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Carbon Dioxide Based Heated Gas Propulsion System for Nano-Satellites

Dan Lev and Jacob Herscovitz
Technion – Israel Institute of Technology
Haifa, Israel; +972-52-429-1745
Dan.R.Lev@gmail.com

ABSTRACT

The use of carbon dioxide as propellant makes small satellite maneuverability safe, affordable and attractive to use for academic space missions. We present a preliminary conceptual design of a propulsion system based on carbon dioxide propellant. The design was tailored for the SAMSON 6U nano-satellite constellation. A careful analysis of the gas properties was made, which provided the essential working points and architecture of the propulsion system. Subsequently, a dedicated conceptual design was performed to comply with the propellant working points and basic satellite requirements. The main components, such as the propellant storage tank and thruster nozzle, were defined and designed. Overall, the system mass is 1,777 gr of which the propellant is 310 gr. The system can generate thrust of 80 mN and ΔV of 20 m/s. Finally, we present an operational analysis of the system, defining the operational constraints and performance. A full mission simulation was run, utilizing the propulsion system characteristics while satisfying mission requirements. The final design fully complies with the mass, volume and performance requirements.

INTRODUCTION

One of the major caveats of propulsion systems for small satellites is the lack of available volume [1]. As most small satellites are compact and space-limited, their propulsion systems must as be compact. This includes the propellant that should be stored in the vicinity of the thrusters and take the minimum possible space without degrading system performance.

In addition, the propellant should be such that minimum auxiliary components or material should be used for the propulsion system – plenum tank, pressure regulators, heaters, valves, propellant tank material, piping etc.

The simplest type of propulsion system, which is usually the lightest in mass and most volume-effective, is the cold gas propulsion system [2]. Cold gas propulsion systems harness the internal energy of gas by releasing it through a diverging nozzle out to space [2]. The name “cold gas” implies that no chemical process, weather chemical reaction or catalytic decomposition, is used to increase the internal, thus kinetic, energy of the ejected gas. Additionally, since no chemical reaction takes place most gases may be used in cold gas propulsion systems, subject to system requirements [3].

To date many types of propellants are used in cold gas propulsion systems: Nitrogen (N_2) [4], Sulfur Tetrafluoride (SF_6) [5], Butane (C_4H_{10}) [1], Argon (Ar), [6] and even Krypton (Kr) [7,8]. Each propellant is chosen according to the advantages and constraints it brings to the propulsion system. For example, butane may be stored at low pressure in a saturated state; therefore requiring a light-weight propellant tank. However, butane is a flammable substance in ambient air environment and should be handled with care. Also, butane needs to be heated up to gas phase before it is ejected out of the spacecraft; therefore requiring a heating chamber and the investment of the corresponding heating power.

Carbon Dioxide (CO_2) may serve as propellant in propulsion systems with severe volume limitation thanks to some of its attributes [8,9] (Table 1). Figure 1 shows carbon dioxide’s p-v diagram. Carbon dioxide’s critical temperature is about 31°C, property that allows for an effective compression of the propellant within the temperature envelope of most small satellites, especially during launch when the satellite is under vibrational mechanical stresses. Three gas densities of interest are also marked in the figure. For example, propellant tank sized as 1U (volume of 1 liter) may store up to 500 grams of carbon dioxide in saturated vapor state.

Table 1. Basic properties of carbon dioxide (CO2).

Property	Value
Mol. Form	CO_2
Atomic Number	44.01
Critical Temperature (T_{cr})	30.98°C
Critical Pressure (P_{cr})	74.73 bar

In addition, carbon dioxide is an available substance, cheap and inert (nonflammable, non-explosive and nontoxic). These attributes make carbon dioxide an attractive candidate for academic institutes and can be handled by non-qualified personnel, such as University students.

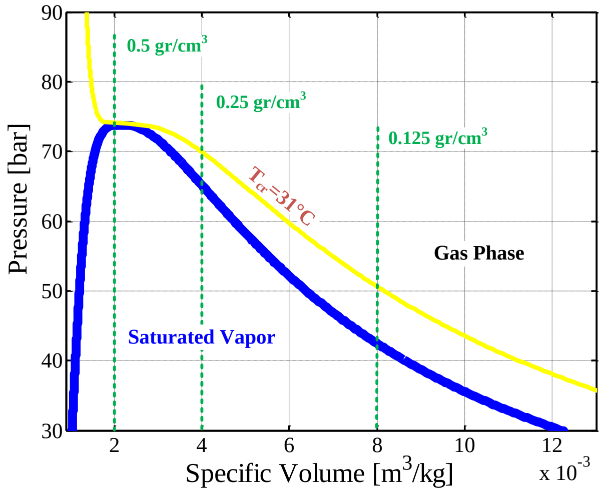

Figure 1. A p-v diagram of carbon dioxide (CO2). Gas densities of interest are marked with dotted green lines. The critical temperature is marked with a solid yellow line.

Nevertheless, when propulsion system operation is required the gas should be heated to temperatures over the critical temperature to make sure that the propellant is ejected solely in gas phase and does not liquefy before system operation comes to a halt.

For this reason we conceptually designed a propulsion system that utilizes carbon dioxide and can meet stringent volume requirements. To do so a representative future mission, the SAMSON cubesat constellation was chosen.

MISSION PLATFORM AND REQUIREMENTS

The Space Autonomous Mission for Swarming and geOlocation with Nano-satellites (SAMSON) project is an effort to evaluate 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, comprises the design, development, integration and mission execution of all three nano-satellites [10]. Each satellite is a 6U CubeSat with a target nominal launch mass of less than 8 kg. The satellites are planned to be launched in 2018 and the mission duration is to be at least 12 months.

To perform cluster keeping and orbit maneuvers each of the three nano-satellites necessitates some means 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 delivering space propulsion systems. Due to each satellite's low mass (<10 kg) and small size (6U) its propulsion system needs to be compact and light weight. At the same time, to maintain the three satellites in accurate formation and orbit, sufficient maneuverability is required (ΔV= 20 m/s). The propulsion system specification requirements are listed in Table 2.

Table 2. SAMSON satellite propulsion system specification requirements.

Satellite / Mission Property	Constraint / Guiding Line	Remarks
ΔV	20 m/s	One year mission
m_d	8 kg	Delivered Mass
m_{ps}	< 2 kg	System wet mass
Max Volume	10cm×10cm×21cm	For entire system
Electrical Power	< 10 W	
Tank Material	Titanium	Designer preference
Thrust	20 mN	Per thruster

Given the satellite propulsion system requirements a designated CO_2-based propulsion system was conceptually designed.

PROPULSION SYSTEM ARCHITECTURE

Propulsion System Layout

Since CO_2 must be in gas phase when ejected from the satellite the propulsion system must consist of basic elements such as heaters, pressure regulator, valves, propellant tank, piping, pressure transducers and a thruster nozzle. The propulsion system layout is presented in Figure 2.

The philosophy of the propulsion system architecture and operation is as follows: First, propellant is stored in the propellant tank in liquid or saturated states. 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 regulator so to reduce the pressure to several bar. Finally, the thrusters are fired by opening the individual solenoid valve of each thruster.

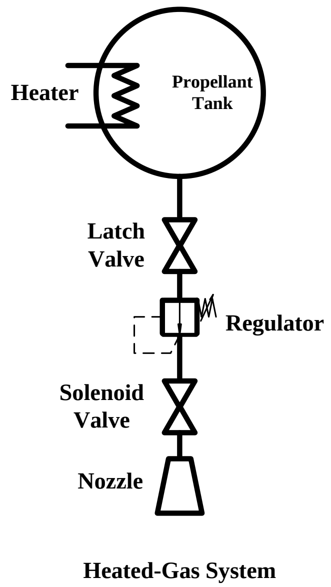

Figure 2. Basic layout schematic of the carbon dioxide based heated-gas propulsion system.

Nozzle Design

The thruster nozzle is the component responsible for converting the propellant internal energy into directed kinetic energy. Nozzle design has to be such that the thrust produced by the nozzle meets the requirement of 20 mN. First, the nozzle expansion ratio (A_e/A_t) was set at 400 where the throat and exit diameters are d_t = 0.25 mm and d_e = 5 mm respectively. The throat diameter was chosen according to manufacturing limitations.

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 under the valid assumption of an isentropic nozzle [11]. Consequently, to achieve the required thrust of 20 mN the nozzle inlet pressure has to be set at 2.135 bar. For the above-mentioned values a thruster mass flow rate of 0.03 gr/sec was calculated.

Nozzle characteristics are listed in Table 3.

Table 3. Nozzle characteristics.

Property	Value
Throat diameter	0.25 mm
Exit diameter	5 mm
Inlet pressure	2.135 bar
nozzle exit to nozzle inlet pressure ratio	13,700
Mass flow rate	0.03 gr/sec

Detailed calculation of nozzle geometry and pneumatic features, including sensitivity to variations in inlet pressure and manufacturing tolerances, can be found in ref.12.

Propellant Tank Design

First, given the propulsion system requirements, and using the rocket equation (Eq.1), the required propellant mass was computed.

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

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 computed minimal specific impulse, using Eq.2, is Isp_{min}=67 sec. Using the calculated value of the minimal specific impulse the required propellant mass is 310 gr.

$$\mathrm{Isp} = \frac{1}{g_0} \sqrt{\frac{2\gamma}{\gamma - 1} \times \frac{R_0 T_c}{Mw}} \left[1 - \left(\frac{p_e}{p_c} \right)^{(\gamma-1)/\gamma} \right]^{\frac{1}{2}} \qquad (2)$$

Following the propellant mass calculation an operational nominal temperature of 80°C was chosen. 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 this operational temperature was based on meticulous analysis performed in which the advantages and disadvantages of a wide range of operational temperatures were considered [12].

Propellant tank dimensions were calculated to meet the following requirements: (1) propulsion system volume allocation, (2) calculated required propellant mass, (3) chosen operational temperature and (4) propellant tank required material yield strength.

The titanium propellant tank is capsule-shaped, able to contain maximum pressure of 155 bar, at the operational point, with a safety factor of 4. The propellant tank weighs approximately 465 gr.

The propellant tank characteristics are listed in Table 4.

Table 4. Propellant tank characteristics.

Property	Value
Shape	Capsule
Material	Ti-6Al-4V
Outer length (end-to-end)	207.5 mm
Outer diameter	74 mm
Stored propellant mass	310 gr
Operational temperature	80°C
Mass	465 gr
Maximum pressure	155 bar

Final System Design

Following the nozzle and propellant tank design phases detailed system design was possible. According to the components specified in Figure 2 commercial off-the-shelf components were determined and allocated, along with the propellant tank, within the defined system volume of approximately 2U (see Table 1Table 2).

The final list of the propulsion system components is presented in Table 5.

Table 5. List of propulsion system components mass.

Component	Typical Mass [gr]
Propellant (CO_2)	310
Propellant Tank	465
Pressure Regulator	500
Latch Valve	50
Fill Valve	100
Thruster (Solenoid Valve & Nozzle)	4×50
Brackets, Harness and Piping	100
Pressure Transducers	30
Thermostat	12
Heaters	10
Overall	**1777**

CAD illustration of the carbon dioxide based propulsion system is shown in Figure 3.

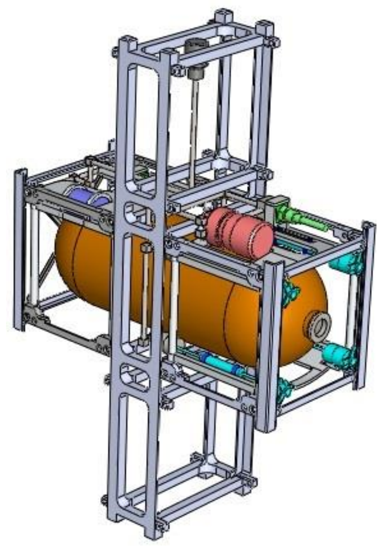

Figure 3. CAD illustration of the carbon dioxide based propulsion system with its components allocated within the 2U designated volume. Only the relevant part of the satellite's chassis is presented.

The overall propulsion system's wet mass is 1,777 gr, which is lower than the maximum allowed value of 2 kg; therefore all design requirements were satisfied.

PROPULSION SYSTEM OPERATION

Launch and Idle States

During launch the system is idle and inactive. However, if the propellant tank is subject to excessive ambient temperatures the pressure in the tank will increase. When the propulsion system reaches the critical point the pressure is equal to the critical pressure that is 74.73 bar. If the ambient temperature further increases to 50°C the pressure will increase to about 104 bar. These values are usually acceptable with launch providers.

When the propulsion system is inactive in space all valves must be closed, heaters off and state monitored (propellant tank pressure and temperature).

Preparation for Operation

When in space and before operation the propellant tank needs to heat until it reaches the operation temperature of 80°C. To do this the heaters should directly heat up the propellant tank, preferably at highest input power of 10 W. However, due to expected thermal losses some of the input power will dissipate into heating the satellite structure, propulsion system components etc.

Since at this stage the design is merely conceptual all net power levels that go into pure propellant tank heating are explored. That is power levels in the range 1-10 W where 10 W represent an ideal case in which all heating power goes into propellant tank heating and 1 W represent the case in which 90% of the heating power is dissipated while only 10% actually heat up the propellant tank. The heating process was calculated using Eq.3.

$$t_{heating} = \frac{\left(c_{v,T_{op}} T_{op} - c_{v,T_0} T_0\right) \times m_p + C_{m,tank}\left(T_{op} - T_0\right) \times m_{tank}}{P_{heat}} \quad (3)$$

where $t_{heating}$ is the required heating duration, $C_{v,T_{op}}$ and C_{v,T_0} 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.

Heating time as a function of net input heating power is presented in Figure 4. It can be seen that as net heating power increases the required waiting time before operation decreases. At the ideal case of net heating power of 10 W the readiness time is approximately 1.5 hour. In a more realistic scenario where the net heating power is approximately 7 W the preparation time is about 2.2 hours.

It can therefore be deduced that propulsion system preparation time is at least two hours. For this reason system operation frequency should be less than every two hours.

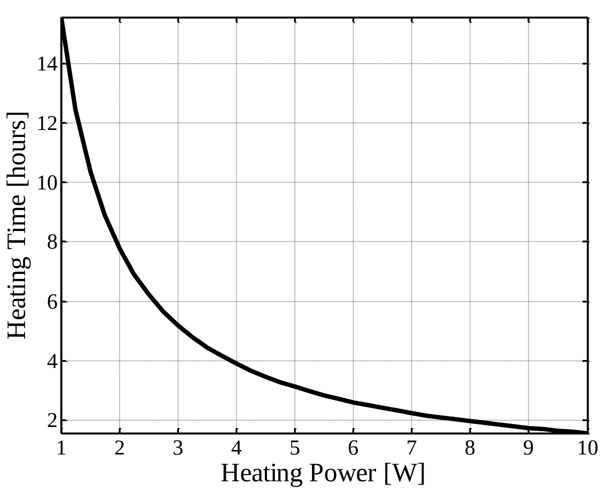

Figure 4. Propellant and propellant tank heating time as a function of net input power. Initial temperature is assumed 10°C while operational temperature is assumed 80°C.

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 away with the released gas. Since the remainder of the propellant is maintained inside a constant volume its temperature decreases. The propellant cooling effect may lead to undesired liquefaction that should be avoided by operating the propulsion system for sufficiently short time durations.

Using basic thermodynamic relations the cooling effect was studied numerically. Figure 5 shows the propellant temperature as a function of time for four initial mass fraction values. The mass fraction is the ratio between the propellant mass, at a given point during the mission, and the initial propellant mass. For example, when the propellant tank contains half of its initial mass, the mass fraction is equal to 0.5.

It can be seen in the figure that at the beginning of mission, when ξ_m=1, the propulsion system can be operated continuously for almost 5 minutes before risk of liquefaction occurs. However, as the propellant is drained out, operation time steadily decreases to less than 2 minutes at ξ_m=0.25.

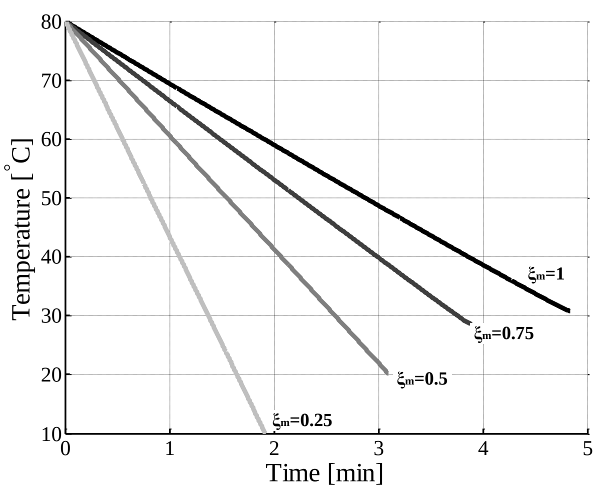

Figure 5. Propellant temperature decrease as a function of time during system operation, for four different mass fraction values (ξ_m).

In addition to the operation duration it is essential to know the propulsion system's impulse capability, or maximum attainable ΔV per operation. Figure 6 shows the maximum attainable ΔV per operation for different mass fractions. In a sense, this figure represents the propulsion system's capability to generate impulses throughout mission lifetime.

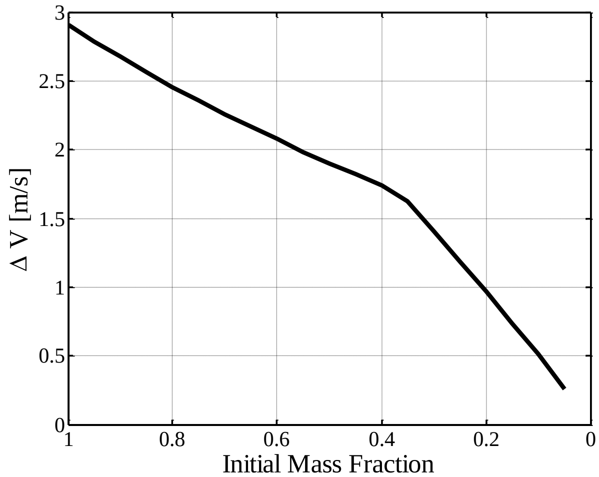

Figure 6. Attainable ΔV as a function of initial propellant mass fraction.

It can be seen in the figure that the attainable ΔV decreases as the propellant tank is drained out. This is in line with the shorter possible operation durations at lower mass fractions already discussed. It is seen in the figure that at the beginning of mission (fully loaded propellant tank) the attainable ΔV is almost 3 m/s. However, as the mission progresses and less propellant is stored in the tank the attainable ΔV decreases down to 1 m/s at a mass fraction value of ξ_m=0.2.

Full Mission Simulation

To illustrate the state of the CO_2 propellant throughout mission lifetime an entire mission, composed of many operation sequences, is presented in Figure 7. The figure shows how in most phases of the mission, when the propulsion system is idle, the propellant is stored in saturated state at pressure of about 45 bar. Before each operation the propellant temperature is raised to the operational temperature of 80°C. When the system is operated and gas is ejected at the required mass flow rate the propellant temperature decreases quickly until liquefaction may occur (when the black curves reach the blue curve).

It is interesting to note that in a large part of the mission the propellant tank is allowed to reach temperatures lower than the critical temperature during system operation. This is because CO_2 is still in pure gaseous state even below the critical temperature; an attribute that gives an important advantage to using CO_2 in cold-gas propulsion systems.

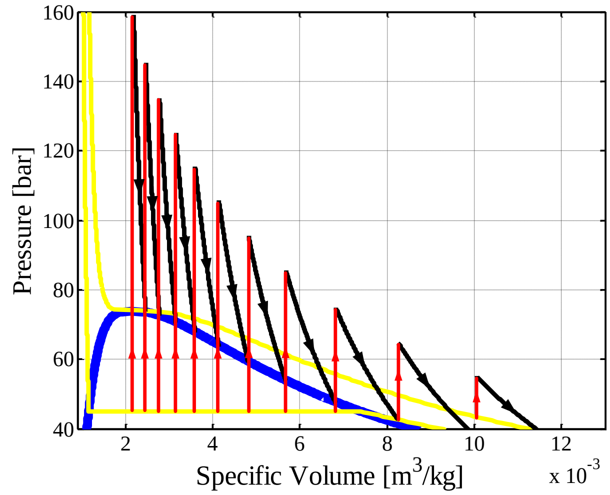

Figure 7. A p-v diagram representing an entire mission sequence. Red lines represent heating sequence. Black lines represent operation sequence, Yellow lines are isotherms (10°C and critical temperature of 31°C).

CONCLUSION

The paper showed the feasibility and relative ease to utilize an inert, affordable and safe propellant, carbon dioxide, for a cubesat mission. A preliminary conceptual design of a propulsion system based on carbon dioxide propellant was presented. The main characteristics of the system are its low volume, low mass and ease of integration into the 6U small spacecraft, which may also be applicable to larger nano and micro satellites, depending on mission requirements.

We showed that the presented propulsion system requires a readiness time duration of several hours, depending on the net heating power invested and the particular thermal design of the spacecraft. We discussed the propulsion system operation duration that depends on the time before propellant liquefaction occurs. At early stages of the mission, when the tank is relatively full, operation duration may reach nearly 5 minutes; whereas at later stages of the mission operation duration may be lower than 2 minutes. Lastly, it was shown that the propulsion system may generate considerable impulse bits, or ΔV, as high as 3 m/s at Beginning of Life (BOL), which is impressive compared to other propulsion systems for cubesats.

Since nano-satellites are becoming popular, and affordable, an increasing number of nano-missions are foreseen – the presented propulsion system may provide many benefits to satellite integrators, given the compliance with their performance requirements.

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