric power accelerates the propellant to produce useful thrust. These higher ejection velocities immediately translate in higher efficiency of the propellant (less propellant needed for the same impulse as a chemical propulsion).

- Electrothermal propulsion systems work with a gas that is heated by passing over an electrically heated surface or through an arc discharge. Then the heated gas is accelerated by gas-dynamic expansion.
- Electromagnetic propulsion systems transform the propellant gas to a neutral plasma. A classic mechanism is to use an arc discharge heating process (similar to electrothermal systems) to reach very high temperatures. Other techniques exist to convert the gas to a neutral plasma (for instance a radio frequency antenna like in the RIT thruster). The resulting plasma is then expelled at high velocity by the interaction of the discharge current with the electromagnetic field. If ions are accelerated either by the Lorentz force or by the effect of an electromagnetic field where the electric field is not in the direction of the acceleration, the device is considered electromagnetic.
- Electrostatic propulsion systems work with a high molecular propellant that can be ionized in different ways (by electron bombardment, in a high frequency electromagnetic field or by extracting ions (or aggregates) from the surface of a liquid metal under the effect of a strong electrostatic field). If the acceleration is caused mainly by the application of a static electric field in the direction of the acceleration (Coulomb force), the device is considered electrostatic. Many electrostatic systems produce ion species (classic ion thrusters and Hall Effect thrusters), necessitating the presence of a cathode to neutralize the plume by releasing electrons near the exit nozzle.

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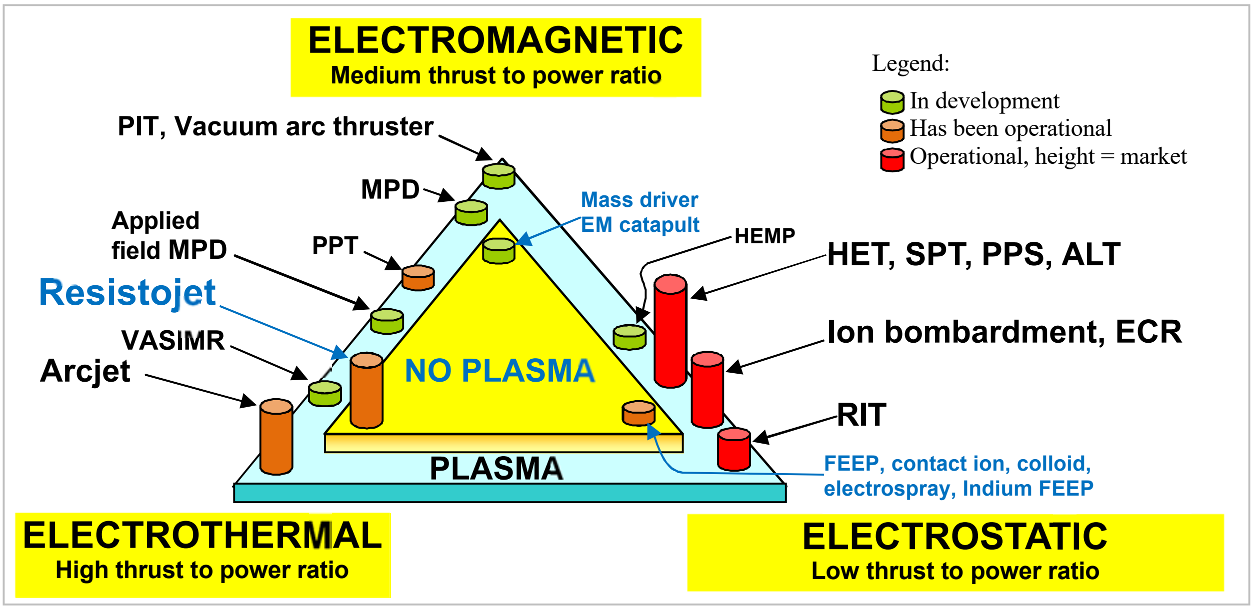

Figure 4: Electric propulsion concepts with their known importance (adapted from Dominique Valentian slide)

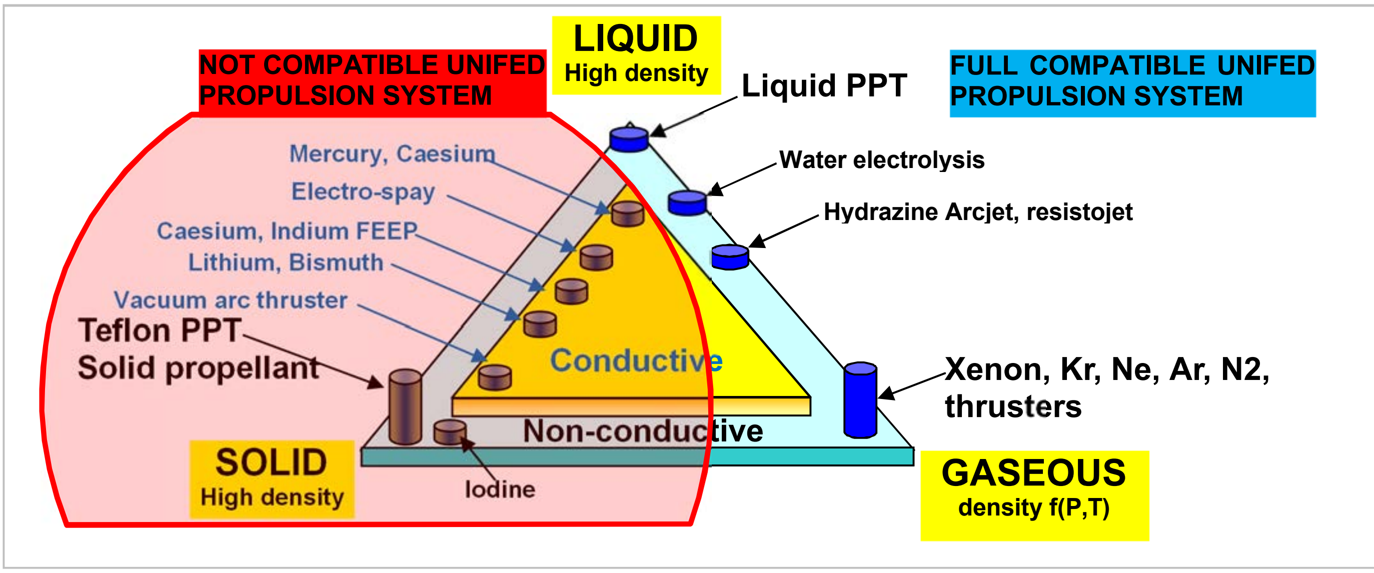

Figure 5: Electric propulsion propellant concepts with their phases, electric conductivity and compatibility with unified propulsion system concept
(a thruster concept not compatible of the unified propulsion system concept is defined as the impossibility for the thrusters to be fed by a common propellant tank)

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