DOI: 10.13009/EUCASS2017-474

Nitrous Oxide Application for Low-Thrust and Low-Cost Liquid Rocket Engine

Tomasz Palacz

AGH Space Systems
Mechanical Engineering and Robotics Faculty
AGH University of Science and Technology, Cracow, Poland
palacztt@gmail.com

Abstract
Nitrous oxide as an oxidizer in liquid rocket engines is considered. General characterization of propellant is performed. Theoretical calculations for such system are made with help of small experimental liquid rocket engine preliminary design as an example. That includes thrust chamber, cooling, feed system and injector design. Comparison is made with other high-energy propellants. Unique factors and design problems that arise are addressed with possible solutions. Review of recent advancements is attempted with vision of further investigations and future applications.

Nomenclature

I_{sp}	=	Specific impulse [s]	A_t	=	Nozzle throat area [m^2]
u_e	=	Actual exhaust velocity [m/s]	A	=	Chamber cross-section area [m^2]
c^*	=	Characteristic velocity m/s]	D_t	=	Nozzle throat diameter [m^2]
P_e	=	Nozzle exit pressure [Pa]	R	=	Specific gas constant [J/kg K]
P_c	=	Chamber pressure [Pa]	ρ_l	=	Liquid density [kg/m^3]
T_c	=	Chamber temperature [K]	T_{aw}	=	Adiabatic wall temperature [K]
ρ_c	=	Combustion gases density [kg/m^3]	T_{wg}	=	Gas-side wall temperature [K]
V_c	=	Chamber volume [m^3]	T_{wc}	=	Liquid-side wall temperature [K]
$\dot{m}$	=	Propellants mass flow [kg/s]	T_{co}	=	Coolant free stream temperature [K]
$\dot{m}_f$	=	Fuel mass flow [kg/s]	T_{cc}	=	Coolant critical temperature [K]
$\dot{m}_o$	=	Oxidizer mass flow [kg/s]	T_{ci}	=	Coolant initial temperature [K]
C_d	=	Discharge coefficient	C_p	=	Specific heat capacity [J/kg K]

1. Introduction

This paper is the result of experience gained by an author in AGH Space Systems student's society, where main propulsion project is small liquid rocket engine able to launch a rocket up to the Karman line. An author, on basis of previous experience with small hybrid rockets researched a concept of bi-propellant liquid rocket engine with NOx. Although, entire propulsion team is involved in this project, sole author of this paper performed presented work. Moreover, broad scope of the subject caused that only theoretical part is discussed, which happens to be an author's domain. Experimental part of Zawisza rocket engine project is not introduced, against initial thoughts, for project is still on-going and some experiments are either not ready or still inconclusive. Additionally, experimental design merits separate discussion and it would render this paper extensively large.
Last decade brought significant advancement into the hybrid rocket engines, especially those utilizing nitrous oxide (NOx) as an oxidizer. However, it has not been used extensively in liquid rocket propulsion to this date. This paper is intended to be the generalized case study in order to diagnose unique properties and design problems. In some issues, author recalls mentioned engine project as an example. Consecutive chapters discuss all aspects of liquid propulsion revealing problems with application of NOx and proposing solutions focused mostly on bi-liquid rockets. Since this paper covers such vast subject its intention is not to be in-depth analysis, but rather highlighting review for further investigations.

DOI: 10.13009/EUCASS2017-474

2. Theoretical considerations

2.1 Physics of nitrous oxide

In room temperature NOx is in subcritical state having critical point at 36.37°C, 7.24MPa. In consequence, NOx can be stored in closed tank with coexisting both liquid and gaseous phases. Favourably, vapour provides high pressure when stored in wide range of temperatures, as shown on Figure 1. What is more, that close to critical point yields NOx with useful behaviour when subjected to pressure drop. After closed tank is opened liquid boils off rapidly producing large amount of vapour, thus supplies extra pressure for the tank. This phenomenon called self-pressurization can be used for feeding oxidizer into the combustion chamber [1].

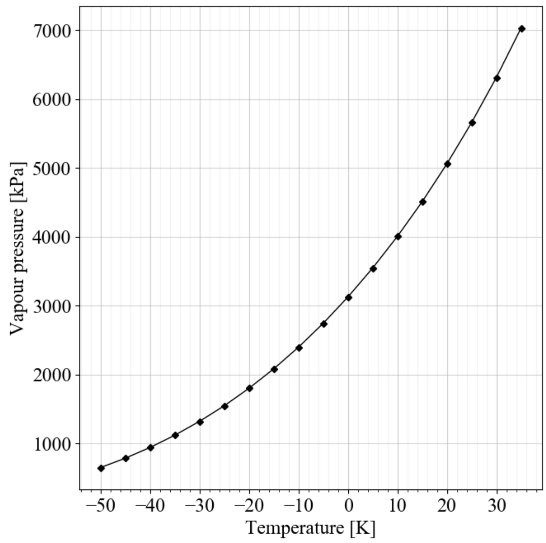


Figure 1: Saturated vapour pressure of stored nitrous oxide in wide range of temperatures

Nitrous oxide is mostly combined out of nitrogen by mass. Therefore, it appears that mixture ratio for NOx as oxidizer should be much higher than ratios for typical LOX, RFNA, nitrogen tetroxide or HTP oxidizers. Moreover, liquid density in near room temperatures is not imposing as given on Fig. 2.

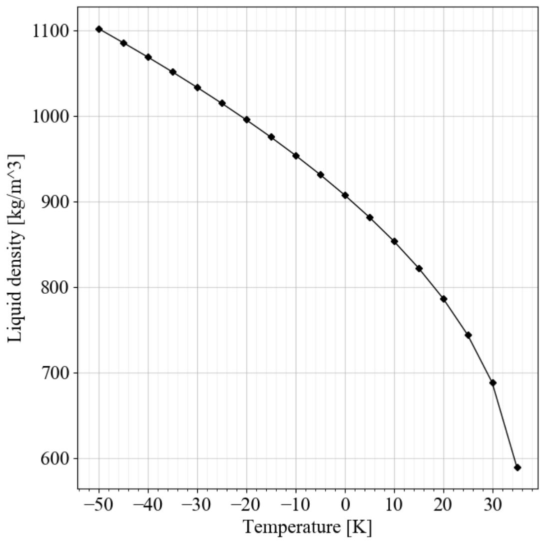


Figure 2: Saturated liquid density of stored nitrous oxide in wide range of temperatures

Material compatibility, non-toxicity and room temperature application promotes NOx as a highly storable oxidizer. In fact, it may be most suitable liquid oxidizer for long term storage, when compared to cryogenic LOX, toxic nitrogen compounds or unstable hydrogen peroxide (HTP). With relative small effort NOx can be kept in pressure vessels in stable conditions and readily operated. Moreover, NOx can be safely used with most materials involved in

DOI: 10.13009/EUCASS2017-474

rocket propulsion, especially materials for tanks, feed system, injector and launch pad infrastructure. This includes metals, plastics and composite materials. Special care must be taken when operating with copper, gold and its compounds, which exhibit catalytic properties. [2] Particular safety precautions must be paid to cover all hazards of storing and handling. Compared to other energetic propellants like HTP, nitrous oxide is relatively safe, but eventuality of deflagration and detonation is present [3]. Accidents with sudden nitrous oxide decomposition were reported [4] [5].
On the other hand, controlled process of decomposition promises to be useful enough to merit serious investigation. Decomposition of NOx results in hot mixture of oxygen and nitrogen reaching ~1650°. Use of catalyst can accelerate decomposition, by lowering the activation energy barrier of approximately 250kJ/mole [6]. In general, this enables use of nitrous oxide as monopropellant in various aerospace applications, which are discussed later.

2.2 Combustion

2.2.1 Propellant performance

In order to characterize efficiency of combustion using nitrous oxide, three fuels were chosen: ethanol (ethyl alcohol), RP-1 and ethylene. Other fuels such as liquid hydrogen or hypergolic hydrazine were not considered in this analysis, because NOx advantageous storability logically imposes that requirement. That being said, it is believed that in some circumstances LH and hydrazine would go well with nitrous oxide.

From application of energy equation for ideal rocket [7]:

$$I_{sp} = \frac{u_e}{g} = \frac{1}{g}\sqrt{2C_pT_c\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{k-1}{k}}\right]} \tag{1}$$

Expressing ratio of specific heats C_p in terms of k, the universal gas constant $\bar{R}$, and the molecular weight of gases $\bar{M}$ we have:

$$I_{sp} = \frac{1}{g}\sqrt{2\frac{k\bar{R}}{(k-1)\,\bar{M}}T_c\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{k-1}{k}}\right]} \tag{2}$$

NASA CEA [8] has been used to determine product composition, specific heats ratio, combustion temperature and molecular weight. Equilibrium product composition is presented in Table 1.

Table 1: Equilibrium gas composition for different propellants given in mole fractions

Propellant					
NOx - Ethanol		NOx – RP1		NOx - Ethylene	
N_2	0.453	N_2	0.518	N_2	0.446
H_2O	0.248	CO	0.167	CO	0.206
CO	0.143	H_2O	0.154	H_2O	0.145
CO_2	0.074	CO_2	0.061	H_2	0.092
H_2	0.065	H_2	0.047	CO_2	0.050
H	0.011	HO	0.025	H	0.031
HO	0.007	H	0.018	HO	0.024
		NO	0.007	NO	0.013
		O_2	0.004		
		O	0.002		

Knowing parameters of chemical equilibrium of combustion, the specific impulse can be calculated using Eq. (2) and nominal value of nozzle pressure ratio. Resultant characteristic parameters of various propellants and mixture ratios are given in Fig. 3.

DOI: 10.13009/EUCASS2017-474

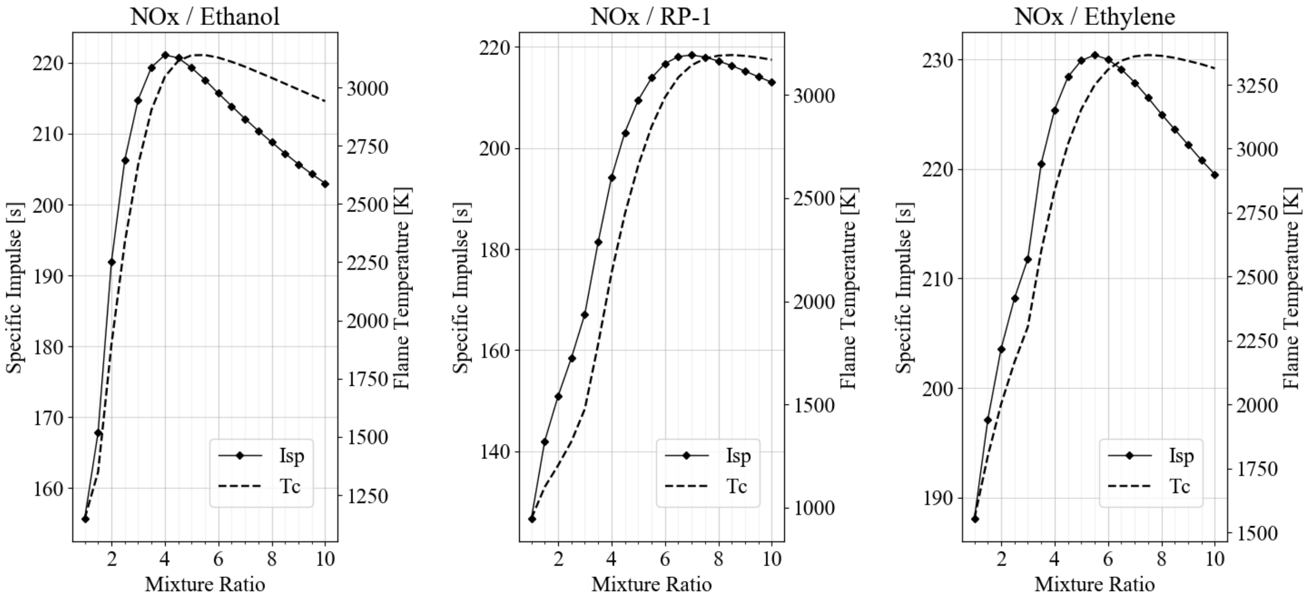


Figure 3: Calculated specific impulse and flame temperature in combustion chamber of different propellants and mixture ratio

2.2.2 Optimal chamber volume for bi-propellant

In order to achieve combustion with near-equilibrium product composition, thus attain calculated specific impulse for given mixture ratio, the size of the combustion chamber must be taken into the consideration. For that purpose the transit time Δt can be defined as:

$$\Delta t = \frac{\rho_c V_c}{\dot{m}} \tag{3}$$

If volume of combustion chamber is too small, then transit time Δt will not be enough to accomplish propellant mixing, vaporization and chemical reaction and some of the propellant will be carried away through the nozzle unburned. On the other hand, if combustion chamber is excessively large, it will result in redundant mass, increased heat transfer to the chamber walls and significant losses due to friction. To compare different engine designs chamber size is described by the characteristic length L^*, defined as:

$$L^* = \frac{V_c}{A_t}, \tag{4}$$

and related to transit time Δt by [9]:

$$\Delta t = \frac{L^*}{\sqrt{kRT_0}} \left(\frac{k+1}{2}\right)^{\frac{k+1}{2(k-1)}} \tag{5}$$

It can be seen from that equation that optimal chamber size depends strongly on the transit time Δt. Although in practice L* is often based upon previous successful engine designs, that were similar in operation, there are some approaches to determine minimum acceptable chamber volume.

Spalding [10] considered processes in liquid-propellant combustion chamber that contributes to value of the time Δt. He based upon assumption that the slowest is vaporization of the liquid drops and formulated one-dimensional model to analyse behaviour of propellant in the chamber. For dimensionless axial distance from the injector ξ he obtained such value $\xi = \xi_2$, at which propellant droplet vanishes (droplet radius reaches 0). Theoretically, at this point complete combustion is attained and further increase in chamber size will result in lowered performance. For ξ_2, Spalding obtains:

$$\xi_2 = \frac{\chi_0 + \frac{3}{10}}{2+S}, \tag{6}$$

where χ_0 is ratio of initial droplet velocity (injection velocity) to final gas velocity in the chamber and S is droplet drag. This drag can be expressed by:

DOI: 10.13009/EUCASS2017-474

$$S = \frac{Pr}{2B}, \tag{7}$$

$$B = \frac{C_p(T_0 - T_f)}{h_f}, \tag{8}$$

where, Pr = Prandtl number of the gas, C_p = specific heat of the gas, T_0 = gas stagnation temperature, T_f = saturation temperature of liquid, h_f= heat of vaporization of the liquid. Finally, for the characteristic length of the chamber [11]:

$$L^* = \xi_2 \frac{r_0^2\left(\left(\frac{2}{k+1}\right)\left\{1+\left[\frac{(k-1)}{2}\right]M_2^2\right\}\right)^{\frac{k+1}{2(k-1)}}\sqrt{kRT_0}}{\left(\frac{k}{C_p\rho_l}\right)ln(1+B)} \tag{9}$$

In this equation, M_2 = Mach number of fully burned gases in combustion chamber. Now, Spalding states that L^* can be evaluated and it strongly depends on injection velocity χ_0 and inlet drop size r_0. This model comes to agreement with other empirically determined values for typical propellants given for example by Altman, *et al.* [12].

This analysis assumes that chemical reaction takes place sufficiently fast, so when vaporization of propellant is over ($\xi = \xi_2$), combustion is already complete. Nitrous oxide, however, needs time to decompose into nitrogen and oxygen beforehand. This process will take place after vaporization and before reaction with fuel gases and in general goes by this equation:

$$N_2O \rightarrow N_2 + \frac{1}{2}O_2 + Heat \tag{10}$$

Since this reaction is exothermic, the speed of vaporization should actually increase locally and mitigate additional effect on ξ_2 of decomposition reaction.

Table 2: Typical values for combustion chamber characteristic length for various propellants

Propellant combination	**L* [cm]**
LOX / RP-1	100 - 130
LOX / LH2	75 - 100
LOX / GH2	55 - 75
LOX / Ammonia	75 - 100
LOX / alcohol	120 - 150
Nitrogen tetroxide / hydrazine	75 - 90
Nitric acid / hydrazine	75 - 90

Two cases of propellant combinations of NOx/RP-1 and NOx/Ethanol were considered for L* calculations using Spalding's model. Combustion pressure assumed is 2 MPa. Results are presented in Table 3. In general, results are in fine agreement with comparable values for LOX given by Huzel and Huang [13] and Sutton *et al.* [14]. Spalding's theory is therefore good starting point for preliminary design of the bi-liquid rocket engine using nitrous oxide as an oxidizer.

Table 3: Calculated values of characteristic length for combustion pressure = 2MPa

Propellant combination	**L* [cm]**
NOx / RP-1	114.3 – 132.8
NOx / Ethanol	125.6 – 167.8

DOI: 10.13009/EUCASS2017-474

2.3 Cooling

2.3.1 Convective heat transfer

If a rocket engine is to be used longer than few seconds appropriate cooling is required to prevent engine failure. With NOx as an oxidizer well known principles apply when it comes to various cooling systems like ablation or radiation. Regenerative cooling is the mostly used cooling system in rocket propulsion and can be applied in all sorts of the propulsion systems. Therefore, this part will cover mostly regenerative cooling and problems that arise with design and application of this system when utilizing nitrous oxide.

The convective heat transfer, which states between 70 – 90 % of total heat transfer in a regeneratively cooled thrust chamber, can be described with following equations:

$$\frac{1}{H} = \frac{1}{h_g} + \frac{1}{h_c} + \frac{t}{k}, \tag{11}$$

$$q = h_g(T_{aw} - T_{wg}) = h_c(T_{wc} - T_{co}) = H(T_{aw} - T_{co}), \tag{12}$$

where: H = overall heat transfer coefficient, h_g = gas side heat transfer coefficient, h_c – coolant (liquid) side heat transfer coefficient, t = chamber wall thickness, k = thermal conductivity of chamber wall material and q = heat flux. These factors impose severe limitations upon configuration of cooling system and thrust chamber design. Overall schematic of convective heat transfer is given in Figure 5.
Additional strongly limiting factor is total heat capacity of the coolant. It performs crucial role in design of cooling system.

$$Q_{max} = \dot{m}C_p(T_{cc} - T_{ci}) \tag{13}$$

In order to cool down hot gas side wall to a safe temperature, h_g must be kept as low as possible and at the same time h_c must be sufficiently high. Then, coolant is able to take the excess heat from the combustion chamber walls. Usually, there is not much to do with h_g, unless we employ some other kind of cooling, like ablation or film cooling. To alter coolant side heat transfer coefficient h_c, changes have to be made to coolant passage configuration like pressure and velocity distribution. There has been significant progress in optimization of liquid side using high aspect ratio fins as passages cross-section [15] [16].

Nevertheless, as shown by Eq. (13), there is certain level of heat flux up to which it can be tolerated by the coolant and which is governed by coolant mass flow.

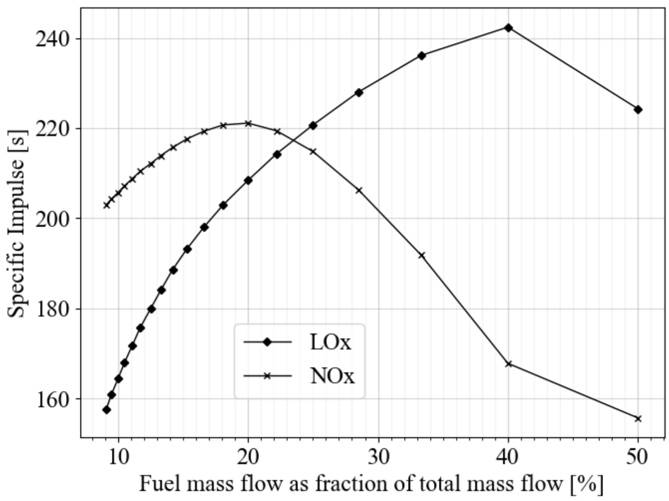


Figure 4: Comparison of fuel available for cooling as fraction of total mass flow with LOx and NOx oxidizers to attainable specific impulse

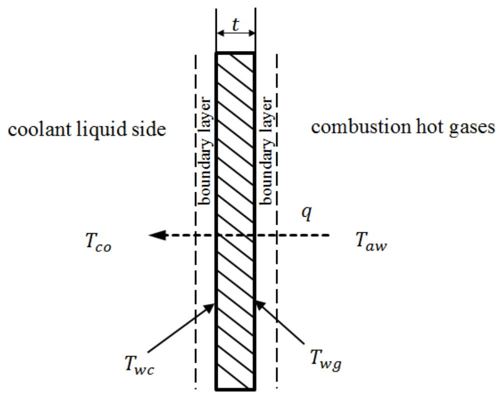


Figure 5: Heat transfer schematic in regenerative cooling in liquid rocket engine

It can be clearly seen from Fig. 4, that use of nitrous oxide leaves very small part of total mass flow for the fuel, which in most cases is the coolant. Therefore, maximum heat capacity becomes problematically small and there is a substantial risk of coolant vaporization and forming a vapour film on the wall, which drastically reduces heat transfer

DOI: 10.13009/EUCASS2017-474

coefficient. In fact, failure of wall material occurs even before coolant is able to reach its critical temperature, whereas temperature of the coolant near wall is higher than bulk coolant temperature. Although, forming of vapour bubbles in coolant passages, which is known as 'nucleate boiling', significantly increases heat flux, exceeding certain level will lead to rapid failure. [17]

2.3.2 Heat capacity issue

As a part of preliminary design of the thrust chamber of experimental liquid rocket engine Zawisza the heat transfer analysis was performed for various scenarios of operating points. For purpose of this paper, this analysis will serve as a case study. Combustion chamber of Zawisza engine is made of copper with thermal conductivity t about 350 W/(m K). As a cheap and readily available alternative material carbon steel is fine proposition with its $t = 40$ W/(m K), but much larger allowable wall temperature. As the fuel in Zawisza ethanol is employed, which has relatively good cooling capabilities, namely heat capacity and thermal conductivity. Calculated fuel mass flow and estimated total heat capacity dependent upon mixture ratio are shown in Table 4 and chamber geometry is given in Figure 6.

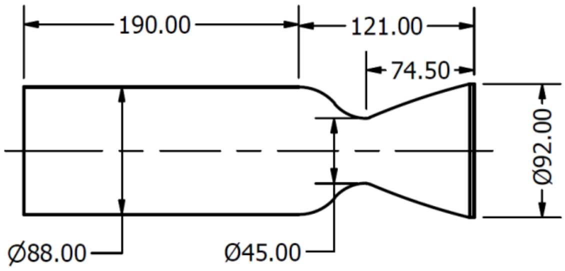
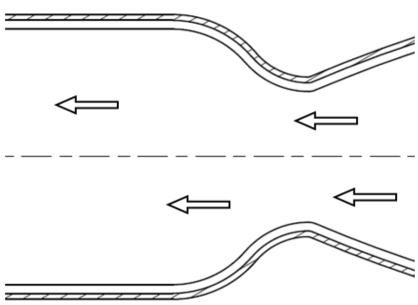


Figure 6: Experimental liquid rocket engine Zawisza 1B preliminary design thrust chamber geometry

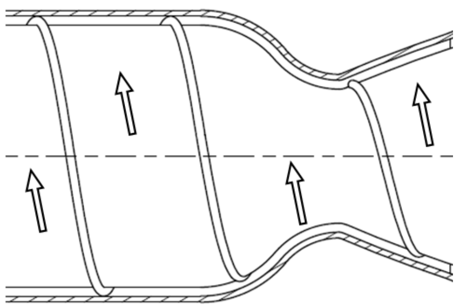
Table 4: Fuel mass flow and maximum total heat capacity available for cooling with varying mixture ratio for Zawisza rocket engine

MR	$m_f\ (kg/s)$	$Q_{max}\ (kW)$
1.0	1.00	493.0
1.5	0.80	394.4
2.0	0.67	328.7
2.5	0.57	281.7
3.0	0.50	246.5
3.5	0.44	219.1
4.0	0.40	197.2
4.5	0.36	179.3
5.0	0.33	164.3
5.5	0.31	151.7
6.0	0.29	140.9

Two cases of co-axial cooling passages configuration were taken into the account and are given in Fig. 7. First was *annular* jacket at which whole fuel is passed into one single passage around chamber. Only ribs in a form of thin rods brazed to the chamber in the axial direction are present in the passage, but serves solely structural role as separation of external shell from the chamber. Second configuration removes ribs and implements helical copper *coil* around perimeter of the chamber going from nozzle exit to the injector assembly. In general, *coil* arrangement exhibits enhanced liquid side coefficient h_c, thanks to increased flow velocity and proves to be more flexible in design.



a) Annular arrangement with axial flow of coolant



b) Coil arrangement with radial flow of coolant

Figure 7: Two cooling arrangements for Zawisza thrust chamber schematically showed nearby nozzle throat

DOI: 10.13009/EUCASS2017-474

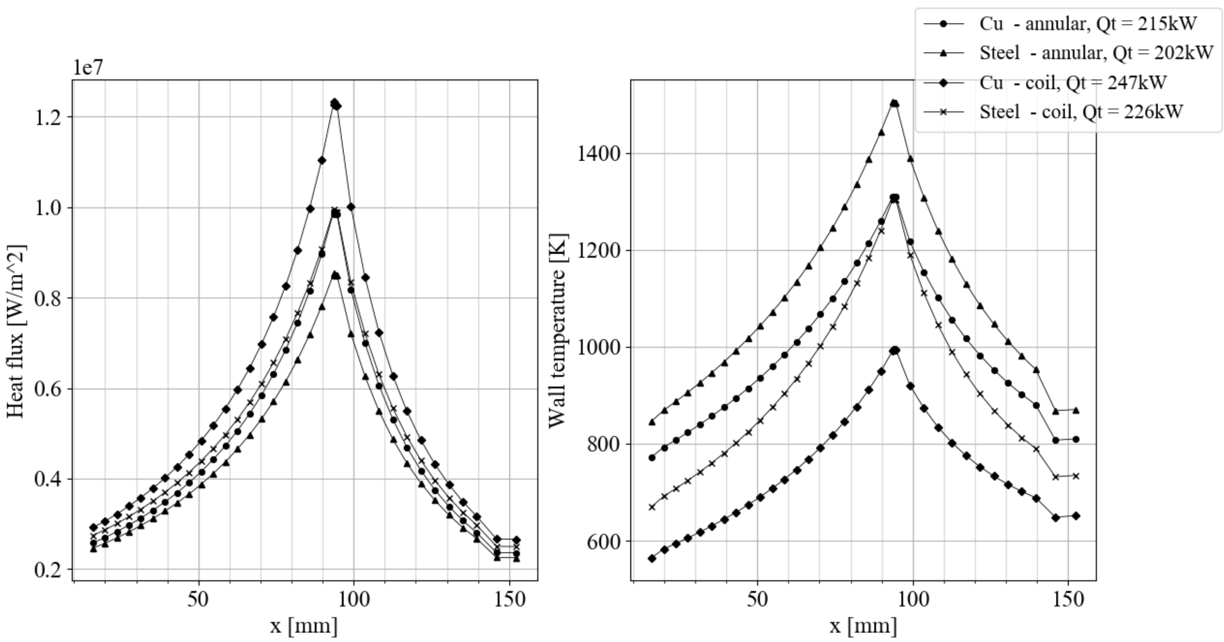
For numerical analysis of convective heat transfer, calculation of hot gas side coefficient $h_g$is crucial. Bartz [18] gives following equation:

$$h_g = \left[\frac{0.026}{(D_t)^{0.2}}\left(\frac{\mu^{0.2}C_p}{Pr^{0.6}}\right)\left(\frac{P_c}{c^*}\right)^{0.8}\left(\frac{D_t}{r_c}\right)^{0.1}\right]\left(\frac{A_t}{A}\right)^{0.9}\sigma, \quad \textbf{(14)}$$

in which: Pr = Prandtl number, r_c= throat radius curvature and σ is correlation factor defined as:

$$\sigma = \left[0.5\frac{T_{wg}}{T_c}\left(1+\frac{k-1}{2}M^2\right)+0.5\right]^{-0.68}\left(1+\frac{k-1}{2}M^2\right)^{-0.12}, \quad \textbf{(15)}$$

Using Bartz correlation to find heat transfer from the hot gases, heat flux and gas side wall temperature were found over distance from the nozzle exit, where fuel is fed. To attain high specific impulse mixture ratio of 4.5 was assumed, as shown on Fig. 3, which corresponds to 0.36kg/s of fuel mass flow.



a) Heat flux distribution along nozzle axis   a) Maximum wall temperature along nozzle axis

Figure 8: Results of heat transfer analysis for Zawisza rocket engine and mixture ratio = 4.5 presented along chamber axial distance. Nozzle exit is at 0, throat at around 90mm and chamber starts at 150 mm. For clarity, graphs are limited to nozzle section and surroundings.

Wall temperatures on the Figure 8a corresponds to the hot gas side wall temperature, where it will be the highest. Calculated temperatures mostly exceeds allowable material temperature (Cu: 475-675K, Steel: 850-1000K) especially at the nozzle throat, which is located at around 90 mm from the fuel inlet. Another issue is that total heat absorbed is far above 179.3kW, that were set as maximum for given coolant mass flow. Results for different cooling passage cofigurations presented on Fig. 8 describes general corellations between heat flux, wall temperature, total heat absorbed and coolant side configuration. Increasing liquid side coefficient h_c significantly reduces wall temperature, but at the obvious cost of absorbed heat. Increasing resistance of heat transfer through the wall by changing material from copper to steel increases wall temperature, but reduces heat flux. It appears that not much can be done to design proper cooling system with such mixture ratio. Changing h_c and t/k either way, will meet one of two outcomes: too much heat flux or too large wall temperature. Obvioulsy, modifying h_g by adding film, transpiration or ablative cooling or some protective coatings may result in a successful cooling system. However, these add another level of complexity to the rocket engine.

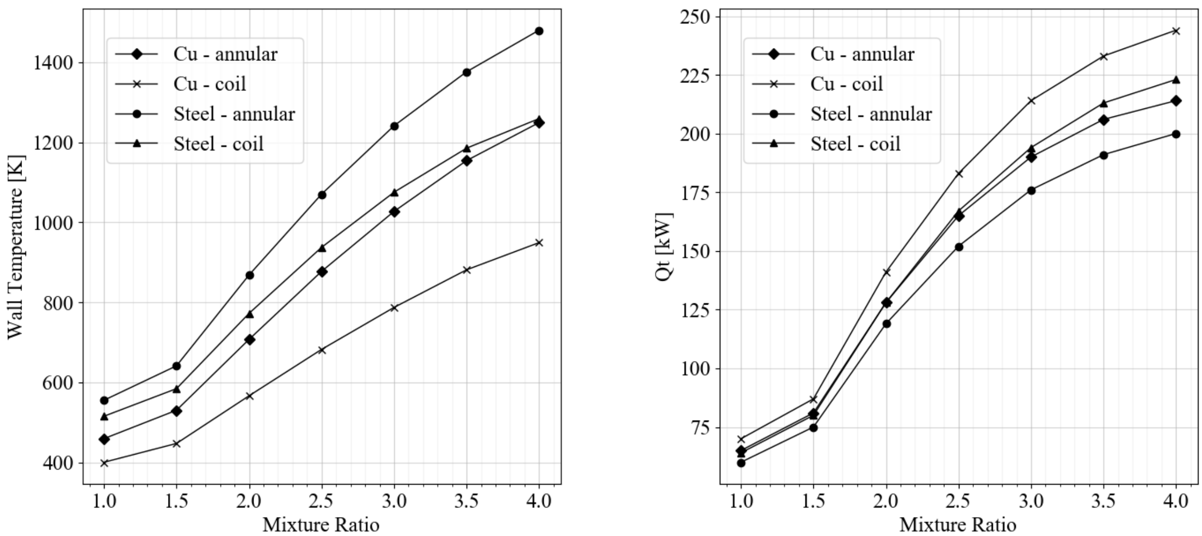
2.3.3 Regenerative cooling alteration

There are some ways to alter this behaviour of regenerative cooling. Perhaps use of high temperature graded wall material may permit finding such liquid side arrangement to meet all cooling requirements [19]. Another interesting choice is nitrous oxide as coolant itself [20]. Still, it has not been thoroughly investigated. Both this methods were not studied further.

DOI: 10.13009/EUCASS2017-474

Table 4 shows that for lesser mixture ratio, more heat capacity is available, at the cost of performance. Figure 3 demonstrate that anywhere between fuel rich ratio 2.25 and optimal 4.5 specific impulse is still above 200 seconds. That requirement is taken as performance check for Zawisza rocket engine. It seems that within range of acceptable drop of specific impulse, meaningful value of heat capacity can be gained. Further analysis of heat transfer behaviour revealed promising dependency upon mixture ratio. As with mixture ratio goes down, combustion goes more and more fuel rich, which significtanly reduces flame temperature. This effect, combined with enchanced total heat capacity provides interesting results. Figure 9 presents relation of wall temperature and heat flux with mixture ratio for previously discussed cooling arrangements. Consulting Figure 9b with Table 4 we can notice that starting from mixture ratio of 3.5 and going down, there are configurations that do not exceed maximum heat capacity. Still, most of them reaches higher that allowable wall temperatures. Although, these temperatures were calculated for the nozzle throat, for which heat flux is few times larger than at the rest of the thrust chamber, where temperatures will surely be lesser, that fact cannot be neglected. Even with this conditions, starting from mixture ratio 2.5 both conditions are met.

It can be observed, that for qualified mixture ratios and configurations there is substantial reserve of heat capacity, but wall temperature requirement is hardly met. That is the case, when h_c has to be enchanced, so that there is satisfactory margin on both temperature and heat capacity conditions. That was the conclusion from preliminary heat transfer analysis for Zawisza rocket engine. From this point, it is matter of optimization of liquid side, which is broadly described in the literature.



a) Maximum wall temperatures at the nozzle throat b) Total heat absorbed during coolant passage

Figure 9: Performance comparison of liquid side cooling arrangements for various mixture ratios measured by total heat absorbed and wall temperature

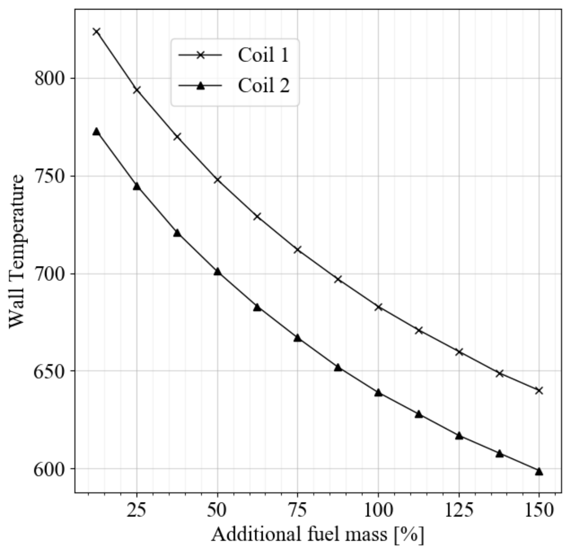
2.3.4 Semi-dump cooling

Reduced mixture ratio solution may be satisfactory, but additional method of achieving regenerative cooling was proposed and shortly studied. Concept of semi-dump cooling derives directly from previously discussed idea. Dump cooling is similar to regenerative cooling, since forced convection in cooling passages is used to keep wall temperatures low [21]. After passage run, coolant is not passed to combustion chamber, but dumped overboard. Usually in such cooling only limited amount of fuel is used for this process [22].

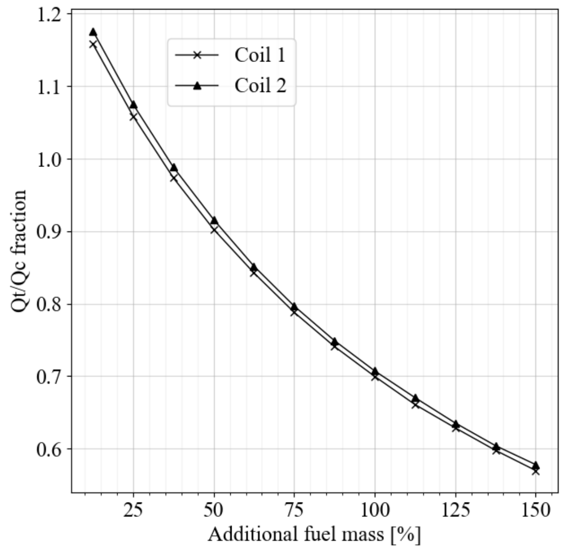
However, we can consider semi-dump cooling, which feeds all available fuel flow to cooling passages, but then dumps only some amount of it. Most part of fuel is passed to the combustion chamber and burned with oxygen. This solution has clear application for NOx regeneratively cooled engine, where high mixture ratio entails insufficient coolant heat capacity. Semi-dump includes enough fuel for the cooling process and then provides precise part of it for high efficiency combustion with optimal mixture ratio. Rest of the coolant is discarded.

Short analysis of semi-dump cooling application for Zawisza was carried out. Liquid side *coil* cooling arrangements were enhanced with increased h_c to take advantage of excessive heat capacity. Fig. 10 gives example results of semi-dump.

DOI: 10.13009/EUCASS2017-474



a) Maximum wall temperatures at the nozzle throat



b) Fraction of maximum heat capacity used

Figure 10: Semi-dump cooling results for two different coil arrangements demonstrating wall temperature and heat capacity margin as a function of additional fuel mass in the system

Downside of this solution is reduced overall performance resulting from ejecting unburned fuel. Specific impulse describing performance of such system may be defined as:

$$I_{sp} = \frac{F_{combustion}+F_{dump}}{g(\dot{m}_o+\dot{m}_f+\dot{m}_d)}, \tag{16}$$

where $\dot{m}_d$= additional fuel mass flow that do not take part in combustion and F_{dump}= thrust attained from discharging dumped fuel at given velocity.

F_{dump} contribution is strongly dependant on type of fuel and cooling configuration. Principal advantage of dump cooling is parallel connection with injector feed system. This feature provides that coolant pressure drop is independent from injector requirements, thus ejected coolant can be superheated and discharged at a reasonably high velocities [23]. Semi-dump whereas, is constrained with injector requirements. This renders dumping thrust minor. However, this requirement should not affect cooling passages much more than in typical regenerative cooling.
Specific impulse as a function of additional fuel mass is presented on Figure 11. Thrust from dumping was neglected.

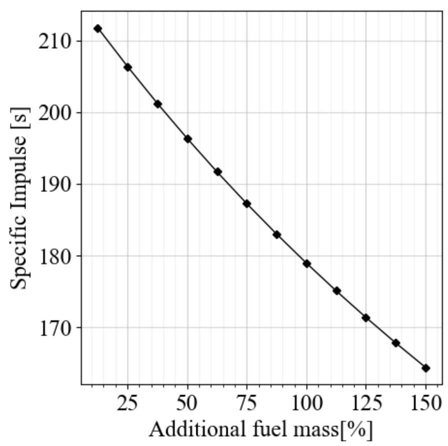


Figure 11: Loss of specific impulse due to addition of fuel mass for cooling which is to be dumped not taking part in combustion process.

Deprivation of specific impulse is substantial, but within range of operating regimes of low thrust liquid rocket engines. For example, semi-dump configuration for Zawisza would result in $I_{sp}$of about 190 seconds.

Semi-dump may be enhanced with alternative cooling patterns. Proposed idea assumes that entire coolant mass could be fed near nozzle throat and passed upwards to the injector assembly. Fuel mass part would go to the combustion chamber and dumped mass part could be passed further to expanding nozzle section cooling passages.

DOI: 10.13009/EUCASS2017-474

This way, remaining coolant may be superheated and discharged as in standard dump cooling, so contributed thrust can no longer be neglected. When fully exploiting dumped mass little to no performance loss can be attained [24] [25].

2.4 Feed system

2.4.1 Vapour pressurization

High vapour pressure of nitrous oxide provides that gaseous phase can be used as the pressurant. This kind of feed system is called VaPak (vapour pressurized) [26]. Vapor pressurization has been investigated since early 60s and it is well known phenomenon. VaPak compromises propellant in a saturated state stored in the closed container. When liquid is extracted from the container, vapour expands and resultant pressure drop causes flash vaporization of small part of a liquid. This way, internal energy of a liquid is used to expel propellant mass and restore pressure to stable condition. In a process, liquid losses temperature, thus equilibrium pressure of the system is usually lowered in time. Simplified schematic of VaPak is presented on Fig. 12.

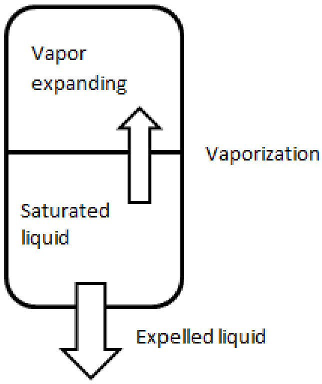


Figure 12: VaPak feed system

Vapor pressurization can be compared to traditional blow-down system, in which prescribed amount of pressurant is closed with propellant in the container. After vessel opening, gas expands and performs work needed to expel propellant mass. However, there is no feature that would attempt to restore lost pressure during expansion, so blow-down systems cannot maintain high pressure throughout the engine burn. VaPak compromises simplicity of blow-down feed system, but at the same time prospects high performance of turbo-pump or external gas pressurization. Nitrous oxide is explicitly used in hybrid propulsion, by both amateur and professionals [27], due to that advantageous feature. Bi-propellant liquid rocket requires little more control over mass flow of the propellant in order to maintain certain level of performance and stability. Since VaPak feed system is natural choice for NOx, the question is what method should be used for feeding fuel.

Nitrous oxide behaviour in VaPak system can be determined knowing initial conditions and system characteristics. Still, VaPak is somehow unpredictable regarding the path it follows till fully expelled state [28]. Typical pressure curve of full expulsion is given by Fig. 13. In order to increase performance, as much NOx mass should be drawn as a liquid with as little pressure drop as possible over expulsion time. More than 90% of NOx liquid mass can be drawn with no less than 70% of initial pressure at the liquid depletion point.

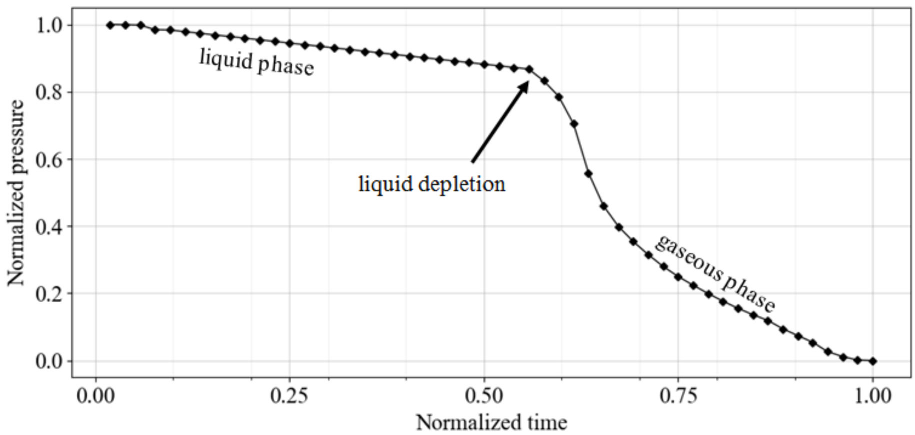


Figure 13: Typical pressure tank history in VaPak NOx expulsion

Pressure history in the NOx tank can be determined by strict initial setup. Such properties as volume, initial pressure, initial temperature, initial liquid mass fraction and mass flow impacts NOx tank state during the expulsion. For example, if ratio of mass flow to initial liquid mass is high, system will produce more steep pressure loss curve and more liquid will be used as pressurizing vapor, what will result in extended gaseous phase of expulsion. If a liquid rocket engine is to work in narrow pressure operating regime, this ratio should be kept small, so to minimize pressure change during liquid depletion.

DOI: 10.13009/EUCASS2017-474

2.4.2 NOx - fuel feed systems coupling

Coupling of NOx VaPak feed system with fuel blow-down system would result in serious OF shift during the engine burn, which is not desirable, as it will reduce performance. What is more important, engine stability could be thrown off balance due to drastic change of operating conditions. One can freely assume that running on optimal mixture ratio for entire burn with that kind of coupled feed is not attainable. However in some cases, system can be optimized to provide best performance for a given purpose. For example, Zawisza uses blow-down for fuel, but with large amount of pressurizing gas. Essentially, we use one big tank for many tests in which there is maximum 40% of the fuel by volume. During test few percent is removed and after that gas pressure is restored. Following this example, coupled blow-down can be calibrated with feed system and injector for running at average optimal mixture ratio and maintaining combustion stability. Still, blow-down coupling will produce significant decrease in performance and tuning of such a feed system to maintain stability may be out of reach.

Alternative is to use classic external gas pressurization, in which inert gas like helium or nitrogen from another vessel is used. It appears, however like a step backwards, since VaPak characteristic property is no external tanks and devices, while gas pressurization requires vessel, regulator and often heat exchanger to improve expansion efficiency. This makes system more complex and with all these features onboard a rocket it seems that there is no real gain from having NOx with VaPak. One have to remember that VaPak have its downsides. If external pressurization is present in the rocket, NOx can be supercharged with inert gas, so VaPak is no longer needed.

Ewig [29] considered use of separate VaPak systems for each propellant for bi-liquid engine. Fuel candidates for VaPak include hydrogen, methane, ethylene and so on. Ewig identified fundamental problems with this solution, namely different mass expulsion ratio (due to pressure difference) in consequence of different physical characteristics of each propellant and mismatch of liquid – gas transition. The latter would result in highly unstable OF shift. Ewig proposed few solutions including electrical or physical links between propellant tanks with some sort of closed-loop controller.

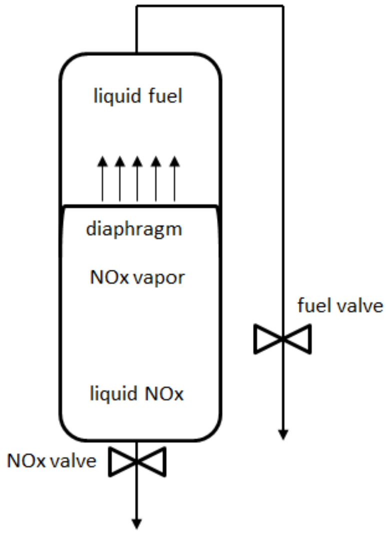


Figure 14: Single VaPak feed system for bi-propellant rocket

Similar solution is to use NOx vapor to pressurize also the fuel. That will include coupled tanks with a diaphragm to separate propellants as presented on Fig. 14. It is believed, that such solution would result with equal pressure in the tanks and similar response of propellant flow characteristics to the system state. Nature of that response is not well known, since there is little flight tests done with VaPak. Therefore, some sort of controller probably should be introduced to correct for any deviation. It appears that this feed coupling could result with maximum simplicity. Similar system to given on Fig. 14 is prepared for Zawisza rocket flight model and conclusion is that bi-propellant VaPak is optimal solution between blow-down and external gas combinations. That said, system coupling still merits serious research.

2.5 Injection

2.5.1 Two-phase mass flow estimation

Since nitrous oxide is stored in saturated state in the tanks, drawing saturated liquid results in two-phase flow downstream. All feed system components like valves, bends and joints will result in pressure drop and given that it falls sufficiently below saturation pressure, flash vaporization will take place and flow may contain gas bubbles. Even more drastic vaporization can occur at the injector, which should permit right amount of NOx flow through the orifices, while atomizing the liquid into small droplets. However, proper injector design is especially hard due to two-phase flow, compressible liquid and high vapour pressure of nitrous oxide. Some work has been done to characterize the flow in injector orifices, presented on Fig. 15.

Dyer et al. considered flow through the orifice of the injector and proposed following equation for predicting nitrous oxide mass flow [30]:

$$\dot{m} = C_d A\left(\frac{1}{1+k}\dot{m}_{SPI} + (1-(\frac{1}{1+k})\dot{m}_{HEM}\right), \quad \textbf{(17)}$$

DOI: 10.13009/EUCASS2017-474

where, $\dot{m}_{HEM}$ is mass flow predicted by Homogeneous Equilibrium Model and $\dot{m}_{SPI}$ is mass flow predicted by Single-Phase Incompressible model. It can be seen, that this equation is trade-off between both these models. HEM assumes thermodynamic equilibrium of the liquid and vapour phases in the orifice. However, it has been noticed that HEM does not predict mass flow well. Dyer et al. proposed explanation that residence time in orifice inflicts that prediction, so flow has virtually no time to reach equilibrium, due to limited mass transfer rate between the phases. They reasoned that the flow depends on ratio of vaporization time and residence time and should fall between entirely liquid SPI prediction and two phases in equilibrium HEM prediction. They proposed parameter k to be:

$$k = \frac{\tau_b}{\tau_r} = \sqrt{\frac{P_1 - P_2}{P_v - P_2}}, \tag{18}$$

where: $\tau_b =$ vapour bubble growth time, τ_r= residence time, $P_1, P_2, P_v =$ upstream, downstream and vapour pressures respectively. Authors report very good agreement of mass flow prediction with collected results from static test stand.

2.5.1 Atomization and mixing

Key to optimal injector design is to fully understand jet forming accompanied by droplet atomization, vaporization and mixing. Waxman et al. conducted series of tests with carbon dioxide as NOx analog with high speed video in order to investigate atomization and mechanical breakup of NOx jet [31]. They noticed that use of impinging injector significantly increases atomization. This conclusion is not so obvious, as one may assume. Since nitrous oxide can be injected with flashing or non-flashing mode, which refer to rate of vaporization that can be modified by supercharging. In non-flashing mode, corresponding to supercharged NOx (or chamber pressure above saturation pressure) mechanical breakup of droplets dominate flash atomization. Waxman et al. also noticed correlation between L/D ratio of orifice and atomization performance. That fact somehow corresponds with previously discussed mass flow prediction model. Also experience from NOx used in hybrid rockets states that high L/D and small orifice diameter promotes vaporization and increases C_d.

Little has been done with other injector arrangements and influence of NOx jet on fuel jet and vice versa. Popular unlike-impinging and coaxial elements merit research. It is worth to mention, that on preliminary stage, hybrid (pintle type) injector was tested for Zawisza engine. Although initial cold flows promised good operation, engine was unable to attain proper combustion and exhibited highly unstable injection. Problem was not easy to be identified, due to limited resources. Cold flow tests were not performed in operating regime of pressure drop. Moreover, series of detrimental factors were discovered in the feed system, so it may not be entirely injector problem. Still, cause of this malfunction remains unknown and will be addressed in the future.

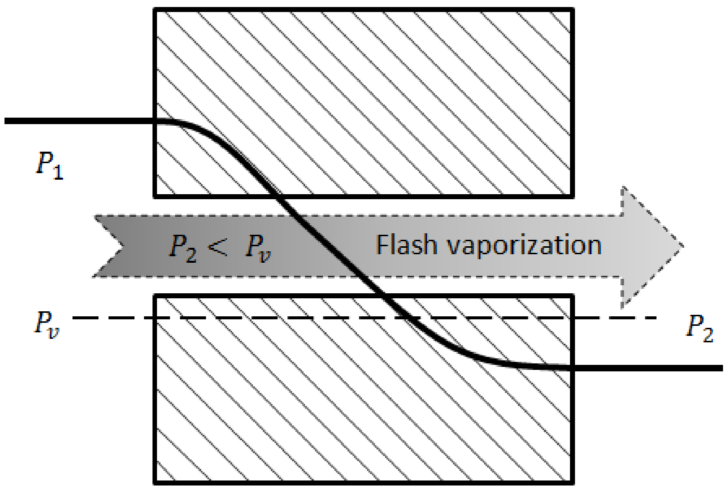


Figure 15: Two-phase flow of nitrous oxide through the injector orifice

3. Application

Previous chapters discussed properties of nitrous oxide, brought up problems with integration or compared with other propellants. Important part was also to propose solutions to specific problems and highlight advantages of NOx in particular situations.

Decomposition was mentioned before, but it is better to discuss it in terms of particular application. Similarly to hydrazine, nitrous oxide is able to decompose on various catalysts such as iridium, cobalt, gold, platinum and copper. Popular catalysts for hydrazine such as Shell 405 were successfully tested [32]. Exothermic reaction given by Eq. (10) can be used for ignition of another propellant. Decomposed mixture rich with free oxygen will ignite all sorts of fuel including hydrocarbons or even solid block of hybrid rocket [33]. Catalytic decomposition can be used in monopropellant or bi-propellant propulsion. Highly storable and compatible with common structural materials NOx is attractive propellant for long term space missions. Successful on orbit storage has been reported with no problems [34]. Non-toxicity reduces cost, safety precautions and improves integration. In the same time self-

DOI: 10.13009/EUCASS2017-474

pressurization can be leveraged to drastically simplify feed system and storage. Putting these factors together and basing on comparison with other space propellants made by Zakirov et al. [35], we can conclude following applications:

- low thrust bipropellant propulsion - i.e. upper stages
- storable military units
- small lifting rockets
- propulsion for small satellites – i.e. resistojets
- reaction control systems (RCS) – cold gas thrusters
- orbital maneuvering systems (OMS) – mono- or bi- propellant
- auxiliary power units – gas generators (APU)
- orbital station keeping

Although nitrous oxide as propellant offers lower performance, overall system efficiency (measured as payload mass on orbit or Δv for manoeuvres) can be better, thanks to significant decrease of complexity. Power savings can be made, for example with resistojets, when compared to traditional thermal decomposition. What is more, combining three applications, namely cold gas thruster, monopropellant and bipropellant, a multi-mode space application (Figure 16) can be achieved with even more simplification to space propulsion system. NOx is low-energetic propellant and due to low density it is rather not suitable for heavy lifting rockets. Self-pressurization however, may permit for efficient small lifters and upper stages. Also VaPak makes sense only with use of relatively low chamber pressure.

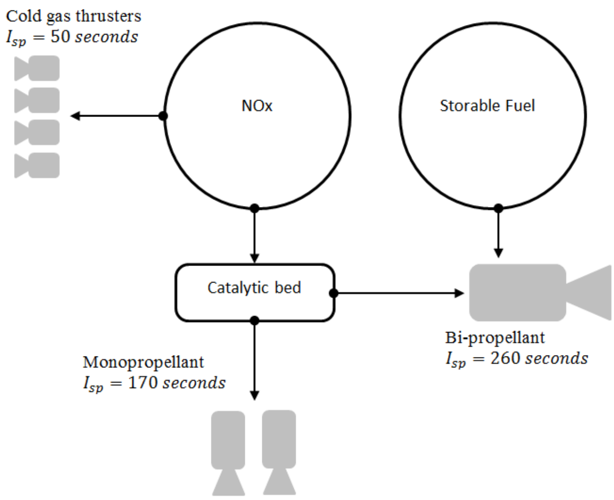


Figure 16: Multi-mode space propulsion system based on nitrous oxide

4. Conclusion

Nitrous oxide promises to be useful propellant in particular applications, especially for space propulsion. Although it does not offer best performance as other high-energy oxidizers, it may bring improvement for overall system. There are however, few issues that need to be investigated in order to make NOx applicable in liquid propulsion. Laboratory research followed by flight testing could result in efficient alternative for low thrust, multi-purpose propulsion. Next years may see increased interest in that propellant, especially in terms of commercialization of space linked with cost reduction and reusability.

Acknowledgements

The author would like to express appreciation to the people at AGH Space Systems, especially propulsion team members who worked on Zawisza project. Special thanks also goes to the Board and supporters of this association. This work was funded by President of AGH University of Science and Technology prof. Tadeusz Słomka. The author would also like to acknowledge Vice-President of AGH prof. Anna Siwik for support, without which this publication wouldn't be possible.

DOI: 10.13009/EUCASS2017-474

References

[1] Zimmerman, J. E., Waxman, B. S., Cantwell. B. J., 2013. “Review and Evaluation of Models for Self-Pressurizing Propellant Tank Dynamics.” *49th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*

[2] Zakirov, V., Sweeting, M., Goeman, V., Lawrence, T., 2000. “Surrey Research on Nitrous Oxide Catalytic Decomposition for Space Applications.” *14th AIAA/USU Conference on Small Satellites*

[3] Space Propulsion Group Inc., 2012. http://www.spg-corp.com/nitrous-oxide-safety.html

[4] Merill, C., 2008. Nitrous Oxide Explosive Hazards, Air Force Research Laboratory, http://www.dtic.mil/dtic/tr/fulltext/u2/a489459.pdf

[5] Daly, G., Knight, C., Mason, K., 2013. Scaled Composites Accident, http://www.knightsarrow.com/rockets/scaled-composites-accident/

[6] Atkins P.W., L. L. Jones, 1997. “Chemistry: Molecules, Matter, and Change.” W. H. Freeman and Company, New York

[7] Hill, P. G., Peterson, C. R., 1965. “Mechanics and Thermodynamics of Propulsion”, Addison-Wesley Publishing Company, Reading, Massachusetts

[8] Gordon, S., McBride, B. J., 1994. “Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Application: Analysis.” NASA

[9] Hill, P. G., Peterson, C. R., op. cit.

[10] Spalding, D. B., January 1959. “Combustion in Liquid-Fuel Rocket Motors.” *Aero Quarterly*

[11] Ibid.

[12] Altman, D., J. M. Carter, Penner S. S., Summerfield M., 1960. “Liquid Propellant Rockets.” Princeton, N. J.: Princeton Aeronautical Paperbacks, Princeton University Press

[13] Huzel, D. K., Huang, D. H., 1967. “Design of Liquid Propellant Rocket Engines.” NASA, Washington, D.C.

[14] Sutton, G. P., Biblarz, O., 2001. “Rocket Propulsion Elements.” John Wiley & Sons, New York

[15] LeBail F., Popp M., 1993. “Numerical analysis of high aspect ratio cooling passage flow and heat transfer.” *29th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*

[16] Pizzarelli, M., Nasuti F., Onofri M., 2013. “Trade-off analysis of high-aspect-ratio-cooling-channels for rocket engines.” *International Journal of Heat and Fluid Flow*

[17] Huzel, D. K., Huang, D. H., op. cit.

[18] Bartz, D. R., 1957. “A Simple Equation for Rapid Estimation of Rocket Nozzle Convective Heat Transfer Coefficients” *Jet Propulsion*

[19] Halchak, J. A., Cannon, J. L., Brown, C., 2016. “Materials for Liquid Propulsion Systems.” NASA Marshall Space Flight Center

[20] Lemieux, P., 2010. “Nitrous oxide cooling in hybrid rocket nozzles.” *Progress in Aerospace Sciences*

[21] Pavli, A. J., Curley, J. K., Masters, P. A., Schwartz, R. M., 1966. “Design and Cooling Performance of Dump-Cooled Rocket Engine.” NASA

[22] Huzel, D. K., Huang, D. H., op. cit.

[23] Ibid.

[24] Ibid.

[25] Pavli, A. J., op. cit.

[26] Ewig, R., 2009. “Vapor Pressurization (VaPak) Systems History, Concepts, and Applications.” Holder Consulting Group

[27] Calabro, M., 2011. “Overview in Hybrid Propulsion.” *Progress in Propulsion Physics 2*

[28] Ewig, R., 2009. “Vapor Pressurization …” op. cit.

[29] Ewig, R., 2009. “Active Control of Vapor Pressurization (VaPak) Systems.” Holder Consulting Group

[30] Dyer, J., Doran, E., Dunn, Z., Lohner, K., Zilliac, G., Cantwell, B., 2007. “Modeling Feed System Flow Physics for Self-Pressurizing Propellants.” *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*

[31] Waxman, B. S., Cantwell, B. J., Zilliac, G., 2012. “Effects of Injector Design and Impingement Techniques on the Atomization of Self-Pressurizing Oxidizers.” *48th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*

[32] Herdy, R., 2006. “Nitrous Oxide / Hydrocarbon Fuel Advanced Chemical Propulsion: DARPA Contract Overview.” *TFAWS*

[33] Zakirov, V., Sweeting, M., Sellers, J., Lawrence, T., 2001. “Nitrous Oxide as Rocket Propellant.” *Acta Astronautica*

[34] Internet: https://directory.eoportal.org/web/eoportal/satellite-missions/u/uosat-12

[35] Zakirov, V., Sweeting, M., Sellers, J., Lawrence, T., op. cit.