TECHNOLOGY ROAD MAP FOR THE DEVELOPMENT OF A EUROPEAN STAGED COMBUSTION ROCKET ENGINE FOR REUSIBLE LAUNCH VEHICLES

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ABSTRACT

The increasing global enthusiasm for space exploration hinges on pivotal technological advancements and breakthroughs. The European Union (EU) faces the imperative task of defining its role in these aspirations and outlining key areas for national technology development to bolster future capabilities in space exploration. Over the past decade, launcher development has experienced a resurgence in dynamism, marked by rapid technical breakthroughs driven by private entities such as SpaceX and Blue Origin.

Reusable launch vehicles (RLVs) have emerged as a compelling option for the future, promising significant cost savings compared to expendable launchers. Notably, newly developed rocket engines, with a focus on reusability, employ closed cycle technologies such as the expander cycle (Blue Origin BE-7), staged combustion cycle (Blue Origin BE-4), and full-flow staged combustion cycle (SpaceX Raptor). The selection of the staged combustion cycle technology is justified by its numerous benefits, including high fuel efficiency, the ability to achieve high thrust levels for the initial part of the flight, overall engine performance, higher specific impulse (ISP), lower carbon footprint, and more.

While the Full-Flow Staged Combustion (FFSC) cycle appears particularly promising for RLVs, Europe currently lacks representation in staged combustion technology. This work aims to propose a comprehensive technology and engine maturation plan, addressing critical and enabling technologies, along with specific maturation needs for RLVs. The plan will emphasize the importance of test campaigns at both subsystem and system/engine levels, considering aspects such as test configurations (especially integrated systems like powerpack or powerhead), demonstration scale, and test rig capabilities and constraints.

Funded under ESA Contract No. 4000142002/23/NL/RK, this roadmap aims to guide the decision-making processes of the European Space Agency, focusing on technologies where European academia and industry are leading or well-positioned to lead.

NOMENCLOTURE

Isp	Specific Impulse [s]
ε	Expansion Ratio [-]
MR	Mixture Ratio [-]
SM	Separation Margin [%]
AM	Additive Manufacturing
AxSSS	AxStream System Simulation
BC	Boundary Conditions
CMC	Ceramic Matric Composites
DLR-SART	German Space Agency – Space Launcher Systems Analysis
ESR	Engine System Requirement
EU	European Union
FADEC	Full Authority Digital Electronic Control
FFSC	Full Flow Staged Combution
FRPB	Fuel Rich Preburner
GH2	Gaseous Hydrogen
GOX	Gaseous Oxygen
HPFTP	High Pressure Fuel Turbopump
HPOTP	High Pressure Oxygen Turbopump
HTHL	Hoziontal Takeoff Horizontal Landing
IOPP	Integrated Oxygen Power Pack
IPH	Integrated Power Head
L-PBF	Laser powder bed fusion
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MCC	Main Combustion Chamber
MCCP	Main Combustion Chamber Pressure
ORPB	Oxygen Rich Preburner
RLV	Reusable Launch Vehicle
SLME	SpaceLiner Main Engine
SIW-S	SoftInWay Switzerland
SLO	SpaceLiner Orbital Stage
TCA	Thrust Chamber Assembly
VTHL	Vertical Takeoff Horizontal Landing
VTVL	Vertical Takeoff Vertical Landing

1. INTRODUCTION

SIW-S in partnership with DLR-SART is currently busy with a De-risk study [2] for the next generation LOX-LH2 full-flow staged combustion (FFSC) engine for Reusable Launch Vehicle (RLV) applications. The SpaceLiner [1] was selected as the RLV application for the FFSC engine study. The High-Level Requirements (HLRs) for the SpaceLiner Main Engine (SLME) was generated based on the 1st or booster stage performance requirements.

The SLME cycle design has been developed for the past several years by DLR-SART. The results of the updated cycle design were cross-checked between DLR-SART's tools, and SIW-S tool AxStream System Simulation (AxSSS) for 0D simulations. Furthermore, AxSSS was used to advance the system simulation and increase accuracy of the results by including additional 1D sub-system characteristics like pre-burner duct losses, and turbomachinery performance maps.

The boundary conditions (BCs) generated by the cycle analysis was used for the SLME sub-system pre-liminary designs. The task of (sub-system design) component sizing which includes the turbomachinery, pre-burners, Main Combustion Chamber (MCC) and thrust nozzle was completed by SIW-S. The outcome from the component sizing and engine integration was used to inform the final task of the technology road map.

Several critical technologies were identified but the HPOTP and ORPB were of particular interest hence the emphasis placed on these sub-systems. A major outcome of the study suggests the need for an integrated HPOTP and ORPB in-line with the MCC is investigated. An idea that has been previously investigated by DLR-SART and most recently, utilizing a different propellant, employed on the Raptor engine.

2. PROPULSION SYSTEM

2.1. SpaceLiner application

SpaceLiner was identified as the European RLV for which the main engine (SLME) will be developed. SpaceLiner launchers have different variant concepts including VTHL, VTVL, and HTHL with multi-stage systems [1]. Figure 1 shows SpaceLiner 7 with SLMEs. The concept is to use the same main engine for all stages with differing thrust nozzle configurations.

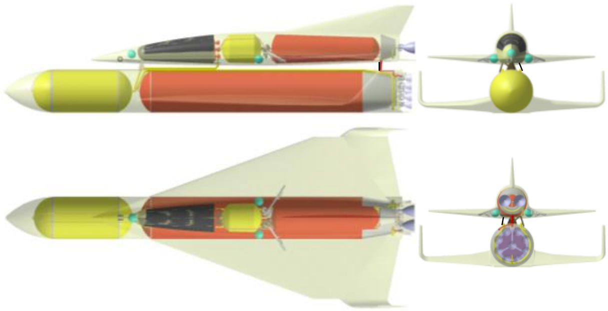

Figure 1: Sketch of SpaceLiner 7 launch configuration with passenger stage with its booster stage at bottom position and orbital stage of SLO in insert at top showing the SLME arrangement in the lower right figure [13]

2.2. SpaceLiner Main Engine Requirements

High-level Requirements (HLRs)
Some of the HLRs that were defined for the study are listed in Table 1.

Table 1: HLR examples for SLME

HLR no.	Description
HLR 1	Propellant combination should be LOX-LH2 in suitable MR-range.
HLR 2	Thrust level should be 2200 kN in vacuum condition.
HLR 3	Thrust level should be throttleable at least in range 93% - 107%
HLR 4	Engine should be capable of [25] flight-mission reuses.
HLR 5	Design of engine components should consider state-of-the-art low-cost manufacturing technologies.
HLR 6	Engine should use FADEC and electric actuators when possible and collect operating data in HMS.
HLR 7	Reliability of engine should reach [1-1.e-4] and availability should reach [1-1.e-4]
HLR 8	Engine should reach Initial Operational Capability (IOC) in [2035]

Engine System Requirements
In addition to the HLRs mentioned above, it is necessary to mention some key Engine System Requirements also used for the study.

Operational domain
The calculated operational domain is shown in Fig. 3. O1 is the nominal performance and design point for the nozzle geometry with a MR = 6 and MCC pressure of 16 MPa. The O/F mixture ratio (MR) in the MCC is throttled between 5.5 (O3) and 6.5 (O2) for nominal thrust performance (93% to 107%). The MCCP is kept between 15 MPa and 17 MPa. E1 to E8 are extreme points that defines the safe limits of IPH operation.

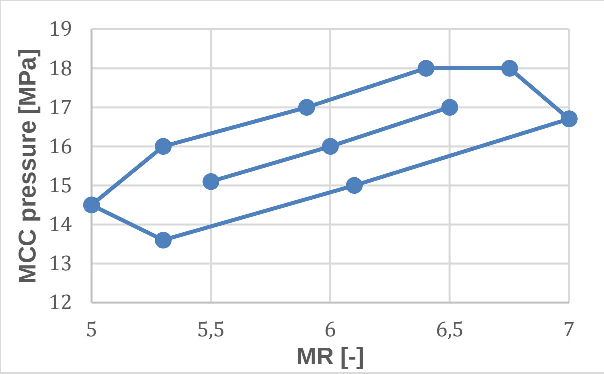

Figure 2: Operational domain for SLME

Expansion Ratio (ER)
Each variant requires different configurations of nozzle expansion ratios depending on the stage number. Based on previous studies done by DLR-SART, the nozzle with an expansion ratio of ε=33 was selected for the de-risk study which is a trade-off of engine performance (thrust and Isp) at different altitudes as can be seen in Fig. 4. This ER is similar to several of the SpaceLiner variants' booster stage ERs.

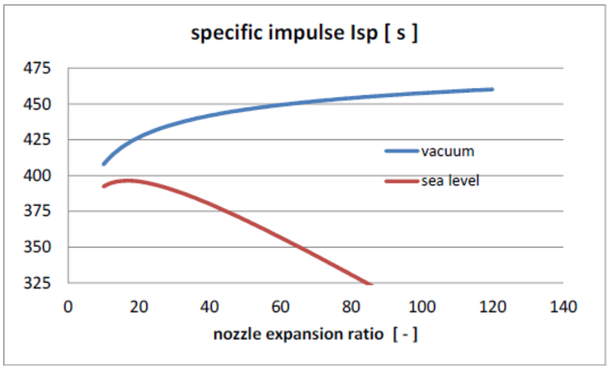

Figure 3: SLME Isp performance as a function of Nozzle Expansion Ratio. [2]

Integrated Power Head (IPH) envelope
The IPH which includes the assembly of turbomachinery, ducts, valves, and pre-burners must fit within a cylinder extruded upward from the nozzle exit diameter. The booster's ER provides the smallest physical envelope which means the IPH will easily pass the envelope requirement for the upper stage nozzles with larger ERs hence larger exit diameters.

Deep throttling
Some of the launcher concepts have deep-throttling requirements of approximately 35% of sea-level thrust (≈740 kN). This is an exceptional requirement for the IPH, falling well outside the regular operational domain depicted in Fig. 3.

Turbine Inlet Temperatures
To designing a robust engine cycle, the pre-burner exit temperature (similar to turbine inlet temperatures) are kept below 760°C to avoid requiring special materials, thermal coatings, or cooling methods. This also increases likelihood of the final design achieving the reusability and lifetime requirements. [2]

2.3. Full flow staged combustion cycle analysis

Figure 4 shows the cycle design for the SLME and Table 2 list the corresponding labelled sub-system. [2]

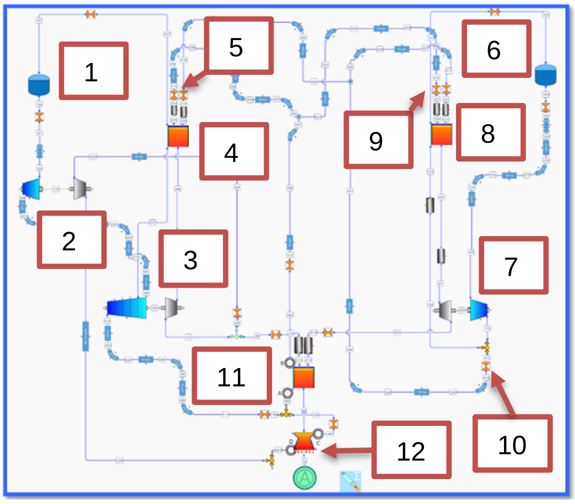

Figure 4: Cycle design in AxStream System Simulation

Table 2: SLME Sub-systems

Diagram label	**Sub-system description**	Fig. 4 label no.
LH2 TANK	LH2 Rocket liquefied gas storage	1
LPFTP	Low Pressure Fuel Turbo Pump	2
HPFTP	High Pressure Fuel Turbo Pump	3
FRPB	Fuel Rich Pre-Burner	4
FPBCV	Fuel Rich Pre-burner Control Valve	5
LOX TANK	LOX Rocket liquefied gas storage	6
HPOTP	High pressure Oxidiser Turbo Pump	7
ORPB	Oxygen rich pre-burner	8
OPBCV	Oxygen rich pre-burner control valve	9
MOV	Main Oxidiser Valve	10
MCC	Main Combustion Chamber	11
Nozzle (NOZ)	Thrust nozzle (incl. regenerative cooling)	12

2.4. Sub-system Boundary Conditions

An iterative design process between the cycle analysis and sub-systems' design points were performed until BCs converged. The turbomachinery performance maps including off-design behaviour were utilized in the cycle design to determine the power balance between the pumps and turbines. The flow control valves before each pre-burner are used to maintain sufficient turbine power by controlling the MRs. Using this strategy to maintain MCCP throughout the operational domain is achieved by allowing higher turbine exit temperatures.

3. SUB-SYSTEM DEFINITION & REQUIREMENTS

The following sub-system technical specifications are results obtained from the De-risk study [2] relevant for test facility planning and selection.

3.1. Turbomachinery

LPFTP

The LPFTP preliminary design is shown in Figure 5.

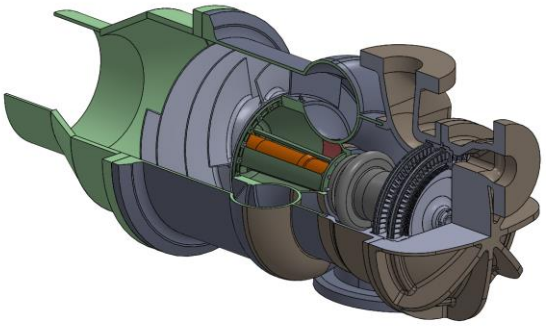

Figure 5: SLME LPFTP preliminary CAD design

Table 3: LPFTP (Pump) specifications

Point	O1	O2	O3	DT
Mass flow rate [kg/s]	75.7	76.2	75.9	43.4
Pump Power [kW]	1766	1981	1769	1690
Pressure ratio [-]	8.5	9.4	8.5	7.8

Table 4: LPFTP (Turbine) specifications

Point	O1	O2	O3	DT
Mass flow rate [kg/s]	9.9	10.6	9.8	7.8
Power [kW]	1755	1962	1769	1752
Axial length [mm]	55.2			
Maximal diameter [mm]	224			

LPFTP Rotordynamics & rotor supports

The LPFTP has a 1st critical frequency above the nominal operational rotor speed with sufficient separation margin (SM) to satisfy API 684 [4]. Therefore, no rotor natural frequencies are intersected during startup, shutdown, and deep throttling procedures. Furthermore, no critical damping is required for the supports to move through a critical speed. This design should simplify the rotor development process. The critical speed map is shown in Figure 6 and the rotor support requirements are listed in Table 5.

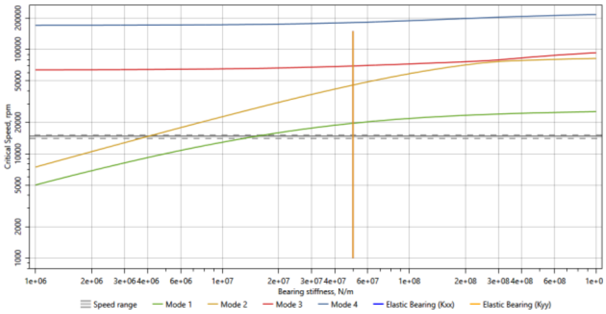

Figure 6: LPFTP rotor critical speed map

Table 5: Dynamic coefficients for the flexible supports of the LPFTP

Bearing	Kxx, N/m	Kyy, N/m	Kzz, N/m	Cxx, Cyy, N*s/m	Czz, N*s/m
Front	5E+7	5E+7	1e+8	645	1289
Rear	5E+7	5E+7	1e+8	645	1289

HPFTP+FRPB

The HPFTP + FRPB assembly is shown in Figure 7. The technical specifications of the pump, turbine and pre-burner are listed in Table 6, Table 7, and Table 8 respectively.

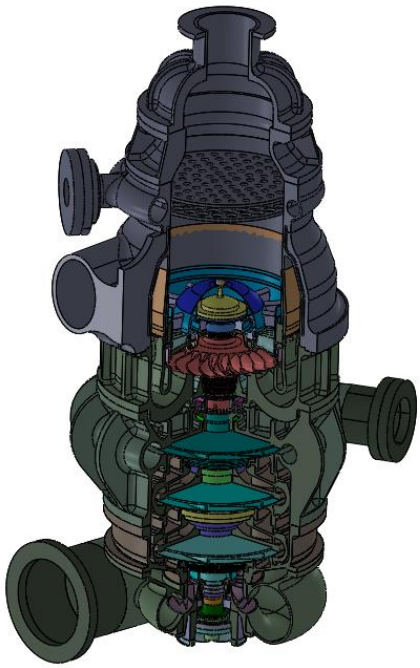

Figure 7: SLME HPFTP preliminary design CAD

Table 6: HPFTP (Pump) specifications

Point	O1	O2	O3	DT
Mass flow rate [kg/s]	75.7	76.2	75.9	43.4
Power [kW]	38710	41030	37994	15024
Pressure ratio [-]	24.6	23.5	24.2	14.4

Table 7: HPFTP (Turbine) specifications

Point	O1	O2	O3	DT
Turbine Inlet Temp	760	815	715	660
Mass flow rate [kg/s]	98	102	96	48
Power [kW]	38269	41219	37296	13855

Table 8: FRPB specifications

Point	O1	O2	O3	DT
Turbine Inlet Temp [K]	760	770	755	660
Cooling jacket H2 MFR [kg/s]	0.385	0.388	0.386	0.221
LOX MFR [kg/s]	37.6	40.5	35.3	15.8
LH2 MFR [kg/s]	60.5	59.99	60.99	32.5
Injector type	Coaxial			

HPFTP Rotordynamics & rotor supports

The operational speed lies between the 2nd and 3rd critical frequencies. The SSME HPFTP operates in the same condition [3]. This is not ideal, but the trade-offs of heavier design and different rotor architecture do not seem worthwhile vs. the effort to properly control and verify the support properties.

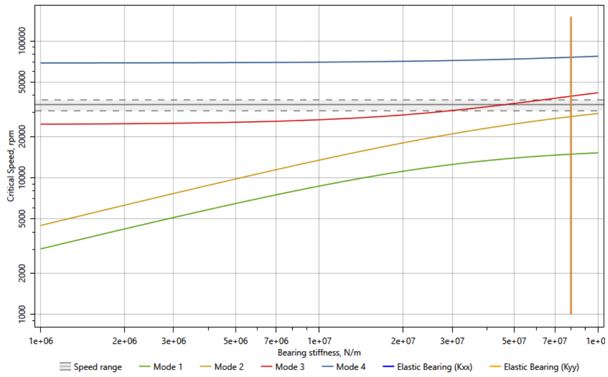

Figure 8: HPFTP rotor critical speed map

Table 9: Dynamic coefficients for the flexible supports of the HPFTP

Bearing	Kxx, N/m	Kyy, N/m	Kzz, N/m	Cxx, Cyy, N*s/m	Czz, N*s/m
Front	8E+7	8E+7	1.5E+8	445	835
Rear	8E+7	8E+7	N/A	445	N/A

HPOTP + ORPB

The HPOTP+ORPB assembly can be seen in Figure 9. The technical specifications for the pump, turbine and preburner are listed in Table 10, Table 11, and Table 12 respectively.

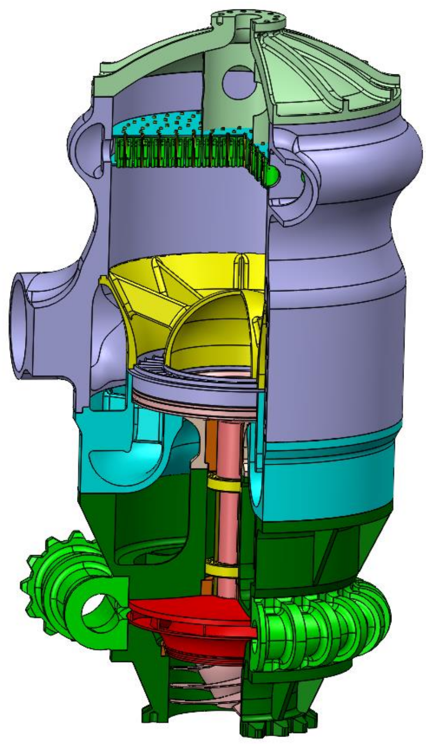

Figure 9: SLME HPOTP preliminary design CAD

Table 10: HPOTP (Pump) specifications

Point	O1	O2	O3	DT
Mass flow rate [kg/s]	441	486	409	180
Power [kW]	15128	17746	14076	3159
Pressure ratio [-]	60.5	71.1	57.1	21.1

Table 11: HPOTP (Turbine) specifications

Point	O1	O2	O3	DT
Turbine Inlet Temp [K]	760	770	755	660
Mass flow rate [kg/s]	406	443	371	163
Power [kW]	15154	17775	14097	

Table 12: ORPB specifications

Point	O1	O2	O3	DT
Turbine Inlet Temp [K]	760	770	755	660
Cooling jacket H2 MFR [kg/s]	5.36	5.85	4.92	2.16
LOX	403	440	369	162

MFR [kg/s]				
LH2 MFR [kg/s]	2.97	3.27	2.71	1.06
Injector type	Coaxial			

HPOTP Rotordynamics and rotor support

The operational speed lies between the 1st and 2nd critical frequencies. This necessitates careful design control of the support stiffness and critical damping requirements. The API 684 separation margins are satisfied for the preliminary design however deep throttling would require the rotor to operate dangerously close to the 1st critical frequency. Either the critical damping must be verified to be robust, or a trade-off must be considered resulting in a heavier turbo pump design.

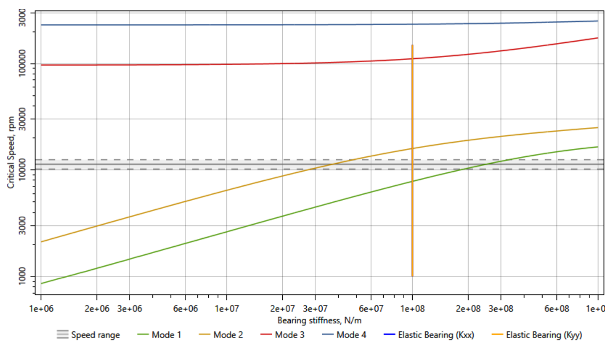

Figure 10: HPOTP rotor critical speed map

Table 13: Dynamic coefficients for the flexible supports of the HPOTP

Bearing	**Kxx, N/m**	**Kyy, N/m**	**Kzz, N/m**	**Cxx, Cyy, N*s/m**	**Czz, N*s/m**
Front	1E+8	1E+8	1E+8	1682	1682
Rear	1E+8	1E+8	1E+8	1682	1682

3.2. MCC & Nozzle (Thrust Chamber Assembly)

Table 14 lists the results for the regenerative cooling system on the Thrust Chamber Assembly (TCA).

Table 14: Regenerative cooling requirements for thrust nozzle.

Parameters	**Unit**	**Value**
Coolant Inlet conditions for Segment 1		
Mass Flow Rate	kg/s	65.33
Inlet Pressure	MPa	30.102
Pressure losses	MPa	2
Inlet Temperature	K	45.31
Coolant Inlet conditions for Segment 2		
Mass Flow Rate	kg/s	12.12
Inlet Pressure	MPa	29.749
Pressure losses	MPa	0.65
Inlet Temperature	K	45.31
Coolant Inlet conditions for Segment 3		
Mass Flow Rate	kg/s	2
Outlet Static Pressure	MPa	0.042
Inlet Temperature	K	Segment 2 Outlet Temperature

4. SLME DESIGN

Figure 11 and Figure 12 shows the top view of the IPH and ducting of the SLME concept developed by SIW-S and DLR-SART for the De-risk study [2]. The assembly as depicted in the image stands about 3.30 m tall and has a nozzle exit diameter of about 1.78 m in diameter. These two dimensions generally define the outer envelope of the engine. However, the HPOTP and to a lesser extent the HPFTP breach the ESR to remain within the exit diameter of the nozzle skirt.

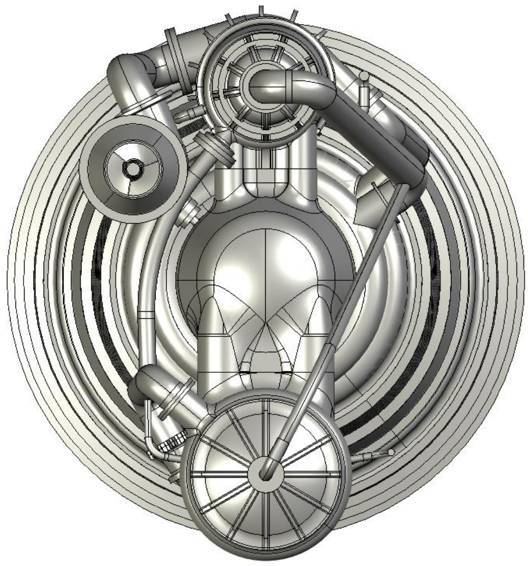

Figure 11: SLME integrated power pack top view

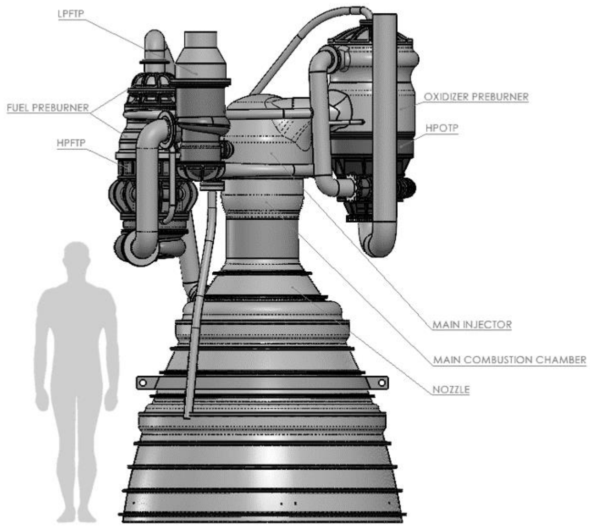

Figure 12: SLME integrated power pack side view with nozzle ε=33.

5. INTEGRATED LOX POWER PACK (IOPP)

The De-risk study revealed the HPOTP is severely sensitive to the inlet conditions in terms of cavitation. The LOX feed line from the LOX tank requires several duct bends to reach the bottom of the HPOTP inlet. The proximity of the bend to the HPOTP inlet is also a concern for asymmetric flow conditions and cavitation. Furthermore, the HPOTP duct to the ORPB induces additional pressure losses.

It is proposed to develop an integrated oxygen pump, preburner, and turbine in-line with the MCC. The general concept is shown in Figure 13. The flow and pressure requirements are the same as in Table 10, Table 11, and Table 12. IOPP aims to significantly lower the inlet pressure requirements which means lower LOX tank pressure and mass. The SpaceX Raptor engine utilizes a similar concept but with methane fuel. DLR-SART also proposed a similar concept for a LOX/LH2 engine in 2018 [13]. The SSME "Derivative engine" was also similar but importantly not in-line with the MCC [14].

In addition to the cavitation advantages, it would aid in the removal of several heavy components like the ducts between LOX tank and HPOTP, between the HPOTP and ORPB, and bulky HPOTP turbine outlet manifold (connected to the MCC).

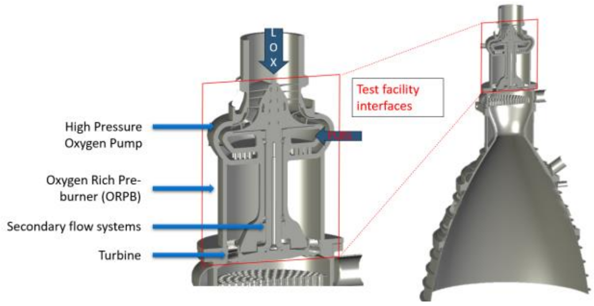

Figure 13: Integrated HPOTP + ORPB in-line with MCC or Integrated Oxygen Power Pack (IOPP)

6. CRITICAL TECHNOLOGIES

The critical technologies section will be focussed on IOPP. The LPFTP and HPFTP are relatively "standard" technologies, and many examples can be found of similar sub-systems being developed. Furthermore, they happened to have very similar architectures to SSME which makes benchmarking and development somewhat more predictable.

6.1. Materials and manufacturing

New material developments and manufacturing methods are one of the most effective ways to improve the engine performance. Material selection is usually based on five general factors. [12]

1. Size of the engine
2. Engine duty cycle (expendable or reusable).
3. The propellants
4. Turbine drive cycle
5. Stage type, booster or upper

For IOPP, the hydrogen exposure at elevated pressure and low temperature is limited in the injector inlet manifold but must be considered. The pump impeller will be exposed to LOX and the turbine GOX. The cooling systems will be primarily exposed to LOX during operation. This also means the bearings will be cooled, purged, and lubricated with LOX.

Oxygen rich compatibility

Materials not compatible with oxygen rich environment can quickly ignite and burn. This is a major development challenge. Oxygen ignition resistant alloys should be used throughout IOPP. Designing for high pressure, hot oxygen rich environments requires more considerations than just the material choice. The design considerations are however closely related to the material properties. The configuration of the geometry significantly influences the flammability and ignitability of the material. Ignition can occur due to several factors like particle impact, rapid pressurization, resonance heating, mechanical impact, friction heating, etc. NASA has a guide of oxygen compatibility assessments [20].

Hydrogen rich compatibility

Hydrogen environmental embrittlement damage, internal hydrogen embrittlement, or hydrogen reaction (hydrogen formation) must be considered. Internal hydrogen embrittlement was mitigated by the SSME by using single crystal PW1480 blades for the turbines however this can be a procurement challenge in low budget, rapid development programs. For IOPP, single crystal superalloy might not be required because all H2 should be combusted before it reaches the turbine. Single crystal superalloys are a consideration for the HPFTP turbine. However, for the injector inlet manifold LH2 compatibility will be considered. Coatings like gold or copper or iron-based overlays are optional if required. [12]

High temperature superalloys

By maintaining Turbine inlet temperatures below 600 °C (1112 °F) for nominal thrust conditions, a relatively common super-alloy like Inconel can be used. For the case of 107% thrust, the HPFTP TIT reaches a maximum of 540 °C (1004 °F) which is well within the limits of Inconel 7-series. The typical tensile strength of 713C is shown in Figure 14. [18] If single crystal superalloy like PWA1484 is used, the temperature limits are even higher, approximately 870 °C (1600 °F). [19]

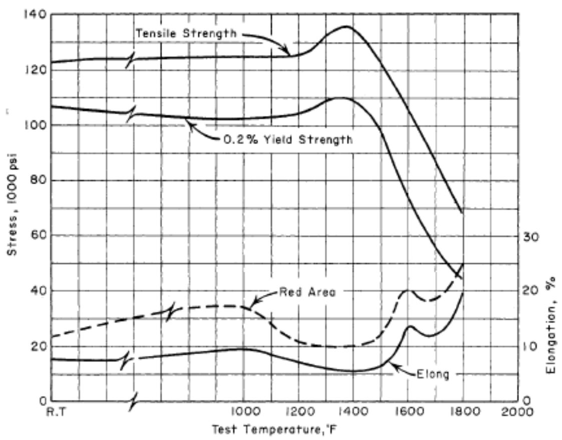

Figure 14: Typical tensile properties of as cast, vacuum melted, vacuum cast alloy 713C. [18]

Large ΔT transient loads

For the burner structures like casings, liner, baffles etc. where large temperature changes are expected (>Δ1000°C), a common choice is another Nickel based superalloy, Inconel 718 because of its excellent low and high temperature mechanical properties. There are other considerations, but Inconel 718 is offered as a common example. Alternative material options are discussed in the next sections for injector and burner structures.

Additive manufacturing (AM) & Ceramic Matric Composites (CMC)

Compared to traditional manufacturing methods, AM and CMC are rapidly developing technologies. In the context of IOPP, there are opportunities for employing this type of technology in the design. Due to the potentially compact design and temperature requirements of the bearing, large thermal gradients are anticipated. Hence, alternative materials like ceramic matrix composites (CMC) might be considered in areas where temperatures exceed the capabilities of typical high temperature alloys.

Additive manufacturing (AM) and Ceramic Matrix Composites (CMC) are both major topics on their own and all the considerations for use in rocket engine applications cannot be covered in this paper. CMC is considered attractive mainly due to the thermal properties, especially high-temperature applications. Figure 15 shows various AM methods and their performance in terms of deposition rate vs. feature size. Other considerations are bed sizes (and height), surface finishes, material strength, pre & post-processes, LOX, GOX, LH2 & GH2 compatibility, repeatability of the process, residual stresses and distortion, governing standards, certification and many more. Some advantages to use AM is the reduction in lead times and ability to produce complex designs for improved performance previously not achievable with traditional manufacturing.

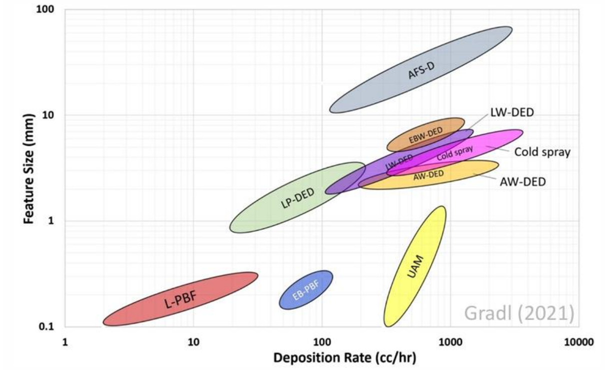

Figure 15: AM deposition rate vs. feature size for various AM methods. [16]

Figure 16: Example applications produced with AM method Laser powder bed fusion (L-PBF). [16]

CMC has been tested for preburner injectors for example in 2014 by DLR-SART [15] which showed promising results. The hot-fire test is shown Figure 17. CMC has also been tested for turbine blades and disks.

Figure 17: 30 bar hot-run of an integrated CMC thrust chamber assembly including ceramic injector. [15]

AM development for combustion chamber is progressing well in Europe. For example, Ariane group has had success with its hot-fire tests and plans to use the technology in its future launch programs. [11]

6.2. Turbomachinery development

Pump and turbine development

Pump and turbine testing at component or sub-scale level requires additional test facilities. In the USA the Marshal Space Flight Centre (MSFC) has a variety of testing facilities for turbo pumps, turbines and nozzles.

- It has a continuous flow water facility providing a controlled simulated environment.
- MSFC also has a blowdown system providing a controlled simulated environment to the inlet and exit of turbines. [7]

Rotor assembly

Harmonic vibration and rotor imbalance are major sources of energy through the system and is one of the key contributors to high-cycle fatigue (HCF). Spin testing and rotor balancing are standard rotor development procedures and commercial services are readily available with companies like Schenck-Rotec. [21] These processes are a combination of development, quality control, and production processes.

Rotor resonance frequencies (and multiples) must be verified. The rotordynamics simulations are highly dependent on stiffness and damping coefficients for accurate modelling. Depending on the support structure's architecture, the equivalent structural stiffness and damping must be verified through testing. Blade harmonics must also be verified. Spectrgrams produced from tests can be compared (and overlaid) with simulation Campbell diagrams to verify rotor frequencies.

IOPP will most probably have two overhung rotors which poses some additional challenges for rotordynamics and balancing in terms of stability and deflection. The dissimilar in Coefficients of LTEs can be a challenge and needs to be verified in rotor test setups.

Pump cavitation

Cavitation has detrimental effects on pump performance and service life. Furthermore, cavitation can cause significant vibration that can negatively impact the rest of the IPH performance. Figure 18 shows a simplified example of a Pump Cavitation test facility schematic [5]. The vertical orientation is preferred to avoid hydrostatic influences at the impeller inlet. The water is heated to simulate similar fluid properties for the inlet conditions of LOX and LH2.

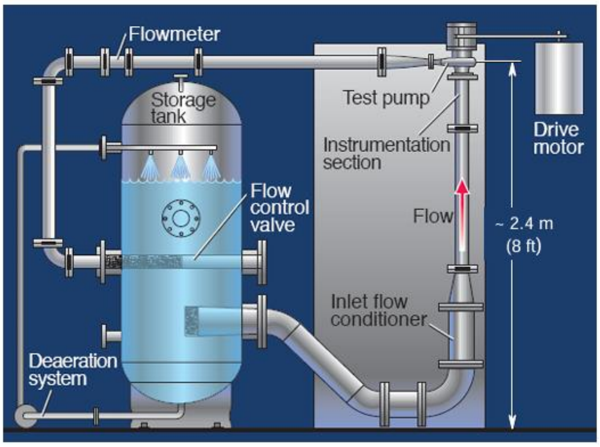

Figure 18: Cavitation test facility schematic [5]

Dynamic seals & Secondary flow systems

Characterizing the seal performance is critical during the development phase to verify the impact on spool balance which in turn effect bearing life, pump efficiency, bearing lubrication, and internal thermal management. From a development perspective much of the secondary flow used for cooling and lubrication is a loss for efficiency and needs to be optimised. This requires component/sub-system V&V efforts as early as possible in the project which usually requires sophisticated test setups. The challenge with testing secondary flow structures is creating a representative and worthwhile test. A careful trade-off between part-assembly and full-assembly must be made in terms of risk, budget, and schedule. It is preferable to do these type of test setups "in-house" and during the "Development" phase of the project.

An advantage of the FFSC is that it has an ORPB. The HPOTP does not have the same risk as being driven by a FRPB when leakage between the fuel and oxygen can lead to fire as was the case with the SSME during development. That dynamic seal in the SSME HPOTP required a significant amount of development effort, mass, and space.

Bearings
The required loads and speeds for the IPH bearings are not out of the ordinary. The cryogenic and high temperature requirements are relatively common requirements for the industry. The challenges around bearings are maintaining the bearing requirements in terms of pre-load, lubrication, purging, cooling, and fits which lie more with the structural and secondary flow design. Excessive relative thermal deformation is detrimental to proper bearing operation. V&V efforts in representative environments are required for bearings. Multiple bearing test setups will be required because it is so difficult to simulate the final working conditions.

Custom bearings can be subcontracted from bearing specialists like ADR-ALCEN for example. Integrated bearings can be considered for increased performance, lightweighting and smaller packages [10]. Custom high-performance bearings are relatively commonplace from a procurement perspective however they can have very long lead times (up to a year or more) for first small batch orders. The challenge is getting to a preliminary design mature enough to select and order bearings to have in time for bearing verification tests. There is a limited selection of material for the balls, races, and cages.

6.3. Oxygen Rich Preburner (ORPB)

In the European context, experience with Oxygen rich preburners is limited. The recent success of engines like the SpaceX Raptor which uses a FFSC engine with ORPB (and integrated turbomachinery) has proven the viability of the technology. Europe has an opportunity to leap ahead and become a technology leader by learning from what has been done in addition to utilizing LOX-LH2 propellants with the in-line architecture is unique. Score-D, a European 200t thrust engine reached PDR level in development.

Injectors
The injector selection affects the injector head total diameter and cooling requirements. IOPPs space for burner is more constrained than the traditional configuration. The preburner has a typical gas turbine configuration and must also compete with the shaft tunnel running through the centre. The usual injector risks have to be tested. [22]

1. Late ignition resulting in increased transient temperature and pressure loads.
2. Low-frequency pressure oscillations in micro and macro structures.
3. High-frequency pressure oscillations in the preburner.
4. Thermal stratification within expected ranges.
5. Preburner performance robust and within expected margins in steady-state and transient.

6.4. Thrust nozzle

Regenerative cooling requires small flow channels as well as multiple metal types throughout the nozzle structure which imposes several manufacturing, quality control, and test challenges for the development. Examples of design considerations for nozzles include: [8]

- Thinner hotwalls to balance cooling with increased heat fluxes.
- Balancing coolant channel dimensions (incl. surface roughness) with pressure drop profiles.
- Ability to produce robust joints at increased bond joint temperatures.
- Ability to inspect the bonding of the closeout to the channel lands.
- Reduction in assembly build hours and manual processing.
- Reduction in lead time for materials or processes.
- Various options for materials and combinations (i.e. monolithic, bimetallic and multi-metallic).
- Direct build and/or simplified attachment of manifolds.
- Increased system performance through nozzle weight reduction or hydraulic performance.

The MCC & nozzle also known as Thrust Chamber Assembly (TCA), comprises of several technology categories that require V&V during development. The structure is complex and subject to extreme dynamics loads, high pressures and large thermal gradients. It is a large thin structure with excessive amount of joining surfaces, often between dissimilar metals. This makes the structure vulnerable to fatigue and cracking in areas that are very difficult or unable to inspect properly between flights.

Injector head characterization
Injector performance tests can be performed before hot tests to verify pressure drops, leakage, burst tests, and spray characteristics.

Sideloads
Sideloads are a type of dynamic loading that is one of, if not the greatest risk for the nozzle's structural integrity. Sideload testing has been performed in the past at DLR's P6.2 test stand. Similar tests must be considered. [9]

Figure 19: Various manufacturing technologies considered for channel wall nozzle fabrication (CWNF) [8]

7. Test facilities

For most of the sub-system test requirements, the facilities in Europe are available however some of the flow requirements would require significant upgrades. The availability and limitations of European test facilities are an important consideration for the technology development roadmap. The SLME FFSC sub-system full-scale flow requirements are high.

- **The SLME LOX requirement is up to 440 kg/s at 350 bar.**
- **The SLME LH2 requirement is up to 80 kg/s at 360 bar.**

DLR Lampoldshausen has several test facilities. [24]

- The P3.2 facility can be used for preburner standalone tests but is limited to about 300 kg/s for LOX.
- P8 for testing injection systems up to 330 bars.
- The P5 facility is for staged combustion hot engine tests.

Figure 20 shows SCORE-D integrated into P5. The SLME engine will easily fit within the test rig dimensions shown.

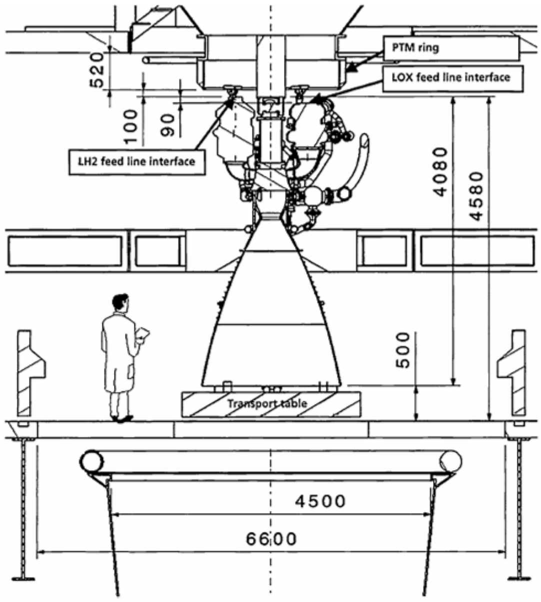

Figure 20: Integration of SCORE-D in P5 test cell. [22].

At Snecma, Vernon rocket test facility,

- PF52 for cryogenic engines and sub-systems ideal for standalone turbopump tests shown in Figure 21. [23]

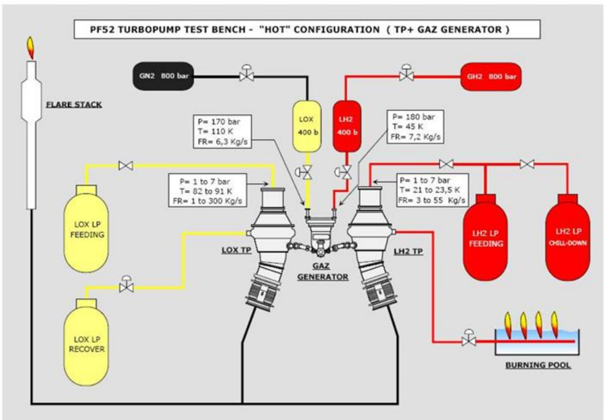

Figure 21: "HOT" configuration for PF52 turbopump test bench at Snecma, Vernon. [23]

7.1. Turbomachinery test facilities

Several component and sub-system level test facility requirements have been discussed.

- Rotor spin chambers and overspeed tests
- Balancing facility and rotor signatures

- Pump characterization including cavitation profiling with continuous hot water system.
- Turbine characterization with blowdown facilities.
- Bearing tests; lubrication, dynamic loads, pre-load, heat generation, lifetime.
    a. Component level or shaft assembly
    b. Included in full rotor assembly rotordynamics tests, spin chamber
    c. Final assembly hot tests, main test facility
- Static structure vibration tests, random and directional harmonics. Shaker facilities
- Material compatibility and corrosion tests, material science lab work
- Material mechanical property tests, material science lab work.
- Secondary flow system (SFS) including seals: Custom sub-system assembly tests.
- Final turbopump assembly tests bench.

7.2. Preburner test facilities

- Injector and injector head performance, cold flow test bench
- Injector head performance, sub-scale fire tests bench.
- Preburner casing pressure tests bench
- Preburner performance, full-scale fire tests bench
- Integrated cooling system test bench

7.3. MCC & Nozzle testing

Some of the test benches envisaged could be combined into a single facility but will still required functional specific setups.

- Cold flow test bench, possibly blowdown
- Sideload test bench (DLR, P6.2 [9])
- Shaker for random and directional vibration tests.
- Pressure test bench
- Thermal cycle test bench

8. Technology Road Map

A technology road map to develop a 200t SLME demonstrator in 8 years to TRL 7 is in Figure 22. A mid-scale approach is presented which would produce a 100t to 120t mid-scale engine. The road map also presents the need to begin with a small-scale technology development program with the aim to produce a fully operational FFSC engine demonstrator. As indicated on the road map each sub-scale version can branch off to further mature into a product dedicated to its appropriate thrust range. The plan is founded on developing each sub-system, listed in Table 2, in parallel.

The following IPH grouping or units of sub-systems can be developed by lead partners until M18 (SIR):

1. LPFTP + HPFTP + FRPB
2. HPOTP + ORPB
3. MCC + Thrust Nozzle (TCA)

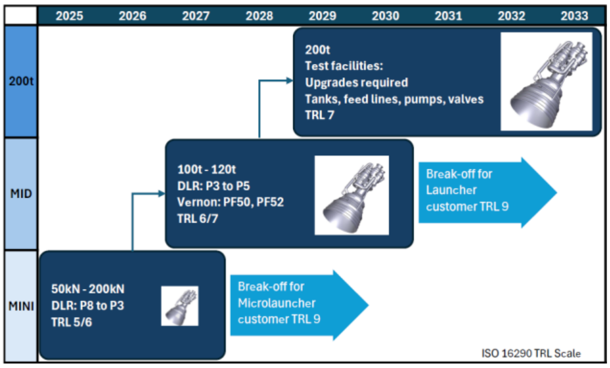

Figure 22: Technology Road Map

The road map for the MINI presented in the is a sub-part and/or sub-scale version of the SLME FFSC engine. The purpose of the small scale and mid-scale engines is partly to verify and validate the overall engine concept and critical technologies before major investment is committed for upgrading test facilities (or while test facilities are upgraded in parallel). One of the key challenges is to have a launcher (or series of launchers) in partnership whilst developing the propulsion systems.

To achieve the ambitious 3-year schedule for the MINI, the goal is to use as much modern manufacturing methodologies as possible, like AM and CMC. This should shorten iterations between prototypes during development and it is the goal to find large margins in the design and application of ceramics.

9. Conclusion

Europe's rocket test facilities currently cannot supply all SLMEs required flow rate and pressures. The required test facility upgrades (including auxiliaries, tanks, sound suppression etc.) would most probably be a significant portion of the program budget. Several development paths are possible depending on risk appetite and funding available. The most aggressive technology road map (3-year program) is presented which aims to develop only the sub-systems required for an integrated hot test at P8 and/or P5. This sub-scale and sub-part strategy should negate significant test facility upgrades. After the critical technologies like FFSC and Oxygen rich sub-systems have been validated, additional funding and programs can be justified.

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