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Mars CO_2 Powered ISRU Propulsion System

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Mars atmosphere is made up mostly of carbon dioxide (more than 95% by volume), and its average ambient surface temperature of 210 K, coupled with large diurnal temperature swings can facilitate the formation of dry ice during the night. Because of its low critical temperature, frozen carbon dioxide (dry ice) can be heated at constant volume to modest temperatures on the order of 400 K to produce a supercritical fluid for use as an *in situ* rocket propellant. Building from previous undergraduate engineering capstone design projects, this paper has investigated the blow down propulsive behavior using a 1 mm diameter sonic nozzle, supplied with less than 1 kg of supercritical carbon dioxide fluid as propellant. The tests showed that these systems can produce specific impulse levels in excess of 100 seconds for very short duration, and that heating by a small solar array could yield useful thrust enhancements for longer durations, but the effect depends on the energy added per unit mass of propellant. In addition, supersonic nozzles have been designed using the method of characteristics for this non-ideal gas fluid in order to further explore the behavior of this unconventional rocket propellant.

Nomenclature

A	=	area
A_e	=	exit area
a	=	local speed of sound
C_+	=	left running characteristic
C_-	=	right running characteristic
g	=	gravity
I_{sp}	=	specific impulse
$\dot{m}$	=	mass flow rate
p_a	=	ambient pressure
p_e	=	exit pressure
r	=	radial distance
u	=	velocity component in the axial direction
u_e	=	exit velocity
u_{eq}	=	exhaust velocity
V	=	velocity magnitude
v	=	velocity component in the radial direction
x	=	axial distance
θ	=	flow angle
$\lambda_\pm$	=	slope of right-running and left-running characteristic lines
μ	=	Mach angle
ρ	=	density

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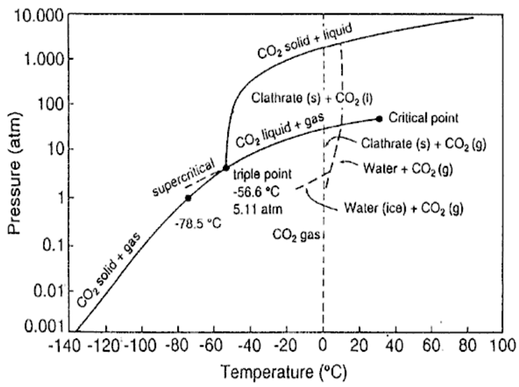
I. Introduction

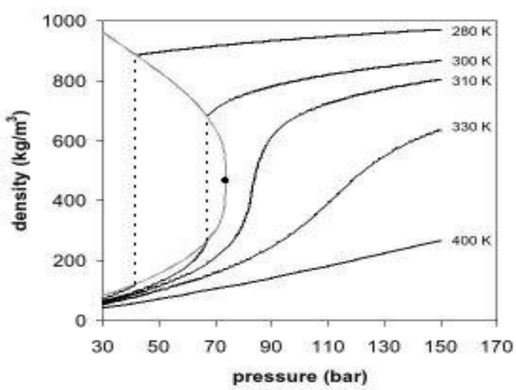
Present solar-powered Mars surface exploration vehicles can only traverse relatively flat terrain at mid-latitudes during Martian summer. The upcoming Mars Science Laboratory rover will use radioisotope thermoelectric generators for power enabling year around operation, but it is only expected to traverse approximately 20 km during one Mars year. Many of the surface locations of greatest scientific interest are widely dispersed and some are in terrain that is too hazardous for a rover. Consequently, high-thrust systems are needed to propel surface vehicles in and around hazardous terrain and to cover large distances. The ideal surface propulsion system should use local Mars resources as a propellant source to permit multiple operations, and the most readily available Mars resource is its atmosphere. Since the atmosphere consists mostly of carbon dioxide, the goal of this research has been to examine the feasibility of a reusable high-thrust propulsion system utilizing solid carbon dioxide condensed out of the atmosphere and heated subsequently to produce a supercritical fluid for rocket propellant. Zubrin *et al*[1] have investigated compressed carbon dioxide as a Mars surface propellant previously, but their study focused on high temperature performance, requiring significant levels of heating. We are concerned here with producing useful propulsive thrust that can be achieved using thermoelectric refrigerator/heat pump systems that could be powered using energy from a small solar array.

Phase diagrams for carbon dioxide are shown in Figure 1. Observe that the critical temperature is 31^o C (304.2 K), and the density of the solid phase of pure carbon dioxide is greater than 1,000 kg/m^3 at Mars ambient pressures (< 10 mb, not shown[*]).

Fig. 1. Phase Diagrams for Carbon Dioxide



a) Pressure-temperature phase diagram (in presence of water)



b) Density-pressure phase diagram

[*] The density of saturated solid carbon dioxide at 6 mb (148^o K) is approximately 1600 kg/m^3.

We note that non-condensable gases and water are present for the Mars surface application, assuming that a thermoelectric cooler would be used to condense carbon dioxide directly out of the atmosphere. From Figure 1a, we see that the solid phase for pure CO_2 at a pressure of 10 mb, exists at temperatures below -120^o C (153 K). While Mars nighttime temperatures can dip below this threshold, thermoelectric cooling can be employed to condense dry ice inside of a tank at higher ambient temperatures. Subsequently, the collected solid CO_2 contained in the tank can be heated at constant density to temperatures above the 304.2 K critical temperature, converting the carbon dioxide into a supercritical fluid (Figure 1b). A variety of details related to the condensation, capture and heating of atmospheric carbon dioxide require further study, but for the present investigation, we have assumed that dry ice has been collected in a tank and we have been concerned with the propulsive performance produced by expanding heated supercritical carbon dioxide fluid through a nozzle.

An earlier undergraduate engineering capstone design team built a pressurized carbon dioxide thrust system incorporating a sonic nozzle within a test stand. Their apparatus did not measure thrust and mass flow rate with sufficient accuracy to determine specific impulse, but they observed (visual and audible) nozzle flow behavior suggesting that the expanding carbon dioxide continued to behave like a single phase gas well beyond the equilibrium two-phase (solid-vapor) limit.

Since Mars surface pressures are very low, it is possible to operate rocket nozzles using very low tank pressures, but the phase transition behavior that readily permits supercritical fluid production also produces *equilibrium* gas expansions that revert quickly to two-phase flows. Hence, the possibility of exploiting non-equilibrium expansion of carbon dioxide gas beyond phase limiting conditions is an important consideration for its possible future use as an *in situ* propellant. Supersonic Mach two nozzles should produce near-optimal specific impulse and thus have been designed as part of this investigation. Supercritical carbon dioxide nozzle flows cannot be modeled using simple ideal gas models; thus motivating the use of the method of characteristics for a non-ideal gas.

A great deal has been written about the structure of jets produced by highly underexpanded nozzles. Adamson and Nicholls[2] investigated Mach disk formation resulting from supersonic expansion of a sonic flow into still air. Crist, Sherman and Glass[3] reported on experiments that included highly underexpanded carbon dioxide flow from a sonic nozzle, but they did not measure thrust. Recently, Kim, Kang, Otobe and Setoguchi[4] have investigated nonequilibrium condensation effects in underexpanded jets. By and large, those studies have attempted to use ideal gas compressible flow models as much as possible.

We were concerned primarily with the performance of supercritical carbon dioxide fluid as a propellant for Mars surface applications where ambient temperatures are low and only small amounts of (solar-derived) electrical power can be available to heat the fluid to convert the solid phase to a supercritical fluid phase. We were interested in the degree to which modest amounts of electrical power, produced by a solar array, could be used to augment propulsive performance during hypothetical winged daytime flight.

Carbon dioxide is not expected to behave like an ideal gas for these high-pressure, low-temperature operating conditions and it is expected that dry ice will freeze out during expansion. Since supersonic nozzle flows can be sustained at Mars for propellant tank pressures on the order of 10 kPa, and surface molecular mean free paths vary between 2.5 and 12.5 μm[5], some very interesting non-equilibrium flow possibilities exist. A great deal of work has been devoted to studying these types of flows (See Bartell[6]), but budget limitations have prevented experimental investigation of that flow regime.

II. Numerical Design of Supersonic Carbon Dioxide Nozzles

Supersonic carbon dioxide nozzle designs are not amenable to standard linear methods of characteristics using ideal-gas-based, Prandtl-Meyer functions. Instead, a method of characteristics approach for a non-ideal gas nozzle design was used. Due to the expected high stagnation pressures, non-ideal gas considerations for the subsonic and nozzle throat geometries, other than consideration of phase transitions, can be neglected. That is, any smooth contraction contour upstream of the throat will produce sonic flow at the throat. Non-ideal fluid behavior must be considered for the supersonic portion of the nozzle where the hyperbolic governing equations can be solved numerically using the method of characteristics.

Recently, Aldo and Argrow[7] studied plug nozzle designs for non-ideal, dense-gas Bethe-Zel'dovich-Thompson (BZT) fluids. They developed a method of characteristics approach based on earlier work by Zucrow and Hoffman[8] who developed an Euler predictor-corrector scheme to solve the real gas characteristic and compatibility equations simultaneously. That approach has been modified here for the specific case of supercritical carbon dioxide. However, the resulting "streamline contour" predictions neglect viscous interaction with the nozzle wall and hence must be corrected for boundary layer effects.

For throat diameters of 1, 2, and 4 mm, with specified nominal density, viscosity, and speed of sound at the nozzle throat, this study has determined that fully turbulent Reynolds numbers characterize the flow in the sonic section. At the anticipated Reynolds numbers, the streamwise velocity within the boundary layer is, in general, an unsteady swirling flow. Rogers and Davis[9] collected and summarized the various empirical methods for estimating time-averaged supersonic boundary layer growth more than 50 years ago. In nozzle design it is often assumed that the boundary layer thickness at the throat is negligible due to the very favorable upstream pressure gradient. Indeed, the favorable pressure gradient inhibits the growth of the boundary layer, but Rogers and Davis pointed out that experiments have shown that the boundary layer thickness at the throat is negligible only when the design exit Mach number is above 3. In the present design, the boundary layer thickness at the throat was estimated using a displacement thickness model from McCabe[10]. Subsequently, the average boundary layer growth along the nozzle contour was estimated using the methods of Rogers and Davis[9], then iterated to adjust for displacement thickness pressure effects. These boundary layer calculations were then combined with the non-ideal gas method-of-characteristics-based streamline contours to complete the supersonic nozzle design.

Thrust and specific impulse can be estimated assuming that the stagnation temperature and pressure correspond to the tank conditions, neglecting any line pressure drops. Throat flow conditions can be estimated by assuming isentropic expansion from the (time-varying) tank conditions. The subsonic nozzle unit used in our experiments had an entrance diameter of 19.05 mm (0.375 in.) and three throat diameters were considered. The 2 mm throat diameter analysis is presented here, assuming that the initial (stagnation) pressure was 31.03 MPa (4500 psia) and the stagnation temperature was 450.7 K. The nozzle design analysis that follows has assumed that the fluid is pure carbon dioxide and the tank pressure and temperature are constant.

Carbon dioxide properties were evaluated using the on-line NIST properties[11]. Below the critical pressure and temperature (73.78 bar, 304.2 K) the nozzle flow will be a (non-ideal) gas and the isentropic flow conditions for the supersonic portion of the nozzle are shown in Table I.

Table I: NIST-based carbon dioxide properties for supersonic nozzle flow

(P_o = 31.01 MPa, T_o = 450.7 K)

P (MPa)	T (K)	ρ (kg/m^3)	M
15.51 (2250 psia)	390.1	299.83	1.01
14.13 (2050 psia)	382.0	282.48	1.09
12.76 (1850 psia)	373.2	263.93	1.19
11.38 (1650 psia)	363.4	244.11	1.29
9.997 (1450 psia)	352.5	222.97	1.41
8.62 (1250 psia)	340.2	200.24	1.53
7.24 (1050 psia)	326.1	175.99	1.67
5.86 (850 psia)	309.5	149.78	1.83
4.48 (650 psia)	289.51	121.59	2.01
3.79 (550 psia)	277.55	106.5	2.12

A second-order-average-property Euler Predictor scheme was used to solve the following characteristic and compatibility equations for a real gas, where the radial and axial coordinates are r and x; the flow angle with respect to the axial coordinate direction is θ, and the characteristic surfaces are oriented with respect to the flow direction via angle μ. The local flow velocity components are u and v, and the local speed of sound is a. The $C_\pm$ characteristic equations are:

$$\left(\frac{dr}{dx}\right)_\pm = \lambda_\pm = \tan(\theta \pm \mu), \quad (1)$$

and the C$_\pm$ compatibility equations are:

$$\left(u^2 - a^2\right)du_\pm + \left[2uv - \left(u^2 - a^2\right)\lambda_\pm\right]dv_\pm = \frac{a^2 v}{r}dx_\pm . \qquad (2)$$

The resulting supersonic nozzle contour design for a 2mm throat diameter is shown in Figure 2. Variations in the flow properties within the inviscid and boundary layer regions are shown in Figures 3 and 4.

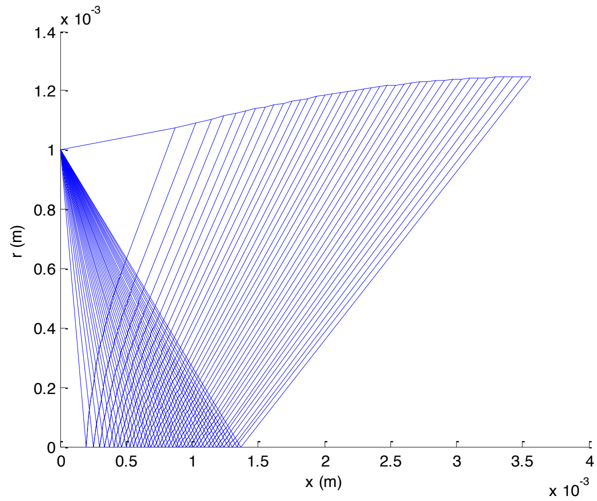


Fig. 2. Supersonic nozzle contour (upper-half)

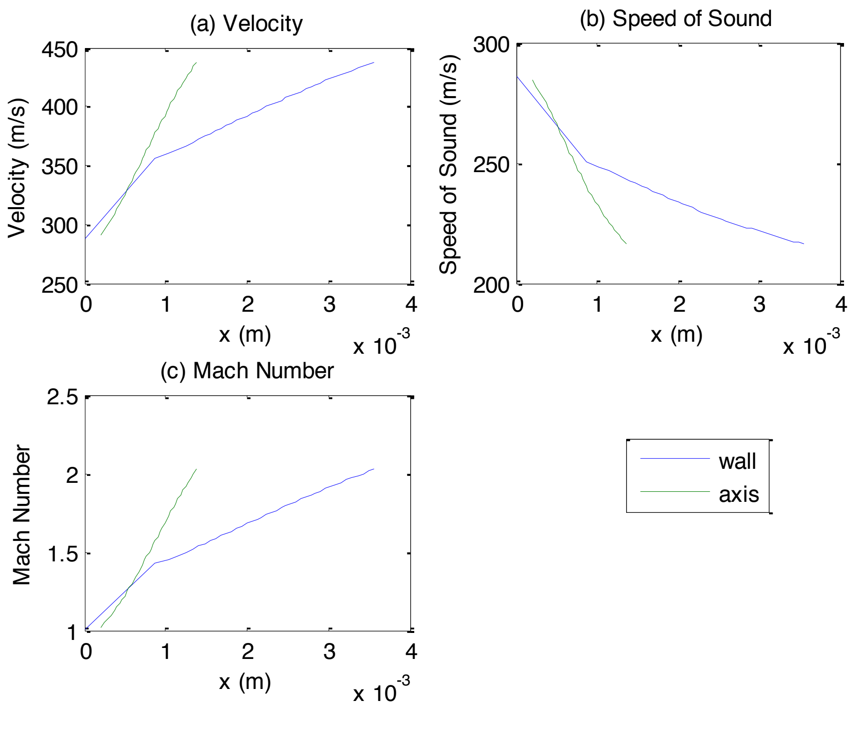


Fig. 3. a) Velocity distribution along the nozzle wall and axis; b) Speed of Sound distribution; c) Mach Number distribution.

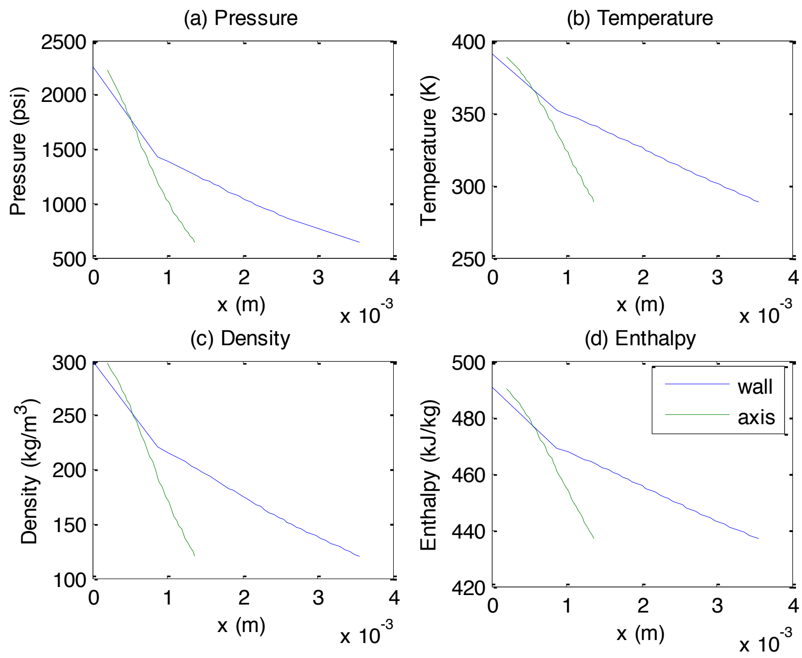


Fig. 4. Variation of nozzle thermodynamic properties in supersonic section

The thrust and specific impulse can be calculated using the equations below:

$$\dot{m} = \rho V A \tag{3}$$

$$u_{eq} = u_e + \left(\frac{p_e - p_a}{\dot{m}}\right) A_e \tag{4}$$

$$Thrust = \dot{m} u_{eq} \tag{5}$$

$$I_{sp} = \frac{u_{eq}}{g} \tag{6}$$

Using eqs. 3-6, as well as the additional boundary layer correction analysis we get the results shown in Table II. We see that the nozzle diameter has little influence on predicted specific impulse.

Table II: Nozzle Performance Predictions

Throat DIA	**1 mm**	**2 mm**	**4 mm**
Re	3,200,000	6,400,000	12,800,000
Corrected Exit Area m^2	$1.25\text{x}10^{-6}$	$4.98\text{x}10^{-6}$	$1.99\text{x}10^{-5}$
Thrust N (lb)	35.23 (7.92)	140.7 (31.6)	562 (126.)
Specific Impulse sec	52.7	52.6	52.6

III. Experimental Apparatus

A. Tank Design

A flanged steel pressure vessel was designed to facilitate rapid charging of dry ice. The flanged end incorporated a silicon O-ring and six grade-8 bolts to fully seal against the expected 310 bar (4500 psi) tank pressures. Tank heating was accomplished using an end-mounted (via NPT threads) internal plug heater in order to heat directly the carbon dioxide within the steel tank.

B. Test Stand Design

The test stand is shown schematically in Figure 5. We have utilized a sliding rail suspension system, avoiding the binding and frictional losses encountered by a previous design. The sliding rail system constrains the thrust-generated displacements to the axial direction. Extreme care was taken to insure proper alignment of the guide rods in order to prevent any radial resultant forces from affecting the load cell axial thrust measurements. During experiments, small changes in the suspended center of mass, due to carbon dioxide ejection, produce small (correctable) shifts in measured thrust. The rail-mounted sliding units were connected to the propulsion system using supports mounted directly beneath the two primary mass elements in order to minimize torsional loads. The two sliding rail units were linked using a rectangular plate that centered the lower supports, and included a vertical end plate that served as the rigid load cell force plate. An endplate bracket (partially visible in the schematic) was utilized in order to accommodate the aft-mounted heater connections and properly engage the load cell.

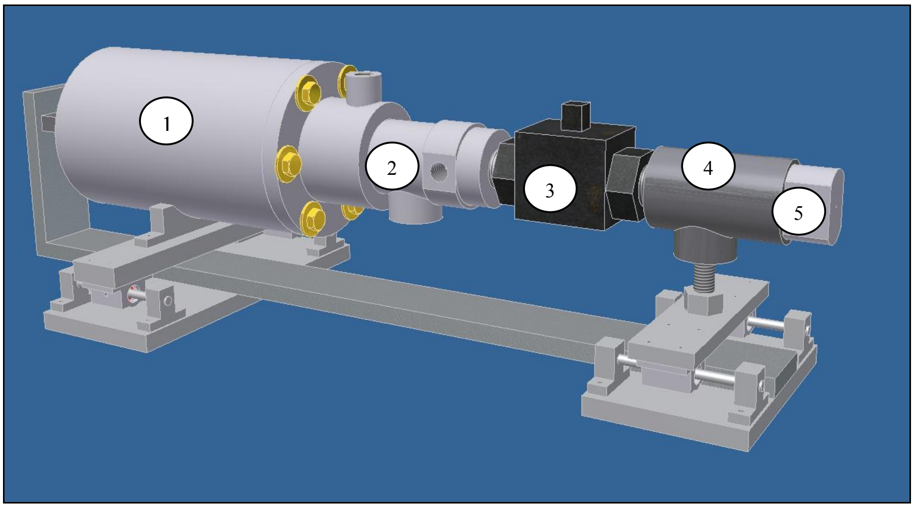


Fig. 5. Schematic of Test Stand: *1. Pressure tank; 2. aluminum instrumentation fitting; 3. Valve; 4. nozzle holder; 5. nozzle.*

C. Sensor System

A single Omega PX4202-6KG5V pressure transducer, located in the aluminum fitting between the valve and the tank, was used to monitor the varying total pressure produced during propellant discharge. Thrust was measured using an aft-mounted aluminum "S-Beam" Load Cell. The load cell was calibrated using standard weights and tested at different operating temperatures to ensure accuracy and repeatability. Data acquisition was accomplished using LabView software through an NI USB-6210 interface.

The pressure transducer was calibrated using a LabView program and a test apparatus that was constructed for that purpose. The simple pressurization tests demonstrated that the transducer met the manufacturer's specification of +/- 0.25% of full-scale accuracy. Thus, for its specified maximum pressure of 6000 psia (414 bar), a 0.25% error is equivalent to a resolution to within one atmosphere. Total pressures were well above one atmosphere at the end of each experiment.

IV. Experimental Results

This project was supported by a small capstone design project award, provided by Old Dominion University's Batten College of Engineering and Technology, and thus was limited to an extremely small budget. The propulsion system was sized so that measured propulsive forces could be resolved using an inexpensive load cell and test durations on the order of one minute could be achieved. This resulted in small nozzle throats and the resulting supersonic nozzle fabrication required very fine precision machine tools that were not available in time to meet our deadlines. Consequently, preliminary testing was confined to measuring the specific impulse performance of a 1 mm diameter sonic nozzle. Blow down tests were conducted for underexpanded sonic nozzle flows utilizing both a heated tank and an unheated tank for the purpose of establishing the possible effectiveness of utilizing solar energy to augment propulsive performance. However, early tests showed that the heat capacity of the tank and nozzle assembly resulted in appreciable heating of the compressed carbon dioxide gas during the low-pressure portions of the blow-down tests.

The basic procedure for these tests included crushing commercial dry ice using a mallet and forcing the solid CO_2 into the tank. Initially the dry ice sublimated to cool the stainless steel tank. After the tank was cooled, pulverized dry ice injection was continued until the tank was filled. No attempt was made to remove residual air contained in the tank and, because of relatively high humidity, an undocumented but small amount of frost sometimes was present on the commercial dry ice. The dry mass of the entire propulsion system (carried on the rail system) was measured prior to charging and, after the charged tank was sealed and reinstalled in the thrust unit test stand, the total mass was measured again in order to determine the actual CO_2 propellant mass. Typical carbon dioxide charge masses were between 0.7 and 0.8 kg. The system was heated subsequently using the silicon wrapped plug heater and monitored so the temperature did not exceed 344 K. Once the system was heated to the desired temperature, the valve was opened and the CO_2 escaped through the nozzle.

A typical blow down test is shown in Figure 6. That blow down test is representative of a series of tests conducted in Old Dominion University's Space System Engineering Laboratory, showing the limitations of the experimental test rig.

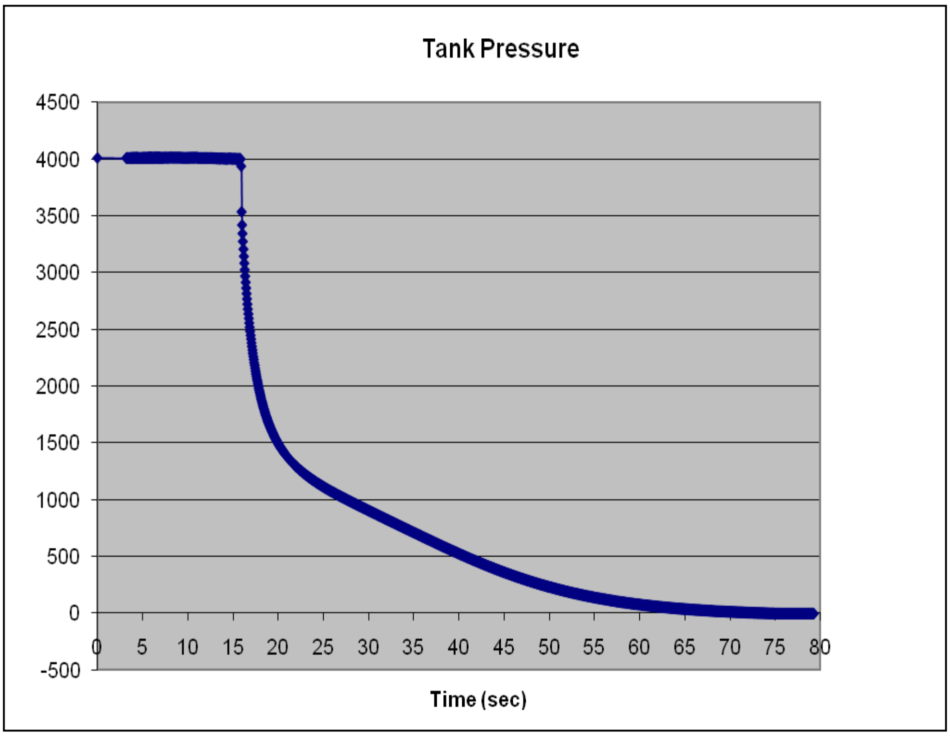


Figure 6. Representative blow down test a). Pressure (*psig***)**

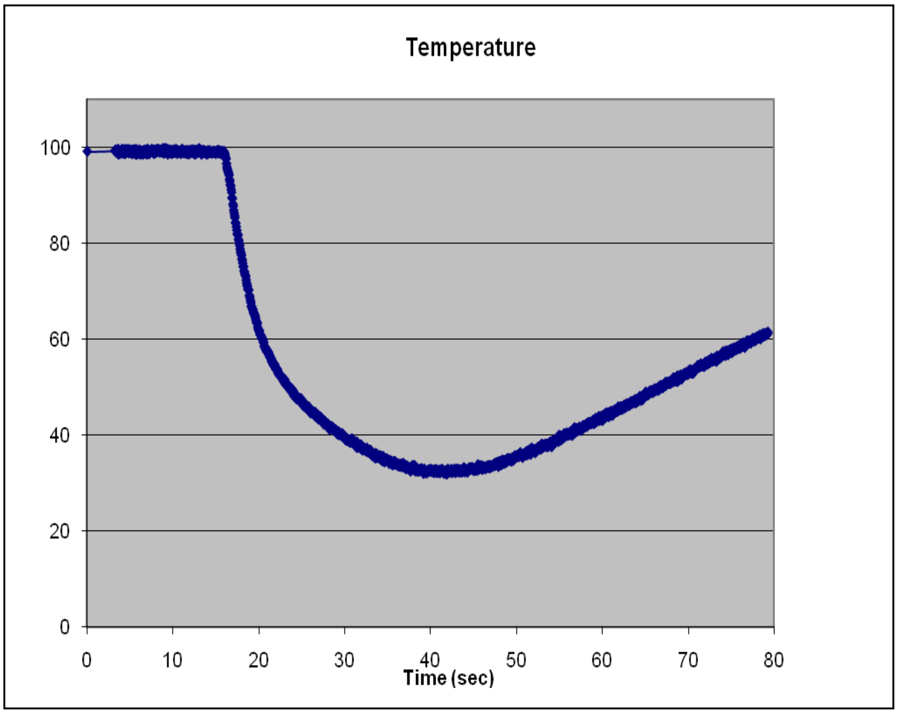


Figure 6 b). Temperature (*Celsius*).

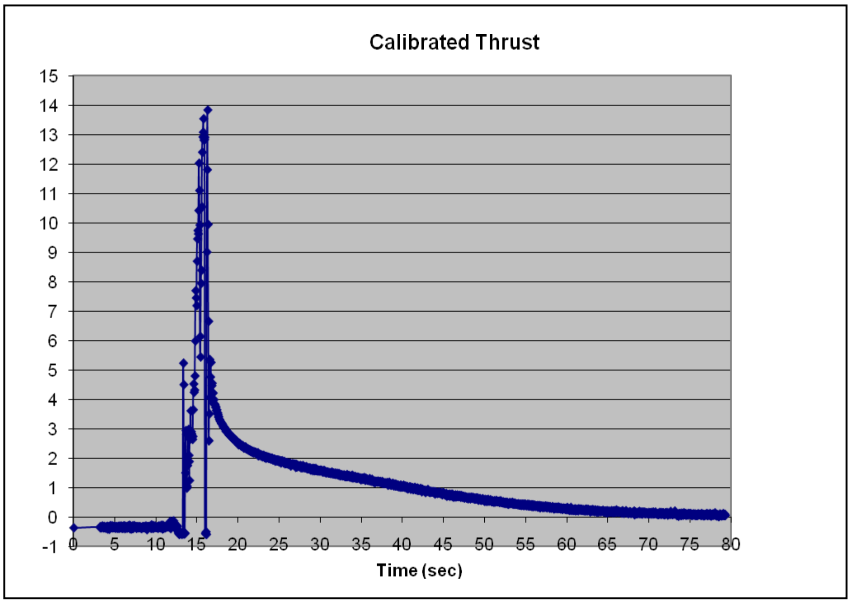


Figure 6 c). Measured thrust (lb_f,) *calibrated at test temperature*.

The test results in Figure 6 were for a test that was conducted when the integrated electric plug heater was turned off just prior to the blow down test. The erratic initial thrust behavior, exhibited in Figure 6c, is due to the primitive valve system utilized in this test. Blow down was initiated by pushing the test apparatus against the thrust load cell, while opening the *half-turn to open* valve with a very large wrench. When the valve was opened, the wrench was removed and the test rig mass experienced transients resulting from the release of the manual forces and the rapidly varying nozzle thrust.

The temperature history, shown in Figure 6b, shows that the thermal mass of the test rig has acted to heat the expanding carbon dioxide fluid in the pressure tank as the temperature of the compressed carbon dioxide remaining in the tank dropped well below the initial tank temperature. The temperature history shows that heating has become significant 15 to 20 seconds after the valve was opened. Blow down tests conducted with the electric plug heater operating during the experiment, produced similar (but more rapid) tank temperature increases, and the linear temperature vs time slope exhibited toward the end of both blow down tests could be used to estimate the heating rate produced by the thermal mass of the test apparatus. That is, at the later times when the carbon dioxide nozzle flow rates were quite small, the temperature history suggests that gas heating by the apparatus is nearly constant. When the plug heater was operated during the blow down test, the more rapid increase in gas temperature with time, resulting from the additional 205 W heating rate, produced a higher slope and a simple lumped system model was used to estimate the heating produced by the apparatus during the later stages. The estimated combined heating from the thermal mass of the apparatus and the electrical heater was approximately 1 kW at the later times, after the tank pressure had dropped below 20 bar. Hence, the sonic nozzle performance at times greater than 50 seconds, shown in Figure 6, is indicative of the thrust performance for a low-pressure carbon dioxide propulsion system, augmented by heating rates that might be delivered by a solar array during mid-day Mars operation.

The influence of heating on the specific impulse performance is demonstrated in Figure 7. In that figure, we have compared the specific impulse estimated by assuming adiabatic expansion starting from an initial state inferred by smoothing the valve and load cell transients with between the specific impulse calculated by computing the average thrust between successive load cell measurements and dividing that average thrust by the estimated mass flow rate (change in carbon dioxide mass computed between two successive measurements divided by the time interval between measurements). The rapid cooling produced by fluid expansion from the highest pressures occurs so rapidly that no appreciable heating occurs. However, as the fluid temperature (and pressure) drops to levels substantially below the stainless steel (and other hardware) temperature, that heating has a very beneficial effect in producing substantially higher thrust. That effect warrants additional investigation. Since the energy added per unit mass of residual carbon dioxide is much greater when the mass is small, that effect requires further investigation.

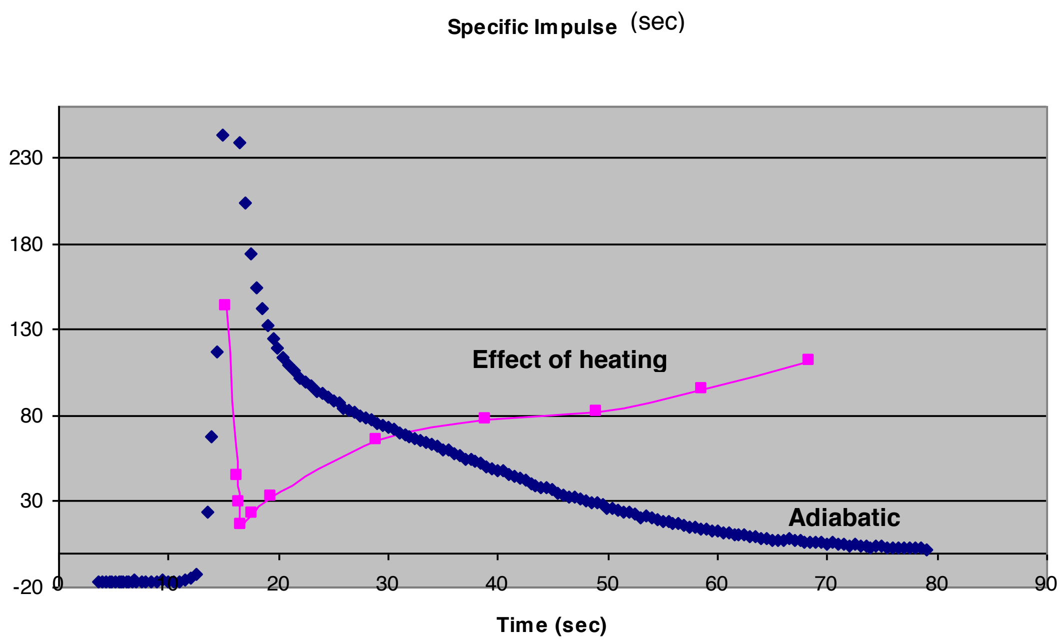


Fig. 7 Comparison of adiabatic specific impulse estimates with tank-heated specific impulse

The sonic nozzle flows were photographed in order to document the flow behavior of supercritical, under expanded sonic carbon dioxide flows. Figure 8 shows the flow produced by supercritical carbon dioxide expansion at a pressure and temperature substantially above the critical state (P_{crit} = 73.8 bar; T_{crit} = 304.2 K), whereas Figure 9 is the under expanded flow of carbon dioxide as the tank conditions approached the critical state.

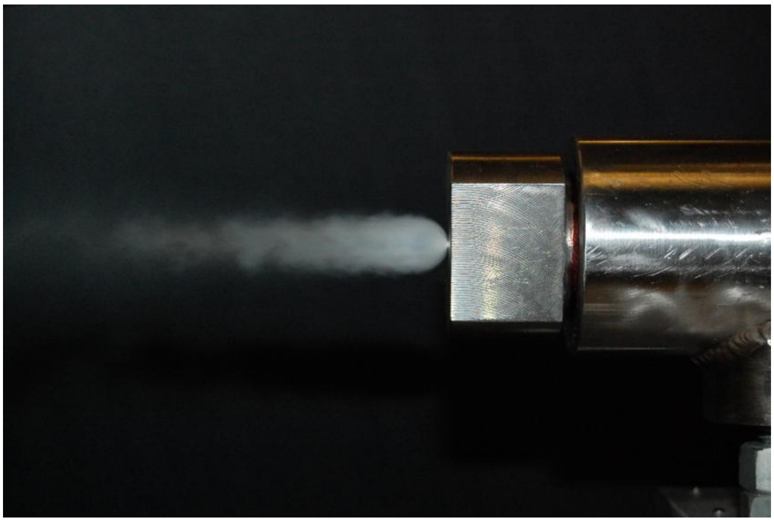


Figure 8. Under expanded sonic nozzle flow behavior at 111 bar (T_o = 340 K)**.**

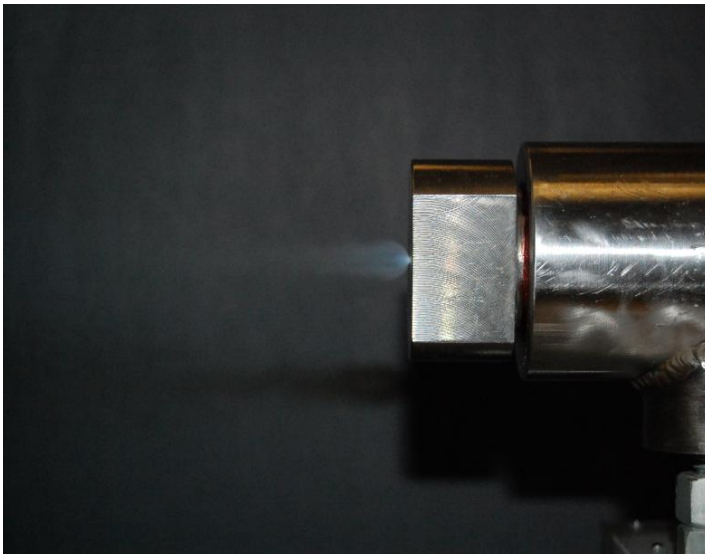


Figure 9. Under expanded sonic nozzle flow behavior at 77 bar (T_o = 320 K)**.**

V. Conclusions

Our combined numerical and experimental studies investigating the feasibility of freezing carbon dioxide out of the Mars atmosphere and heating the resulting solid phase at constant volume to temperatures above 304.1 K can indeed produce an *in situ* propellant for Mars surface applications where small amounts of solar electric energy, typical of contemporary surface vehicles, is the only available energy source.

We have developed Mach two nozzle designs for supercritical carbon dioxide flows that could be produced using Mars atmosphere as a feedstock. The nozzle design assumed that the initial stagnation temperature and pressure conditions could be maintained during propulsion operations. While that operating condition is not considered to be realistic for an actual Mars system, it has produced a somewhat surprising result indicating that non-ideal gas effects result in substantially smaller nozzle exit areas than would be expected for an ideal gas with a ratio of specific heats typical for low-pressure carbon dioxide. Supersonic experimental studies could be justified that explore this behavior since the reduced size could permit substantially higher exit Mach numbers, exploiting Mars' low ambient pressures.

Our sonic underexpanded nozzle experiments have shown that the relatively high specific impulse thrust produced when carbon dioxide is propelled by its very high initial supercritical fluid pressures are of short duration and would be useful primarily for impulsive "kicks" that might be required for mechanical manipulations of geologic material or for freeing a bound device. In addition, the specific impulse behavior of the expanding carbon dioxide at lower pressures has been enhanced by heating but the degree of enhancement is related to the energy added per unit mass of residual carbon dioxide and therefore requires additional investigation.

During the later stages of the thrust experiments, when the carbon dioxide propellant temperature was substantially lower than the stainless steel tank temperature, as a result of presumed adiabatic expansion, heat transfer from the tank to the remaining fluid became significant. Tank heating could be isolated by comparing the unheated propellant temperature history with the temperature history when employing the plug heater to supply a known amount of energy to the fluid. At pressures below approximately 35 bar, heating by the tank was sufficient to prevent a serious examination of potential non-equilibrium flow behavior because precise energy balance calculations weren't possible. Supercritical carbon dioxide propellant capabilities should be investigated further in order to explore possible high Mach number, non-equilibrium flows for greater specific impulse performance in the Mars environment.

Acknowledgments

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