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I ABSTRACT

This paper describes the current developments regarding cryogenic rocket engine technology at Delft Aerospace Rocket Engineering (DARE). DARE is a student society based at Delft University of Technology with the goal of being the first student group in the world to launch a rocket into space. After the launch of the hybrid engine powered Stratos II+ sounding rocket in October 2015, DARE decided to investigate highly efficient liquid rocket engine technology . In this context DARE initiated the cryogenic project with the goal of developing a liquid rocket engine using liquid oxygen and liquid methane as propellants with a nominal thrust in the order of 10kN. Eventually this engine shall power a future sounding rocket into space.

As an intermediate step, a 3 kN class engine is being developed. Subsystem development tests and possibly a hot-fire test campaign on this engine are planned for 2016 and its development intends to provide DARE with the required experience and knowledge to develop large scale liquid rocket engines. The engine is developed in cooperation with Heliaq Advanced Engineering and is designed to meet the requirements of the second stage engine of the ALV reusable launch vehicle.

The design is a pressure-fed engine, regeneratively cooled using the liquid methane fuel. After passing the coolant channels, the methane is injected into the combustion chamber in gaseous state together with the liquid oxygen in a co-axial manner. The engine is ignited by means of a pyrotechnic igniter using an ammonium perchlorate based propellant. During a test sequence, the propellants are stored in insulated run-tanks and are pressurized using helium.

This paper describes the project objective, the current progress on the design and production, and finally four proposed research topics that are indented to be conducted at the faculty of Aerospace Engineering of Delft University of Technology.

II INTRODUCTION

Delft Aerospace Rocket Engineering (DARE) is a student society of Delft University of Technology in the Netherlands. DARE has the goal to allow its students to do practical work on rocket related technology next to their studies. In this way the society adds significantly to the educational tract of the students where it concerns rocket engines and rocket subsystems. Within the society, work is conducted on all three types of chemical rocket propulsion: solid, hybrid and liquid. Furthermore there is experience on electronic systems, recovery mechanisms, and active stabilization of the rocket in flight. The current progress of the society culminated in the launch of the Stratos II+ rocket in October 2015 to the European record of 21.5 km altitude for student rocketry.

DARE has the goal to eventually reach the edge of space at 100 km. To achieve this a high efficiency propulsion system is needed, and therefore the development of a cryogenic liquid rocket engine has started. The intended goal is to use liquid oxygen and liquid methane in a pressure-fed, regeneratively cooled engine. The methane is used as the coolant fluid. The project uses the experience gathered by DARE members during the Stratos II project and builds upon the lessons learned, and uses (test) equipment acquired during that project. The project objective and its connection to Stratos III and industry is explained further in section III.

The project started in 2015 and its members have up till now developed conceptual and detailed designs for a number of subsystems, while for others hardware has already been produced. The hardware consists of a cryogenic propellant tank, while detailed design has been done on the rocket engine cooling channels, the test setup feed system design, and the igniter design. This research is elaborated upon in section IV.

To give the project a good scientific basis it is proposed that a number of research projects are conducted at Delft University in conjunction with DARE, and possibly with industry. These research projects are intended to take the form of Master of Science thesis projects that focus on different aspects of the rocket engine. The proposed topics are: (1) Modeling of liquid rocket engine instabilities and design to avoid these instabilities. (2) Design and modeling of the injector and studying its relation to combustion instabilities. (3) Modeling and validation by testing of the cryogenic propellant tank pressurization system and pressurant injector design and (4) the modeling and validation of a torch igniter system. All of these proposed research topics are explained in section V.

III PROJECT OBJECTIVE

III.I Liquid Rocket Development within DARE

On the 16th of October 2015, the Stratos II+ sounding rocket broke the European Altitude Record with its launch from the CEDEA Test Range in the south of Spain. Despite the enormous success of this achievement, the rocket's apogee of 21.5 km fell significantly short of the missions goal of flying halfway to space with a target altitude of 50 km. Amongst other issues, this shortfall in apogee underlines the need for Delft Aerospace Rocket Engineering (DARE) to develop a more efficient and powerful propulsion system in order to achieve its goal of reaching space. Previous DARE rockets have been powered by low performance solid rocket motors, with an average sea level specific impulse of 100 s or, in the case of Stratos II+ by medium performing hybrid rocket motors, with an average sea level specific impulse of 185 s. Within DARE however, knowledge and experience with liquid rocket engines is very limited [14]. The most significant success with liquid rocket systems within DARE was the successful static test campaign of the Deimos engine in 2013 [5]. The Deimos engine, fueled by nitrous oxide and ethane, however produced insufficient thrust and achieved an average specific impulse of only 175 s. Furthermore, the system proved to be unsuitable to be adapted in a flight system due to its low thrust to weight ratio. No significant progress in the field of liquid rocket propulsion systems has been achieved within DARE since then.

To improve on this situation, a new project was initiated in November 2015 with the goal to develop and statically demonstrate a high performance LRE with a thrust level comparable to that of the Stratos II+ rocket. The intended design thrust level will be between 10 and 15 kN (sea level). To increase the practical value of the engine, in the context of space access, the engine shall utilize a propellant combination of liquid oxygen and liquid methane. Liquid oxygen offers a significantly increased specific impulse and density compared to nitrous oxide, previously used by DARE [18]. However, liquid oxygen offers significant challenges due to its reactivity and cryogenic properties. As such a main goal of the project is to gain practical experience in the design, logistics and operations of liquid oxygen systems. In this context, cooperation with industry experts such as TNO and Airliquide has been initiated.

Liquid methane is a non-toxic cryogenic fuel. It offers a higher specific impulse compared to kerosene, whereas it is easier to store than hydrogen. Perhaps its most promising merit is its relatively high coking temperature of 950 K, compared to the coking temperature of kerosene of 560 K. The coking temperature is defined as the temperature at which the fuel molecules start to polymerize, forming longer polymer chains, resulting in a more viscous liquid that might cause deposits on the inside of the fluid lines. This high coking temperature prevents carbon deposits to form in the regenerative cooling channels, which are typically observed in kerosene fueled rockets. This makes liquid methane very suitable for reusable rocket engines [4]. As such, DARE wants to utilize liquid methane as fuel for the cryogenic liquid rocket engine project. Due to the limited availability and relatively high cost of purified methane, liquified natural gas (LNG) is considered as the preferred fuel.

After an initial feasibility study, it was found that the cost and complexity involved in the logistics of both liquid methane and LNG in relatively small quantities (meaning less then 1000 kg) are excessively high. As such, it is very likely that initial hot fire tests in the context of the cryogenic liquid engine project of DARE will be conducted with an alternative hydrocarbon fuel such as refined kerosene, propane or ethyl-alcohol. Of these alternative fuels, refined kerosene offers higher efficiency. However from a logistics perspective, ethyl alcohol is more straightforward to acquire. As such, it is likely that the initial hot-fire-tests will be performed with a propellant combination of liquid oxygen and ethyl-alcohol. To further reduce the risks involved in initial hot fire testing, an engine with reduced thrust will be tested first. The thrust of this initial demonstrator engine has been selected to be 3 kN (sea level).

III.II Application in the ALV-2 Launch Vehicle

The Austral Launch Vehicle (ALV) is an international project with the goal of developing a cost optimized reusable launch vehicle. The basic concept behind the ALV is the clustering of fly-back boosters with fold-able wings. A schematic mission overview is shown in figure 1. Initial studies showed that this concept has the potential to provide a modular, simple and flexible small satellite launch vehicle [25]. The first stage of a launch vehicle is far larger than the upper stages (around 75% of the launch mass) and is also much easier to recover due to the relatively low speed. The use of flyback boosters (using wings and an air breathing engine) further limits firing of the rocket engines to the ascent, saving as much of this valuable resource for future missions as possible. The focus of the ALV project is the development of the ALV-2 small satellite launcher. This vehicle is aimed directly for small satellites with a mass ranging between 1 and 10 kg. The second stage of the ALV-2 vehicle produces a thrust of 3 kN. As such, the first demonstrator engine developed by DARE is comparable to the ALV-2 second stage and it is envisioned that the ALV-2 second stage engine will be a derivation from the DARE cryogenic demonstrator engine.

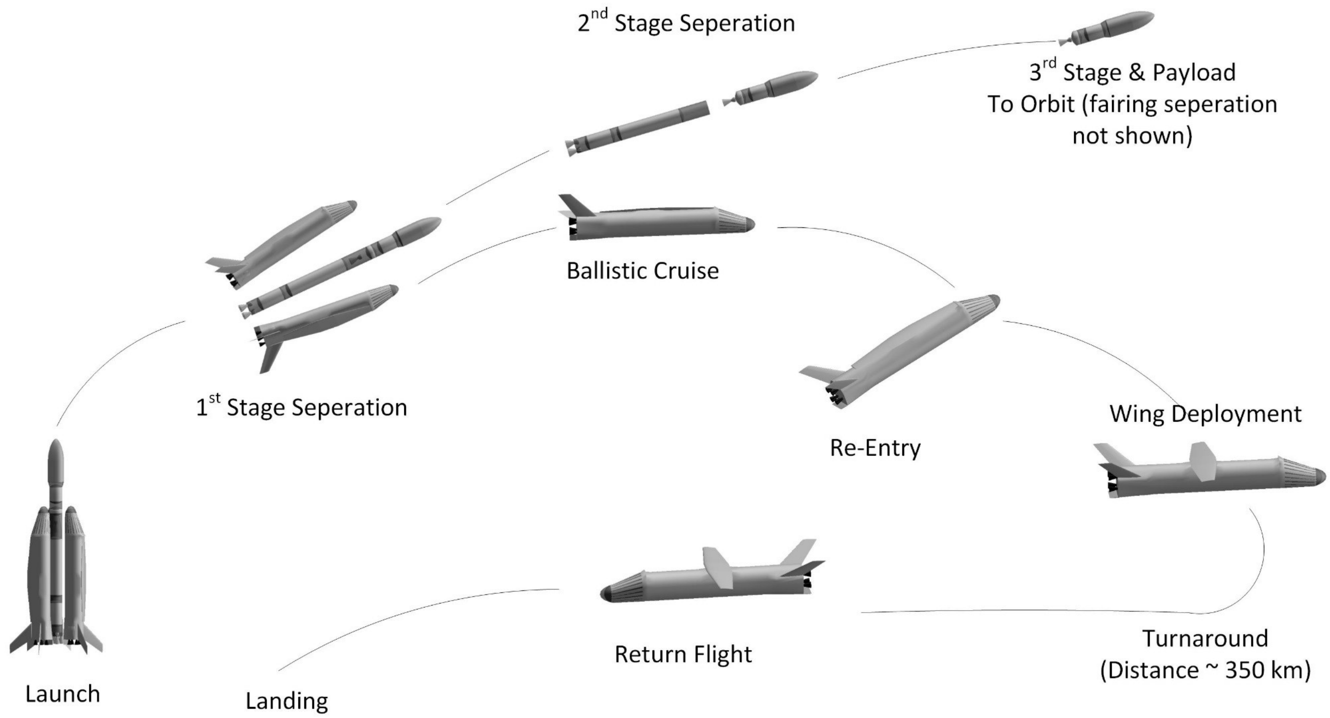

Figure 1: Schematic overview of the ALV-2 Launch Vehicle mission concept.

IV CURRENT PROGRESS

IV.I Run Tank Development

Initial investigations on the liquid oxygen tank have been performed. The tank has been designed as a cylinder with simple flat bulkheads, made of aluminum AL-6082 T6 for an operating pressure of 55 bar. The tank material was selected out of a range of cryogen compatible materials mainly because of its of-the-shelf availability in the desired diameter (250mm). Two concepts of how to fix the bulkheads to the cylinder wall where developed. The first concept featured an arc-weld connection between the tank wall and the bulkheads. The second concept featured a radial bolt connection with the sealing achieved by means of a single Fluorinated Ethylene Propylene (FEP) covered spring loaded o-ring. In the arc-weld design, sealing was achieved with the weld connection. For both designs, significant production constraints were imposed. Due to the limited production facilities that were available, the wall thickness was constrained to 4 mm. Due to the same limited production techniques, the tank bulkheads were also designed with a significant safety margin. A test run-tank, with a volume of 40L has been designed for both concepts. The resulting mass estimates, predicted from the CAD software, are 12.8 kg for the arc-weld design and 12.9 kg for the bolted design. The welded tank has already been produced with help from Linde gas, who performed the welding. Construction of the bulkhead design has been started but has not yet been completed. The possibilities of a composite tank have also been investigated. To mitigate the risk of fiber-matrix delamination due to thermal stresses, a glass fiber dry wound concept was selected. The required layup pattern was determined using the PresVes tool of the Delft based company Advanced Lightweight Engineering. PresVes takes the tank shell structure and defines a winding pattern for a given burst pressure and polar opening area. Dry Fiber winding is a technique where filaments are placed on the tank liner such that when the tank is loaded with internal pressure, all filaments are loaded in tension only. By making use of friction forces, no matrix is needed. PresVes finds the correct contour for this technique while making sure that it can be manufactured with a tumble winder. The composite design under consideration has a liner of aluminum 6082-T6 with a wall thickness of 2 mm and uses E10 glass fibers wrapped around the liner to provide the strength needed to withstand the pressure loads. A Finite Element model of the tank was created, which modeled both the structural and thermal properties of the tank when loaded by a pressure and a thermal load. The resulting mass estimate for a 40L tank was 1.8 kg and hence significantly lower compared to the aluminum designs. However, thermal effects, such as potential delamination of the fibers from the matrix or cracking of the liner, still need to be evaluated. As such, it is at this moment not clear if the composite design is a feasible option.

IV.II Cooling Channel Design Tool

A preliminary analysis was performed on the design of the regenerative cooling channels. Two numerical models have been developed to analyse the design: A simple and fast one-dimensional model developed in Python, and an extensive analysis using the CFD library OpenFOAM. The one-dimensional analysis tool is called OMECA (One-dimensional Methane Engine Cooling Analysis) and has been made publicly available [9]. The CFD analysis using OpenFOAM has also resulted in additions to the OpenFOAM project, such as a library to interpolate fluid properties at runtime [10]. For an extensive discussion of the CFD analysis you are refered to the thesis and paper of L. Denies [11, 12].

The OMECA code calculates the thermal equilibrium at many axial stations in the nozzle. Semi-empirical relations are used to estimate the convective heat transfer coefficients of the hot combustion gases and the coolant. These are the Bartz equation given in equation 1 described in [1] and the Taylor relation, given in equation 3 described in [31], respectively.

$$\alpha_h = \frac{0.026}{D_t^{0.2}} \frac{\mu_0^{0.2} C_{p,0}}{\mathrm{Pr}_0^{0.6}} \left(\frac{P_0}{c^*}\right)^{0.8} \left(\frac{A_t}{A}\right)^{0.9} s \tag{1}$$

With s being defined as:

$$s = \left[\frac{1}{2}\frac{T_{t,h}}{T_0}\left(1+\frac{\gamma-1}{2}M^2\right)+\frac{1}{2}\right]^{-0.68}\left[1+\frac{\gamma-1}{2}M^2\right]^{-0.12} \tag{2}$$

$$\mathrm{Nu}_b = 0.023\mathrm{Re}_b^{0.8}\mathrm{Pr}_b^{0.4}\left(\frac{T_b}{T_t}\right)^{(0.57-1.59D/x)} \tag{3}$$

The methane in the cooling channels is assumed to have uniform bulk properties at each station. The thermodynamic and transport properties are modeled with the GERG-2004 equation of state [19].

A preliminary design of the cooling channels for the 3 kN demonstrator engine has been made. From the thrust requirement, a methane mass flow of 0.2712 kg/s can be derived. This flow is available for regenerative cooling. As reference material, NARloy-Z was taken to represent a high-temperature copper alloy. Its properties were taken from [15]. NARloy-Z has a temperature limit of approximately 800 K, above which it loses structural strength rapidly. For this reason, a design was sought which limited the maximum wall temperature to 800 K. Straight channels in the axial direction were analyzed for reasons of simplicity. In the interest of re-usability, a chamber-saddle-jacket design was investigated. In such a design, the chamber, throat saddle and outer jacket are separate parts that are not rigidly connected. This allows for thermal expansion and eliminates or reduces fatigue constraints. For a copper chamber, a chamber wall thickness of 3 mm is sufficient to withstand the 50 bar inward pressure of the coolant. Figure 2 shows the heat flux, wall temperature and geometry of a possible regenerative cooling design. The solid black line shows the thrust chamber contour, while the dotted-dashed line shows the incoming heat flux estimated with the Bartz equation. 44 cooling channels are spread around the engine, varying in width so as to keep the rib thickness at 1 mm. The blue dotted line shows the variation in channel depth that is necessary to keep the wall temperature (in green) below the limit of 800 K.

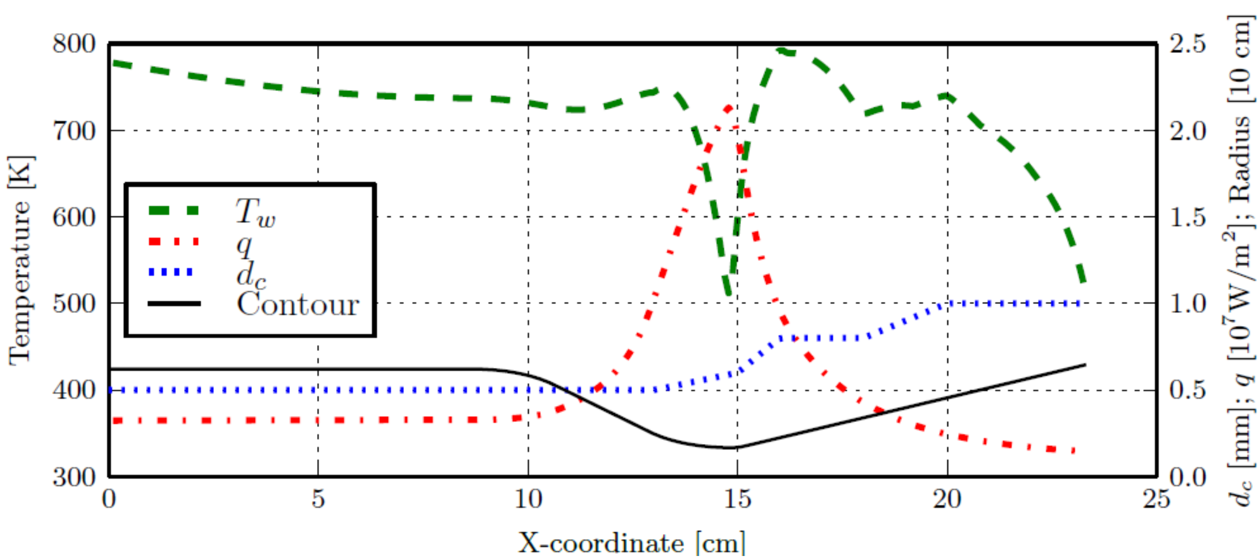

Figure 2: Chamber Contour, Channel Depth, Heatflux and Wall Temperature of the described cooling design

This analysis shows that it is possible to keep the copper chamber wall below the allowable temperature with the given methane coolant flow. It further shows that a decrease in channel depth to 0.5 mm is necessary near the combustion chamber. This is due to the heating up of the coolant as it flows from the nozzle towards the combustion chamber. The sharp drop in temperature near the throat is due to the decrease in radius and associated decrease in cooling channel hydraulic diameter. This increases the coolant velocity and therefore the convective cooling.

The high outlet temperature illustrates the difficulty of performing regenerative cooling for a small liquid rocket engine. This difficulty is due to the relatively large surface area with respect to the amount of coolant. In addition to the copper chamber, it was investigated whether it was feasible to create an aluminum chamber design. Due to the low allowable temperature of the latter (around 500 K) and the small engine size, this was deemed not feasible. Since the methane rises to approximately this temperature while cooling the engine, it is not adequate to keep the aluminum sufficiently cool.

IV.III Test Setup Feed System Design

Within the project a feed system design has been drawn up for the ground test setup of the cryogenic engine. This first design is shown in figure 3. It is a schematic feed-system design made to determine the kind and amount of components required, and to determine how they need to be connected. Based on this design the acquisition of feed system parts has been started.

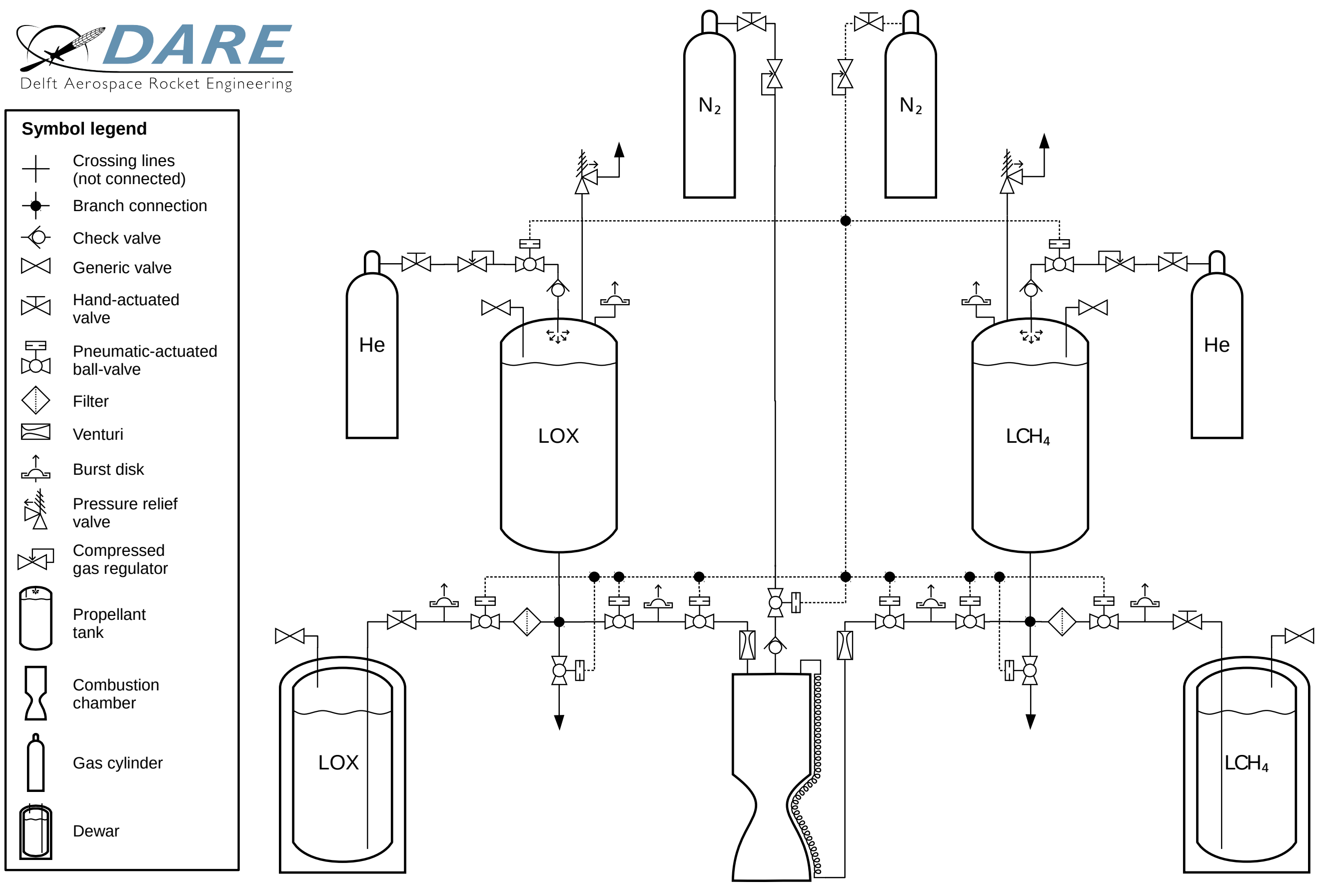

Figure 3: Schematic design for the DARE ground test setup cryogenic feed system. The connections to a dump-line for all pressure relief and dump ports, as well as the placement of sensors, are not indicated in this diagram.

The whole feed system has a symmetric build-up with the LOX on one side and LCH4 on the other. The lines for both systems are virtually the same, with the exception that the methane is led through the cooling channels of the chamber. Next to these two propellant lines there is one high pressure nitrogen purge line going into the combustion chamber for post-burn purging. A low pressure nitrogen line will be used for pneumatic valve actuation.

Both propellants will be stored in a cryogenic dewar prior to testing. Filling of the run-tanks will be done by opening the valves between dewar and run-tank and opening the vent-valve on top of the propellant tank. Within the line a filter is placed to prevent particulates entering the system from the dewar side. Level control of the propellant tank can be done by mass measurement of the tank and possibly the use of a dip-tube. Other methods of liquid level measuring such as thermal sensors and optical sensors are however being investigated as the mass measurement may be inaccurate, and the dip-tube method is not preferred as it would mean liquid oxygen would flow via the dump line.

Pressurization of both propellant tanks will be done using gaseous helium supplied from standard industrial gas cylinders. The usage of nitrogen as pressurant in ground tests has been discarded as nitrogen condenses into the LOX and dissolves into the LCH4 [23]. Often the added cost of helium versus nitrogen is cited as reason to use nitrogen in ground tests, however the cost difference between them has been deemed insignificant, especially when compared to the costs of the full system and the propellants.

Both main lines are equipped with two main valves in series. One of these will act as the main valve during regular operations, while the second is there to provide redundancy in an emergency, when the feed system needs to be closed. All sections of the line that can entrap cryogenic propellant, such as between these two valves, are equipped with a pressure relief system.

Both lines include a cavitating venturi as last element before the combustion chamber. These elements have been placed there to provide a simple way to stabilize the mass flow into the chamber, and as a precautionary measure to decouple the feed system from combustion chamber instabilities.

At the moment of writing, no decision on the exact sensors has been made. However, in order to fully characterize the behavior of the fluids in the feed system and the performance of the engine, a number of digital pressure transducers will be used. Critical locations of these pressure transducers are on the two run tanks, on the combustion chamber, on the cooling jacket inlet and on the injector inlet of both the liquid oxygen dome and the fuel manifold. Mass flow measurements of the propellants will most likely be obtained via turbine type flow meters. An alternative mass flow measurement could come from pressure sensors measuring the static pressure in the cavitating venturi inlets and throats. Additionally, averaged mass flow measurements will be obtained by suspending the run tanks on load cells.

IV.III Injector Element Design

Due to its high mixing efficiency for both liquid-liquid and liquid-gaseous injector schemes, the swirl coaxial element is selected [17]. The design of the injector element is based on the guidelines published by Bazarov [2]. In this approach, the pressure drop over the entire element is determined via the principle of maximum flow for swirling liquids. As described, the massflow of fuel and oxidizer in the feedsystem is controlled via cavitating venturies. As such, the injector elements do not need to be sized to yield a specific mixture ratio and only need to have sufficient pressure drop for combustion stability. An overview of the injector element design parameters is presented in table 1.

Table 1: Injector Element Design Input

Parameter	Unit	Value
Total engine thrust	kN	3
Combustion pressure	bar	40
Number of elements	#	6
Thrust per element	N	500
Estimated Specific Impulse	s	280
Oxidizer flow rate	g/s	136.8
Fuel flow rate	g/s	45.2

Using reference data based on the RD-0110 engine, a liquid oxygen injector velocity of 5 m/s has been selected at suitable starting value [24]. This injection velocity, in combination with a pressure drop over the injector element of 6 bar, results in a required discharge coefficient C_d of 0.154. With the discharge coefficient of the swirl chamber determined, Other parameters can be obtained via figure 4.

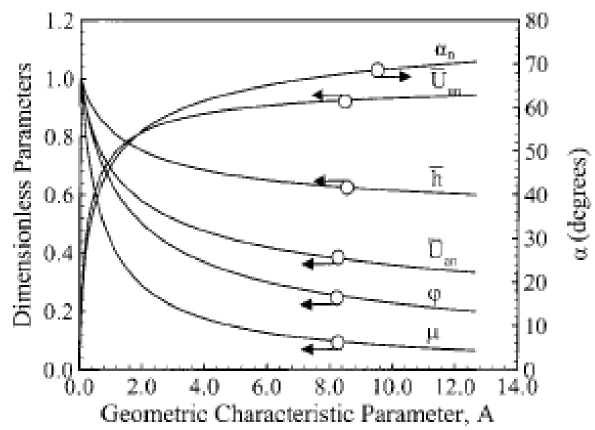

Figure 4: Swirl Injector half fan angle α, discharge coefficient μ, liquid film ratio ϕ, nozzle swirl velocity $\bar{U_{un}}$ and Nozzle Axial Velocity $\bar{U_{an}}$ as function of characteristic coefficient A. Obtained from [2]

In order to allow for a total massflow of 136.8 g of liquid oxygen, yields an oxygen outlet nozzle radius of 2.76 mm. With a discharge coefficient of 1.54, Bazarov presents a prediction of geometric characteristic coefficient A equal to 6, with A being defined as:

$$A = \frac{A_n R_{in}}{A_{in} R_n} \tag{4}$$

With A_n being the total area of the injector exit nozzle, R_{in} the radial location of the center of inlet passages, A_{in} the total area of the inlet passages and R_n nozzle exit radius. The radial location of the center of the inlet passages is determined by:

$$R_{in} = \bar{R_n} R_n \tag{5}$$

Where Bazarov suggest a value of $\bar{R_n}$ of 0.8 for open combustion cycles (pressure fed and gas generator cycles). Using the value of 0.8, the radial position becomes equal to 2.21 mm. Initially, a total of 3 tangential inlet channels are selected.This yields an inlet channel radius is equal to 0.582 mm. Bazarov proposes an l/d ratio of the tangential inlet channels of 2, yielding a tangential inlet channel length of 2.32 mm. Furthermore, Bazarov recommends a length of the vortex chamber to be equal to 2 times the radial location of the inlet passages and a nozzle length equal to the nozzle diameter, yielding a vortex chamber length of 4.42 mm and a nozzle length equal to 5.52 mm.

For the gaseous methane stage of the injector, a total massflow of 45 g/s and a similar pressure drop of 6 bar is required. Following the same procedure, and taking into account a wall thickness of 0.5 mm between the oxygen and the methane post, we get a required discharge coefficient of 0.151, yielding a characteristic geometric coefficient of $A = 6$. Following this procedure yields a fuel tangential inlet radius of 1.06 mm (for three channels), the tangential inlet radial location distance of 2.484 mm and a fuel post length of 5.52 mm.

IV.IV Pyrotechnic Igniter Design

To mitigate the risk of a hard-start during the initial tests, a robust pyrotechnic igniter will be utilized. To avoid the necessity for students to produce the energetic pyrotechnic propellant, it has been decided to acquire commercially produced propellant grains. In order to ensure ignition and to design for maximum reliability it is decided to set the requirement that the igniter should be able to heat the complete propellant mass flow above its auto ignition temperature. The work of Steward [28] and Younglove [34] is used to calculate the propellant heat capacity at various temperatures after which these values are numerically integrated to obtain the total required energy to heat the propellants. As the methane fuel will be used to regeneratively cool the engine prior to being injected into the combustion chamber, it is assumed that the methane will have vaporised once it is injected into the chamber. This assumption may however not be valid at engine start, due to the fact that there is no combustion in the main chamber yet and that the engine is likely to be prechilled prior to start. For ignition it is thus assumed that the methane is still a liquid. The liquid oxygen will still be liquid and needs to undergo a phase change. The properties of the propellants at injection are summarized in Table 2.

In order to determine the temperature to which the propellants need to be raised at which auto-ignition occurs the work of Shchemelev [26] is used. Shchemelev reports that oxygen-methane mixtures readily self ignite at temperatures above 500 degrees Celsius at atmospheric pressure. At elevated pressures the auto ignition temperature tends to decrease [32], but the exact mixture ratio at ignition is uncertain. Therefore the required temperature of the propellants for ignition is taken to be equal to 800 K as a safe first estimate.

Table 2: Propellant properties at the injector face

Parameter	Unit	Value
Oxidiser injection temperature	K	90
Oxidiser injection phase	-	liquid
Fuel injection temperature	K	112
Fuel injection phase	-	liquid

In order to determine the energy delivered per unit of mass by APCP propellant, the enthalpy change resulting from the combustion can be determined using a chemical equilibrium software tool like RPA or NASA's CEA. Because the exact composition of the commercially available propellant grains is unknown it is however not easy to obtain the enthalpy change. From the propellant safety data sheets of the Cesaroni Pro-X grains [6] a composition of 85% APCP and 15% HTPB is deducted, yielding an O/F ratio of 5.67. The igniter heating power P_i above the auto ignition temperature of 800 K is then determined using the equation:

$$P_i = \dot{m}_i (H_i - H_{ref}) \tag{6}$$

Where H_i is the enthalpy of the gas leaving the igniter in $\mathrm{kJkg^{-1}}$, H_{ref} is the enthalpy of the gases at the reference temperature in $\mathrm{kJkg^{-1}}$ and $\dot{m}_i$ is the mass flow of gas leaving the igniter in $\mathrm{kgs^{-1}}$. Using RPA the chemical composition and temperature of the flow exiting the igniter nozzle into the main combustion chamber is calculated. For the igniter an initial chamber pressure of 1.5 MPa is assumed. For the igniter exhaust nozzle an area ratio $\frac{A_e}{A_t}$ of 1.1 is assumed. Using these values for the given APCP composition in RPA yields a nozzle exhaust temperature gas temperature of 2385 K. Taking the auto ignition temperature as the reference temperature and working out equation 6 for all the major species in the APCP grain yields the results as shown in Table 3.

Table 3: AP + HTPB propellant heating power above 800 K from 2400 K from major gas species. Pc = 1.5 MPa, O/F = 5.67, $\frac{At}{Ae}$ = 1.1

Species	ΔH [kJ/mol]	Molar fraction	ΔH [kJ/kg]
H2O	75.74	0.3433	1442
CO2	92.99	0.1012	213
HCl	54.52	0.1782	266
CO	56.15	0.1780	356
H2	52.23	0.1030	2668
N2	55.59	0.0903	179
Total		0.994	5127

Naturally not all energy will be available to heat the main propellants. Some heat will be lost in the igniter itself but also a significant amount of energy is converted to velocity as the gases exit the igniter housing and flow into the main combustion chamber. The amount of energy converted into velocity is however expected to quickly change as the main combustion chamber is brought up to pressure once propellant injection starts. It is thus very hard to peg a solid number to the amount of energy available without performing detailed analysis.

By dividing the total energy required by the energy delivered by the propellant, the required mass flow out of the igniter is obtained. The results are summarized in Table 4.

Table 4: Required igniter power and mass flow

Parameter	Unit	Value
Required ignition power	W	$312\dot{1}0^3$
APCP propellant enthalpy	Jkg^{-1}	$5.127\dot{1}0^6$
Required mass flow	kgs^{-1}	0.0608

Based on the required mass flow and the duration of about 1 second for which this mass flow needs to be delivered, the Cesaroni Pro38 Vmax propellant grain was selected. Cesaroni Vmax propellant grains are readily available in the Netherlands. The Vmax propellant has a high specific impulse and burns relatively clean due to the lack of metal additives. This makes the Vmax propellant suitable for the pyrotechnic igniter. Key properties of this propellant grain type are listed in Table 5.

Table 5: Cesaroni Pro-38 Vmax propellant grain key properties as provided by Cesaroni. Source: Cesaroni [30]

Parameter	Unit	Value
Propellant weight (sic)	g	61.2
Burntime	s	0.69
Outer diameter	mm	31
Outer diameter incl. liner	mm	33
Inner diameter	mm	11
Length	mm	58

The igniter is modeled using ideal rocketry theory [29]. The igniter is connected to the main combustion chamber via a convergent-divergent nozzle. The mass flow due to combustion and the mass flow out of the engine are determined as functions of pressure. By setting up a mass balance and by applying time discretization, the contained mass and hence pressure of the engine are determined. Under the assumptions of ideal rocketry the mass flow $\dot{m}$ in kg/s through a nozzle with a throat area A_t in m^2 is given by the relation:

$$\dot{m} = \frac{P_c A_t}{c^*} \tag{7}$$

Where P_c is the chamber pressure in Pa and c^* the characteristic exhaust velocity of the propellant in $\frac{m}{s}$. It is assumed that the burn rate or regression rate r of the APCP propellant is a function of pressure p and can be approximated using the following equation:

$$r = ap^n \tag{8}$$

The power function coefficient n typically has a value between 0.3 and 0.5 [29]. Cesaroni was able to provide a theoretical characteristic exhaust velocity, chamber temperature and regression power function coefficient for the Pro38 Vmax grain on the condition that these figures are not made public. Because limited information is available on the exact propellant composition some assumptions are made in order to accommodate these deficiencies. It is assumed that both the chamber temperature and characteristic exhaust velocity are fixed at the supplied values for the duration of the burn.

Using the described igniter model the following results shown in Table 6 and Figures 5 and 6 were obtained. With the throat diameter being the primary design variable, its value was chosen to allow for a slightly higher mass flow than required to accommodate for thermodynamic losses and a slow start-up.

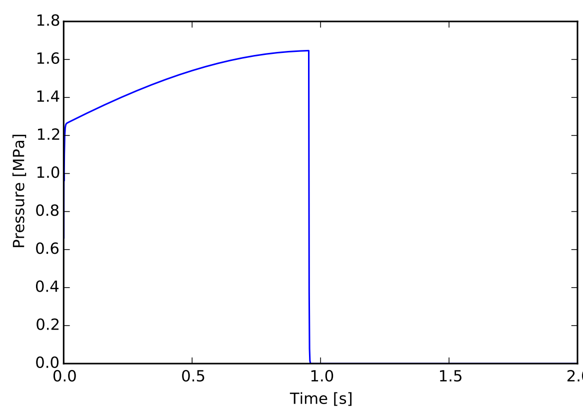

Figure 5: Igniter chamber pressure

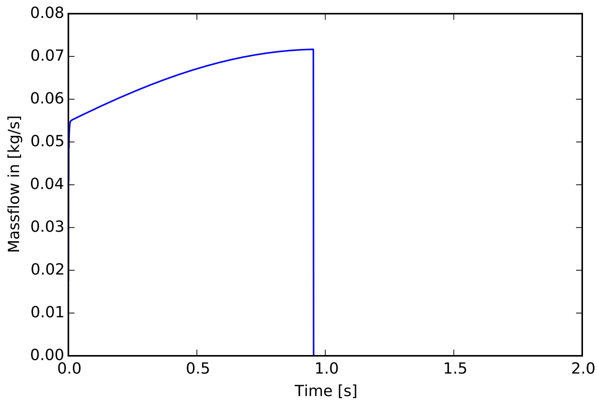

Figure 6: Igniter nozzle mass flow

Table 6: Igniter model simulation results

Parameter	Unit	Value
Maximum pressure	MPa	1.646
Throat diameter	mm	9
Maximum massflow	kg/s	0.0717
Average massflow	kg/s	0.0652
Burntime	s	0.959

The use of ideal rocket theory neglects pressure losses occurring due to boundary layers, chemical kinetics and two phase flows. Furthermore it is difficult to predict the exact start-up transients. These facts, added to the uncertainty of the exact chemical composition of the Pro38 Vmax propellant, make the validity of the predictions doubtful. As such, it was chosen to design the igniter to provide sufficient power to ignite the full propellant flow. Despite this safety factor, a number of tests are currently planned for the igniter to validate that it delivers the required massflow before it will be used to ignite an engine.

In addition a detailed analysis of the effects of startup on the igniter flow are required. While simulations predict that the igniter on its own will only reach an operating pressure of 1.646 MPa, the main chamber pressure will rise to its designed operating point around 4 MPa once ignition is achieved. While the low power function coefficient n for the burning rate of the APCP should prevent any adverse effects from an increased igniter chamber pressure the exact influence of an elevated back pressure on the igniter is to be determined.

V PROPOSED RESEARCH TOPICS

V.I Combustion Stability Modeling

Combustion instabilities are an important engine phenomenon to be avoided in the development program of any liquid rocket engine. The pressure and heat release oscillations associated with combustion instabilities lead to a significant increase in heat transfer to the combustion chamber structure. This occurs because these oscillations destroying the boundary layer which normally exists in a stable combustion environment. The increase in heat release often leads to structural damage and even a complete failure of the engine.

Theoretical analysis of combustion instabilities is often lacking in the development process of a liquid rocket engine due to the complex nature of combustion instabilities. This lack in knowledge often leads to destructive instabilities appearing late in the engine development stage. To prevent this, a one-dimensional high frequency combustion stability model will be developed. This model will be able to give predictions about the stability of the given engine design and provide the designer with suggestions as to which parameters will provide better stability.

For the construction of this model, several stability analysis methods which are used in the past will be analyzed for their applicability. The first method is the time lag model developed by Crocco [7]. The time lag model assumes that there is a finite time lag between when the propellants are injected and when they are combusted. The acoustic model describes the propagation of acoustic pressure waves in a given chamber geometry and the acoustic mode frequencies of the chamber. This model was first applied to describe combustion instabilities by Lord Rayleigh [22] when he describes the fundamental mechanisms behind the driving of an instability. Implementing empirical relations which describe the combustion response of individual combustion processes will allow for a more in-depth analysis of the driving mechanisms behind an instability. However, The applicability of these empirical relations is limited.

The one dimensional model will be validated by comparing the results to existing one dimensional programs like ROCCID [20] and GIM [21]. A further increase in fidelity will be reached when the individual parts of the model are compared to test data. Parameters like the injection droplet size and distribution, propellant evaporation rate and propellant mixing rates can be confirmed by the execution of cold flow and hot fire tests.

V.II Injector and Thrust chamber Design

The coaxial swirl injector has been selected as preferred element due to its high efficiency for liquid-gas configurations, as well as liquid-liquid injection configuration. An initial design for the injector elements, in the liquid oxygen-gaseous methane case, has been made using the methodology proposed by Bazarov [2]. However, many design parameters, including propellant velocity and swirl ratio, intra element mixing length and inter element spacing are not properly addressed in the initial design. To evaluate these design parameters, a Reynolds Averaged Navier Stokes model will be developed. Currently DARE has a RANS code for numerical combustion implemented in ANSYS CFX. This code however has numerous shortcomings. In its current state, the code uses single step chemistry, yielding excessively high temperature predictions. Furthermore it assumes gaseous injection of both propellants and no real-gas effects are simulated. As such it is advised to upgrade this model by implementing a real gas equation of state, multi-step chemistry and to include liquid injection and vaporization model. The improved numerical combustion model should be utilized to optimize the injector geometry. Parameters such as oxidizer post recess, injection velocity and injection velocity ratio should be optimized for the injector elements. It is planned to perform cold flow tests on the injector using liquid nitrogen as analogue for the liquid oxygen and gaseous nitrogen as analog for the methane. These tests should be performed to validate that the injector has a sufficient pressure drop. Furthermore these tests shall be used to validate the liquid spray parameters, such as Sauter mean diameter, utilized in the RANS model.

V.III Tank Pressurization Design

For the design of the liquid propulsion system a choice has been made to focus on pressure driven expulsion of the propellant from the tanks. This option was selected because at the scale of the engine it does not yet lead to insurmountable increases in tank mass, while the development of a pump system would need to start from scratch within DARE.

To pressurize the tank and to expel the propellant from it, a pressurant gas needs to be introduced into the tank. There are various ways to do this, but the most common method is the usage of a pressurization gas such as helium stored under high pressure in an on-board tank [29]. When looking at the system under consideration the propellant tanks would operate at pressures around 5MPa or possibly even higher. Preliminary calculations show that this means that the pressurant tank is comparable in size to a propellant tank. A good knowledge on the amount of pressurant required and a possible mass optimization of it is therefor key to reducing the empty mass of the engine system.

The pressurant gas is sprayed into the propellant tank via a pressurant diffuser. As is already discussed by Huzel [18], the most common shape of the injector is that of a radial diffuser, which injects pressurant in such a way that disturbance of the propellant surface by the incoming gas is minimized. The idea is that this minimizes heat transfer from the gas to the propellant and that it as such minimizes the amount of pressurant required. The radial injector produces a steady inflow of gas with a large vertical temperature gradient in the ullage gas, but negligible temperature gradient in the radial direction [8].

In a NASA technical note from DeWitt, Stochl and Johnson on the effectiveness of different pressurant injectors from 1966 [13] the effect of using different pressurant injectors is discussed and backed up with experiments. This paper is of particular interest for this project as involves a tank of similar scale as what can be expected for DARE. The tank they used was 0.68m diameter and 2m high. Tests were conducted with liquid hydrogen as propellant and gaseous hydrogen as pressurant. Their results indicated that a straight tube injector resulted in the least amount of pressurant required. They attributed this to (1) the evaporation of propellant when it came in contact with the pressurant gas and (2) the radial temperature profile created by this injector. The temperature gradient was such that the gas was warm at the tank centreline and cold near the walls. The cold gas temperatures at the wall resulted in low heat transfer from gas to the wall, and so to lower amounts of pressurant required. This is all in stark contrast with the common usage of the radial injector and the minimization of radial temperature gradients. Possibly the heat loss to the tank walls is of more importance with small tanks due to a higher area to volume ratio when compared to large, orbital-launcher tanks.

Huzel [18] also mentions the usage of a so called Ranque-Hilsch-vortex-tube as pressurant injector. A vortex tube is a simple and light mechanical construction without moving parts that splits an incoming gas flow into a cold and a warm gas stream [16]. Huzel suggests to use this system to protect cold cryogenic tank structural members from the incoming hot pressurant gas. However, the system is considered for this project instead to introduce a large thermal gradient within the ullage gas, with the goal to minimize the heat transfer from the gas to the wall, as is proposed by DeWitt [13].

Considering all this, it is decided that a research project is conducted to investigate the pressurized discharge of propellant from a cryogenic propulsion tank by means of a pressurant gas. The investigation will be done by means of numerical modeling of the processes within the tank, which will be validated by cold-flow tests conducted with the system. The effect of different injector types will be looked at, with special focus on the usage of a vortex tube to maximize radial temperature gradient in an effort to minimize heat losses to the tank wall.

V.IV Torch Igniter Development

While pyrotechnic igniters are a well proven method to ignite liquid rocket engines they suffer from some drawbacks. Pyrotechnic igniters require stringent safety requirements and need to be serviced for each engine firing [18, p. 121]. A spark-torch igniter consumes a small amount of the main propellant in a separate combustion chamber before introducing the hot combustion products into the main thrust chamber. Because no solid propellant is used the stringent safety requirements can be dropped. In addition a spark-torch igniter can provide a large number of ignitions easing engine testing [18, p. 121].

In order to come to an efficient spark-torch design detailed modeling of the ignition process is required. As already noted in Section V.II the current DARE RANS code is overestimating temperature due to simplified chemistry. Work at DLR suggests that similar simplified chemistry models also impart accuracy when modeling ignition [33].

NASA reported good results in employing the National Combustion Code (NCC) for designing a LOX/Methane torch igniter [3]. Airbus demonstrated good results in modeling the ignition of LOX/CH4 burning gas generator using their Rocflam II code [27]. Both works provide hints at what is required to develop a well performing RANS for modeling ignition. The main differences with DARE's code are the inclusion of more elaborate chemistry models and droplet tracking.

It is therefore decided to investigate the possibility of implementing more elaborate chemistry models and droplet tracking models specifically for modeling unsteady processes such as ignition. The resulting code shall have to be verified against previously verified code such as the NCC or Rocflam or against experimental data before it can be used to come a final torch igniter design.

VI CONCLUSIONS

This paper described the current activities performed and planned by DARE in the field of cryogenic liquid rocket propulsion. Currently DARE has no significant experience in the field of liquid rocket propulsion. A research and development project has been initiated to change this situation. Eventually the knowledge gained in this project shall lead to a sounding rocket design capable of reaching sub-orbital space. As a first step, a smaller pressure fed engine with a thrust of 3 kN will be developed. This engine is similar in size and performance as the proposed second stage engine of the ALV small satellite launch vehicle. Some first steps in this ambitious endeavor have been taken. Multiple designs for the cryogenic propellant tanks are made, and a aluminum test tank has been filled with liquid nitrogen to evaluate thermal behavior. A 1 dimensional numerical tool, using adapted empirical relations based on those proposed by Bartz has been developed to aid in the design of the regenerative cooling channels. A detailed architecture of the test setup fluid feedsystem has been designed. Ignition of the engine is to be achieved by means of a pyrotechnic igniter, using the commercially available Pro38 Vmax propellant. A first order design of this igniter has been made using ideal rocket theory solid propellant burn rate estimates. The swirl coaxial injector will be designed using the methods proposed by Bazarov, which are based on the Ambramovich theory of maximum flow.

Despite these initial steps, a multitude of questions remain. In order to asses the combustion stability characteristics of liquid rocket engines, a unsteady one dimensional CFD model will be developed. To determine and optimize the steady state performance of the engine, a detailed 3 dimensional CFD model will be made in a commercial tool such as CFX or Fluent. To optimize the pressurant consumption of the cryogenic run tanks, detailed theoretical and experimental investigations using a Hilsch tube will be performed. Finally a torch igniter, using gaseous oxygen and gaseous methane will be developed to mitigate the risks and cost involved with solid propellants. Once developed, this torch igniter will be reusable, reducing the time and costs of tests, and making the eventual engine also more suitable for re-usability.

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