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Heated Gas Propulsion System Conceptual Design for the SAMSON Nano-Satellite

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Abstract

This paper presents a new cold gas concept of a heated gas propulsion system for the 6U SAMSON nano-satellite. In this type of system, the entire propellant tank is heated to an operational temperature prior to system operation and let to cool down immediately afterwards. The current analysis shows that it is possible to meet mission requirements by using 310 gr of CO_2 contained in a capsule-shaped propellant tank. The study shows that the required thrust of 80 mN can be obtained by choosing a nozzle geometry with aspect ratio of 400 and nozzle diameter of 0.25 mm. The resulting specific impulse of such a configuration is approximately 67 sec. It also analyzes the benefits of operating the propulsion system at various operational temperatures between 40°C and 80°C. The analysis shows that low operational temperatures lead to a relatively lightweight propulsion system, short operation readiness durations, and low attainable ΔV per operation. On the other hand, high operational temperatures lead to a heavier system, longer operation readiness duration, and high attainable ΔV per operation.

1. Introduction

The Space Autonomous Mission for Swarming and geOlocation with Nano-satellites (SAMSON) project is an endeavor to test new algorithms for cluster keeping and geolocation using a three nano-satellite cluster in Low Earth Orbit (LEO). The project, led by the Asher Space Research Institute (ASRI) at the Technion, involves the design, development, integration, and mission execution of all three nano-satellites and algorithms (Gurfil, P. et al. 2012). Each satellite will be a 6U CubeSat and will have a target nominal launch mass of less than 8 kg. The satellites are planned to be launched in 2017, with mission duration of at least 12 months.

To perform cluster keeping and orbit maneuvers, each of the three nano-satellites requires some means

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of propulsion. Therefore, it was decided to incorporate a propulsion module in each satellite. The propulsion system design, development, manufacturing, and qualification were assigned to Rafael for its expertise and experience in space propulsion systems. Due to each satellite's low mass (<10 kg) and small size (6U), its propulsion system needs to be compact and lightweight. At the same time, to maintain the three satellites in accurate formation and orbit, sufficient maneuverability is required ($\Delta V = 20$ m s^{-1}). The formulated propulsion system specification requirements are listed in Table 1.

To satisfy the propulsion system requirements and guidelines, an adequate choice of propulsion system type must be made. The propulsion system types considered were: cold gas, monopropellant-based, bi-propellant-based, and electric propulsion suitable for nano-satellites. The different propulsion system types were compared, and their advantages and disadvantages weighed one against the other. A fully detailed description of the propulsion system evaluation campaign is described in Lev et al. (2014). The evaluation campaign includes the basic design of various types of propulsion systems tailored specifically to meet the requirements of the SAMSON nano-satellite. Following the evaluation, it was decided that the propulsion system in each satellite should be based on cold gas propulsion. Such a propulsion system consists of a propellant tank, pressure regulator, valves, pressure and temperature transducers, and a thruster. The thruster is essentially a diverging nozzle through which the propellant flows and accelerates.

Cold gas propulsion systems have been, and still are, used in nano-satellites, micro-satellites, and even on some of the newest communication satellites, thanks to their simplicity, high reliability, and low cost (Mauthe et al., 2000; Gibbon et al., 2000; Smith et al., 2004; Adler et al., 2005). Traditionally, cold gas systems are divided into two main groups: pressurized gas systems and liquefied gas systems (Figure 1). Pressurized gas systems operate with gaseous propellants. A sufficient amount of gas is pressurized inside the propellant tank to meet the required ΔV. From the propellant tank, the gas is let out through a series of valves and pressure regulators that bring it to the required operating pressure. Consequently, the gas is let through a nozzle and out to space. In the liquefied system, the propellant is stored in liquid phase while the pressure is the vapor pressure at the storage temperature. Propellant is let out from the tank into a plenum, where it is vaporized through expansion (Arestie et al., 2012) or external heating (Assad, 2012). Once the propellant is in gaseous phase, it is let through the nozzle and out to space. Propellant vaporization and the usage of the plenum make the liquefied gas systems more complex than the pressurized gas systems. However, liquefied systems include lower storage volume and pressure, which enables the use of lighter and smaller components.

This paper presents a third, non-conventional, cold gas propulsion system concept, followed by a description of the propellant selection process. Next, the propulsion system design properties and general system

Table 1. SAMSON satellite propulsion system specification requirements

Satellite / Mission Property	Constraint / Guiding Line	Remarks
ΔV	20 m s^{-1}	Required maneuverability per satellite for a one year mission
m_d	8 kg	Delivered mass
m_{ps}	< 2 kg	Propulsion system mass (including propellant)
Volume	< 100 mm × 100 mm × 210 mm	
Propellant Type	Non-Toxic, Non-Flammable, Non-Explosive	Student project constraint
Electrical Power	< 10 W	For all operational phases
Propellant Tank Material	Titanium	Designer preference due to experience at Rafael
Thrust	80 mN (20 mN per thruster)	For the entire propulsion system (4 thrusters/nozzles in total)
Temperature Envelope	10°C – 40°C	
Propellant Tank Pressure	As low as possible	Designer preference

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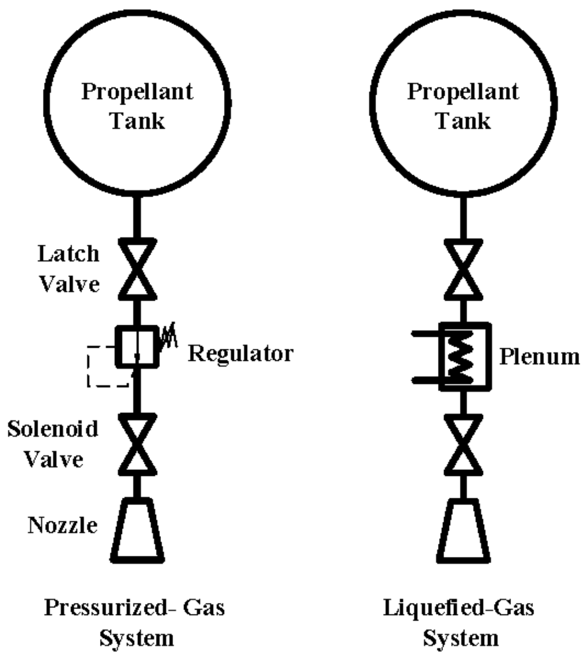

Figure 1. Basic schematics of the two common variations of cold gas propulsion systems. Left: Pressurized gas system; Right: Liquefied gas system.

architecture are presented. Last, the system operational point selection trade-offs are analyzed and conclusions are drawn for an optimized mission system operation scheme.

2. The Heated Gas Propulsion System

2.1. Incentive

To meet the constraints and preferences specified in Table 1, two conceptual designs of traditional cold gas propulsion systems were performed, the first one based on pressurized gas, and the second, based on liquefied gas technology. Both propulsion systems were found compatible with the mission constraints. The details of the designs of the two propulsion systems are not elaborated upon thoroughly in this article, although a brief overview is presented.

For the defined constraints, a solution to a pressurized gas propulsion system can be found only for heavy gas propellants, such as xenon and krypton, because of the high pressure formed in the propellant tank when using lighter gases, such as nitrogen. Although xenon was found to be suitable, due to its low critical temperature of 16°C, it might liquefy during operation; therefore heating is required for such a propulsion system. In addition, xenon is relatively expensive, and might affect the cost of a low-budget student project. To avoid both the problem of designing additional pipeline heating, and high prices of xenon, the xenon-based pressurized propulsion system concept was dropped. Finally, the only solution found when using krypton was for tank pressure of about 200 bar (Lev et al., 2014). This high pressure value was found to be somewhat problematic for a student project, with the possibility of being rejected by the launch provider.

Liquefied gas propulsion systems required overcoming the problem of spitting out liquid propellant that does not vaporize completely, thus resulting in an inferior specific impulse and thrust for these cases (Gibbon et al., 2001). Specifically, it requires knowledge of the amount of propellant filling the intermediate chamber, referred to as the plenum, between the propellant tank and the nozzle. Unless the propellant mass in the plenum is well known, a reasonable estimate of the generated impulse per maneuver is impossible to obtain. For this reason, a knowledge gap was identified with regard to how to control and secure the amount of propellant mass flowing into the plenum. Moreover, the thrust of such propulsion systems has a non-constant, descending profile as a function of time. This is due to the gas emptying process that leads to a quick pressure drop in the plenum. Such an event, in the case of a mission such as SAMSON, might inflict uncontrollable asymmetrical moments on the satellite due to unbalanced thrust on all thrusters, and even worse, it might degrade the performance of the cluster control algorithms.

2.2. Concept

In light of the above, a different propulsion system was sought. To meet the above requirements, a novel cold gas-based propulsion system concept was developed (Figure 2). First, propellant is stored in the propellant tank in liquid form. Each time before operating the propulsion system, the entire propellant tank is heated until the propellant is fully vaporized. The system is then operated by opening a latch valve and letting the gaseous propellant pass through a pressure

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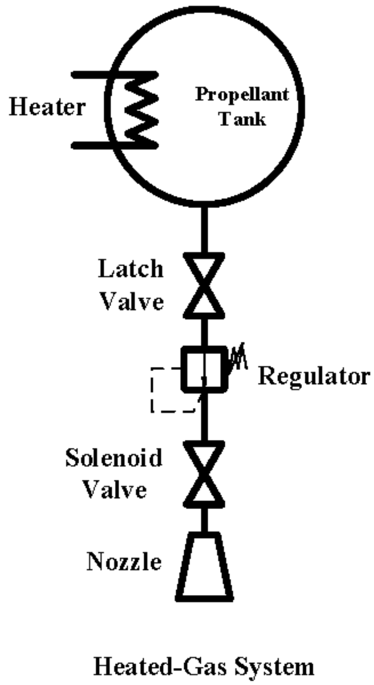

Figure 2. Basic schematic of the heated gas propulsion system.

regulator, to reduce the pressure to several bar. Finally, the thrusters are fired by opening the individual solenoid valve of each thruster.

The use of the heated gas propulsion system holds a few key advantages over the two previously described conventional systems:

a) Most gases, near their critical temperature, have a compressibility factor below one ($Z \leq 1$). This fact can aid in reducing tank pressure, when the propellant is fully vaporized.
b) The propulsion system has to be heated only before operation, which leaves the propellant in liquid form during most stages of the mission. When the propellant is liquid, it is self-pressurized (that is, the pressure in the tank is the vapor pressure that corresponds to the cold tank temperature). This is especially significant during launch, when the satellite and its subsystems are most vulnerable due to the harsh environmental conditions. The lower the pressure, the lower the chance for possible faults.
c) Since the entire propellant is heated and in gaseous form during operation, thruster firing can be steady and continuous for up to hundreds of seconds. The required firing time can be controlled by controlling the propellant temperature prior to thruster firing.

At the same time, the heated gas propulsion system suffers from several shortcomings relative to the conventional pressurized gas and liquefied gas propulsion systems:

a) Heating of the entire propellant tank and propellant prior to each thruster operation may lead to power loss through the various structural elements in the satellite, thereby reducing the power efficiency of such propulsion system. This will require a meticulous thermal design process that will make sure the propulsion system is well thermally insulated.
b) Heating of the entire propellant tank and propellant prior to each thruster operation reduces firing readiness level. Depending on the heated gas operation temperatures and heating power, it may takes tens or hundreds of minutes to preheat the system before operation is permitted. This may pose a problem for some formation flight missions that usually include frequent small impulse maneuvers.
c) A heated gas system configuration requires constant temperature monitoring, along with repeated on-off cycles of the propellant tank heater. This makes the heated gas system more complex and susceptible to technical faults relative to conventional cold gas propulsion systems.

3. Propellant Selection Process

After the basic architecture of the heated liquid propulsion system has been determined there is a need to identify a suitable propellant for the SAMSON satellites. The optimal propellant would be mostly liquid

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with low vapor pressure when not heated, that is within the satellite temperature envelope. This is to keep the propellant tank at lowest possible pressure during most mission phases when the propulsion system is inactive, thus saving power. On the other hand, the propellant should have a critical temperature as close as possible to the high end of the satellite's temperature envelope. This is so that when the propulsion system is operational, its temperature stays as close as possible to the temperature of the satellite, and to restrain any parasitic heating from heating the rest of the satellite. In other words, excessive heating should be avoided by choosing a suitable propellant.

Last, as listed in the requirements in Table 1, the propellant should be non-toxic and non-flammable, for safety reasons. The propellants considered are listed in Table 2.

It can be deduced from the table that except for CO_2, N_2O, Xe, and SF_6, all propellants fail to satisfy the safety constraints. It can also be seen in the table that SF6 has a critical temperature that is too high, and will require excessive system heating. Last, xenon can be excluded due to its high price, as already specified, and its relatively low critical temperature. Therefore, it can be concluded that the two most suitable propellants are carbon dioxide (CO_2) and nitrous oxide (N_2O), both with an atomic mass of 44 gr mol^{-1} and similar thermodynamic properties. Of the two propellants, carbon dioxide was chosen, due to its slightly lower critical temperature of 31°C.

It is important to note that the critical temperature of the selected gas is within the specified temperature envelope (10°C–40°C). This means that the propellant may be supercritical during storage and even during launch. However, it will be shown in this paper that since the propulsion system's operational point is higher than 40°C, the propellant tank's design is able to withstand all storage and launch pressure values. Last, high gas leakage rates may occur with supercritical gas in storage, which could pose limitations on storage duration.

4. Nozzle Design

The thruster nozzle is the component responsible for converting the propellant internal energy into directed kinetic energy. The goal of the nozzle design phase is to determine nozzle geometry and expected mass flow rate. The former is essential to achieve the required propulsion system thrust while taking into consideration material production limitations and past experience. The latter is required to compute the remaining propellant cool-down time, therefore determining the maximum allowed continuous propulsion system operation duration before propellant liquefaction occurs.

The nozzle is separated from the propellant tank by a pressure regulator that maintains constant inlet pressure independent of propellant tank pressure and remaining gas mass. As such, the nozzle design phase

Table 2. List of All Considered Propellants for the Heated Propulsions System of the SAMSON Satellites

Name	Mol. Form	T_c [°C]	P_c [bar]	Atomic Number	Comments
Carbon Dioxide	CO_2	90.98	74.73	44.01	
Nitrous Oxide	N_2O	36.45	73.41	44.01	
Xenon	Xe	16.58	58.42	131.3	Expensive
Sulfur Hexafluoride	SF_6	45.57	37.55	146	
Acetylene	C_2H_2	35.25	63.23	26.04	Flammable, Unstable
Chlorotrifluoromethane	$CClF_3$	28.75	39.42	104.5	Toxic
1,1-Difluoroethene	$C_2H_2F_2$	29.75	45.39	64.03	Flammable, Toxic
Ethane	C_2H_6	32.21	49.45	30.07	Flammable
Fluoromethane	CH_3F	44.27	59.58	34.03	Very Flammable
Hexafluoroethane	C_2F_6	19.75	30.70	138	Toxic
Tetrafluoroethene	C_2F_4	33.85	39.92	100	Toxic, Explosive
Trifluoroacetonitrile	C_2F_3N	37.95	36.58	95	Toxic
Trifluoromethane	CHF_3	25.85	48.84	70	Toxic
Tetrafluorohydrazine	N_2F_4	35.85	38	104	Toxic, Explosive

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was carried out independently from the propellant tank design phase while assuming known nozzle inlet pressure. The value of the inlet nozzle pressure is determined by the choice of pressure regulator, which is an off-the-shelf component.

As a guideline, the nozzle expansion ratio (A_e/A_t) should be as high as possible, to achieve high exhaust velocity. Consequently, the expansion ratio was set at 400, where the throat and exit diameters are d_t = 0.25 mm and d_e = 5 mm respectively. The values were chosen to satisfy manufacturing constraints, and are similar to values reported in previous work (Nicolini et al., 2004; Zakirov et al., 2001). Given the nozzle expansion ratio, the nozzle exit to nozzle inlet pressure ratio (p_e/p_c) was computed and found to be 13,700, using Eq. 1 under the valid assumption of an isentropic nozzle (Turner, 2006):

$$\frac{A_e}{A_t} = \left\{ \frac{\left(\frac{\gamma-1}{2}\right)\left(\frac{2}{\gamma+1}\right)^{\left(\frac{\gamma+1}{\gamma-1}\right)}}{\left(\frac{P_e}{P_c}\right)^{\frac{2}{\gamma}}\left[1-\left(\frac{P_e}{P_c}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right]} \right\}^{\frac{1}{2}}, \qquad (1)$$

where γ is the specific gas ratio, which is 1.28 in the case of carbon dioxide.

The nozzle inlet pressure, p_c, on one hand must be within the chosen pressure regulator specifications (Moog, 2014), and on the other hand, must be such that the thrust generated by one nozzle is 20 mN. To properly determine pc, the relation for thrust, is used. In Eq. 2, F is the thrust and C_F is the thrust coefficient calculated to be 1.917. Figure 3 presents the value of

$$F = C_F A_t P_c = \left\{ \left\{ \left(\frac{2\gamma^2}{\gamma-1}\right)\left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{(\gamma-1)}} \left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{(\gamma-1)}{\gamma}}\right]\right\}^{\frac{1}{2}} + \frac{P_e A_e}{P_c A_t} \right\} A_t P_c, \qquad (2)$$

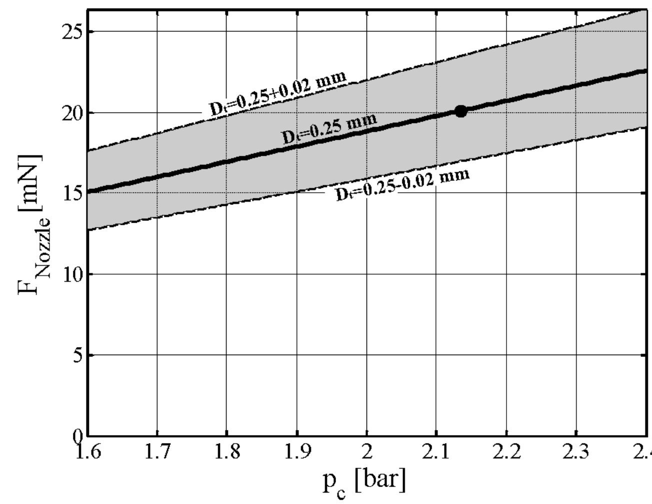

Figure 3. Thrust generated by one nozzle against nozzle inlet pressure for throat geometrical inaccuracies (dt=0.25±0.02 mm). The nominal operation point is denoted by a black circle.

F as a function of p_c. It can be deduced from the figure that to achieve the required thrust of 20 mN, the nozzle inlet pressure has to be set at 2.135 bar. In addition, it can be observed from the figure that, for a known throat diameter, nozzle inlet pressure variations of 10% result in acceptable thrust variations of no more than 10%. Similarly, reasonable manufacturing inaccuracies of 0.02 mm in throat diameter result in thrust deviation of about 15%. Of course, geometrical inaccuracies can be detected in lab prior to launch, and the pressure regulator reset to readjust the nozzle inlet pressure, thereby obtaining the required thrust.

Once the nozzle geometry is determined and nozzle inlet pressure is set, the propulsion system mass flow rate can be computed, using Eq. 3:

$$\dot{m} = A_t P_c \left\{ \gamma \left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{(\gamma-1)}} \frac{M_w}{R_0 T_c} \right\}^{\frac{1}{2}}, \qquad (3)$$

with the result $\dot{m}$=0.03 gr sec^{-1} per nozzle and $\dot{m}_{tot}$=0.12 gr sec^{-1} for the entire propulsion system. T_c is the nozzle inlet gas temperature, M_W is the atomic mass, and R_0 is the universal gas constant (8.314 J mol^{-1} K^{-1}). Here, it was assumed that all four thrusters/nozzles are simultaneously operational, which cor-

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responds to the greatest propellant mass flow rate leaving the propellant tank. The rate at which propellant leaves the propellant tank is crucial in heated gas system operation, as it dictates the temperature drop rate of the remaining propellant, and hence, the maximum allowed continuous system operation duration.

All values calculated in this chapter are typical for low thrust cold gas systems with similar thrust requirements (Smith et al., 2004; Nicolini et al., 2004; Zakirov et al., 2001).

5. System Architecture

The above-described system architecture was realized using commercial CAD software by choosing off-the-shelf components that can operate at the required temperature envelope and expected pressure range. The expected pressure range was estimated to be in the range of the critical pressure of carbon dioxide. The chosen propulsion system components were valves, pressure regulators, thruster nozzles, and piping. All components were allocated to allow maximum remaining space for the propellant tank.

The remaining space allows for the allocation of a capsule-shaped propellant tank, as shown in Figure 4. The outer dimensions of the propellant tank are diameter of 74 mm and overall length of 207.5 mm. These parameters pose as hard design constraints, due to the limited available propulsion system volume. As such, the rest of the system characteristics are derived from these values. The required propellant mass, however, shall be derived purely from propellant and mission considerations (specific impulse, ΔV, and spacecraft mass). The tank thickness, and thus its inner diameter, is determined by structural considerations and the ability of the propellant tank to hold the heated gas pressure. It is calculated during the operational temperature analysis and presented hereafter.

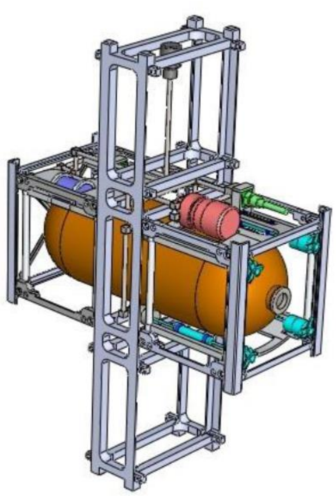

Figure 4. CAD illustration of the heated gas propulsion system with its components allocated within the 2U designated volume. Only the relevant part of the satellite's chassis is presented.

To calculate the inner tank diameter one needs to know the tank pressure, which depends on the propellant mass and temperature. The required propellant mass can be derived from the rocket equation shown in Eq. 4:

$$m_p = \left[\frac{\Delta V}{e^{Isp_{min} \cdot g_0}} - 1\right] m_d (1 + 0.25), \qquad (4)$$

where m_p is the required propellant mass, m_d is the delivered mass of 8 kg, ΔV is the required velocity increment of 20 m s^{-1}, and g_0 is the gravitational constant of 9.81 m s^{-2}. The 25% propellant mass addition is for contingency, to confront possible lack of propellant issues, and is a system requirement. Isp_{min} is the minimal specific impulse the propulsion system can produce. Isp_{min} corresponds to the lowest temperature within the temperature envelope, which is 10°C, to account for the worst case scenario in which the propellant has the minimal temperature allowed.

The minimal specific impulse can be calculated from Eq. 5 (Turner, 2006):

$$\mathrm{Isp} = \frac{1}{g_0} \sqrt{\frac{2\gamma}{\gamma - 1}} \, \mathrm{x} \, \frac{R_0 T_c}{Mw} \left[1 - \left(\frac{P_e}{P_c}\right)^{(\gamma - 1/\gamma)}\right]^{\frac{1}{2}}, \quad (5)$$

where the pressure ratio, p_e/p_c was computed during the nozzle design phase. The computed minimal specific impulse is Isp_{min}=67 sec. For comparison, the value of the ideal minimal specific impulse is 71 sec, and is achieved when perfect nozzle expansion is obtained when the nozzle exit area to throat area asymptotes infinity ($A_e/A_t \rightarrow \infty$).

Using the calculated value of the minimal specific impulse, the required propellant mass is 310 gr. The

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propellant tank thickness must be such that it holds the propellant tank internal pressure at the highest operational point, which is the highest tank temperature. Therefore, to adequately select the operational point, the ability to store the required propellant mass at different temperatures should be examined.

6. Operational Point Selection

The operational temperature, also denoted here as operational point, is the temperature the propellant tank and propellant are brought to prior to operating the propulsion system by releasing the propellant through the nozzle. The choice of operational point affects the following system characteristics and performance: (1) propellant tank pressure and wall thickness (thus, its mass); (2) propellant tank heating duration (thus, system operation readiness); and (3) propellant cool-down time before liquefaction occurs during its release out of the nozzle (thus, maximum allowed continuous operation duration).

The impact of the operational point on each one of these parameters is analyzed hereafter. This analysis explores values of operational temperature between 40°C, allowing a 9°C margin from the critical temperature, and 80°C, which is considered the highest allowed temperature before the propellant tank starts affecting the rest of the satellite.

6.1. Propellant Tank Pressure and Wall Thickness

It was shown in previous sections that the propellant tank should contain 310 gr of CO_2 and has outer dimensions of 74 mm in diameter and an overall length of 207.5 mm. Propellant tank thickness should be sufficiently large to hold the internal pressure, yet small enough so to allow sufficient volume for the propellant. To find the minimal sufficient tank thickness, a simple computer algorithm was constructed, shown in Figure 5.

The algorithm computes the minimal sufficient propellant tank wall thickness as follows. First, a zero millimeter wall thickness is assumed. From tank outer dimensions constraint and assumed wall thickness, the inner volume is computed. Given the previously calculated propellant mass with the available volume, and

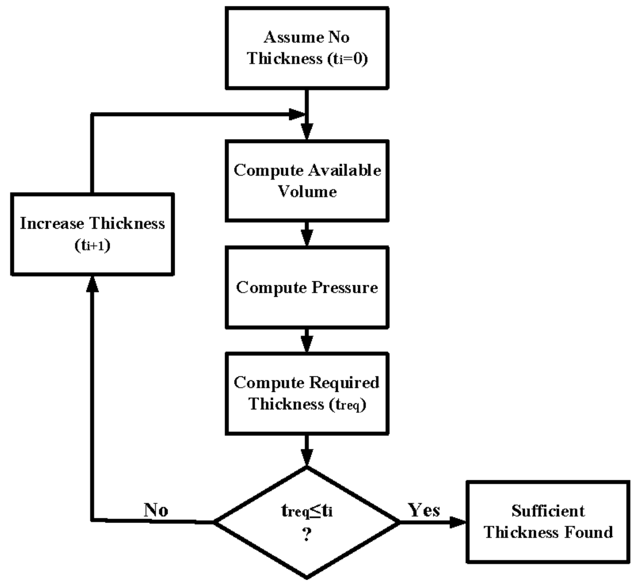

Figure 5. The computer algorithm constructed to compute the minimal sufficient propellant tank thickness for each operational point.

therefore propellant density, the pressure can be deduced (Span, R. W. et al., 1996). This is done by considering the operational temperature and using the non-ideal gas law, $p_{tank}=Z\rho_p R_0 T_p M_w^{-1}$, where p_{tank} is propellant tank pressure, Z is the compressibility factor, ρ_p is the propellant density and T_p is propellant temperature. Once the tank pressure is computed, the minimum required propellant tank wall thickness can be computed using

$$t_{h=}SFx\left(\frac{p_{tank}{}^{r}}{2\sigma_y}\right), \quad (6)$$

where t_h is the thickness of the hemispherical sections of the propellant tank, SF is the safety factor set at the value of 4, r is the propellant tank radius and σ_y is the tank material (Ti-6Al-4V) yield strength, which is 827 MPa. Since in capsule-shaped pressure containers the hemispherical and cylindrical sections experience different stresses, the cylindrical section wall thickness should be double that of the hemispherical sections ($t_{Cyl}=2t_h$) (Roark, Raymond J. et al., 2002).

The propellant tank pressure values and propulsion system total mass, along with propellant tank cylindrical section wall thickness, are plotted against the operational point in Figures 6 and 7, respectively. It can be observed from the figures that an increased operational

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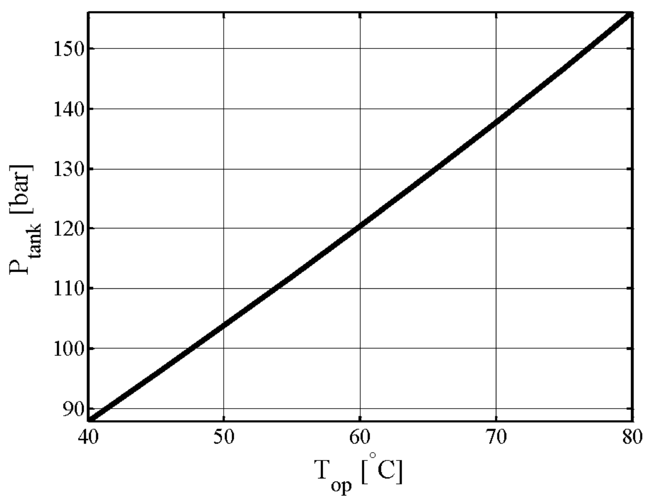

Figure 6. Propellant tank pressure against operational point temperature.

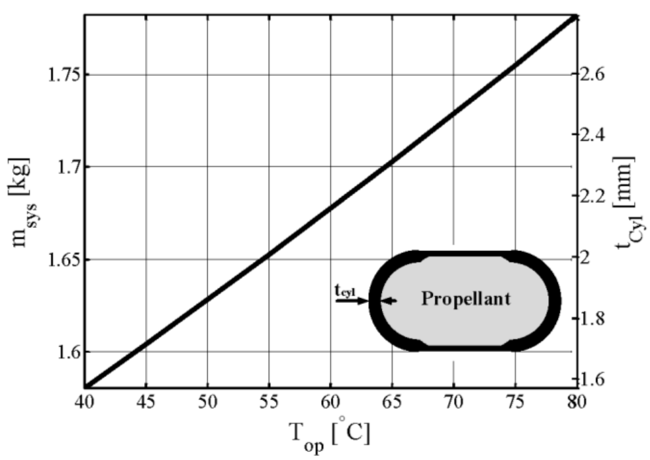

Figure 7. Propulsion system mass and propellant tank cylindrical section wall thickness against operational point temperature. (Total system mass was computed assuming that all dry system components, not including the tank, weigh 1 kg.) Representative illustration of the propellant tank structure (not to scale) is shown on the bottom right corner.

temperature leads to higher tank pressure, and requires a thicker, thus heavier, propellant tank. At the highest considered operational temperature (80°C), tank pressure is above 150 bar, which is an acceptable figure for conventional spacecraft propulsion systems. At the lowest operational temperature (40°C), which is the highest end of the initial system temperature envelope, the propellant tank pressure is below 90 bar. It is important to note that during launch, this would be the highest possible tank pressure. At the same time since CubeSats are usually launched in a “piggy back” fashion, along with heavier satellites, launch providers might raise the demand for reduced propellant tank pressure to protect the primary payload. This issue, however, should be cleared out with the specific launch provider.

In addition, one can see that the total propulsion system mass is little affected by the choice of the operational point, and varies between 1.58 kg and 1.78 kg. Here, it was assumed that the propulsion system dry mass, not including the propellant tank, is 1 kg. This assumption is based on typical values of the required propulsion system components and experience in designing space propulsion systems (Daniel, Brack, et al., 2013). The values are presented in Table 3. In any case, the overall propulsion system mass is estimated to be lower than the maximum allowed value of 2 kg.

Table 3. Typical Propellant Mass Components of the Propulsion System

Component	Typical Mass
Pressure Regulator	500 gr
Latch Valve	50 gr
Fill Valve	100 gr
Thruster (Solenoid Valve & Nozzle)	4×50 gr
Brackets, Harness and Piping	100 gr
Pressure Transducers	30 gr
Thermostat	12 gr
Heaters	10 gr

Last, it is interesting to note how carbon dioxide’s low compressibility factor, near its critical temperature, comes into effect in the suggested heated gas propulsion system. Thanks to the low compressibility factor, it is possible to insert over 300 gr of CO_2 into a capsule-shaped propellant tank 207.5 mm long and 74 mm in diameter, which can fit in 2U volume at a relatively modest pressure.

6.2. Propellant Tank Heating Duration

Prior to operating the propulsion system, the propellant needs to be heated up to the operational point, a process that takes time. For this reason, the propulsion system operation readiness depends on the choice of the operational temperature and heating power. To calculate the propellant and propellant tank heating duration of the worst case scenario with a fully loaded

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propellant tank before first operation, the following equation may be is used:

$$t_{heating} = \frac{\left(c_{v,T_{op}} T_{op} - c_{v,T_0} T_0\right) \times m_p + C_{m,tank}\left(T_{op} T_0\right) \times m_{tank}}{P_{heat}}, \tag{7}$$

where $t_{heating}$ is the required heating duration, $C_{v,Top}$ and $C_{v,T0}$ are the propellant specific heat capacities at the operational and initial temperatures, respectively, $C_{m,tank}$ is the propellant tank specific heat capacity, T_0 is the initial propellant tank temperature before heating, m_{tank} is the propellant tank mass, and P_{heat} is the net input heating power. In the above mathematical relation, P_{heat} represents the net heating power that is invested in heating the propellant and propellant tank. Note that due to conductive and radiative thermal losses, higher power levels should be invested to obtain the net heating value. This makes heated gas-based propulsion system design more challenging than cold gas based systems, as the required stringent thermal design may limit the spacecraft design process. Since thermal loss mechanisms are dictated by meticulous satellite thermal design, which takes place in more advanced stages of a project, the present analysis considers only the net power that heats the propellant tank and propellant.

The required heating duration for three different heating power levels, 2 W, 5 W, and 10 W, is presented in Figure 8. It can be observed in the figure that, as expected, the highest the operational temperature correlates to the longest the heating duration. In the same manner, lower heating power requires a longer heating duration. When 10 W is used to heat the propellant system, operation readiness is in the range of 50–100 min. However, generating as much as 10 W might be quite demanding for a 6U CubeSat, considering electrical demand from other onboard components; therefore, heating power might be limited to lower levels. If 5 W or less is invested in propellant heating, heating duration is higher than 100 min, and might even be as high as 450 min (7.5 hours) at the highest investigated operational temperature (80°C),

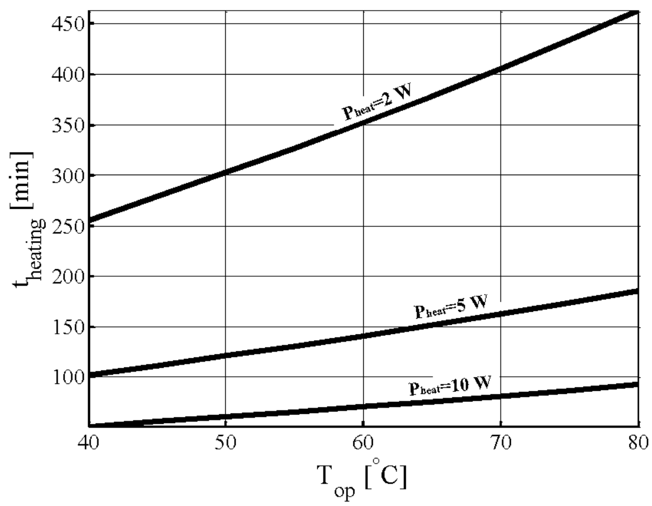

Figure 8. Propellant heating duration, from an initial temperature of 10°C, as a function of the operational point.

which may be unacceptable for various types of missions.

In summary, system operation readiness is greatly affected by the choice of operational point and available power. From the readiness point of view, it is preferred to select the lowest allowed operational temperature.

6.3. Propellant Cool-Down during System Operation

During operation, the propellant is ejected through the nozzle and out to space. The release of propellant out of the propellant tank is also responsible for the removal of energy in the form of internal energy carried by the released gas. Since the remainder of the propellant is maintained inside a constant volume, its temperature decreases. In addition, as the propulsion system is used multiple times, propellant mass decreases throughout mission lifetime; therefore, each system operation begins with lower propellant mass than the former operation. This leads to more rapid propellant temperature decrease from each operation to the next. The propellant cooling effect may lead to undesired liquefaction that should be avoided by operating the propulsion system for sufficiently short time durations, or at sufficiently high operational temperatures.

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To quantify the maximum allowed operation time durations before propellant liquefaction occurs, for different operational temperatures, one needs to determine the remaining propellant temperature, inside the propellant tank, as a function of time during system operation. This was done by solving Eqs. 8–11:

$$v(t) = \frac{V_{tank}}{m_0 - \int_{t_0}^{t} \dot{m}dt}, \tag{8}$$

$$u(t) = u_0 - c_v(t)T(t)\frac{\int_{t_0}^{t} \dot{m}dt}{m_0 - \int_{t_0}^{t} \dot{m}dt}, \tag{9}$$

$$c_v(t) = \frac{\partial u(t)}{\partial u(t)}, \tag{10}$$

$$T(t) = f\big(u(t), v(t)\big), \tag{11}$$

where v is the specific volume, V_{tank} is the propellant tank inner volume, m_0 is the initial propellant mass, t is the time which is also the independent variable of the equations, u is the propellant internal energy, c_v is the propellant specific heat, and T is the propellant temperature. The subscript X_0 refers to the point in time at which the propellant release process begins. Eq. 8 describes the change of the propellant specific volume with time as propellant is released from the propellant tank and the remaining propellant mass decreases. Eq. 9 describes the change of specific energy with time, due to the fact that the released gas removes energy from the remaining propellant. Eq. 10 shows the definition of specific heat capacity at constant volume, and is a property of the thermodynamic state of the propellant. Eq.11 describes the method of obtaining the propellant temperature. The function *f* (u(t),v(t)) represents the extraction of the propellant temperature from the relevant CO_2 thermodynamic lookup table database (NIST, 2016). Since the gas release process is non-adiabatic, the thermodynamic state of the propellant changes with time. Nevertheless, the thermodynamic state may be defined by any two specific thermodynamic quantities. To calculate the temperature, this analysis used the specific volume (v) and specific internal energy (u).

Since Eqs. 8–11 are coupled and require the use of lookup tables, they were solved numerically. The temperature was computed as a function of time for the explored operational temperatures of 40°C to 80°C. In addition, propellant temperature was calculated for various propellant mass fractions, of the propellant mass (m_p) at the beginning of system operation. Each calculation was stopped when liquefaction temperature was reached. For all cases, it was assumed that propellant tank heating is turned off during propulsion system operation. In addition, according to propulsion system requirements during firing, all four nozzles are fired; therefore, maximum propellant mass flow rate is released from the satellite and out to open space.

Propellant temperature as a function of time for two operational temperatures of 40°C and 80°C, and for four cases of initial propellant mass fraction (ξ_m) of 1, 0.75, 0.5 and 0.25, is presented in Figure 9. It can be seen in the figure that the propellant temperature decrease depends on the initial mass fraction; the fuller

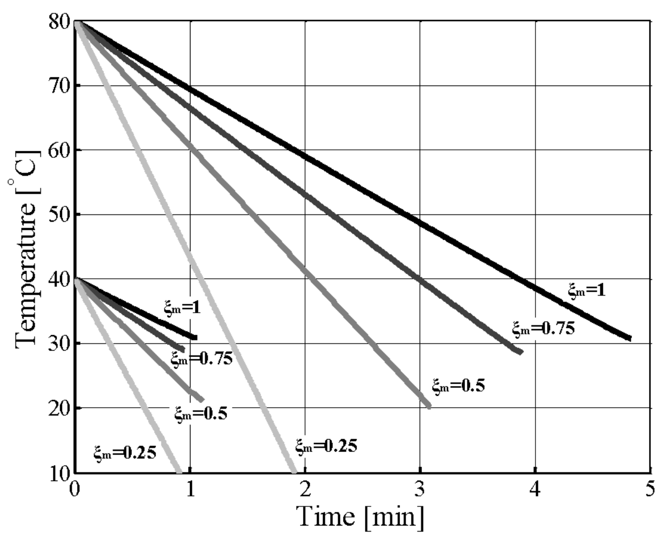

Figure 9. Propellant temperature decreases as a function of time during system operation for operational temperatures of 40°C and 80°C, and for four different initial mass fraction values (ξ_m).

the propellant tank at the beginning of each operation, the slower the temperature decreases. For all cases, liquefaction is reached in less than five minutes, and heavily depends on both the choice of operational temperature and on the initial mass fraction. This fact limits the maximum attainable impulse of each system operation. In addition, the decrease of operational dura-

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tion with initial propellant mass fraction leads to operation constraints throughout the satellite lifetime. At the mission's beginning of life, the maximum attainable impulse may be significantly larger than the impulse when little gas remains in the propellant tank near the end of the mission. It can also be observed in the figure that an operational temperature of 80°C enables propulsion system operation duration more than twice as long as with an operational temperature of 40°C. The choice of the operational temperature affects the maximum attainable impulse.

To explore the possible limitations on the propulsion system's impulse at each operation sequence, the ΔV per operation was computed for different operational temperatures and initial mass fractions. The results are presented in Figure 10. It can be seen that at mission beginning of life, the attainable ΔV depends

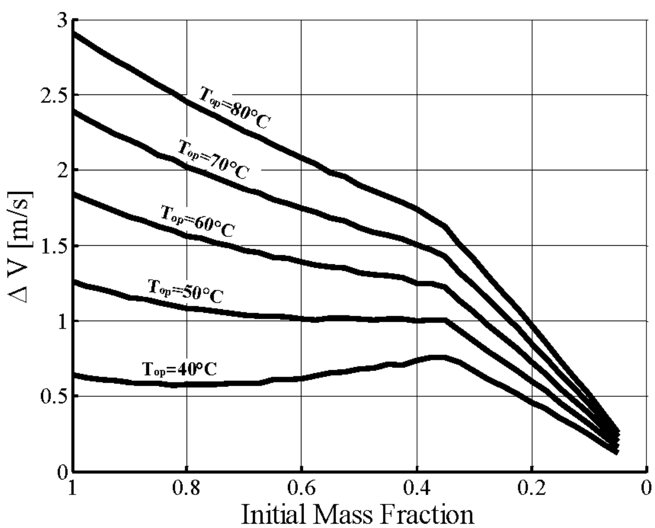

Figure 10. Attainable ΔV as a function of initial propellant mass fraction for a variety of operational temperatures.

on the operational temperature. At 80°C, ΔV of almost 3 m s^{-1} can be achieved, while at 40°C, this figure drops to about 0.6 m s^{-1}. Moving along the X-axis is analogous to operating the system multiple times throughout the mission lifetime and slowly emptying the propellant tank. The more times the propulsion system is used, the initial mass fraction is lower. For most values of operational temperature, the maximum attainable ΔV decreases as the propellant tank loses more and more gas. Additionally, as the mass fraction decreases, at low operational temperatures, a shift in the curves' slope is seen. The reason for this behavior is the change in the liquefaction temperature. At high initial mass fractions, CO_2 is either in a supercritical or saturated state; therefore, liquefaction occurs when the critical temperature of 31°C is reached. As less gas fills the propellant tank and the initial mass fraction decreases, CO_2 can be in a subcritical and fully gaseous state; therefore, the liquefaction temperature decreases.

The decreasing-increasing behavior depends on the propellant tank emptying process as it progresses, relative to the saturation slope of CO_2. This is illustrated in Figure 11, where propulsion system operation for five different initial mass fraction values, is graphed on the p-v diagram of CO_2.

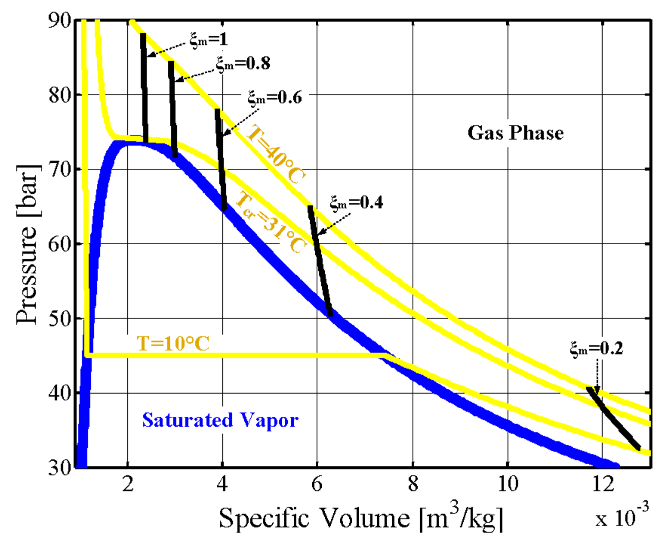

Figure 11. A p-v diagram representing propulsion system operation for five different initial propellant mass fractions for propulsion system operational temperature of 40°C.

Finally, at mass fractions of about ξm=0.35, the liquefaction temperature drops below 10°C, which was chosen as the minimal allowed temperature for the propulsion system, and thus, operation is halted. This is the point at which the T=10°C isotherm intersects with CO_2's saturation curve. Any further operation at lower initial mass fractions is truncated by the T=10°C isotherm. This is best illustrated by the ξm=0.2 operation curve in Figure 11.

The implementation of liquefaction-prevention control may be conducted by monitoring the propellant pressure and temperature inside the propellant

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tank during system operation. Both the propellant pressure and temperature are used to define the thermodynamic state of the gas and determine whether it is saturated or purely in gaseous state.

Throughout the mission lifetime, the propulsion system may operate several dozen times until all propellant mass is depleted. An example of a lifetime firing cycle is presented in Figure 12, where each point represents the beginning of one firing sequence until

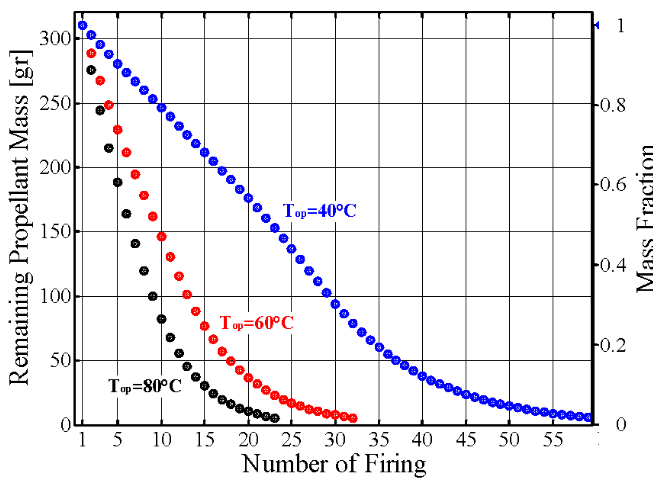

Figure 12. Propellant mass depletion as a function of propulsion system firing number for minimum and maximum operational temperatures explored.

the system is halted before liquefaction occurs. The firing represented by the subsequence point starts with the remainder of the propellant. It can be seen from the figure that at the beginning of lifetime, at an operational temperature of 80°C, about 35 gr of propellant is released, generating almost 3 m s^{-1} as already discussed, while at an operational temperature of 40°C, the amount of propellant released is about 7.6 gr, generating about 0.6 m s^{-1}. This fact leads to almost 60 firing cycles required to perform the mission if the operational temperature is 40°C, whereas fewer than 25 firing cycles are required if an operational temperature of 80°C is implemented.

In summary, propulsion system operation duration depends on the choice of operational temperature: the higher the operational temperature, the greater the attainable impulse and fewer firing cycles required to perform the mission. It is also important to note that even if sufficiently high operational temperature is chosen the attainable impulse decreases throughout mission lifetime as the propulsion system is operated time and again.

7. Conclusions

This paper has presented and analyzed a new concept of a heated gas propulsion system for the 6U SAMSON nano-satellite. It showed that it is possible to meet mission requirements by using 310 gr of CO_2 contained in a capsule-shaped propellant tank. It further showed that the required thrust of 80 mN can be obtained by choosing nozzle geometry with aspect ratio of 400 and nozzle diameter (d_t) of 0.25 mm. It also showed that the combination of nozzle geometry and the type of propellant lead to specific impulse of about 67 sec.

Following the preliminary calculations and determining basic system architecture, this article discussed the importance of the choice of the operational temperature. It showed the effect of operational temperature on three different propulsion system attributes or performance characteristics: (1) propellant tank pressure and system mass; (2) operation readiness; and (3) maximum allowed continuous operation duration. It showed that high operational temperature of up to 80°C increases propellant tank pressure above 150 bar, extends operation readiness duration, and allows for higher ΔV in each operation, up to nearly 3 m s^{-1}. On the other hand, at low operational temperature of 40°C, close to carbon dioxide's critical temperature (31°C), propellant tank pressure drops below 90 bar, and system readiness duration decreases, yet the maximum attainable ΔV per operation is below 0.75 m s^{-1}.

This analysis concludes that high operational temperature is suitable for missions in which long maneuvers are required and long readiness duration allowed. At the same time, low operational temperature is more suitable for missions where satellite agility is required (i.e., a mission in which minimal maneuverability is allowed, yet short readiness time is required). It should be noted that system operation readiness duration is more sensitive to propellant net heating power than to operational temperature; therefore, it is recommended to provide higher heating power in cases where operation readiness duration is of the essence.

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In conclusion, the heated gas system serves as a viable propulsion solution for CubeSats, as it is affordable, compact, and versatile. This type of system may accomplish a variety of missions, while meeting stringent requirements.

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References

Adler, S. et al. (2005): Low-Cost Cold-Gas Reaction Control System for the Sloshsat FLEVO Small Satellite, *J. Spacecraft and Rockets,* Vol. 42 (2), pp. 345–351.

Arestie, S. et al. (2012): Development of a Modular, Cold Gas Propulsion System for Small Satellite Applications. *J of Small Satellites,* Vol. 1 (2), pp. 63–74.

Assad, A. (2012): Remote Sensing, in *Advanced Techniques and Platforms,* Rijeka, Croatia: InTech.

Brack, D. et al. (2013): SAMSON Student Project Report, Bachelor Degree Final Project, Technion – Israeli Institute of Technology.

Gibbon, D. et al. (2000): The Design, Development and In-orbit Performance of a Propulsion System for the SNAP-1 Nanosatellite, presented at the 3rd Intl. Conf. on Spacecraft Propulsion, Cannes, France, October 10–13, Paper ESA SP·465.

Gibbon, D. et al (2001): Low Cost Butane Propulsion Systems for Small Spacecraft, presented at the 15th AIAA/USU Conf. on Small Satellites, Logan, UT, US, Paper SSC01-XI-1.

Gurfil, P. et al. (2012): The SAMSON Project – Cluster Flight and Geolocation with Three Autonomous Nano-satellites, presented at the 26th AIAA/USU Conf. on Small Satellites, Logan, UT, US, Paper SSC12-VII-2.

Lev, D. et al. (2014): Cold Gas Propulsion System Conceptual Design for the SAMSON Nano-Satellite, presented at the 50th AIAA Joint Propulsion Conf., Cleveland, OH, July 28–30, Paper AIAA-2014-3759.

Mauthe, S. et al. (2000): The Design and Test of a Compact Propulsion System for CanX Nanosatellite Formation Flying, presented at the 19th AIAA/USU Conf. on Small Satellites, Logan, UT, US, Paper SSC05-VI-5.

Moog (2014): Single Stage Pressure Regulator (Xenon) Data Sheet. Available at: http://www.moog.com/products/propulsion-controls/spacecraft/components/regulators/single-stage-regulator-gas-/.

Nicolini, D. et al. (2004): Thrust Characterization of the Cryostat Cold-Gas Propulsion System, presented at the 40th AIAA Joint Propulsion Conf., Fort Lauderdale, FL, July 11–14, Paper AIAA-2004-4150.

NIST (2016): US National Institute of Standards and Technology (NIST). *Thermophysical Properties of Fluid Systems.* U.S. Secretary of Commerce on behalf of the United States of America. Available at: http://webbook.nist.gov/chemistry/fluid/.

Roark, Raymond J. et al. (2002): *Roark's Formulas for Stress and Strain.* New York: McGraw-Hill.

Smith, P. et al. (2004): Cryosat Cold Gas System and Component Development, presented at the 40th AIAA Joint Propulsion Conf., Fort Lauderdale, FL, July 11–14, Paper AIAA-2004-3859.

Span, R. W. et al. (1996): A New Equation of State for Carbon Dioxide Covering the Fluid Region from the Triple Point Temperature to 1100K at Pressures up to 800MPa, *J. Phys. Chem. Ref. Data,* Vol. 25, pp. 1509–1596.

Turner, M. et al. (2006): *Rocket and Spacecraft Propulsion,* 2nd ed. Chichester, UK: Springer.

Zakirov, V. et al. (2001): Specifics of Small Satellite Propulsion: Part 2, presented at the 37th AIAA Joint Propulsion Conf., Salt Lake City, UT, US, July 8–11, Paper AIAA-2001-3834.

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