SSC01-XI-6

Specifics of Small Satellite Propulsion: Part 1

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Abstract. Small satellite propulsion is a subject of unique constraints and requirements. Based on University of Surrey experience in small satellite building and operation, these features are listed and explained. Available volume is often identified as the most severe constraint for a small satellite with power and cost being the other two major constraints. Mass is often only of secondary importance for small satellites.

Propulsion dry mass fraction for a spacecraft grows upon the system scaling-down. For small spacecraft propulsion fraction can easily exceed 85%. In such a case, a combination of independent systems for multi-functional propulsion mission scenarios would aggravate the situation. Moreover, specific impulse is not a factor reflecting small satellite propulsion system performance since spacecraft velocity change is also a function of propulsion dry mass fraction.

New conceptual and design solutions are suggested for small satellite propulsion with respect to its specific constraints and requirements. Features of future advanced, low-cost propulsion system for small satellite are described.

Nomenclature

F – thrust, N

$f_{pd} = \dfrac{M_{pd}}{M_{PS}}$ – propulsion dry mass fraction

g – acceleration of gravity, 9.81m/s^2

$I_{sp} = \dfrac{F}{\dot{m} g}$ – specific impulse or thruster-specific impulse, s

I_{ssp} – system-specific impulse, N s/kg

I_{tot} – total impulse, N s

I_{totcg}, I_{totres} – total impulses for cold-gas, resistojet modes respectively, N s

m – propulsion mode (cold-gas, resistojet, monopropellant, bipropellant, etc.)

$m_r = \dfrac{M_{rp}}{M_{PS}}$ – mass ratio

M_f – final vehicle mass, kg

$M_i = M_{rp} + M_p$ – initial vehicle mass, kg

M_{pd} – propulsion dry mass, kg

$M_{PS} = M_{pd} + M_p$ – propulsion system mass, kg

M_{PSm}, M_{PScg}, M_{PSres} – propulsion system masses for different single-modes, kg

M_p – propellant mass, kg

M_{rp} – mass of the rest of the spacecraft (payload, structure, etc.) excluding propulsion, kg

t_b – burn time, s

V_p – propellant volume, m^3

V_{sp} – propellant storage specific volume, m^3/kg

ΔV – vehicle velocity change, m/s

ρ_p – propellant storage density, kg/m^3

Table 1: UoSAT small satellite's classification. (The costs define "affordable access to space")

Constraints	Nano-	Micro-	Mini-
Propulsion Volume, L	<1	7	50
Power (orbit average), W	6	14	180
Cost, ×£1,000,000	0.6	2.0-3.0	5.5
Mass, kg	1-10	10-100	100-500

Introduction

Small spacecraft propulsion is a logical step in modern space exploration technology advancement. The necessity of its development is based on historical premises.

Outstanding advancements in microelectronics achieved since 1960s have radically changed Mankind's lifestyle. It would be difficult to name the sphere of human activity that hasn't been affected. A remarkable progress has been achieved on spacecraft. Miniaturisation of electronic hardware has led to the development of inexpensive small satellite bus (Table 1) weighing only a few kilos. Similar tendency has been observed for satellite payloads and ground station equipment. Spin-off microelectronics – computer software has led to the development of advanced protocols for autonomous satellite operations that significantly reduce the satellite in-orbit operations cost. All of these have been contributing towards the development and exploitation of low-cost small satellites.

On the other hand, the modern launchers restrict the amount of payload delivered to Low Earth Orbit (LEO) to about maximum 100 tonnes and dictate the cost of >$10,000/kg. [1] Often the launcher lifting capacity exceeds the mass of primary payload. Therefore, in the early days, dummy masses were placed within the payload fairing to inject the payload into a desired orbit. Later, when the advancements in microelectronics led to the development of small satellites that have taken the place of dummy masses. Since such "piggyback" rides may be offered "free-off-charge" or at reduced price, small satellites have become cost effective tools for space exploration and allowed "affordable access to space".

While small satellites become more advanced the plans regarding their applications become more ambitious. Currently small satellites are used for remote sensing, communications, and science missions. Future applications will include small satellite constellations, proximity operations, and interplanetary missions. These missions imply access to a wide range of orbits. Meanwhile, this access is determined by available launch opportunities. The limited number of such opportunities restricts the variety of satellite orbits. Furthermore, a "piggyback rider" has to go to the same orbit as a primary payload. Thus, a number of affordable orbits is very limited for a small satellite with no propulsion on board. Therefore, a propulsion system is required for a small spacecraft to develop its capability for more ambitious missions by exploiting the availability of low-cost launches through expanding the variety of accessible orbits.

Because of its size small satellite propulsion system is a subject of specific constraints and requirements that limit its performance.

This paper is devoted to the conceptual design of the advanced, low-cost small satellite propulsion system. In order to design such a system the specifics of small satellite constraints and requirements have to be learned.

Constraints

Launched as a secondary payload, small spacecraft is a subject to unique constraints. As soon as it fits within a margin between total payload lifting capacity of the launcher and primary payload, its mass is of secondary importance for a "piggyback rider" because launch cost for small satellite is usually fixed (independent of spacecraft mass). Typically for heavy launchers (such as, for example, the *Ariane* family of launchers) with lifting capacity of several tonnes, a few extra kilos of auxiliary payload mass margin is only a fraction of percent of primary payload mass. This value is of the same magnitude as uncertainty of primary payload mass. At the same time this mass can comprise a whole spacecraft propulsion system or a small satellite. Unfortunately the similar logic cannot be applied for small satellite volume. This is because the space under the fairing is usually so tight that even the primary payload needs to be optimised to fit in. Hence, volume is often the most severe constraint for small spacecrafts due to the shortage of space available under the fairing. Therefore, small satellites are usually designed to be compact. Tight envelope, in turn, imposes constraints on small spacecraft subsystems such as propulsion and power. Since a propulsion system relies on power generated onboard the spacecraft the last one also becomes another major constraint. Space limited, the existing power systems (typically using Ga/As or Si solar arrays and Ni-Cd batteries) are capable of supplying small satellites with limited power. Deployable solar panels would increase the small satellite power budget as well as its complexity (Sun pointing, deployment mechanisms, etc.) and cost. Constrained by available space and power, small satellite propulsion systems are often limited by cost. This is a major constraint for small satellite propulsion, since it prevents using the latest high-performance technological achievements in the area. With application of modern, high-performance space propulsion technologies, the cost of a small spacecraft can be easily doubled, tripled, etc. For most of the small satellite missions this cost rise is unacceptable since it defeats the purpose of "affordable access to space". Low cost involves many different aspects such as: inexpensive propulsion system components, hardware and propellants; minimum labour; "safety overheads" and service, and limited testing. Expensive "safety overheads" are usually associated with application and handling of toxic, flammable, and explosive propellants. Flight qualification testing is a long and expensive process. Its cost can be easy comparable with the cost of whole small spacecraft or even a number of them. In this case, limited qualification testing is a compromise between spacecraft and its propulsion qualification costs.

Along with the constraints, a small satellite propulsion system is a subject of common and unique requirements.

Requirements

A propulsion system must provide spacecraft with necessary propulsion functions to fulfil its mission. For near-Earth missions, propulsion functions required for a spacecraft are:

- Attitude Control - keeping a spacecraft pointed to the desired direction.
- Orbit Maintenance (station-keeping) - keeping a spacecraft in the desired mission orbit.
- Orbit Manoeuvring - moving a space vehicle to another desired orbit.

Future interplanetary and rendezvous missions require additional propulsion functions:

- Landing to the celestial body surface (for example, landers, rovers, and probes).
- Launch from celestial body surface (for example, sample return mission).

Propulsion systems are expected to deliver high performance, and remain reliable throughout their mission. It should be easy to integrate into a spacecraft, service and maintain. Often a small satellite has already been built and "waits" for suitable launch opportunities, or during its production it is reassigned to another launch. Therefore, it is desirable that small satellite propulsion system be flexible to cope with changes in the mission scenario.

In the case when multiple propulsion functions have to be covered for a small spacecraft, another important issue arises. The propulsion dry mass fraction for a spacecraft grows at scaling-down. For small spacecraft, it can easily exceed 85% for single-mode propulsion (see Table 2 and Figure 1). In such a case, a combination of independent single-mode propulsion systems would aggravate the situation since their dry masses add. Thus, specific impulse is not a factor reflecting small satellite propulsion system performance since spacecraft velocity change is a function of propulsion dry mass fraction as well.

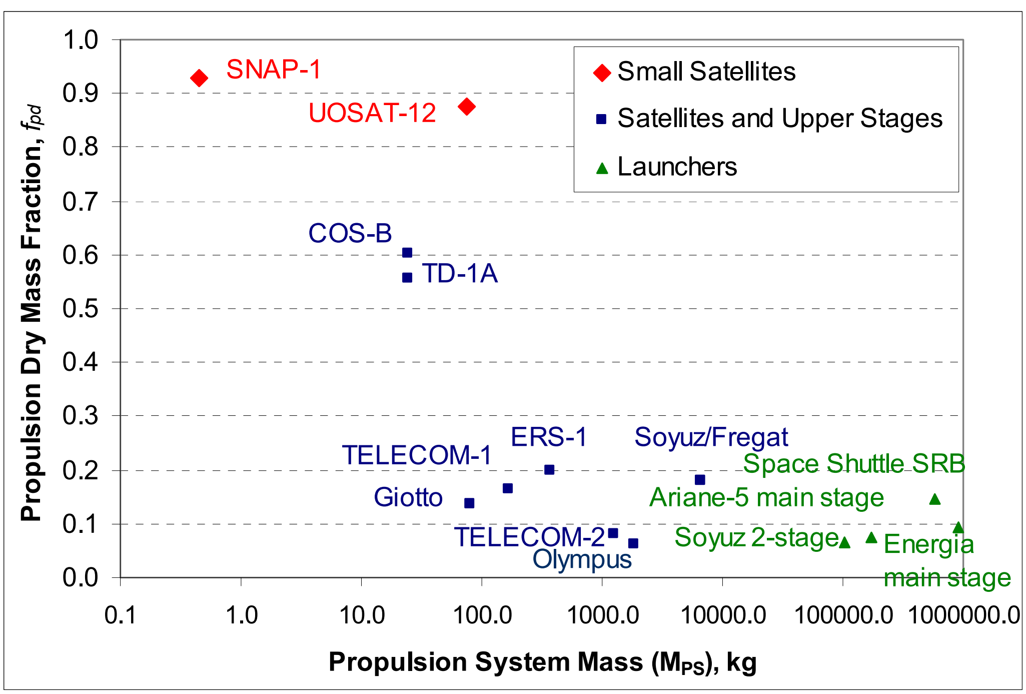


Figure 1: Propulsion system dry mass fraction.

Aware of the constraints and requirements, the system of decisive factors can be established for the advanced small satellite propulsion system. Before this system is determined the feasibility of thruster-specific impulse for small satellite propulsion system performance evaluation has to be reassessed.

I_{sp} vs. I_{ssp}

An *ideal rocket equation* derived in the early days of rocketry by Konstantin Tsyolkovsky is the law of motion for jet propulsion:

$$\Delta V = -I_{sp} g \ln\left(\frac{M_f}{M_i}\right) \qquad \mathbf{(1)}$$

ΔV and I_{sp} are traditionally used for characterisation of space vehicle propulsion. ΔV represents the performance of propulsion system answering how far the vehicle can travel. Thruster-specific impulse (or specific impulse) is a thruster performance factor showing the thruster's perfection in using the propellant for thrust generation. In other words, it shows how efficient (or how economical) the propellant is used. In this respect, specific impulse for space vehicle is akin to mileage for ground motor vehicle. The higher the specific impulse, the more efficient the propulsion technology. Although representing the thruster performance factor, often, specific impulse is misinterpreted for propulsion system performance factor. This misinterpretation assumes constant propulsion dry mass fraction ($f_{pd} = const$), or consequently $\Delta V = f(I_{sp})$. In fact, space vehicle velocity change is a function of multiple variables: $\Delta V = f(I_{sp}, f_{pd}, m_r)$, or

$$\Delta V = -I_{sp} g \ln\left(\frac{m_r + f_{pd}}{m_r + 1}\right) \qquad \mathbf{(2)}$$

Solving this equation for $m_r = const$ total space vehicle velocity change is plotted in Figure 2 as a function of specific impulse and propulsion dry mass fraction.

In the case presented in Figure 2, for the same ΔV the equation 2 has a multiple solution. This solution is a trade between specific impulse performance and propulsion dry mass fraction. In the figure higher specific impulses correspond to higher propulsion dry mass fractions, lower I_{sp} – to lower f_{pd}. There is, however, a drastic decrease in the change of the corresponding propulsion dry mass fraction upon the decreasing ΔV. For example, in the case of ΔV = 200m/s, 3 times reduction in specific impulse corresponds to required 0.59 decrease in f_{pd} while in the case of ΔV = 50m/s, the same reduction in I_{sp} corresponds to only 0.17 decrease in f_{pd}. As it was mentioned earlier the f_{pd} exceeds 0.85 for small space vehicles. The second case, therefore, is more likely to correspond to the small space vehicle than the first. This suggests that for a small space vehicle I_{sp} performance shift is less significant than f_{pd} decrease. In other words, mass optimisation of small vehicle propulsion system design can be at least as effective as thruster's performance improvement.

Application of propulsion hardware or propellant providing lower I_{sp} can be beneficial if it comes along with the reduction in f_{pd}. Elimination of propulsion subsystem(s) is one of the possible design solutions. Significant f_{pd} decrease can be achieved, for example, by application of self-pressurising propellants. In this case propellant extra mass taken as compensation for mass savings due to elimination of expulsion system onboard can not only reimburse for I_{sp} reduction but also provide higher ΔV.

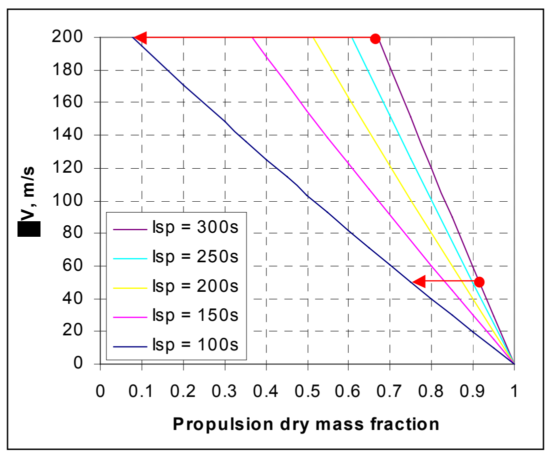


Figure 2: $\Delta V = f(I_{sp}, f_{pd})$ and $m_r = 4$

Table 2: Propulsion systems on small satellites.

Satellite	Class	Mass	Propellant	M_p	ρ_p	M_{PS}	V_{sp}	f_{pd}	I_{ssp}	ΔV
		kg		kg	kg/m^3	kg	m^3/kg	%	Ns/kg	m/s
SNAP-1	Nano	6.5	Butane	0.033	578	0.46	0.000134	93	42.9	3
UoSAT-12	Mini	312	Nitrogen	6.9	230	55.52	0.000617	88	85.3	12
			Nitrous Oxide	2.4	745	19.72	0.000186	88	137.3	9
			N_2/N_2O	9.3	280	75.24	0.000504	88	99.0	21

Note: UoSAT-12 mini-satellite carries two independent experimental propulsion systems onboard: nitrogen cold-gas and nitrous oxide resistojet.

In general, if propulsion dry mass fraction approaches to 1, then:

$$\lim_{f_{pd}\to 1}\left[\ln\left(\frac{m_r + f_{pd}}{m_r + 1}\right)\right] \to 0 \qquad \Rightarrow \qquad \Delta V \to 0$$

(if I_{sp} is finite)

Whence, at the reduced scale size the increase in propulsion dry mass fraction is responsible for the reduction of space vehicle velocity change. For this reason specific impulse cannot be used as a propulsion system performance factor for small space vehicles. The above analysis also implies that there is an optimum value of specific impulse for small space vehicles.

"System-specific impulse" is recommended as a propulsion system performance factor. System-specific impulse is defined as: [2,3]

$$I_{ssp} = \frac{I_{tot}}{M_{PS}} \quad \left[\frac{Ns}{kg}\right] \tag{3}$$

For a system with constant thrust magnitude,

$$I_{tot} = F\, t_b = I_{sp}\, M_p\, g \tag{4}$$

Combining equations 2, 3, and 4, the relationship between system-specific impulse and ΔV can be established as following:

$$\Delta V = -\frac{I_{ssp}}{(1 - f_{pd})}\ln\left(\frac{m_r + f_{pd}}{m_r + 1}\right) \tag{5}$$

Because system-specific impulse considers not only the specific impulse of the propulsion technology but also the mass of propellant onboard and propulsion system mass, it is capable to describe the effect of propulsion dry mass fraction for small spacecraft and effect of power system mass increase for an electric propulsion system.

In order to define the feasibility range for system-specific impulse equation 3 can be modified to:

$$I_{ssp} = I_{sp} g \frac{M_p}{M_{pd} + M_p} = I_{sp} g \left(1 - f_{pd}\right) \tag{6}$$

Substitution of $M_p = \rho_p V_p$ for propellant mass and simultaneous division of numerator and denominator by M_{pd} yields:

$$I_{ssp} = I_{sp} g \frac{\rho_p V_{sp}}{1 + \rho_p V_{sp}} \tag{7}$$

where $V_{sp} = \dfrac{V_p}{M_{pd}} \quad \left[\dfrac{m^3}{kg}\right]$ is propellant storage specific volume indicating volume of propellant per propulsion dry mass.

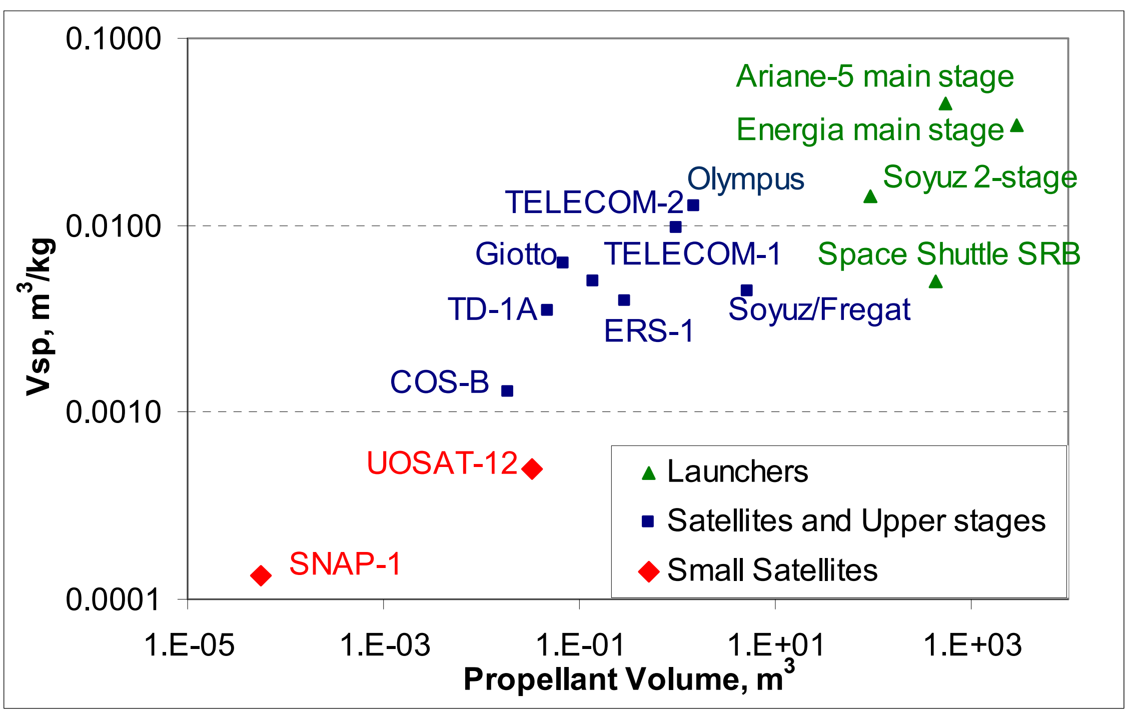
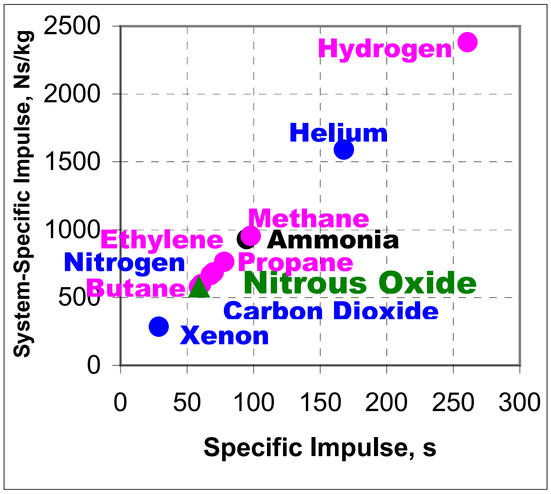
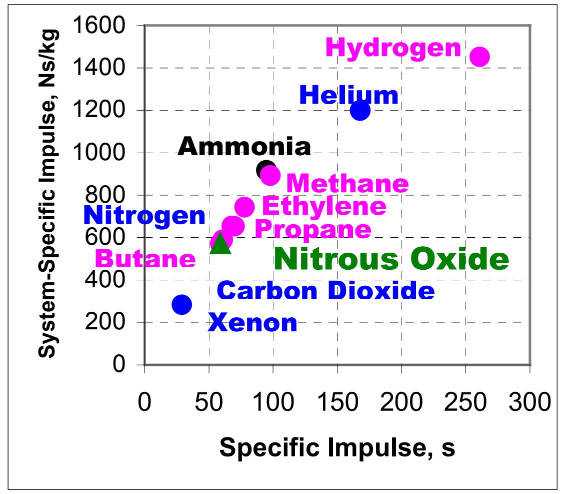


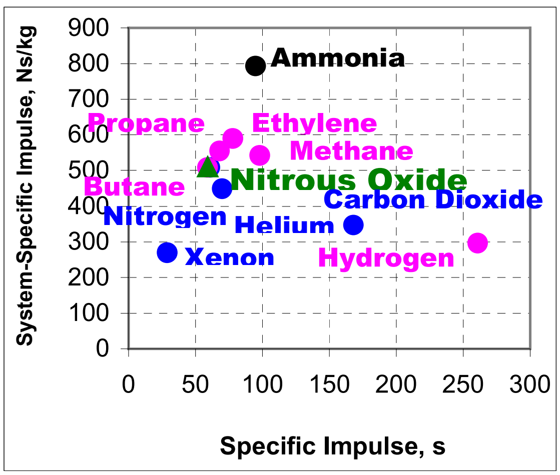
Figure 3: Propellant storage specific volume versus size of space vehicle propulsion system.



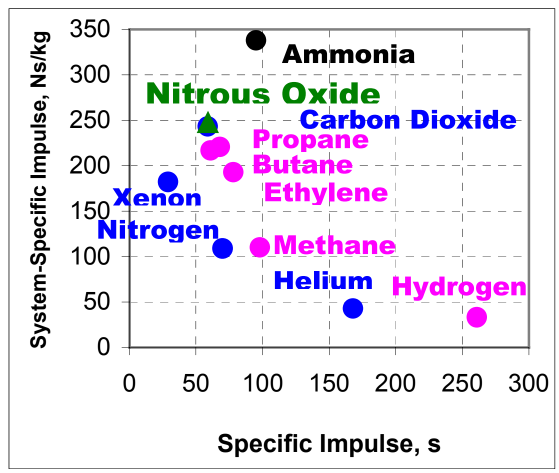
a) V_{sp} = 1 (1000-litre per kg)



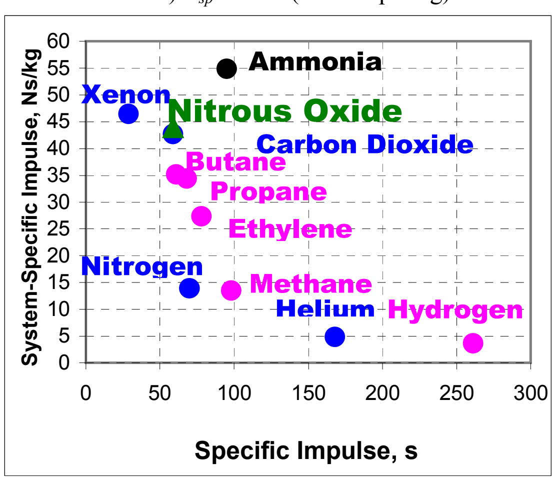
b) V_{sp} = 0.1 (100-litre per kg)



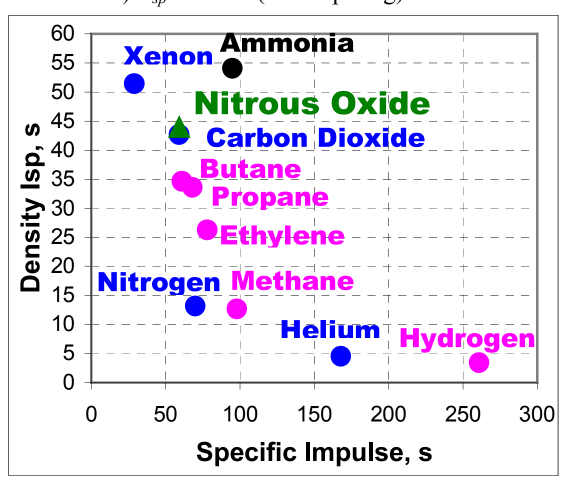
c) V_{sp} = 0.01 (10-litre per kg)



d) V_{sp} = 0.001 (1-litre per kg)



e) V_{sp} = 0.00011 (110-cm^3 per kg)



f) Specific Gravity

Figure 4: Theoretical performance comparison of cold-gas propellants. (nozzle expansion ratio = 200)

The higher V_{sp}, the less dry propulsion mass is associated with storing the propellant amount. High V_{sp} is typically associated with big space vehicles, low with small. This tendency is illustrated in Figure 3. In the figure small satellites have the smallest propellant storage specific volumes since their propulsion dry mass fraction is high. For bigger satellites and upper stages propellant storage specific volumes are higher. Launchers (main stages) represent the case of the most efficient use of dry mass fraction for propellant housing, except for strap-on boosters that usually accommodate recovery system. Magnitude of propellant storage specific volume for small satellites value has been estimated as $0.001 < V_{sp} < 0.0001$.

Another expression for propellant storage specific volume can be derived by combination of equations 6 and 7.

$$V_{sp} = \frac{1 - f_{pd}}{f_{pd}\rho_p} \quad \textbf{(8)}$$

This expression reveals the connection among propellant storage specific volume, propulsion dry mass fraction, and propellant density. Higher propellant densities support smaller propellant storage specific volume. This is why the application of denser propellant on small satellites is beneficial.

An example in Figure 4 demonstrates the advantage of system-specific over thruster-specific impulse and density Isp. While application of specific impulse and density Isp for comparison of propulsion technologies gives "frozen" picture, the use of system-specific impulse reflects the dynamics of system performance variation as a function of propellant amount onboard.

Figure 4a illustrates correlation between system-specific impulse and specific impulse. In this idealised case (i.e. big spacecraft with a large amount of propellant in the tanks) propulsion performance is defined by thermodynamic properties of the propellant so that the propulsion performance can be judged by specific impulse. In reality, however, this number for propellant storage specific volume is practically unattainable, especially for non-liquefied gases, due to insufficient strength of the known construction materials.

With the reduction in tank size the thermodynamic advantage starts to fade (see Figure 4b). Upon the further decrease in tank size Figure 4c, the application of denser propellant becomes beneficial. Liquefied ammonia takes the first place followed by hydrocarbon gases: ethylene, propane, methane, and then: nitrous oxide, butane, and carbon dioxide, helium, hydrogen, xenon. However, thermodynamic properties contribution still remains. At the further reduction in tank size, the advantage of denser propellants is pronounced (Figure 4d). Liquefied ammonia, nitrous oxide, carbon dioxide, propane, butane, ethylene and xenon have definitely superior performance over non-liquefied: nitrogen, methane, helium, and hydrogen. Much further reduction in tank size shows certain domination of dense propellants application for small space vehicles. In Figure 4e ammonia is closely followed by xenon, and then: nitrous oxide and carbon dioxide. This result is very similar to the one that can be obtained by application of density I_{sp} (Figure 4f).

Material strength analysis suggests that $V_{sp} > 0.005$ m^3/kg are hardly feasible for non-liquefied gases while their application for $V_{sp} \leq 0.001$ m^3/kg (see Figure 4d) is unfavourable due to low propulsion system performance. Therefore, the recommended propellant storage specific volume range for non-liquefied cold-gas propulsion system application would be $0.001 \leq V_{sp} < 0.005$ m^3/kg. While COS-B and TD-1A satellites using non-liquefied cold-gas propulsion are within this range the small satellites are out (see Figure 3). For this reason the application of non-liquefied cold-gas propulsion is inefficient for the small satellites.

Multi-Mode Propulsion System

For flexible small satellite missions, multi-mode propulsion systems are essential. Multi-mode propulsion systems are designed to offer a range of thrust and total spacecraft velocity change options to meet specific mission objectives, e.g. orbit insertion, station-keeping, and attitude control.

System-specific impulse for a multi-mode propulsion system is a sum of total impulses by each mode over propulsion system mass:

$$I_{ssp} = \frac{\sum_{1}^{m} I_{tot}}{M_{PS}} \quad \textbf{(9)}$$

A combination of a number of independent single-mode propulsion systems may, however, satisfy the specific mission requirements as well as a multi-mode system. In this case total propulsion system mass is a sum of the independent single-mode propulsion system masses:

$$M_{PS} = \sum_{1}^{m} M_{PSm} \quad \textbf{(10)}$$

The example of a combination of two independent single-mode (nitrogen cold-gas and nitrous oxide electrothermal) propulsion systems flown on *UoSAT-12* mini-satellite is given in Table 2. For this case:

$$I_{ssp} = \frac{I_{totcg} + I_{totres}}{M_{PScg} + M_{PSres}} \quad \textbf{(11)}$$

Since multi-mode system assumes that the propellant(s) and/or hardware is shared, it may be designed having lower mass than a number of independent single-mode propulsion systems:

$$M_{PS}(\text{multi}-\text{mode}) \le \sum_{1}^{m} M_{PSm} \qquad \textbf{(12)}$$

Whence, system-specific impulse of multi-mode propulsion system may be higher than that of a number of independent single-mode propulsion systems:

$$I_{ssp}(\text{multi}-\text{mode}) \ge I_{ssp}(\sum_{1}^{m} M_{PSm}) \qquad \textbf{(13)}$$

This is why application of multi-mode propulsion system is desirable, especially on small satellites where propulsion dry mass fraction is high.

Selection Criteria

After critical revision of the above discussed requirements and constraints, a list of selection criteria for small space vehicle propulsion system is recommended as follows:

- Space vehicle velocity change requirements
- System-specific impulse
- Propulsion envelope volume requirements
- Power requirements
- Cost
- Propellant storability, non-toxicity, non-flammability, non-explosiveness, compatibility, and availability
- Restartability in orbit (if required)
- Multifunctionality (if required)

Discussion

Having the list of criteria and being aware of the requirements and constraints, features of advanced, low-cost propulsion system design for small satellite can be envisioned.

Since the majority of upcoming future missions imply deployment of small satellites in Earth orbit below 800-km their propulsion system must provide thrust sufficient for atmospheric drag compensation at low power consumption. Present technologies suitable for such missions are cold-gas, electrothermal, and chemical propulsion.

Once the technologies have been identified, the choice of appropriate propellant(s) is of primary importance for propulsion system design. It is implied by propulsion requirements stated earlier, the propellants in use onboard the spacecraft must support all necessary propulsion functions. Since propellants require separate storage and feed systems, it is desirable to have as few of them onboard the spacecraft as possible for system design simplification and propulsion system mass reduction. In this case, a single propellant onboard serving all spacecraft propulsion functions would be ideal. At the same time, the propellant's thermodynamic properties must make high specific impulse performance achievable. Propellant(s) must be storable onboard spacecraft. As indicated earlier, application of dense propellants onboard a small satellite is advantageous for compact propulsion system. Tensile stresses in a small size pressure vessel wall are much smaller than the strength of its construction material. This is why a thin-wall, small pressure vessel is able to withstand high pressure. Therefore, until pressure-induced tensile stresses become the same order of magnitude as material strength, a small satellite propellant tank mass is usually "not sensitive" to pressure rise. In this situation, application of self-pressurising propellants onboard is superior over use of propellant expulsion system because it decreases the complexity and mass of a propulsion system. Application of lower strength and lighter construction materials can also be considered in the design of small size pressure vessel.

Application of non-toxic, non-flammable, non-explosive, and compatible propellant leads to inexpensive design, and further, to low overall cost of a propulsion system. Propellant itself must be inexpensive.

As emphasised above, the propellant(s) selection is critical for advanced small satellite propulsion system design. Furthermore, the requirements for a small satellite propellant (multi-functional, high performance, storable, dense, self-pressurising, non-toxic, non-flammable, non-explosive, compatible, and inexpensive) are quite demanding.

Consideration of properties for a number of physical substances led towards the liquefied gases.

Table 3: Properties of liquefied gases.

Name	Chemical Formula	Storage Conditions		Toxicity	Flammability	Remarks
		Density	Vapour Pressure			
		kg/m^3	bar			
Ammonia	NH_3	609	8.9	T	N	highly reactive
Butane	C_4H_{10}	578	2.2	N	F	non-corrosive
Nitrous Oxide	N_2O	745	52.4	N	N	supports combustion

Notes: Storage conditions are taken at 21°C. T – toxic; F – flammable; N – non-toxic or non-flammable.

Application of liquefied gases (such as, for example, ammonia, butane, or nitrous oxide in Table 3) in cold-gas propulsion (I_{sp} = 95, 60 and 59s respectively) may increase the total spacecraft velocity change up to 3 times that of nitrogen (for the same propellant volume). Ammonia may also be used in resistojet (I_{sp} = 423s) and arcjet (I_{sp} ~ 1000s) thrusters. [4,5] Butane resistojet is expected to deliver I_{sp} ~ 180s. Nitrous oxide can be used in resistojet (I_{sp} = 115s), monopropellant (I_{sp} ~ 200s), hybrid and bipropellant (I_{sp} ~ 320s) thrusters. [6-11]

The application of liquefied gases, however, associated with a few difficulties. Countermeasures against the liquid sloshing inside the propellant tank must be undertaken. Additional heat is required to compensate for the phase change (latent heat or heat of vaporisation). Self-pressurising systems could support only limited propellant flows so that the design must ensure the operation within this limit.

Once the propellant(s) are selected, propulsion design can be drawn. In general, it should be modular and simple for easy service and integration into a small spacecraft. Inexpensive materials must be applied in the design.

Conclusion

Overall, because of its unique constraints and requirements, small satellite propulsion is a challenging area.

Due to the growing contribution of propulsion dry mass fraction, specific impulse cannot be used as a propulsion system performance factor for small satellites. For this reason the classical form of the ideal rocket equation ($\Delta V = f(I_{sp},M_i,M_f)$) was changed for the one using I_{ssp}, m_r, and f_{pd} variables. The new form of ideal rocket equation is convenient for propulsion system design-performance trades. The "system-specific impulse" (I_{ssp}) is suggested as a propulsion system performance factor. It is more accurate determination of the spacecraft propulsion system performance than the commonly used "thruster-specific impulse" (I_{sp}). Propellant storage specific volume (V_{sp}) is introduced to reflect the propulsion system perfection in storing propellant(s).

List of selection criteria for small space vehicle propulsion system is recommended.

The importance of propellant selection for advanced small satellite propulsion was emphasised.

The application of liquefied gases as propellants for use on small satellites is recommended as advantageous.

Acknowledgment

The authors thank David Gibbon and Paul Charman (Surrey Space Centre, UK) for their help.

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