Ground Performance Testing of CubeSat ADCS
Case Study on Delfi-n3Xt

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ABSTRACT

Today's market for small satellites is expanding, but there is little capacity for affordable launches. Launch costs of €50,000 per kg are required to compete with ride-shares. Cost reduction is essential, e.g. through reuse, low cost components, volume production, optimised manufacturing.

Fourteen European companies and institutes are joining forces in a Horizon2020 programme called "SMall Innovative Launcher for Europe" (SMILE). The project aims at designing a launcher for satellites up to 50 kg and a European-based launch facility at Andøya, as well as demonstrating critical technologies on propulsion, avionics, and manufacturing of cost-effective solutions.

For the propulsion, a trade-off between liquid and hybrid engines will be made to realise the objectives of the project, both in terms of performance and economic viability.

Liquid propulsion offers reliability, high performance, throttling capability and easy re-ignition. LOX and kerosene propellants are considered, being low-cost, worldwide available, green, and storable. Ceramic materials ensure high oxidation resistance combined with damage tolerance and insensitivity against thermos-shocks and thermal cycling. Combined with transpiration cooling, this will improve the lifetime, leading to reusable liquid engines.

Hybrid propulsion combines advantages of solid engines (simplicity) and liquid (inherent safety, throttling, and re-ignition capabilities). The chosen combination of propellant (H_2O_2 and HTPB) ensures good performances and limited cost, being safe (handling and operation), green and industrially available. Clustering standardized motors is also considered key to limit the cost (increased volume production and reliability).

Traditional composite manufacturing typically results in expensive structures. New technologies are needed to manufacture composite structures at acceptable cost. The ambition regarding structures is to demonstrate both automated composite manufacturing and 3D-printing for metal parts.

The use of space-qualified avionics components typically leads to expensive subsystems. Experience with CubeSats brought the conclusion that selected Commercial-Off-The-Shelf equipment can survive the launch and work in the space environment. The avionics developments focus on the inertial measurement unit, power distribution system, and on-board computer.

To eject satellites into their final orbit, a deployment system is required. The small satellite market is currently dominated by CubeSats, launched inside a container or CubeSat deployment system. The project envisages a lightweight system for the range of 25 - 50 kg satellites as well as a slimmed-down version of a CubeSat dispenser.

Nowadays, launcher ground operations are still time-consuming and expensive. The objective in SMILE is to design a conceptual ground facility with significant cost reduction of ground/launch processing and operations without compromising the mission.

1 BACKGROUND

The new generation ARIANE 6 and VEGA C launchers will guarantee Europe's independent access to space for the high-end market of satellites in terms of mass and size with a competitive edge in the world market of launchers. These launchers however are significantly less attractive for smaller satellites. The initiative therefore addresses reliable, affordable, quick, and frequent access to space for the emerging market of small satellites up to 50 kg, fulfilling the needs from the European space Research and Technology Development (RTD) community as well as commercial initiatives to put satellites into specific LEO orbits within a preferred time window. Herewith a market niche is addressed, which is projected to grow significantly in the coming decades and presently lack the availability of a dedicated European launcher.

The market for small satellites is expected to increase substantially in the coming years, as shown in market analyses of among others SpaceWorks Enterprises Inc (SEI, Nano/ Microsatellite Market Assessment 2015, August 2014) and shown in Figure 1. The excellent prospects for the small satellite market are confirmed by EuroConsult (Prospects for the Small Satellite Market, Feb 2015) with an estimate of more than 500 small satellites (nanosats, microsats, and minisats) to be launched in the next five years. Currently, the U.S. is the most active country in small satellite deployment with almost half of the 620 satellites launched in the past 10 years with Europe as the second-largest region. Historical analysis suggests the current supply of launch vehicles will not sufficiently serve future nano/microsatellite market demand.

Nanosats and microsats nowadays have to share a ride on a large rocket for a primary customer, which often causes conflicts with respect to the timeline and the orbit properties. Now that smaller satellites become technologically more advanced and mature, a call for 'affordable' dedicated launches is expedient for small satellite operators.

This situation has led to several initiatives of small launchers for various payloads in the range of 1 to 150 kg: India (Reusable Launch Vehicle, ISRO), New Zealand (Electron, Rocket Lab Ltd.) and USA (SuperStrypi, Aerojet Rocketdyne; LauncherOne, Firefly, Virgin Galactic; Lynx, XCOR; ALASA, DARPA). But also within Europe, efforts are ongoing: France (Eole, CNES), Norway (North Star, Nammo/Andøya Space Centre), Spain (Arion, PLD Space), Switzerland (SOAR, S3) and UK (Skylon, Reaction Engines Ltd.).

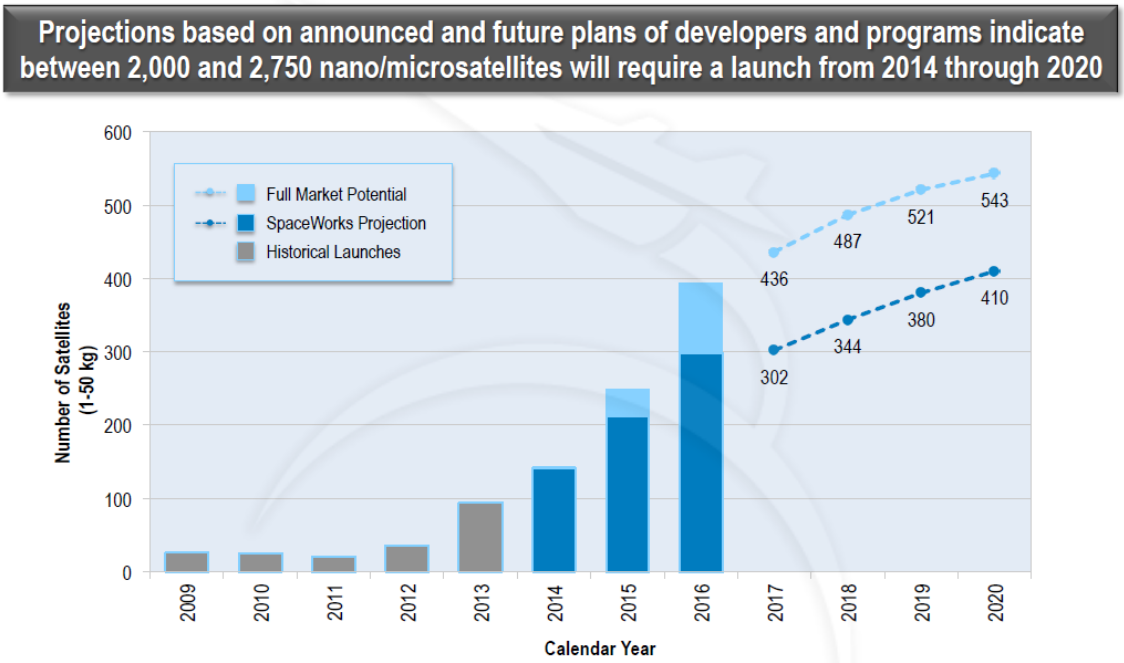

Figure 1. Future launch market for small satellites 1 - 50 kg (courtesy: SEI).

Although the above mentioned launch initiatives focus on the small satellite customer market, none of these focus on delivering the market's "sweet spot" to orbit and focus on specific payload launch ranges (e.g. 1-10 kg or 100kg+). Based on the market analyses the range up to 50 kg payload capacity can be considered the "sweet spot" for a small satellites launcher. Such a launcher will provide a proper launch capability for a single 50 kg satellite (i.e. commercial, scientific, and governmental) as well as for a flexible configuration of multiple smaller satellites (i.e. education, in-orbit demonstration) up to a total mass of about 50 kg. The above mentioned initiatives are in different states of development and are providing no launch services at this moment.

2 SMILE PROJECT

No operational dedicated launcher for small satellites exists today. Small satellites, launched as secondary payload, are entirely dependent of the constraints set by the primary payload, such as launch date and target orbit. Launch costs of less than €50,000 per kg of payload are required in order to compete directly with these piggy-back ride shares which are the current economically viable access to space for small satellites. With a dedicated launcher, a higher cost per kg can be accepted for payloads which need to be delivered timely and accurately to a desired orbit.

Hence, a consortium of 14 partners from 8 European countries are joining forces in a Horizon 2020 work programme to design a dedicated small launcher to be built in and launched from Europe. Together, the consortium coordinated by the Netherlands Aerospace Centre NLR covers all aspects of marketing, developing, and operating a cost-effective launcher with a well-balanced mix of companies, SMEs, and institutes.

The project is called "SMall Innovative Launcher for Europe", SMILE, which started on 1/1/2016 and which will three years. The SMILE project aims at a combined research approach into a new innovative European launcher for an emerging market of small satellites up to 50 kg using a multidisciplinary design and optimisation approach strengthened by the demonstration of critical technologies for cost-effective solutions and complemented with the design of a European-based launch capability from Andøya (Norway).

When targeting commercial launch prices of less than 50,000 €/kg up 50 kg payload capacity, the total maximum cost for a launch shall be well below 2.5 M€. This target cost drives the design, construction, and operation of the launcher. After 2020, it is anticipated that the market for launching small satellites is in the order of several hundred per year and growing. A total capacity of up to 50 launches per year is foreseen. Using a flexible configuration of the launcher-payload interface structure, several combinations of small satellites up to 50 kg can be accommodated.

The launcher will use advances in technology to achieve cost reduction, including design for series production, reusability, and the use of COTS components. Critical technologies enabling affordable and independent access to space will be developed in this project. To be able to meet the target price, the design will be based on existing advanced technologies as a starting point, and drive the development of required new technologies forward as part of the program. The overall objectives of the SMILE project therefore are:

- To design a concept for an innovative, cost-effective European launcher for small satellites.
- To design a Europe-based launch capability for small launchers based on the evolution of the existent sounding rocket launch site at Andøya Space Centre.
- To increase the Technology Readiness Level (TRL) of critical technologies for low-cost European launchers.
- To develop prototypes of components, demonstrating this critical technology.
- To create a roadmap defining the development plan for the small satellites launcher system from a technical, operational and economical perspective.

In order to fulfil the project's objectives the consortium has identified a number of technologies that are capable of upgrading the actual state-of-the-art of this type of vehicles. These include:

- Hybrid engine technology.
- Liquid engine technology with transpiration cooling.
- Turbo-pump technology.
- Advanced low-mass and low-cost materials.
- Series production of low-cost composite structures.
- Printing technology for low-cost metal components.
- Advanced, reliable COTS technology for miniaturised, low-power avionics.
- European-based launch facility.

At the end of the project the target Technology Readiness Levels (TRL) for the critical technologies shall be according to Table 1.

Table 1: Target Technology Readiness Levels (TRL) for the critical technologies

Item	TRL
Launcher concept	2
Hybrid rocket engine	7
Liquid rocket engine	5/6
Advanced materials	3
Automated manufacturing of composites	5
Printing technology	8
Advanced avionics	4
European launch facility concept	2

In order to enhance the continuity of the project's objectives, a roadmap will be set-up by assessing scenarios and critical future steps at technical, financial, and organisational levels. A business development shall include a technology roadmap towards a TRL 9 launcher. Furthermore, it presents a strategy to achieve commercially feasible launch services, including cost – benefit analysis.

Although critical technologies in several areas are encompassed by the SMILE project, the focus in this paper is on novel hybrid and liquid rocket engine technologies by Nammo Raufoss AS and the German Aerospace Centre DLR respectively. Especially, the paper addresses the needs and impacts of these technologies on a small launcher development as well as the foreseen necessary costs reduction. The following objectives are foreseen for critical engine technology development:

- To perform a trade-off between two propulsion technologies in order to obtain the configuration answering the best to the constraints of the project.
- To design the architecture of the launcher's propulsion modules based on the requirements.
- To generate the detailed design of the propulsion modules.
- To select technology for low-cost advanced engine parts.
- To produce prototypes of the selected engine parts.
- To conduct firing tests of the liquid engine.

3 HYBRID ROCKET ENGINE TECHNOLOGY

3.1 Current State

Up to now, only two kinds of engines have been used for operational launchers: liquid engines (such as the European Vulcain II, the Russian RD-180 or the American Merlin 1A) and solid engines. The latter are mainly used as boosters for the big launchers (Ariane 5's SRBs) or for the first stages of medium launcher (Vega's P80, Pegasus system) or sounding rockets.

Liquid engines offer high versatility, through thrust regulation and restart capabilities, and high performance (high specific impulse), but are somewhat limited in thrust and their high complexity (with a turbo-pump feeding the combustion chamber with propellants) makes them quite costly, both in terms of mass budget and development cost. On the other side, while solid engines offer simplicity and high performances in terms of thrust, they have the drawbacks of being inherently hazardous (the oxidizer and fuel are intimately mixed in the grain), uncontrollable (impossible to stop once ignited), and tailored to one specific task.

Hybrid propulsion development started at the same time as for the other two. The goal was to combine the advantages of both types of engine (inherent safety, versatility, being able to throttle, and simplicity) at low cost. Unfortunately, knowledge at that time didn't allow hybrid engines to compete in terms of performance, especially because of a low regression rate of the fuel (leading to only small thrust capabilities, or complex fuel grain geometry).

In the last decade however, hybrid propulsion has matured, mainly through research and technology programs. Full scale flight weight rocket motors are now totally conceivable at low price, and with capabilities and performance allowing a competition with liquid or solid engine.

Nammo Raufoss AS (Nammo), a Norwegian-based defence company, has since 2003 invested in the hybrid rocket propulsion technology. Based on hydrogen peroxide (H2O2), a completely green oxidizer, and HTPB fuel, Nammo has moved the technology forward through the following projects:

- The upscaling of the hybrid technology to a 30kN-class engine under the ESA funded Future Launcher Preparatory Program (FLPP)
- The establishment of a new 500kN Green Propulsion Test Stand
- The development of a throttleable hybrid engine for a Lunar Lander under the European Community funded 7th Framework Program, SPARTAN
- The development of a so-called "Hot Gas Reaction System" (HGRS), a new (mono-propellant) Reaction Control System for Ariane 5ME, Ariane 6 and Vega to replace the hydrazine alternative

The combination H2O2/HTPB offer the advantage of being already available in industrial quantities, while being completely green (only CO2 and H2O produced), safe to handle (nontoxic products) and safe to operate (two propellant completely segregated). Those characteristics, coupled with a simple fluid system, will substantially reduce hybrid propulsion life cycle cost compared to other propulsion systems. Moreover, with the use of a catalyst bed to decompose the H2O2, the engine can be stopped and restarted at will, without the need of an external igniter (which is the case with liquid engine). This could prove crucial for small launchers that want to launch multiple payloads on different orbits.

With Nammo's hybrid architecture, it is possible to develop an engine with performances high enough to suit the needs of small satellites launchers, at a much lower price tag.

3.2 The Unitary Motor: The Building Block Of The Hybrid Rocket Propulsion System

The current state of the hybrid technology at Nammo is represented by the Unitary Motor (UM), a novel concept of hybrid rocket engine developed by Nammo under an ESA-FLPP contract. It uses high concentration hydrogen peroxide (87.5% H2O2) as oxidizer and hydroxyl-terminated polybutadiene (HTPB) rubber as fuel. Its working principle is shown on Figure 2. The incoming liquid oxidizer, with a mass flow of about 11 kg/s, is first decomposed over a catalyst into hot steam and gaseous oxygen to a temperature of 670°C.

It then goes through the injector and enters the combustion chamber in hot gaseous form, where ignition of the hybrid combustion occurs without any dedicated ignition device due to the high oxidizer heat flux, sufficient to vaporize the solid fuel. The vortex flow-field in the chamber generated by the injector helps in maintaining a high heat flux to the fuel surface and in achieving appropriate mixing of the reactants for high combustion efficiency. The hot product gases are then expelled through a nozzle, generating close to 30 kN of thrust.

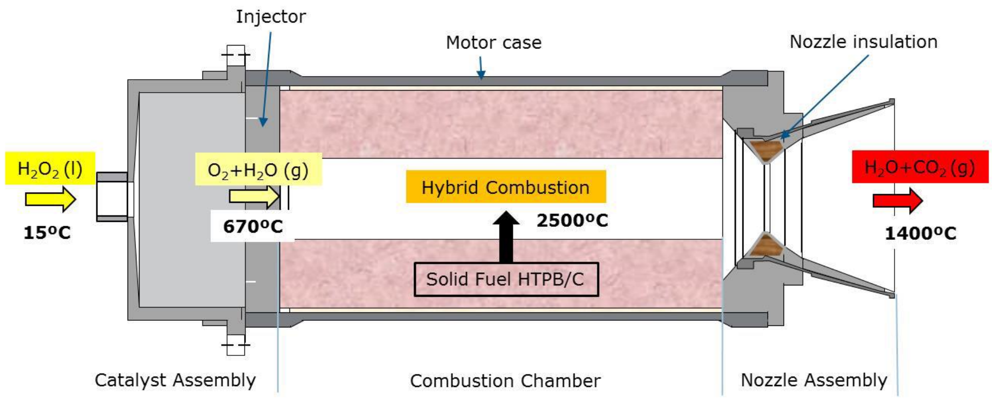

Figure 2. Working principle of the Unitary Motor.

Compared with solid rocket motors, the Unitary Motor designed by Nammo has a rich set of attractive features, even when compared with other versions of hybrid rocket engines with which its shares the inherent properties of hybrid propulsion. These features are:

- Self-ignition increasing engine start reliability and enabling an unlimited restart capability.
- Wide range throttling with limited performance losses.
- Green life cycle and exhaust properties.
- Solid inert fuel and high-density green storable oxidizer.
- High engine combustion efficiency, performance, and stability.
- Simplicity of a single circular port and single feedline configuration.
- Low development and operational costs.

Some of these features are common with liquid rocket engines, but compared with liquid rocket engines, the architecture of the UM is much simpler and the same features are obtained for a fraction of the cost.

The design of the UM has been split in two phases. First, a Heavy-Wall configuration (HWUM) has been designed, manufactured and tested in the fall of 2014. The goal was to assess the up-scaling of the hybrid technology (i.e. inner ballistic, regression rate of the fuel) without the constraint of a flight-weight engine. The HWUM demonstrated great behaviour in terms of both performance and stability from the first test firing (see Figure 3 and Figure 4), and continued to do so throughout the rest of the campaign. This allowed Nammo to complete the HWUM development test campaign in only 6 hybrid firing tests and one iteration on the motor configuration.

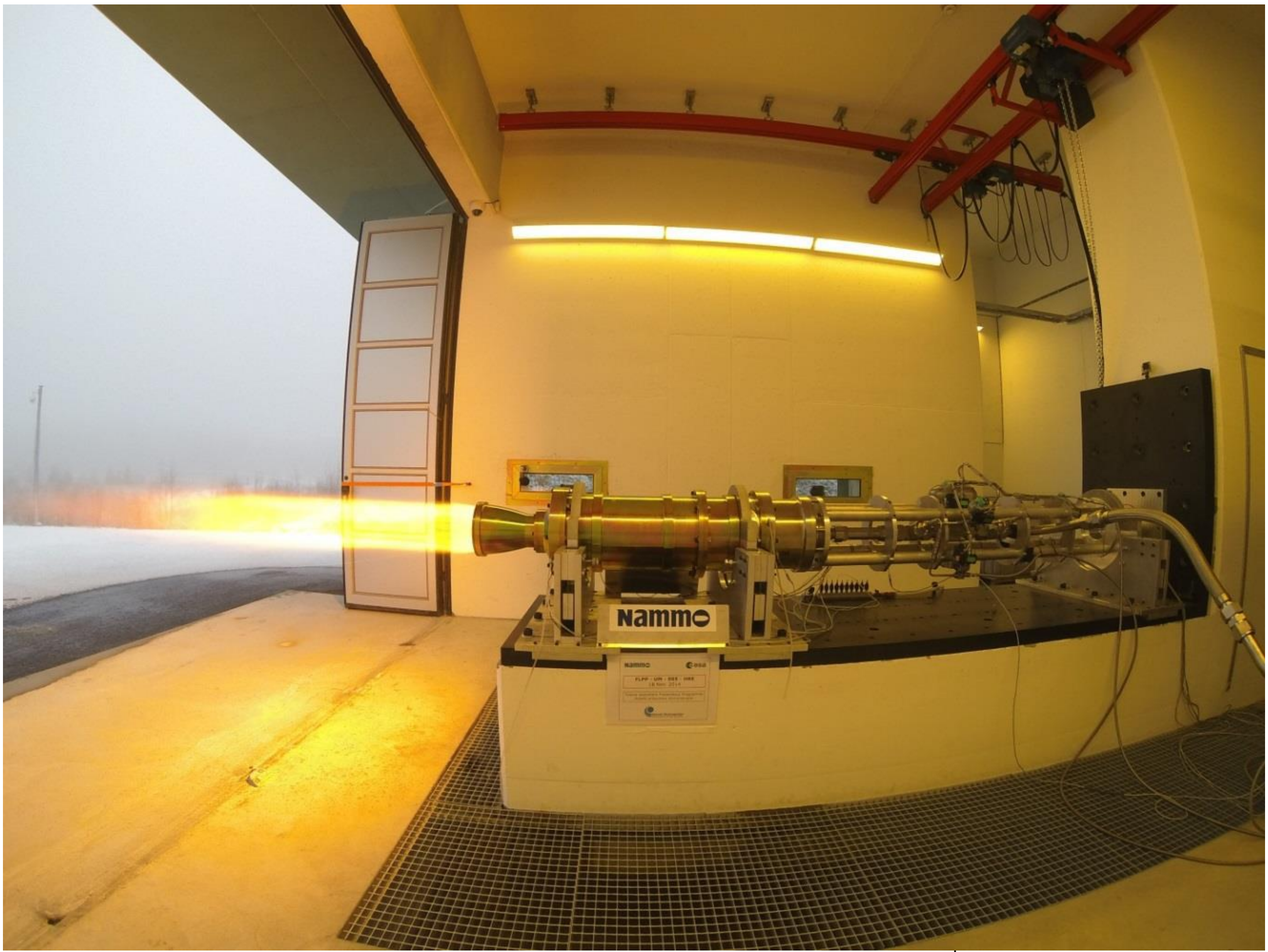

Figure 3. HWUM during 3rd firing on November 18th, 2014.

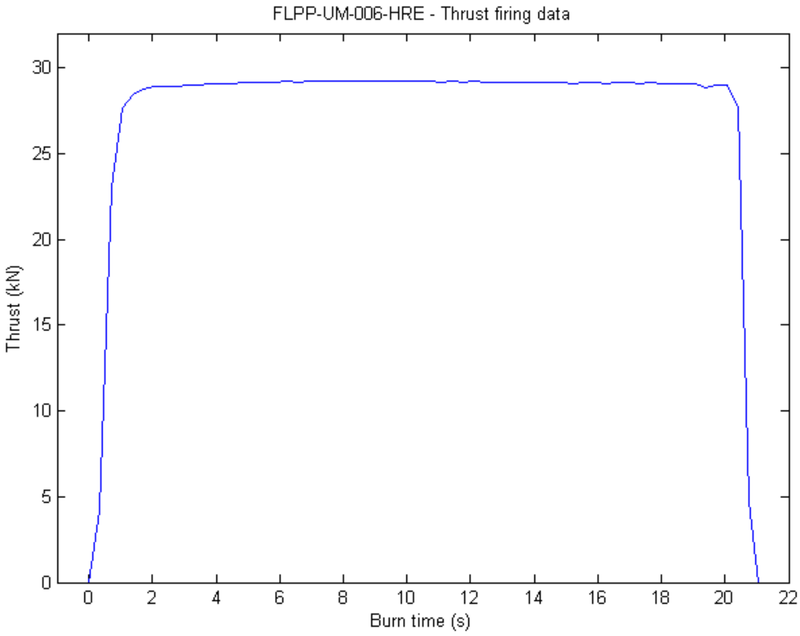

Figure 4. Thrust measured during the 4th HWUM firing, on November 28th, 2014. The measured data has been filtered down to 3 Hz sampling for visualization in this paper.

The HWUM ground tests were concluded with the delivery of a very satisfactory motor design yielding the performance desired (see Table 2) for the next stage in the program.

Table 2: Comparison of the 5th HWUM test experimental results (December 09th, 2014) with the motor design target. In this table, all mean values are averaged over the entire motor burn duration and all values have been rounded independently

Firing	**FLPP-UM-007-HRE**	**Design model target**
Burn Duration	25 s	25 s
Mean oxidizer mass flow	10.8 kg/s	10.8 kg/s
Mean fuel mass flow	1.9 kg/s	1.6 kg/s
Mean oxidizer to fuel ratio	5.75	6.75
Mean chamber pressure	36 bar	35 bar
Mean specific impulse (ground level)	234 s	230 s
Mean engine efficiency	95 %	94 %
Total impulse (ground level)	750 kNs	700 kNs

Based on the results from the Heavy Wall Unitary Motor firings, a Flight Weight Unitary Motor (FWUM) has been designed. This design is currently being manufactured and the test campaign should start in November 2015. The design of the FWUM mainly replaces over-dimensioned parts with optimized parts, but it will also increase the capabilities of the Unitary Motor. Based on discussions with the user community, the capabilities of the UM are adjusted to a larger total impulse capability of 1000 kNs approximately.

Given the demonstrated performance of the HWUM, this can be achieved within an outer diameter of 14 inches, which is the standard sounding rocket payload diameter in use at Andøya Space Centre and Europe in general. The updated design data is given in Table 3. Although also feasible, no attempt has been made to achieve a higher thrust level for the FWUM, but rather a longer burn time. It is increased with 10 sec. from 25 sec. to 35 sec.

Table 3: Main differences between the HWUM and the FWUM

Property	HWUM	FWUM
Total impulse	750 kNs	980 kNs
Outer diameter	305 mm (12 in.)	356 mm (14 in.)
Burn duration	25 s	35 s
Dry mass (without consumed fuel)	>280 kg	<100 kg
Consumed fuel mass	< 50 kg	> 60 kg
Consumed oxidizer mass	~270 kg	~380 kg

A demonstration launch of the FWUM is planned for the fall 2016 on board a prototype Nucleus sounding rocket (based on a single UM) from Andøya Space Center in Northern Norway. The goal of the launch is to reach the space frontier at 100 km altitude.

3.3 Hybrid Rocket Stage For A Micro-Launcher

In order to keep the price of the propulsion system as low as possible, reusability of components is a key feature leading to cost reductions through volume production and increased reliability through automated production. In that sense, the Unitary Motor is thought of as a building block that can be clustered to deliver the required thrust for a micro-launcher.

The North Star rocket family, a Norwegian initiative of sounding rockets and micro-launchers, is based on that principle, with the utilization of two high thrust motors, the UM and its future upgrade the UM2, for the first stages and a third high performance engine with a more moderate thrust requirement and longer burn-time needed to obtain orbit insertion on the upper stage. Figure 5 presents the concepts of the different rockets of the North Star Family and Figure 6 the preliminary design performance of the different propulsion stages.

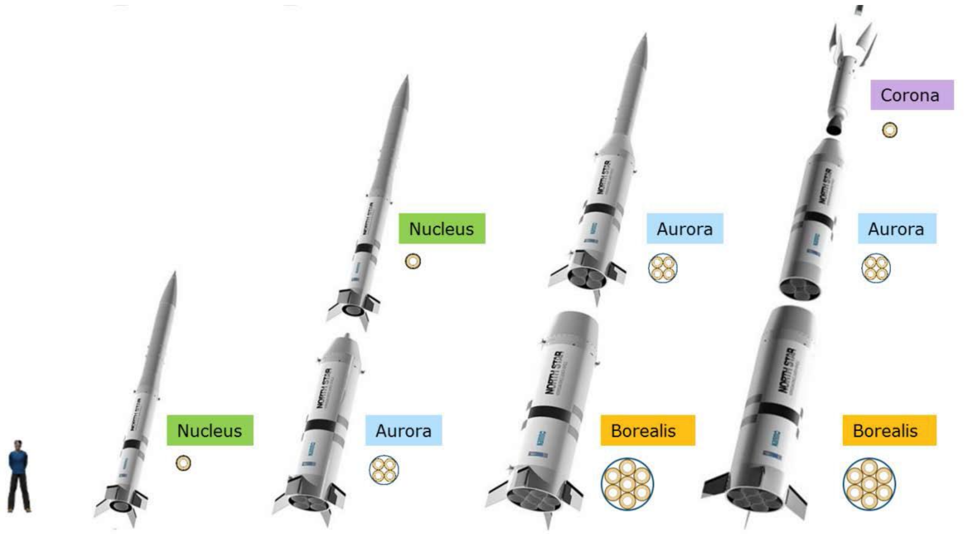

Figure 5. The North Star Rocket Family (source: Andøya Space Center).

Rocket stage	Motorization	Indicative impulse
Nucleus	1x Unitary Motor 1	Thrust: 28 kN Burn time: 35 s Total impulse: 1 MNs
Aurora	4x Unitary Motor 1	Thrust: 114 kN Burn time: 35 s Total impulse: 4 MNs
Borealis*	7x Unitary Motor 2	Thrust: 450 kN Burn time: 64 s Total impulse: 30 MNs
Corona*	1x High Performance Hybrid Motor	Thrust: 5 kN Burn time: 70 s Total impulse: 0.35 MNs

*The characteristics of the stages Borealis and Corona will depend on the flight performance of the Nucleus and later on, the Aurora stages.

Figure 6. The North Star Rocket Family (source: Andøya Space Center).

In SMILE, the same principle will be used with the added value of combining hybrid stages of clustered Unitary Motors with liquid stages. Based on the results and performances obtained during the FWUM test campaign and the demonstration launch, the sizing of the different propulsion stages of the micro-launcher will be achieved by clustering the Unitary Motor.

The fluid feeding system (bringing the liquid oxidizer to the motors) will have to be design and sized accordingly and the performances (i.e. thrust, specific impulse, weight, and size envelope) will be provided to the other members of the consortium for the global design of the launcher. It is strongly believed that both the inherent lower price of the hybrid technology and the clustering of elements enabling a more cost-effective production will be a large contribution in bringing the global cost of the launcher within the required range of 50,000 €/kg.

4 LIQUID ROCKET ENGINE TECHNOLOGY

Liquid propulsion is a well proven technology that can be operated with different types of propellants. Hereby, the choice of propellants is driven by their resulting specific impulse, thrust-levels, and tankage-to-propellant mass ratios. Hence, for lower stages high-density propellants are preferred which yields into both reduced tankage volume and geometrical expansion ratio. For this reason, LOX/kerosene is rather used for first stages than LOX/LH2; in the latter case a combination with solid boosters (e.g. Ariane 5 and Space Shuttle) would be aimed for the launch or the propellants are preferably applied to upper stages as LOX/LH2 offers the highest specific impulse.

In general, liquid propulsion is a reliable technology which is very promising due to its flexibility as the engines can be throttled at a wide range and easily re-ignited. For the current configuration, the combination of LOX/kerosene propellants is considered as very favourable. Kerosene can be easily stored and refuelled, is a cheap fuel, and is available worldwide.

In any case, the propulsion system is the most expensive part of the launcher. Thus, it would be beneficial to retrieve the engines back after a launched mission. Possible solutions might include guided parachutes, propulsion-assisted boosters (like SpaceX), winged fly-back engines (like Adeline from Airbus Defence & Space) or winged fly-back boosters where DLR already did some studies within the FLPP programme funded by the European Space Agency (ESA). Once the engines are retrieved, they have to be inspected in order to have them refuelled and put into operation again.

4.1 Ceramic-based Design

In contrast to solid, hybrid or classical liquid engine approaches, liquid engines based on a ceramic design are very promising candidates with respect to such reusability aspects as they offer:

- Improved lifetime.
- Thermo-shock resistance.
- Thermal-cycling ability.
- Reliability and damage tolerance.
- Reduction in structural weight.
- Oxidation resistance.
- High specific strength at elevated temperatures.
- Low thermal expansion.

Hence, this specific kind of propulsion system using ceramics is well suited and applicable as it can be thermally cycled without degradation which is not the case for metallic approaches. DLR has much experience on liquid rocket propulsion and the DLR Institute of Structures and Design in Stuttgart focuses on ceramic-based designs that are using transpiration cooling. All ceramic materials, such as non-oxide and oxide ceramic matrix composites (CMCs), can be manufactured in-house[4, 5, 6]. The transpiration cooling principle allows increasing the chamber wall lifetime significantly, at the cost of a slight decrease in specific impulse.

Compared to classic metallic solutions, the engine's structural weight can be reduced substantially, depending on the applied ceramic materials[7] and design. In general, transpiration cooling consists of two mechanisms, as depicted in Figure 7: a small portion of the coolant penetrates the combustor walls and thereby convectively extracts heat from the hot wall; in addition, a coolant layer forms at the inner combustor wall that protects the wall from the hot combustion flow.

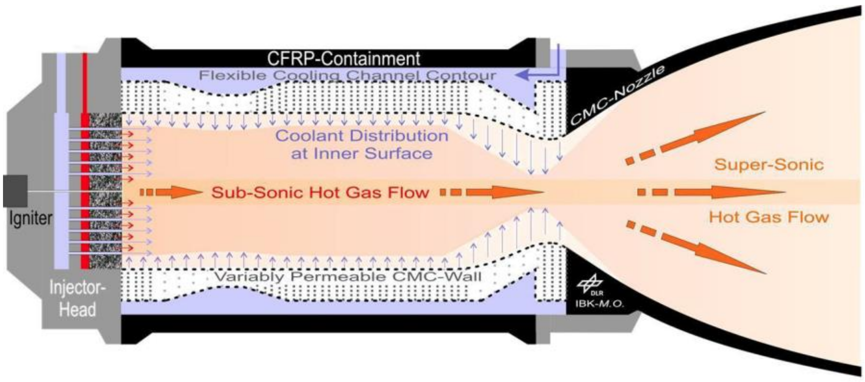

Figure 7. Schematic of a transpiration cooled ceramic thrust chamber.

First initial experiments on transpiration cooled segments for liquid rocket propulsion have been performed at the end of the 1990s. All testing was performed at various high-performance rocket engine test benches of DLR Lampoldshausen, up to 90 bars combustion chamber pressure. They solely focused on hydrogen-oxygen propellants, including cryogenic conditions as well. The development resulted in sophisticated design approaches which were investigated in different projects.

Between 2008 and 2012, four separate test campaigns were performed within the DLR projects *KSK* (ceramic thrust chamber) and *KERBEROS* (Ceramic Design of Experimental Rocket Engines for Upper Stages), as given in Table 4. The different configurations included the variation of wall and nozzle materials, injectors (API: advanced porous injector from DLR Lampoldshausen; TRIK: coaxial injector by DLR Stuttgart), contraction ratio, coolant blowing ratio, characteristic chamber length, etc. Further details can be obtained from[8, 9, 10].

Table 4: DLR ceramic thrust chamber test campaigns 2008-2012

	KSK-KT	**KSK-ST5**	**MT5-A**	**WS1**
Year	2008	2010	2012	2012
Test bench	P8	P8	P6.1	P6.1
Propellant combination	LOX/LH2	LOX/LH2	LOX/GH2	LOX/GH2
Injection temperature (fuel)	≈ 55 K	≈ 55 K	≈ 135 K	≈ 150 K
Injection temperature (oxidizer)	≈ 155 K	≈ 155 K	≈ 125 K	≈ 140 K
Coolant	H_2	H_2	H_2	H_2
Wall material	C/C	Al_2O_3 and C/C	Al_2O_3 and C/C	Various
Nozzle material	Copper	C/C	C/C	C/C
Injector	API	API	TRIK	TRIK
Chamber diameter (d_c)	50 mm	50 mm	50 mm	50 mm
Throat diameter (d_t)	31.6 mm	31.6 mm	20 mm	20 mm
Characteristic chamber length (l*)	0.86 m	0.68 m	1.75 m	1.83 m

Figure 8 shows test operation of the ceramic thrust chamber during the test campaign MT5-A. Especially in combination with the transpiration cooling technique and the use of CFRP housing structures, the engine's structural weight can be significantly reduced. On the other side, sophisticated CMC materials enable replacing ITAR-controlled metal alloys (as the current main material for combustion chambers) in the future.

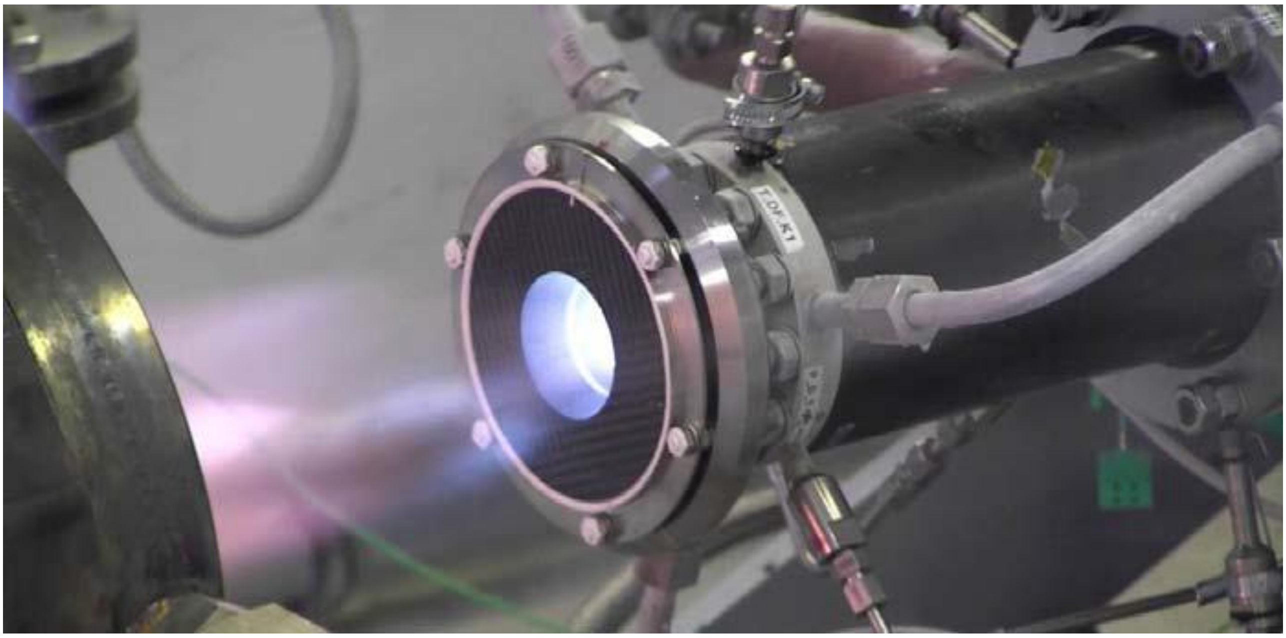

Figure 8. Liquid rocket engine test MT5-A at P6.1 test bench in Lampoldshausen (LOX/GH2).

Furthermore, the general feasibility in GOX/kerosene combustion environment was successfully demonstrated in the EC project ATLLAS (coordinated by ESA and funded within FP6, 2006-2009). All tests were performed at the high-pressure rocket combustion chamber test bench at Technical University Munich (TUM), see Figure 9.

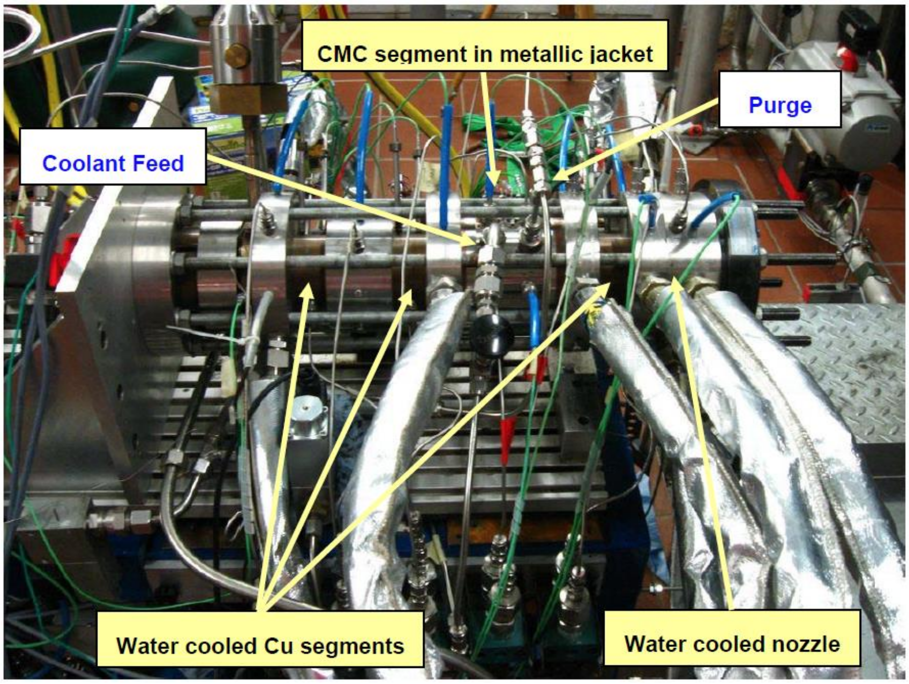

Figure 9. Liquid rocket engine test at TUM test bench (GOX/Jet A-1).

Various CMC materials were tested, whereas oxide CMCs seem to be very suited for this kind of application as the material is able to withstand hot gas oxygen attacks. Figure 10 shows two of the integrated CMC liner materials: C/C (non-oxide) and WHIPOX (oxide). With respect to cooling performance, hydrocarbon-based coolants such as Jet A-1 kerosene turned out to be very efficient.


Figure 10. Ceramic inner liners for TUM test (left: C/C, right: WHIPOX).

4.2 Engine Reusability

Already in the late 1960s, Pratt & Whitney developed the transpiration cooled XLR-129 rocket engine with a chamber pressure of approximately 100 bars. The engine was extensively tested and based on the results; a transpiration cooled design was developed for the Space Shuttle Main Engine (SSME). Transpiration cooling was selected in order to fulfil the NASA criteria of 100 time engine reusability[11].

This engine development of Pratt & Whitney is the only known experimental study dealing with transpiration cooled engine life cycle, durability, and re-usability to date. Based on the published results of Pratt & Whitney and theoretical considerations, the lifetime of transpiration cooled chambers is expected to be at least 10 times higher than that of regeneratively cooled chambers.

It has to be mentioned that at this time, transpiration cooling research was mainly conducted considering metallic materials. In case of local hot spots, such metallic structures tend to melt and cause a catastrophic failure. This is in clear contrast to ceramic-based materials which do not exhibit such behaviour. Additionally, ceramic-based designs enable improved lifetimes due to their positive thermal-cycling ability and thermo-shock resistance.

4.3 Envisaged SMILE Approach

It is expected that existing design approaches could be transferred to LOX/kerosene operation. In doing so, a ceramic-based thrust chamber assembly will be designed. Whereas the injector head might be made via SLM (selective laser melting)-techniques, the combustor component will be designed of ceramic liners actively cooled by transpiration. Here, both fuel and oxidiser are considered as potential coolants. In addition, a ceramic nozzle section is foreseen.

A clustered design is considered which would result in multiple turbo-pump-fed sub-scaled engines, depending on the mission scenario. DLR's engine enables reliable low-cost components to fit into the envisaged target price of 50,000€ per kg of payload with a future potential of reusability.

5 CONCLUSIONS

There is a need for a dedicated and affordable small satellite launcher. A major challenge for the launcher design is to become cost efficient within all technology development areas in order to offer future customer launch prices of less than €50,000 per kg of payload. The SMILE project will take up this challenge by aiming at a combined research approach into a new innovative small launcher for an emerging market of small satellites up to 50 kg using a cost-effective design approach. Cost reduction is achieved by applying reusability of one or more stages, applying commercial industry-grade components and through volume production including cost-optimized manufacturing process. In this paper the cost effectiveness for the rocket engine development is addressed.

For the hybrid rocket engine development this is achieved by the inherent low life-cycle cost of the hybrid technology and the clustering of unitary propulsion elements, the Unitary Motor. Low life-cycle cost is achieved by a simple architecture, the non-toxicity, the inertness, and the availability of the propellants and the overall low development and operational costs. The clustering of the Unitary Motor will also bring the cost down, thanks to a higher volume production for each component. This higher volume could also legitimate an automated production leading to a better reliability of the product.

For the liquid rocket engine development this is achieved by an operation of multiple LOX/kerosene sub-scaled engines based on ceramic materials and a transpiration cooling technique for improved engine lifetime and reuse. In combination with reliable low cost 3-D printed components and the potential use of CFRP (carbon-fibre reinforced plastics) housing structures, the engine's structural weight can be significantly reduced.

The combination of the two hybrid and liquid propulsion technologies will allow the use of the right technology at the right place to offer a launcher delivering the required performance at the lowest price possible. Ultimately, the choice of the propulsion system for all the stages of the rocket will be a trade-off between performance, launch objectives, and cost.

6 ACKNOWLEDGMENTS

This project has received funding from the European Union's Horizon 2020 research and innovation programme under grant agreement No 687242.

Our thanks to all SMILE consortium supporting the project and this paper (Netherlands Aerospace Centre NLR, Nammo Raufoss AS, German Aerospace Centre DLR, Terma, Andøya Space Centre ASC, INCAS, Airborne Technology Centre ATC, Heron Engineering, Innovative Solutions In Space ISIS, 3D Systems LayerWise, PLD Space, Tecnalia, BoesAdvies, and WEPA Technologies).

7 REFERENCES

[1] Verberne C.J., et al. *Development of the North Star Sounding Rocket: Getting ready for the first demonstration launch*, presented at the AIAA 51st Joint Propulsion Conference, Orlando, Florida, 2015.

[2] Boiron A.J. et al. *Hybrid Rocket Motor Upscaling and Development Test Campaign at Nammo Raufoss*, presented at the AIAA 51st Joint Propulsion Conference, Orlando, Florida, 2015.

[3] Vesterås T. et al. *FLPP Large Scale Hybrid Rocket Demonstrator*, Space Propulsion 2014, Cologne, Germany, 2014

[4 Heidenreich B., et al. *C/C-SiC Materials Based on Melt Infiltration - Manufacturing Methods and Experiences from Serial Production*, HT-CMC 8, Xi'an, China, 2013.

[5] Klatt E., et al. *Creep Behavior of Oxide CMC Based on Polymer (OXIPOL)*, HT-CMC 7, Bayreuth, Germany, 2010.

[6] Schmücker M., et al. *Mesostructure of WHIPOX all oxide CMCs*, Composites Part A 34, 2013, doi:10.1016/S1359-835X(03)00100-3.

[7] Kuhn M., et al. *CMC Materials for Combustion Chamber Applications*, 3rd European Conference for Aerospace Sciences (EUCASS), Versailles, France, 2009.

[8] Herbertz A., et al. *Transpiration-Cooled Ceramic Thrust Chamber Applicability for High-Thrust Rocket Engines*, in 48th Joint Propulsion Conference, AIAA-2012-3990, Atlanta, USA, 2012.

[9] Hald H., et al. *Technological Aspects of Transpiration Cooled Composite Structures for Thrust Chamber Applications*, 16th AIAA/DLR/DGLR International Space Planes and Hypersonic Systems and Technologies Conference, Bremen, Germany, 2009.

[10] Herbertz A. et al. *Analysis of Coolant Mass Flow Requirements for Transpiration Cooled Ceramic Thrust Chambers*, Transactions of the Japan Society for Aeronautical and Space Sciences, Space Technology Japan, 12 (ists29), Pa_31-Pa_39, ISSN 1347-3840, 2014.

[11] Mulready D. et al. *Advanced Engine Development at Pratt & Whitney*, Society of Automotive Engineers, Inc., Warrendale, USA, 2001.