System Aspects of European Reusable Staged-Combustion Rocket Engine SLME

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The full-flow staged combustion cycle rocket engine with a moderate 15 to 17 MPa range in nominal chamber pressure called SpaceLiner Main Engine (SLME) has been under investigation by numerical simulations since several years. Originally defined as the baseline propulsion system for the reusable rocket-based high-speed intercontinental passenger transport concept SpaceLiner, the SLME is now also used as a reference for closed cycle LOX-LH2-engines in several studies of future European RLV. A summary overview of the studied reusable cryogenic staged combustion cycle engines in Europe including related launcher applications is provided.

Currently, the Swiss company SoftInway and DLR are jointly performing a de-risk study under contract to the European Space Agency (ESA) to preliminarily consolidate a staged combustion engine design. The launcher system high-level requirements on the main propulsion system as well as the engine system level requirements have been defined by DLR. The commercial AxSTREAM® software tool is implemented for the pre-design of the turbomachinery, preburners and main combustion chamber regenerative cooling. A preliminary engine control logic is established. Consolidated size, mass, and performance data are available by this analysis and are integrated in the engine model. Further, different engine architecture options like two side-mounted integrated power-heads or in-line oxygen pump and preburner are mechanically sized and results are evaluated in the paper.

Nomenclature

c*	characteristic velocity	m / s
I_{sp}	(mass) specific Impulse	s (N s / kg)
M	Mach-number	-
T	Thrust	N
m	mass	kg
ε	expansion ratio	-

Subscripts, Abbreviations

3STO	Three-Stage-To-Orbit
DRL	Down-Range Landing site
FADEC	Full Authority Digital Engine Control
FFSC	Full-Flow Staged Combustion
FRSC	Fuel-Rich Staged Combustion
FTP	Fuel Turbo Pump
HTHL	Horizontal Take-off and Horizontal Landing
IHPRPT	Integrated High Payoff Rocket Propulsion Technology
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MCC	Main Combustion Chamber
MECO	Main Engine Cut Off
MR	mixture ratio
NPSP	Net Positive Suction Pressure
MSFC	Marshal Spaceflight Center (of NASA)
OTP	Oxidizer Turbo Pump
RLV	Reusable Launch Vehicle
RTLS	Return To Launch Site
SLB	SpaceLiner Booster stage
SLME	SpaceLiner Main Engine
SLO	SpaceLiner Orbiter stage
SLP	SpaceLiner Passenger stage
SSME	Space Shuttle Main Engine
TET	Turbine Entry Temperature
TRL	Technology Readiness Level
UDMH	Unsymmetrical Dimethyl Hydrazine
VTHL	Vertical Take-off and Horizontal Landing
VTVL	Vertical Take-off and Vertical Landing
C	chamber
s/l	sea level
vac	vacuum

1 INTRODUCTION

A full-flow staged combustion cycle rocket engine with a moderate 15 to 17 MPa range in chamber pressure called SpaceLiner Main Engine (SLME) has been under investigation by numerical simulations [9 - 15] since several years. Originally defined as the baseline propulsion system for the reusable rocket-based high-speed intercontinental passenger transport concept SpaceLiner, the SLME is now also used as a reference for closed cycle LOX-LH2-engines in several studies of future European RLV summarized in the following section.

The SLME being quite unique until recently in its FFSC-architecture [15], now a surprising number of proposals have been come-up in Europe sharing similar thrust-levels and closed cycle flow schematics. A partially similar staged combustion LOX/methane-engine in a range from 2000 kN up to 2500 kN is now under investigation in France under the name PROMETHEUS-X [4]. A similar high-performance 2200 kN LOX-LH2-engine is under development in China for propulsion of China's heavy launch vehicle core [5].

2 CLOSED CYCLE ENGINES STUDIED FOR RLV CONCEPTS

2.1 SLME in RLV-studies

The SLME has been used as realistic baseline for the next generation of European staged-combustion cycle LOX-LH2 rocket engine and has been implemented by DLR in several studies as RLV main propulsion [e.g. 1, 2, 3]. As the applications are different also the most appropriate nozzle expansion will differ.

2.1.1 SpaceLiner 7 and 8

The key premise behind the original concept inception is that the SpaceLiner ultimately has the potential to enable sustainable low-cost space transportation to orbit while at the same time revolutionizing ultra-long-distance travel between different points on Earth. [7, 8] An important milestone has been reached in 2016 with the successful completion of the Mission Requirements Review (MRR) initiating the concept's maturing from research to structured development [16].

The DLR-proposed SpaceLiner is not the only launcher concept designed for high reusability and multiple mission capabilities. In the U.S. the commercial company SPACE EXPLORATION TECHNOLOGIES CORP. (SpaceX) is pushing developments in similar direction: Using two-stage rocket-powered reusable vehicles for different kinds of missions: to LEO, to Moon and Mars, and as an ultra-fast point-to-point cargo and passenger transport on Earth [17]. Recently, the Starship-launch vehicle [6] has achieved significant progress in its flight testing.

The SpaceLiner general baseline design concept consists of a fully reusable booster and passenger stage arranged in parallel as presented in Figure 1. All rocket engines should work from lift-off until MECO. A propellant crossfeed from the booster to the passenger stage is foreseen up to separation to reduce the overall size of the configuration [14].

The SpaceLiner 7 passenger transport is also technical basis for a two-stage fully reusable satellite launch vehicle. The external shapes will be very similar. The satellite launch configuration is described in more detail in [16].

Recently the definition of SpaceLiner 8 has been started [18]. The key-requirements on the SLME are kept similar but note in Table 1 that the number of engines has been raised from 9 to 10 on the booster side.

2.1.2 Alternative RLV-concepts

System studies of future European RLV configurations with partial reusability of 1st or booster stages in tandem stage arrangement for different return and recovery modes, as well as propulsion options have been under investigation in DLR. Within this study, different propellant combinations and engine cycles were considered to identify the impact and challenges on launcher system level, especially with regards to reusability. In this context, launch systems using LOX/-LH2 and staged combustion engines in both stages were designed. The respective rocket engine was scaled based on the SLME, however with some specific features distinct to the baseline SLME.

A short individual description of many of these RLV-concepts has been included in [15]. Sketches of the vehicles are shown in Figure 2 and technical requirements on these SLME variants are summarized inTable 2.

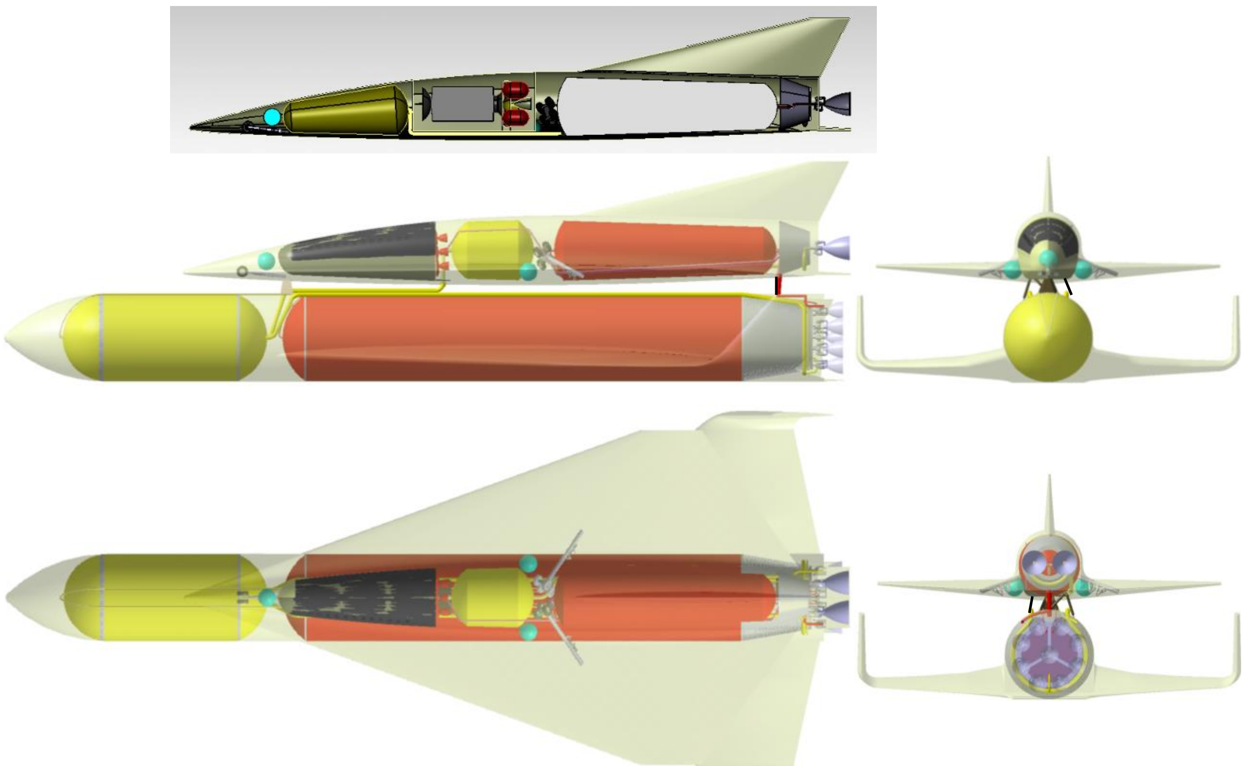

Figure 1: Sketch of SpaceLiner 7 launch configuration with passenger stage (SLP) with its booster stage at bottom position and orbital stage of SLO in insert at top showing the SLME arrangement in the lower right figure

Table 1: SpaceLiner 7-3 and SpaceLiner 8 characteristics of SLME

	SpaceLiner 7-3		SpaceLiner 8	
Stage	SLB	SLP/SLO	SLB	SLP/SLO
Mass flow [kg/s]	513.5 (100%)			
O/F [-]	5.5 – 6.5		5.5 – 6.5	
Throttleability	92.6% - 108.6%	93.3% - 107.36%	92.6% - 108.6%	93.3% - 107.36%
MCC pressure [MPa]	16 (100%)			
Expansion ratio [-]	33	59	33	59
No. of missions	25	25	25	25
No. engines per stage	9	2	10	2

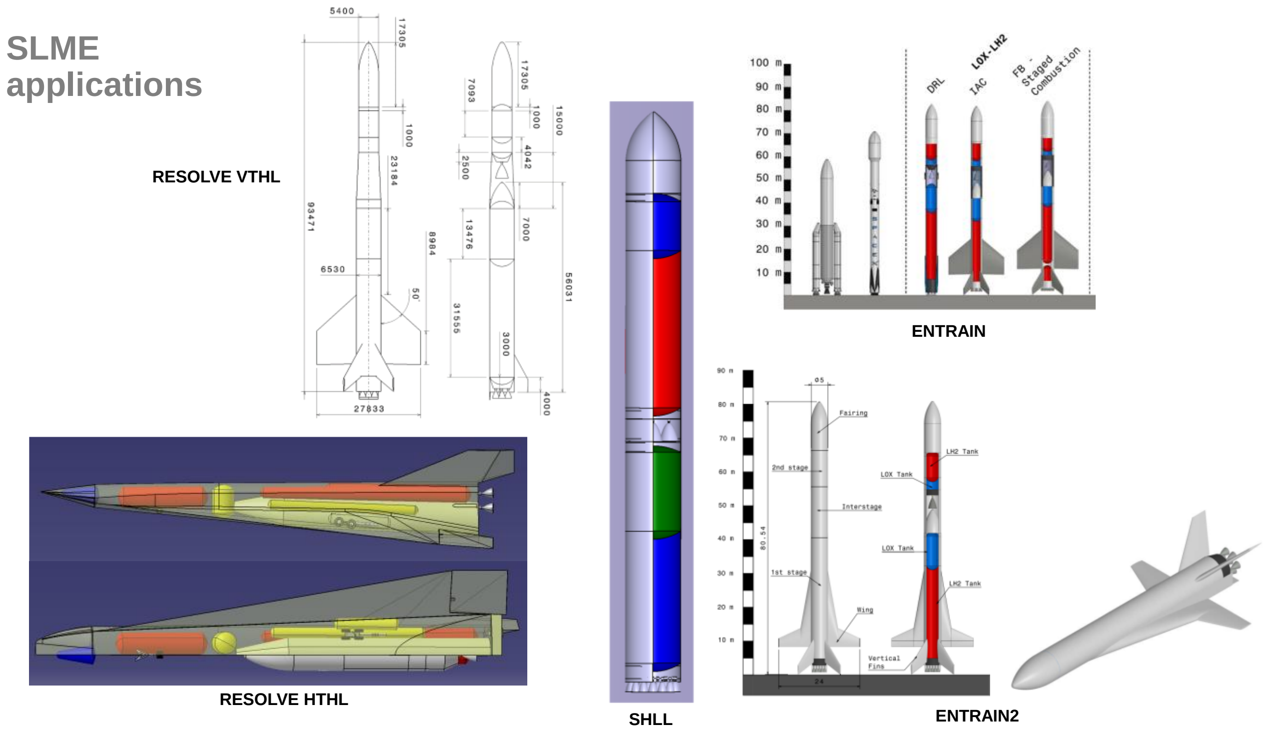

Figure 2: Some alternative RLV-applications of SLME variants

Table 2: Alternative RLV-application characteristics of SLME

Study name	RESOLVE HTHL		RESOLVE VTHL		PROTEIN SHLL		ENTRAIN VTVL		ENTRAIN VTHL		ENTRAIN 2 VTHL	
Stage	BS	US	BS	US	US (LE)	US	BS	US	BS	US	BS	US
Mass flow [kg/s]	555				517	489	145		200 to 240		220	
O/F [-]	6						6					
Throttleability	-	-	-	-	30%	-	30%	-	-	-	-	-
MCC pressure [MPa]	16				20	16	16					
Expansion ratio [-]	32	116	38	120	36	120	23	120	35	120	35	120
Thrust, SL [kN]	1960	-	1933	-	-	-	560	-	757 to 907	-	768	-
Thrust, vacuum [kN]	2209	2321	2225	2323	2200	2200	609	653	858 to 1030	900 to 1079	897	949
Isp, SL [s]	388	-	383	-	389	-	394	-	386	-	386	-
Isp, vacuum [s]	437	460	441	460	434	459	428	459	438	459	438	459
No. of missions	25	25	25	25			6	6	15	15	15	15
No. engines per stage	4	1	5	1	1	6	9	1	5 to 9	1	7	1

Table 3: Other reusable LOX-LH2 SC-cycle engines in applications and in European studies

Study name	**EVEREST**		**FESTIP- FSSC 15 oae** [19]		**FESTIP- FSSC 16 FR**		**SSME**	**RD-0120**	**SCORE-D** [21]
Stage	BS	US	NE un-deployed	NE deployed	BS	US	Space Shuttle	Energia Core	Ground-Demonstrator
Mass flow [kg/s]	384	384	482.3		300		514	439	-
O/F [-]	6		6.6		6.6		6.03	6	6
Throttleability	25% to 120%		-		-		67% to 109%	100% to 106%	-
Cycle	FRSC		FRSC		FRSC		FRSC	FRSC	FRSC
MCC pressure [MPa]	16		24.45		15		20.64	21.9	15
Expansion ratio [-]	29.7	59.6	50	140	65.84	65.84	78	85.7	-
Thrust, SL [kN]	1476	1361	1792	-	1020	1020	1860	1526	-
Thrust, vacuum [kN]	1640	1690	2021	2173	1315	1315	2279	1961	1373
Isp, SL [s]	391	360	379	-	346.7	346.7	366	353	-
Isp, vacuum [s]	435	448	427	460	447	447	452	455	-
No. of missions	25	25	?		?	?	actual 12	1	4 cycles
No. engines per stage	5	2	3		5	2	3	4	-
Study / operat. period	2003-2005		1994-1998				1981-2011	1987-88	up to 2013

Since the total take-off mass of the ENTRAIN RLV is much lower than that of the SpaceLiner (compare around 400 tons to 1830 tons of SpaceLiner), and a further design requirement was to equip both 1st and 2nd stage with the same engine, except for the nozzle expansion ratio, the thrust per engine had to be adapted. The SLME in the 1st stage of VTVL has an expansion ratio of $\epsilon = 23$ and the 2nd stage of $\epsilon = 120$. Furthermore, these engines of the 1st stage have to be throttleable in a range of 33% - 100% in order to allow a soft landing with a T/W ratio close to 1. Such deep-throttling requirement for landing is new compared to the SpaceLiner application. For the VTHL variant without supersonic retro-burn and no vertical-landing burn requirement the nozzle expansion ratio ϵ is set to slightly higher 35.

The horizontal landing (HL) concepts of the RESOLVE-study saw a mission-optimized nozzle expansion in the RLV-stage between 32 and 38 and 116 to 120 in the expendable upper stage.

2.2 Other relevant RLV concepts with SC engines

Table 3 lists the technical data for six other RLV systems utilising SC-engines. Note, only two of them have ever been flown and, hence, reached a TRL of 9: the SSME flown in 135 flights of the US Space Shuttle and the RD-0120 flown twice in the Soviet Energia core stage.

The European RLV-studies FESTIP and EVEREST also considered LOX-LH2 SC-cycle engines. In FESTIP an ambitious chamber pressure of above 24 MPa was assumed, exceeding that of the existing engines at the time SSME and RD-0120. This had been already critically evaluated in [19] together with the assumption of large, in-flight deployable nozzle extension (NE).

3 SLME AS MAIN PROPULSION SYSTEM

Staged combustion cycle rocket engines around a moderate 16 MPa chamber pressure were chosen early in the SpaceLiner definition [7]. This level is not overly ambitious and has already been exceeded by operational values of engines like SSME or RD-0120 (Table 3) and now Raptor. The target of 16 MPa is also a good compromise between European expertise [20] and required performance of future launcher applications. The intended demonstrator SCORE-D [21] was the latest ESA-funded design and experimental work on maturing closed cycle rocket engines. The design chamber pressure would have been 15 MPa (Table 3).

The expansion ratios of the SpaceLiner booster and upper stage engines are to be adapted to their respective optimums; while the mass flow, turbo-machinery, and combustion chamber are intended to remain identical as far as possible and useful. This approach would allow for significant reduction in development-, testing-, and production costs. In certain applications with an expendable upper stage (see section 2.1.2 and Table 2) the reusable booster engines might perform a final mission with additional nozzle extension when approaching its design life-time.

This section summarizes the historical background, lists the newly defined high-level and engine system requirements, followed by describing cycle conditions and latest preliminary sizing of components for the SLME with nozzle expansion ratios 33. The upper stage variant with ε=59 or different RLV-applications as described in section 2 mainly differ in slightly changed nozzle area ratios but will function with very similar internal operating conditions. Therefore, the preliminary component designs are relevant for all these applications.

3.1 Previous SLME Analyses

The best mixture ratio of the SpaceLiner main propulsion system along its passenger mission has been defined by system analyses optimizing the full trajectory. Nominal engine MR control at two engine operation points (6.5 from lift-off until reaching 2.5 g acceleration and 5.5 afterwards) has been found most promising [9].

Two types of staged combustion cycles (one full-flow and the other fuel-rich) have been considered for the SLME and traded by numerical cycle analyses [9, 10]. A Full-Flow Staged Combustion Cycle with a fuel-rich preburner gas turbine driving the LH2-pump and an oxidizer-rich preburner gas turbine driving the LOX-pump remains the preferred design solution for the SpaceLiner [13] [15].

Without any major adaptations to the cycle architecture of SLME, the propellant feed-system and some components have been preliminarily defined and have been described in [14, 15] together with the engine's calculated operational domain.

3.2 Historical designs of LOX-LH2 Full-Flow Staged Combustion cycle engines

Historically, very few staged combustion engines of the Full-Flow sub-cycle have been realized. Only two non-European FFSC engines have ever been developed which are based on different propellant combinations: The RD-270 (8D420) was the first ever full flow staged combustion rocket engine and was designed and produced by Energomash between 1962 and 1970 [22]. The propellants used were UDMH and N_2O_4 and the targeted chamber pressure was 26.1 MPa. The RD-270 was tested between 1967 and 1969 but never flown. This project was discontinued with the abandoning of the UR-700 launcher. The other engine is the SpaceX Raptor (2) [6] based on LOX-LCH4 propellants which is currently in its flight testing with the intended fully reusable Starship&SuperHeavy launcher.

However, in the US the Integrated High Payoff Rocket Propulsion Technology (IHPRPT) research program has spent significant resources on the FFSC-cycle and in particular for LOX-LH2 propellants. The program was instituted as fifteen-year rocket propulsion technology improvement initiative by US Air Force, Army, Navy, and NASA [25]. Eventually, in 2013 an Integrated Powerhead Demonstration reached 100% power level at NASA's Stennis Space Center [27].

Already before, an FFSC derivative engine of the SSME had been proposed [23] to be operated as a highly variable mixture ratio engine, especially for booster applications. A very compact lay-out of the IPH-Ox with annular preburner around the shaft connecting turbine and impeller was intended. References 14 and 23 show the design of the projected SSME "Derivative Engine" ox-rich power head.

Such advanced designs are probably essential for the success of FFSC-types. Reference 26 performs a systematic assessment of all sub-cycle variants of LOX-LH2 staged combustion engines and lists the major thermodynamic advantage of FFSC as "*allowing a significant increase in the powerhead energy release within the same turbine temperature limits. […] The overall effect is approximately a 10% to 15% improvement in chamber pressure of the full-flow cycle over the conventional [FR] cycle combined with lower turbine temperatures.*". However, at the downside [26] mentions "*full-flow dual preburner cycle options have reduced sea level thrust-to-weight due to the significant weight of the oxidizer rich preburner and 02-rich hot gas manifold*".

The latter can be drastically reduced or even completely eliminated if the ox-rich power head is designed as presented in [23, 24] and potentially mounted directly on top of the main combustion chamber as realized with Raptor.

3.3 SLME design requirements

The ongoing de-risk study has been initiated with consolidation and numbering of SLME key design requirements on mission level or High-Level Requirements (HLR) and derived Engine System Requirements (ESR). The performed preliminary sizing has been based on specific numbers. In case these values could change in future variations or trade-offs, the numbers are still preliminary and set in brackets [ ].

3.3.1 High-Level Requirements (HLR)

The SpaceLiner 7 take-off thrust requirement per engine of around 2000 kN at sea-level conditions remains unchanged to [13, 14]. The nominal operational mixture ratio range reaches from 6.5 to 5.5 with MR of 6.5 in the early flight phase and subsequent throttling to 5.5. Other investigated RLV-applications pick simply one of these operating points and keep MR constant along the full mission. Deep-throttling down to 35% of sea-level thrust (≈740 kN) would be mandatory only for the vertical landing of VTVL-concepts. This demanding value has explicitly *not* been included in the HLR-list and should only be added if needed by selected vehicles.

Table 4: High Level Requirements

	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]

The average engine life-time is targeting 25 missions (HLR 4) or cycles with limited refurbishment effort. The SLB engine thus requires an accumulated operational time of 6100 s (1.7 h). The upper stage engine for SLP and SLO is aiming for almost 11600 s (3.2 h) with 2h 20 minutes at a demanding MR of 6.5. These values demonstrate the technical challenges of realizing a safe and cost-efficient reusable rocket engine.

The next generation of partially reusable launchers will see similar operation times and conditions on the RLV-stages but significantly less-demanding environments on the expendable upper stages. In case of VTVL an in-flight reignition capability of up to 4 times per mission would be required while for all other applications a single ignition per mission is sufficient. Multiple ignitions per missions are not an HLR, similar to deep-throttling. Instead a multiple ignition capability of main combustion chamber and preburners is defined as an engine system requirement.

3.3.2 Engine System Requirements (ESR)

The first system requirement is the closed-cycle engine and the second (ESR2) already defines the FFSC as the preferred choice. This approach should allow avoiding the complexity and cost of additional inert gases like Helium for sealing. The additional power of the ox-rich flow enables lower turbine temperatures and hence less stress, translating into longer turbine life, a key factor for reusable rocket engine life (see section 3.2 and [24, 26].

Typical accumulated life time requirement of the SLME are defined in ESR10, derived of the SpaceLiner's ascent reference mission mentioned in [15]:

- Nominal operation time of Booster engine: 245 s with 122 s @ MR=6.5 and 122 s @ MR=5.5 or earlier cut-off
- Nominal operation time of Passenger Stage engine: 463 s with 336 s @ MR=6.5 and 127 s @ MR=5.5

Table 5: Engine System Requirements

	Requirement description
ESR 1	Engine should be a high-performance closed-cycle.
ESR 2	Preferred sub-cycle solution should be Full-Flow Staged Combustion (FFSC) with 2 pre-burners, one oxidizer-rich, one fuel-rich serving main combustion chamber (MCC) only with fluids in gaseous state
ESR 3	Preliminary cycle scheme: see Figure 3
ESR 4	Main combustion chamber pressure in reference operating point should be 16 MPa
ESR 5	Nozzle supersonic area ratio of reference engine for 1st- / booster-stage operations should be around 33
ESR 6	Engine (following ESR4 & ESR5) weight should target T_{vac} / W of 75
ESR 7	Engine should stay compact and it should be avoided that any subsystem, harness, line or turbomachinery component is extending outside an envelope cylinder extruded upward from the nozzle exit plane
ESR 8	Engine should be capable of being adapted to derived version[s] applied to upper-stage operation using nozzle with increased area ratio without major design changes upstream of [nozzle throat]
ESR 9	Engine mixture ratio (MR) should be variable for nominal operating points in the range 5.5 – 6.5 and off-nominal range should extend to at least 5 – 7
ESR 10	Typical engine operation time per mission should be up to 250 s (1st stages application) and up to [500] s (upper stage application) – accumulated life time 6250 s or [12500] s
ESR 11	Expendable variant of upper stage engine could be an option, potentially built out of modified used 1st stage engines reaching accumulated life time [3000] s
ESR 12	Engine should have multiple ignition capability on pre-burners and MCC up [2] times per flight
ESR 13	Maintenance and refurbishment effort of engine between flights should be minimized and not exceed [5 tbc] % of production costs.
ESR 14	Engine propellant-supply interface conditions should be for H2 [0.2 MPa, 20.5 K] and for O2 [0.5 MPa, 90.5 K]
ESR 15	Engine operation should be stable in the defined operational domain and any nominal operation point should be continuously reached within [5 s]
ESR 16	Engine should be capable of providing GH2 at [1.1 kg/s, 10 MPa, 180 K] and GO2 at [1.5 kg/s, 10 MPa, 150 K] for propellant tank pressurization
ESR 17	He-consumption of engine should be minimized and major portion of He should be recovered on ground during purge-procedures prior to chill-down [> 85% tbc].
ESR 18	Startup transient of engine to 100% stable thrust level should take [< 3.5 s]
ESR 19	Shut-down transient of engine from 100% thrust level to [< 10%] thrust level should take [< 5 s]
ESR 20	Engine controls should be fully autonomous and redundantly accept high-level flight control commands
ESR 21	Engine thrust vector should be capable of being controlled by TVC around y- and z-axis within specifications [TBC +/- 8° , tbc 15°/s, TBD °/s2]

The minimum NPSP has been set to 70 kPa for the LH2-boost pump, and to 230 kPa for LOX-inducer pump based on comparable engine designs.

3.4 SLME Functional Architecture

A Full-Flow Staged Combustion Cycle (FFSC) with a fuel-rich preburner gas turbine driving the LH2-pump and an oxidizer-rich preburner gas turbine driving the LOX-pump is the preferred design solution for the SLME (ESR 2). The components and their connections are shown in Figure 3 for the current baseline with FTP split into boost pump driven by separate expander turbine and HPFTP. The HPOTP is a combination of inducer- and impeller-stage driven by the same oxidizer-rich turbine.

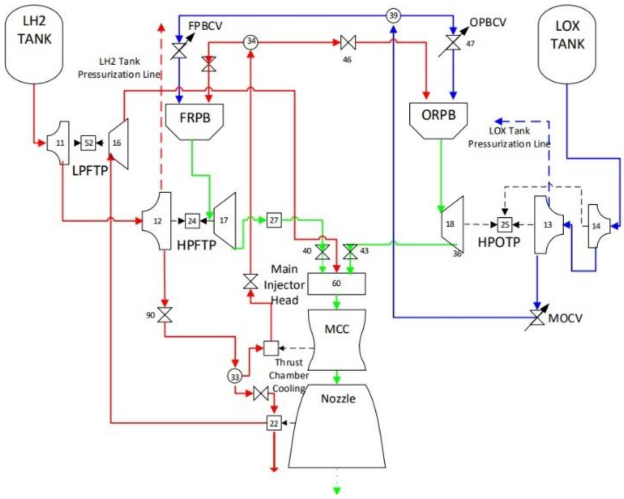

Figure 3: SLME internal flow schematics

Note, the scheme in Figure 3 has been modified to the previous versions presented in [14, 15] with mainly the turbine bypasses eliminated and the principal control valves added. The preliminary definition of the steady-state control logic will be described in section 3.5.6.

Startup transients are not yet considered and might still require bypass flows.

In a Full-Flow Staged Combustion Cycle, two pre-burners whose mixture ratios are strongly different from each other generate turbine gas for the two turbo pumps. All of the fuel and oxidizer, except for the flow rates of the tank pressurisation, is fed to the fuel-rich preburner (FRPB) and the oxidizer-rich preburner (ORPB) after being pressurised by each turbo pump. After the turbine gases created in each preburner drive the respective turbine they are all injected in hot gaseous condition into the main combustion chamber (MCC). The regenerative cooling of the chamber and the nozzle is performed with the hydrogen fuel after being discharged by the HPFTP [9, 10].

3.5 Preliminary subcomponent sizing

Subcomponent sizing and definition is progressing at Phase A conceptual design level supported by the ongoing de-risk study. Refinements are focusing on the turbomachinery designed as an integrated power-head, a suitable regeneratively cooled thrust-chamber lay-out and for the first-time sizing of the high-pressure lines and pipes. The key-objective is a light-weight, long-life, low-maintenance architecture. The subcomponents have been sized for the internal thermodynamic conditions and SLME performance data newly calculated in the full domain for nominal and extreme operating points.

3.5.1 Thrustchamber and regenerative cooling circuit

The geometry of the thrustchamber including chamber and nozzle had been calculated by the DLR tool ncc on the basis of the designed combustion condition (mixture ratio, combustion pressure, fuel flow rate, combustion efficiency) and geometry parameters (contraction ratio, expansion ratio, characteristic chamber length, entry and exit angles of the contour). The booster engine and the orbiter engine have the same geometry in the chamber part including the throat, but not the same in the supersonic expansion part of the nozzle. The nozzle for the orbiter engine does not only have a larger expansion ratio but also a smaller nozzle entry angle. This allows for reduced flow divergence by a smaller exit angle.

The thrustchambers' internal flow contours as presented in [12, 13] do not need to be updated as they still fulfil all requirements.

The thrustchamber cooling baseline has been described in [14, 15]: H2 regenerative and film cooling are combined for the booster engine. Supercritical H2 of the HPFTP discharge adapted to around 30 MPa is split into two separate passes both induced in the supersonic section at expansion 4.5. One counter flow pass (approximately 2/3 of total flow) chills the chamber including the throat area and the other pass chills the nozzle area downstream up to expansion of 16.6. Beyond that section a combination of small bleed and radiation is used for cooling. Fuel for film cooling is supplied from the LPFTP expander turbine at the injector plate's outward ring, further chilling the chamber wall. A thin thermal barrier coating is applied to the wall facing the hotgas to avoid excessive temperatures of the chamber wall material. Thus, thermal stresses and low cycle fatigue effects are reduced, improving the thrustchamber lifetime.

A preliminary thermal analysis of the SLME on the hot-gas side had already been performed using TDK [12, 13]. The program RPA [30] offers a thermal analysis module for different types of thrustchamber cooling methods, including radiation, convective (regenerative) and film cooling. The accuracy is claimed to be sufficient for conceptual and preliminary design studies, as well as for rapid evaluation of different channel variants. [30, 31] The hot gas properties for thermal analysis are retrieved from a quasi-one-dimensional flow model. The heat transfer is simulated in RPA using semi-empirical relations of Ievlev and Bartz. [30, 31] The RPA program had been used for preliminary analyses of the SLME thrustchamber and regenerative cooling circuit at several operation points [14, 15].

In the ongoing de-risk study SoftInWay has performed a new cooling analysis of the main combustion chamber and nozzle. The properties of the working fluids and structural materials are taken from the AxSTREAM® material library that contain different real fluids databases like NIST *Refprop* or *Coolprop*. Zirconium Copper was selected as the internal wall material, since it does not become embrittled at low temperatures and has a high thermal conductivity coefficient.

The one-dimensional analysis was performed using the AxSTREAM® system simulation software. The thrustchamber has been discretized into 18 sections with constant geometrical parameters in each section. Beyond the well-known Bartz-relation, the Gukhman–Ilyukhin heat transfer coefficient model has been considered, however, resulting in low wall temperatures up to 500 K in the hottest areas which has been assessed as not being realistic.

Figure 4 shows wall temperatures along the thrustchamber considering regenerative cooling as well as film cooling. The maximum wall temperatures found remain at less than 800 K in this simulation. Figure 5 depicts the H2 coolant temperature evolution in the channels consistent with the above assumptions.

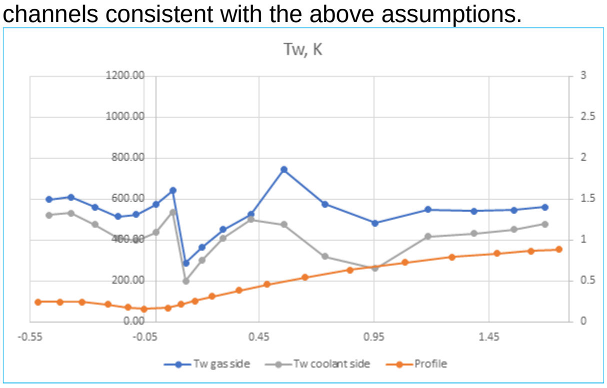

Figure 4: Wall temperature distribution in SLME-33 thrustchamber at O1 obtained from AxSTREAM® analysis

An exploration of different operating points had been previously performed with RPA in order to check on the feasibility of the regenerative cooling concept in the full operational domain, see [14, 15].

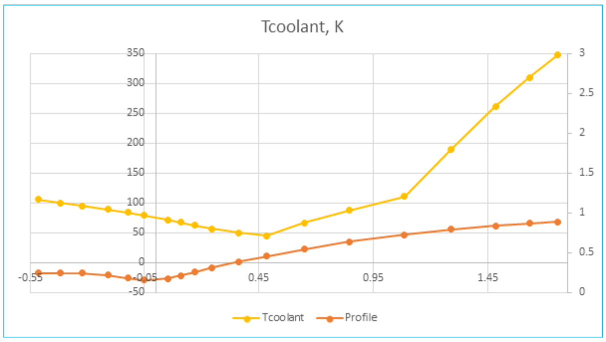

Figure 5: Coolant total temperature in SLME-33 thrustchamber at O1 obtained from AxSTREAM® analysis

In the upper stage version of the SLME with expansion ratio 59 the chamber cooling strategy is very similar to the version with smaller nozzle extension [15]. If confirmed by refined analyses for all nominal operating points, the cooling flow schematic of both SLME-variants could be very similar.

For the main combustion chamber a coaxial injector type is selected similar to other oxygen-hydrogen engines. As a preliminary assumption, 550 coaxial injector elements are selected with a mass flow rate of up to 1 kg/s and flow ratio (ox-rich to fuel-rich) between 3.5 and 4. Note, the injector is operating in gas-gas mode which is simplifying the mixing and enhancing combustion stability.

A spark igniter placed in the center of the injector head is reference for the SLME, similar to the SSME design (Figure 6) and [33].

3.5.2 Integrated Power Head

An Integrated Power Head (Pre-burner + Turbine + Impeller pump) as it has been used on the SSME (Figure 6) is the baseline design for the SLME. However note, the cycle architectures of SSME and SLME differ with the former using two fuel-rich preburners to power the turbines. It is shown below that FFSC makes a difference to the preferred IPH-architecture.

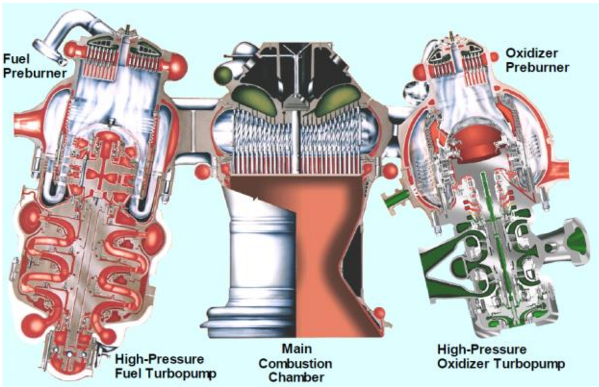

Figure 6: Example of SSME Integrated Power Head assembly attached to combustion chamber [33]

Figure 7 shows in its latest variant the integration of all major components of the Integrated Power Head in the upper section of the SLME and their integration with the combustion chamber injector head following the SSME example. The preliminary layout of [15] is maintained but considering an update of the turbopumps sizing (following section 3.5.4) and for the first time a preliminary definition of mechanical architecture including iterative adaptation of the hot and cold fluid lines (section 3.5.5).

The view of the SLME-33 from above in Figure 7 shows on the right the hydrogen fuel supply and on the left the oxygen flow side.

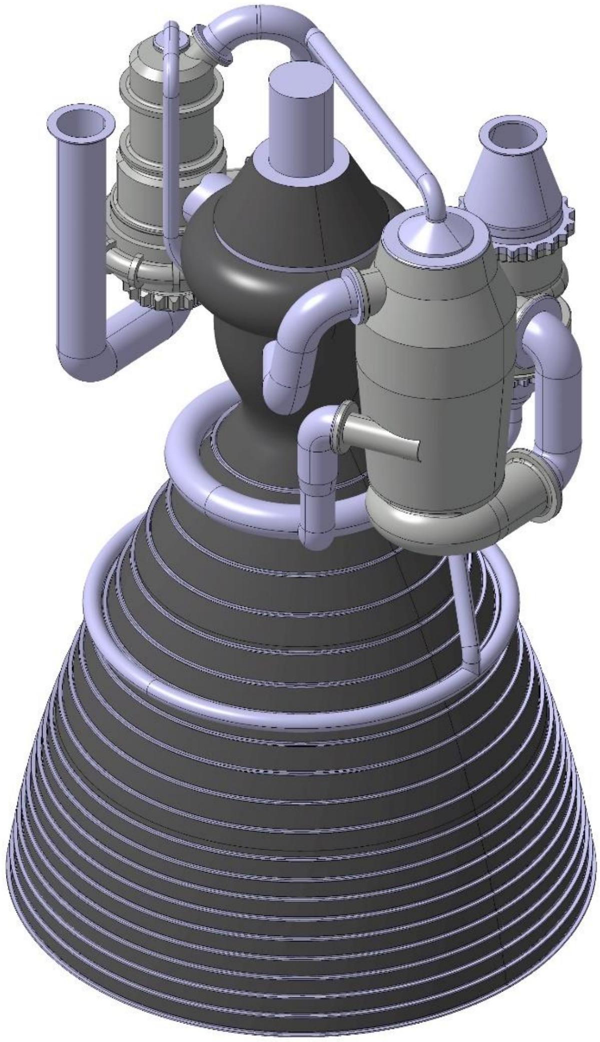

Figure 7: SLME simplified CAD geometry showing arrangement of turbomachinery

3.5.3 Preburners

The SLME preburners are attached to each turbo-pump in the integrated powerhead assembly as visible in Figure 7. The mixture ratios of the fuel-rich preburner (FR-PB) and the oxidizer-rich preburner (OR-PB) are controlled to be less than 1.0 and above 120 so that TET is restricted to acceptable values. The full-flow sub-cycle allows TET remaining in a small range from 740 K to 780 K, even in the extreme domain, without excessively raising preburner pressures. The limitation of the nominal characteristic conditions should enable an engine lifetime of up to 25 flights. Further, this approach gives some margin to significantly raise engine power in case of extreme emergencies by increasing TET beyond the defined limits [9]. However, mission and systems-analyses of the SpaceLiner configuration show that such extreme measures might not even be required due to good robustness and performance margins of the vehicle [16].

In order to preliminarily understand the combustion and those respective interactions, a study was initiated that used CFD simulations with a very simplified, preliminary 2D (axially symmetric) geometry of potential SLME pre-burners. The commercial CFD solver ANSYS CFX was used for the analyses of the fuel-rich preburner with one coaxial injector element assuming different design parameters, such as the impulse ratio between the oxidizer and fuel streams and the mixture fraction were parametrically studied [15].

In the early years of Space Shuttle operations, the MSFC sponsored several initiatives for improving the RLV. The focus was on the SSME, mostly on cost reductions and lifetime improvements. In [23] an FFSC derivative engine of the SSME is proposed with a very compact annular ox-rich preburner around the shaft connecting turbine and impeller/inducer [14, 23].

Such preburner design would allow for significant shortening of the powerhead assembly on the ox-rich side and eliminate some of the heavy high-pressure lines. The SpaceX Raptor makes use of such a lay-out with an assembly mounted right on top of the main combustion chamber. The turbine exhaust gas is directly fed to the injectors and hence into the combustion chamber. Alternative arrangements of the IPH-ox could be of interest as proposed in [24].

This advanced powerhead assembly is potentially of major interest for the SLME and could significantly improve the weight penalty of FFSC mentioned in [26]. Currently, the SLME reference configuration does not yet include such IPH-ox, however, some trades have been performed in the de-risk activities and an engineering sizing was performed by applying suitable propellant residence times. Ignition of LOX with H2 at high mixture ratios is hardly possible (see e.g. [25]). Therefore, a different approach with central hot primary zone at standard MR and cold, circumferential wall zone with O2-flow is proposed. These initially coaxial flows need to be mixed that the temperature field at turbine entry plane is adequately homogeneous. A turbulent mixing zone with forced turbulence for mixing the hot H_2O with O_2 is required. A design with similarities to cross- or reverse-flow gas generators seems to be promising [25, 34]. However, further research will be necessary to reach mature design stage of the concept.

3.5.4 Turbomachinery

The first preliminary definition of the turbomachinery lay-out has been described in references 10 through 14. On the fuel side a boost pump driven by an expander turbine fed from the regenerative circuit is feeding the HPFTP. On the LOX-side a conventional HPOTP with inducer and single stage impeller on the same shaft is proposed powered by a single stage turbine. In case of the preferred full-flow staged combustion cycle the LOX-split pump is eliminated because unnecessary for raising discharge pressure to the fuel-rich preburner level as with the SSME [33].

A newly refined update of the SLME turbomachinery design was performed within the ongoing de-risk study for ESA using the AxSTREAM® platform developed by SoftInWay.

AxSTREAM® is a multidisciplinary design, analysis and optimization software platform that provides fully integrated and streamlined solutions, encompassing the complete turbomachinery design process, all in a seamless interactive user interface. Preliminary estimation of performance and dimension of turbomachinery components are done with the generative design module that is based on an inverse task solver and allows generation of thousands of geometry options within seconds for users to review data and compare at design- and off-design conditions.

According to the cycle flow scheme shown in Figure 3, the following turbomachinery components have been pre-designed: LPFTP pump and turbine, HPFTP pump and turbine and HPOTP pump and turbine. The thermodynamic parameters used for the turbomachines design correspond to the operational point O1 and the SLME cycle design conditions of 2022 as presented in [15].

AxSTREAM® turbomachine internal efficiency accounts for flow path quality, tip and back face leakage losses and disk friction losses without considering mechanical losses.

LPFTP

The boost pump is used to pressurize the hydrogen fuel before its entry into the HPFTP. Increasing the cavitation margins of the fuel-fed system allows decreasing of tank pressure. Fuel from the tank enters the boost pump flow path with a minimum pressure of 0.176 MPa and is pressurized to 1.5 MPa. The maximal casing diameter of the LPFTP is preliminarily set to 444 mm.

An inducer (diagonal) type of wheel is used as the preferred pump concept. The rotational speed and several main geometrical parameters have been varied to generate the design space of feasible configurations that satisfy the required cavitation margin. Figure 8 shows the design in a 3D-view.

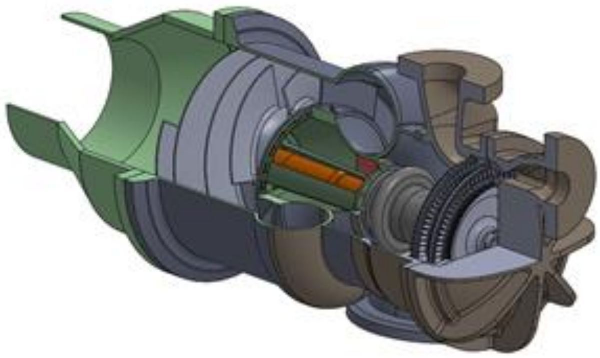

Figure 8: SLME LPFTP preliminary design

Performance parameters of the LPFTP are presented in Table 6.

The turbine is fed from the regenerative circuit with hydrogen gas to drive the boost pump. The expander-type turbine driven by heated hydrogen is designed to cover pump power requirements considering mechanical losses. As it is supposed to use a single shaft turbo-pump, the shaft rotational speed corresponds to the one determined for the pump.

A few turbine designs were considered as potential configurations. The maximum efficiency considering all geometrical restrictions was achieved for double stage impulse design with partial admission about 0.12. With respect to turbine blades, it is preferable to use prismatic shapes as the change of thermodynamic parameters along the relatively short blade height is negligible.

Main performance and dimensions data of the LPFTP turbine are given in Table 7.

Table 6: SLME LPFTP pump preliminary design performance in nominal range O1, O2, O3

Operational Point	O1	O2	O3
Mass flow rate [kg/s]	75.7	76.3	75.9
Power [kW]	1766	1981	1769
Pressure ratio [-]	8.5	9.4	8.5
Axial length [mm]	351.25		
Maximal diameter [mm]	443.73		

Table 7: SLME LPFTP turbine preliminary design performance in nominal range O1, O2, O3

Operational Point	O1	O2	O3
Mass flow rate [kg/s]	9.9	10.6	9.8
Power [kW]	1812	2035	1815
Axial length [mm]	55		
Maximal diameter [mm]	224		

Preliminary selection of materials for the LPFTP are Al 7075, 15-5PH(S15500) and AlLi A356.0 which have been used for the structural pre-sizing. Figure 9 shows FEM-results of studied design options of the LPFTP turbine and pump.

The full scope of rotor dynamic analyses according to API standards [36] was performed for all turbopumps. The critical speed map for LPFTP is shown in Figure 10 and the LPFTP mode shape corresponding to the first bending critical speed is presented in Figure 11.

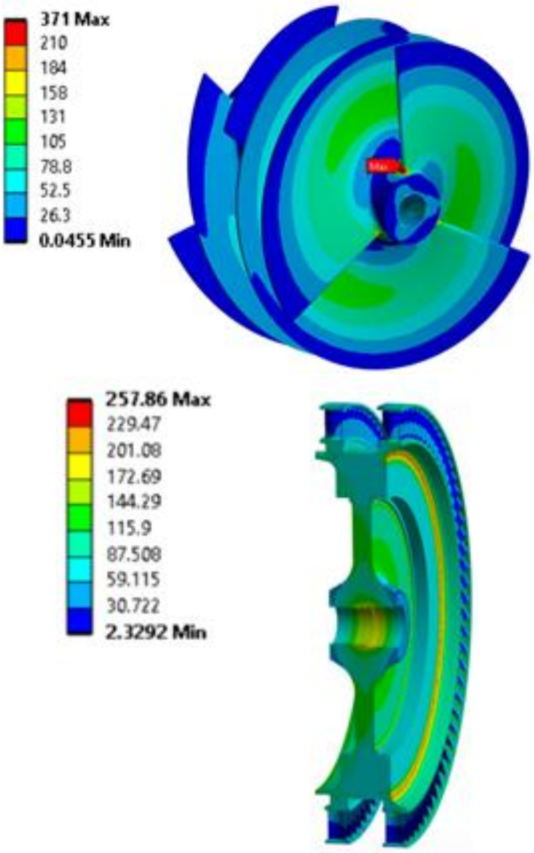

Figure 9: SLME LPFTP inducer blades and turbine rotor studied with FEM

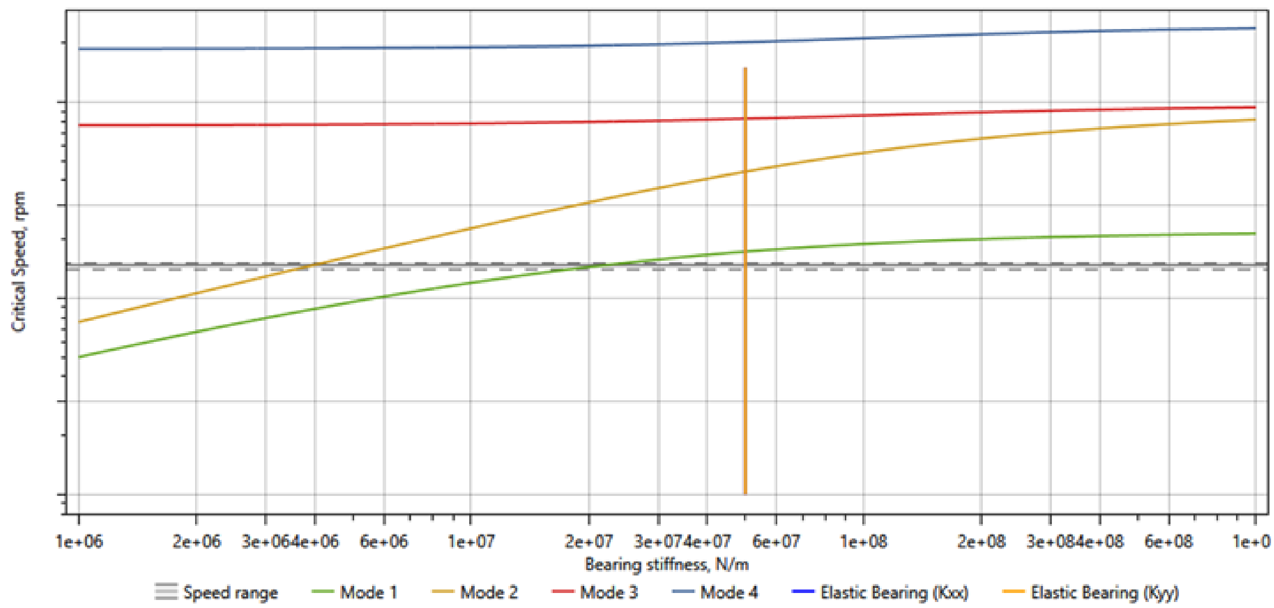

Figure 10: SLME LPFTP critical speed map

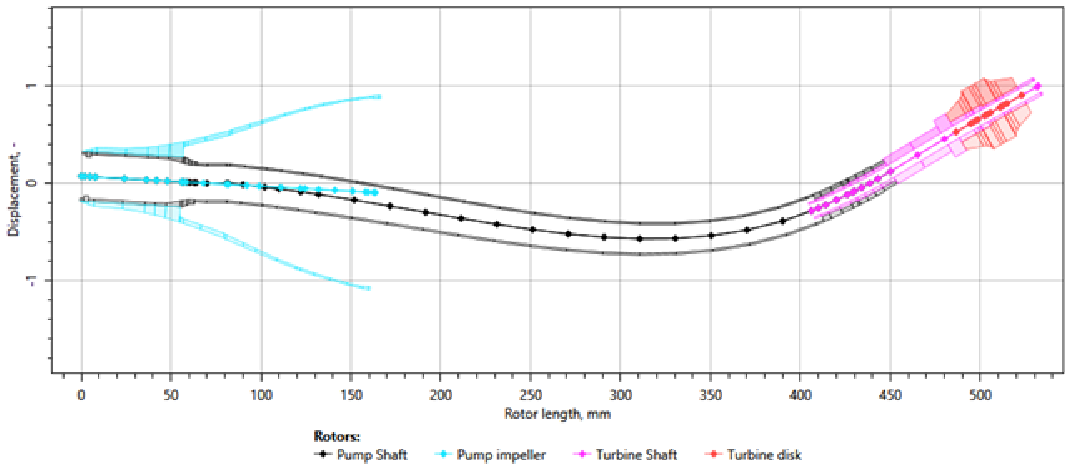

Figure 11: SLME LPFTP mode shape corresponding to the first bending critical speed

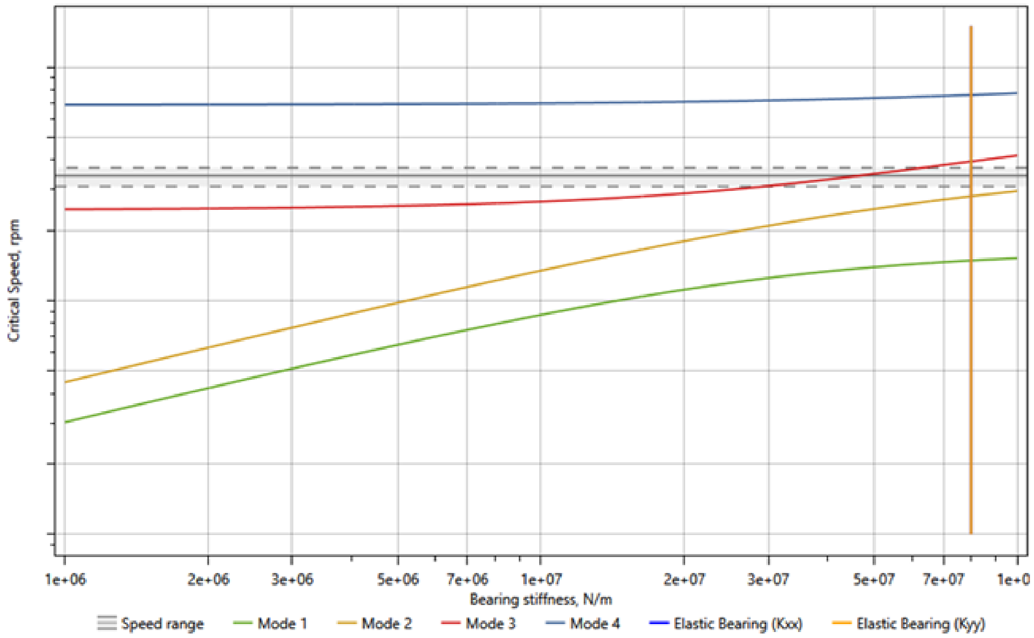

Figure 12: SLME HPFTP critical speed map

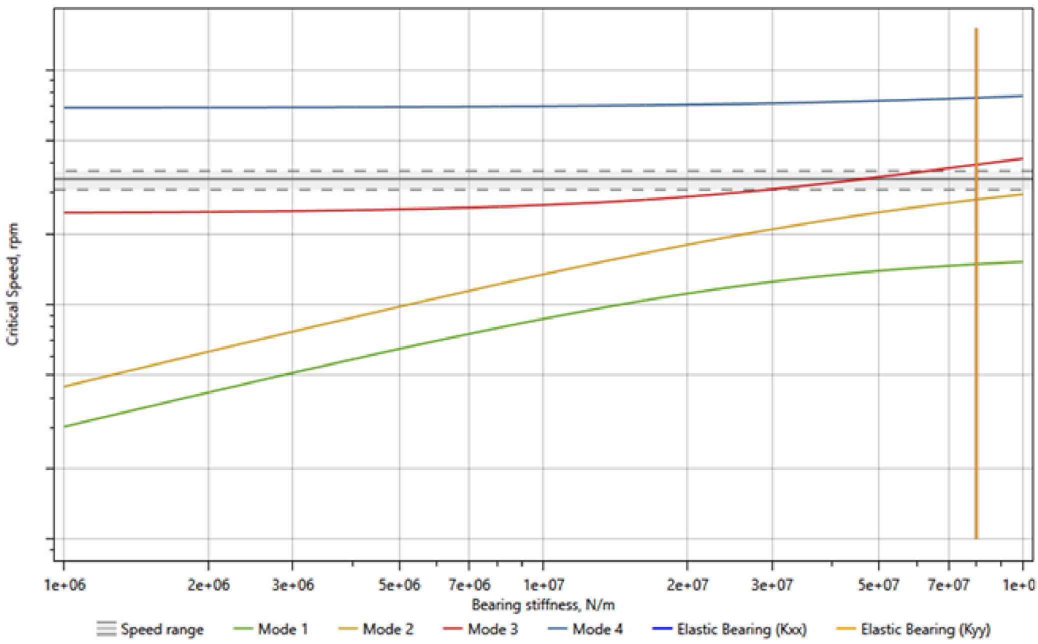

Figure 13: SLME HPOTP critical speed map

HPFTP

The pressurized flow after the LPFTP enters the HPFTP pump. The HPFTP outlet pressures are somehow raised from 2022 conditions and are, depending on the operation point, between 32.2 MPa and 35.8 MPa. Several configurations of the HPFTP pump were initially considered varying the inducer, the vane diffuser presence, and the number of stages. The configuration and parameters are to be selected specifically under consideration of their impact on the cavitation margin, the efficiency, and the dimensions of the designed pumps. Additionally, a casing diameter constraint had been proposed at 500 mm which had to be relaxed to 516 mm for the inlet volute.

During the preliminary performance estimation, no cavitation was observed on the first impeller wheel of any configuration. For this reason, pump configurations without inducer are preferable due to lower axial dimensions and therefore lower mass [15].

It is expected that pump configurations that include a vane diffuser after the centrifugal wheel will help achieve higher efficiency at design point compared to vaneless configurations. However, for off-design conditions the pump performance can be lower due to suboptimal incidence angles at the vane diffuser.

A 3-stage impeller HPFTP pump is considered as the baseline configuration, while satisfying the casing diameter constraint. Figure 14 represents the 3D-drawing of the HPFTP.

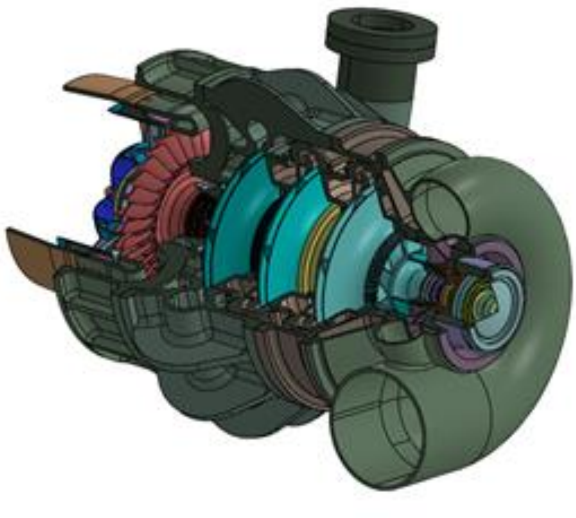

Figure 14: SLME HPFTP preliminary design

All centrifugal wheels contain the same blade profile. This choice helps significantly dropping the manufacturing cost without big impact on the flow path efficiency. Performance data of the HPFTP pump are presented in Table 8.

Table 8: SLME HPFTP pump preliminary design performance in nominal range O1, O2, O3

Point	O1	O2	O3
Mass flow rate [kg/s]	75.7	76.2	75.9
Power [kW]	38710	41030	37994
Pressure ratio [-]	24.7	23.5	24.2
Axial length [mm]	425.2		
Maximal diameter [mm]	516		

The HPFTP turbine is driven by combustion gas from the fuel-rich preburner. Reaction turbine design is selected as the baseline as it provides the highest efficiency for the given isentropic velocity ratio. Maximum diameter restriction, stress limitations and relatively big blade height ($D_m/l \approx 5$) led to a stage design with suboptimal nozzle outlet angles that decreases the flow path efficiency. For flow paths with $D_m/l < 10$ a significant change of the flow parameters (angles) along the blade span is observed, thus, a 3D blade design is preferable.

Table 9 represents the performance data of the HPFTP turbine.

Table 9: SLME HPFTP turbine preliminary design performance in nominal range O1, O2, O3

Point	O1	O2	O3
Mass flow rate [kg/s]	98.1	101.2	95.9
Power [kW]	38710	41030	37994
Axial length [mm]	71.2		
Maximal diameter [mm]	274.6		

The selected materials for the HPFTP are Al 7075, Waspaloy, Inconel718, 15-5PH (S15500).

Calculated stresses in an HPFTP turbine blade resulting from FEM are presented in Figure 15. It should be noted that the maximal stress level is observed locally for the turbine disk. Turbine disk redesign for the planned flow path should be considered further at future detailed design phase.

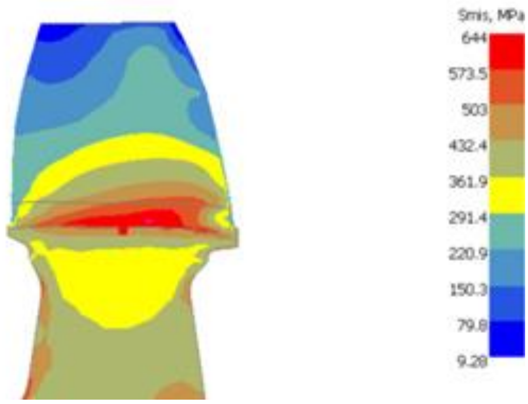

Figure 15: SLME HPFTP turbine blade preliminary design

The critical speed map for the HPFTP is shown in Figure 12.

HPOTP

Oxygen is fed from the tank directly to the HPOTP with minimum interface pressure of 0.75 MPa and is to be pressurized between 28.2 MPa and 32.16 MPa according to latest cycle modeling. Thus, an inducer wheel is required to avoid any cavitation at the impeller stages. Two possible configurations were considered: single and double stage impeller on the same shaft with the inducer [15]. Maximal casing diameter of the HPOTP was initially tried not to exceed 350 mm.

The design with single stage impeller has been selected [15]. Preliminary design of both pumps shows that exceeding the external diameter target can't be avoided. The pump exit volute diameter is calculated at

538 mm. The design of the HPOTP is presented in Figure 16.

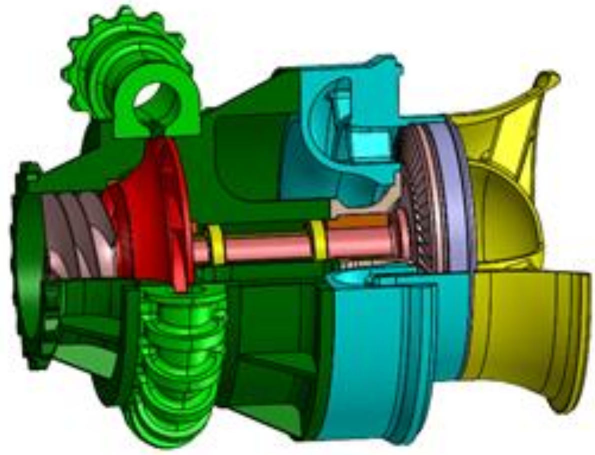

Figure 16: SLME HPOTP preliminary design

Table 10 represents performance data of the HPOTP-pump.

Table 10: SLME HPOTP pump preliminary design performance in nominal range O1, O2, O3

Point	O1	O2	O3
Mass flow rate [kg/s]	445	486	409
Power [kW]	15127	17746	14075
Pressure ratio [-]	60.6	71.1	57.2
Axial length [mm]	197.4		
Maximal diameter [mm]	528.6		

The materials selected for the HPOTP pre-design are Al 7075, Waspaloy, Inconel718 and 15-5PH (S15500). Calculated stresses in the impeller blades are shown in Figure 17.

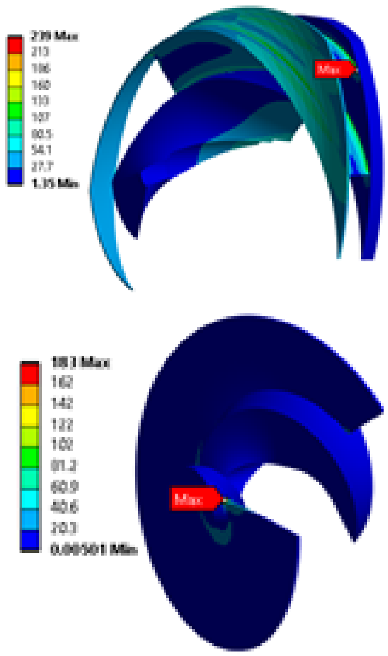

Figure 17: Equivalent stresses in SLME HPOTP pump impeller blades

The HPOTP turbine is driven by the oxidizer rich gases from the oxidizer preburner. The preliminary design of the HPOTP turbine is adequate to meet the pump power requirement. A reaction design of the turbine with 3D blades is assumed to be optimal for the given boundary conditions.

Performance data of the HPOTP turbine is presented in Table 11.

Table 11: SLME HPOTP turbine preliminary design performance in nominal range O1, O2, O3

Point	O1	O2	O3
Mass flow rate [kg/s]	405	442	370
Power [kW]	15154	17775	14097
Axial length [mm]	65.1		
Maximal diameter [mm]	362		

The critical speed map for the HPOTP is shown in Figure 13.

3.5.5 Mechanical Architecture and Fluid Lines

The mechanical architecture of the SLME with arrangement of the complete turbomachinery, the connecting lines and bellows and the components attachment to the thrustchamber has been significantly refined. A classic rocket engine architecture is selected, although, alternatives with clustered turbomachinery serving different thrustchambers without being structurally attached to them [24] might be considered in the future for the multiple-engine launcher stages.

As first step, the fluid lines have been iteratively sized for required cross-sections to keep pressure losses at acceptable levels.

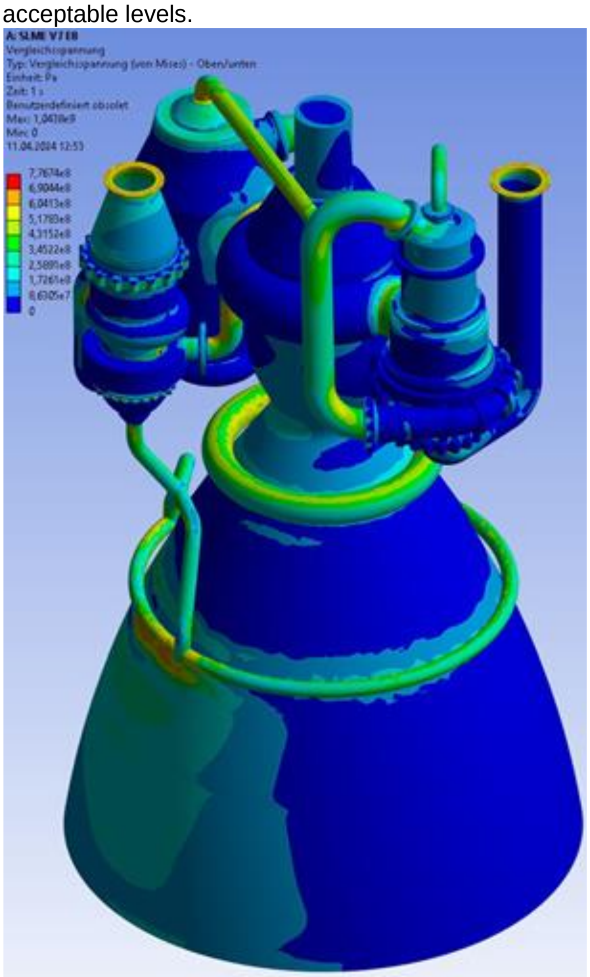

Figure 18: Von Mises stresses in SLME-33 lines at extreme operating point E8

Based on the dimensions and total conditions from cycle analyses, the static loads acting on the structure are calculated in the full domain for nominal and extreme operating points (see section 3.6). The material choice of all lines in the preliminary sizing has been Inconel 625 with consideration of temperature dependent properties. Stresses have been calculated using FEM with the commercial tool Ansys 2023 R1 for dimensioning load cases. An example of the obtained stresses after adjusting the wall thicknesses is shown in Figure 18. Note, only stress levels of lines are meaningful because thrustchamber and turbopump casings are strongly simplified and were not subjected to all relevant loads in the analyses.

Insulation has been applied to the H2-lines up to the supply torus of the regenerative circuit while the oxygen side does not require line insulation. Figure 19 shows the calculated wall temperatures under the theoretical assumption of power packs operating but without any heatloads from the main combustion chamber.

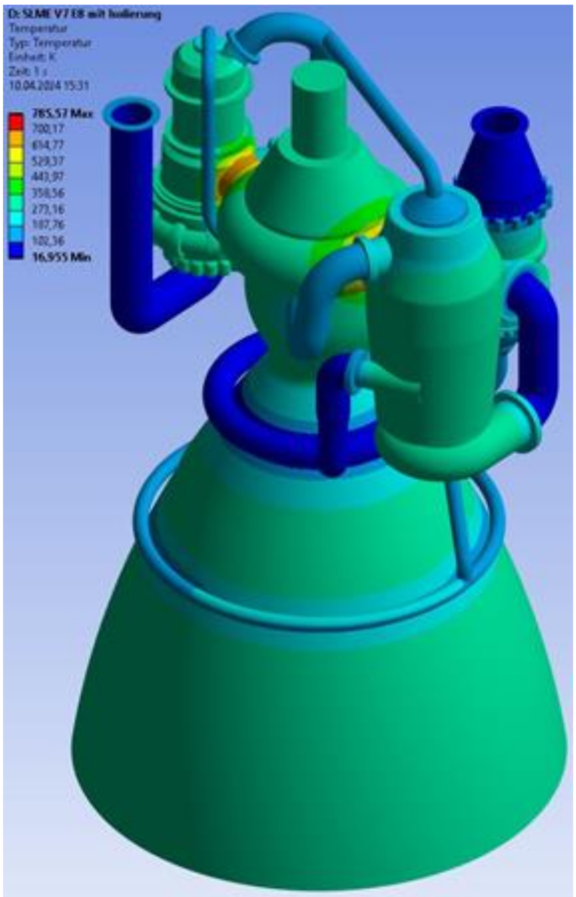

Figure 19: Wall temperatures in SLME-33 lines from thermal analysis

3.5.6 Engine Controls and HM

The SLME engine controls and actuation system is intended to be designed fully electric for maximum safety and manufacturing cost reduction. A FADEC system as in modern aircraft engines centralizes all HM-information and has a redundant data link to the vehicle's flight control and data management and data handling.

Preliminary work is currently performed with the purpose to define an actuator and control logic for the steady-state regime of the SLME. A first cast of a potential control scheme is shown in Figure 20. It is similar to [37] but yet simplified at this stage of work. The engine monitor unit would collect and interpret sensor information and evaluate them. The purpose is to identify component and system level parameters, verify that the observed parameter range is within reasonable limits in terms of durability and safety, and generate data from sensor measurements used for motor control. As such it encompasses the HMS but also acts towards the feedback loop of the control loop.

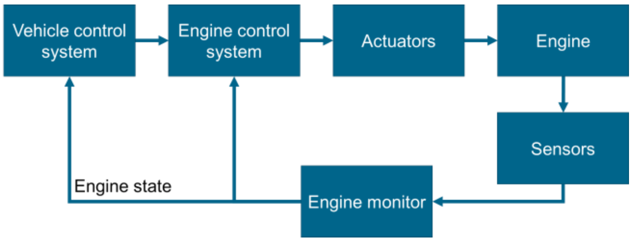

Figure 20: Preliminary control scheme for the SLME

Output from the engine monitor unit will be fed to the launch vehicle or SpaceLiner flight control system and to the engine controller. The former will formulate commands to the engine control system according to mission and vehicle needs and in accordance with high level requirements or when safety measures are to be taken e.g. reacting to engine-out events.

The proper operation of the engine and its components will be controlled by the engine control system for which the baseline concept, similarly to the Space Shuttle Main Engine (SSME) [33], has thrust control by the oxidizer-rich preburner control valve (OPBCV) whereas mixture ratio is controlled by the fuel-rich preburner control valve (FPBCV). Both only act upon oxidizer mass flow rate into the respective preburners as shown in Figure 3. The need for an additional main oxidizer control valve (MOCV) which is in series to the other two valves is under debate and will depend on the actual valve architecture and its operating range. The SSME has a valve and bypass to regulate the regenerative cooling flow. If such devices would be needed in the SLME requires more in-depth studies of the cooling flow characteristics and its dynamics.

Engine status data are provided by a suitable selection of sensor types. As preliminary assumption, combustion chamber pressure shall be monitored as much as temperatures and pressures at the low-pressure fuel pump outlet. These data shall be complemented by a volume flow meter at the low-pressure fuel pump discharge and temperature gauges at the turbine discharges for both high pressure turbopumps to indirectly survey temperature loads on the turbines. Further measures could be pressures of the preburner feed flows at appropriate locations of the thrustchamber and the rotational speed of the turbopumps. This is similar to the proposal in [37] for the SSME, where the standard sensors should be supplemented with additional temperature and pressure measurements for an advanced control scheme, with the aim of extending the life expectancy of engine components.

The HMS independently provides input for the engine emergency control and collection of huge operations data sets for maintenance prediction and support. The latter is to be stored with high sample rate in redundant form 'on-engine' for download after flight. Internal flow conditions, thermal and mechanical load data including vibrations can be used for automated post-flight assessment, implementing machine-learning algorithms. If such an approach is consequently followed already during development testing, a significant improvement in rocket engine reliability and robustness can be expected.

3.6 SLME performance estimation

3.6.1 Calculation models

A computer program used in the early phases of the SLME cycle analysis is lrp2, based on the modular program SEQ [29] of DLR. Since the 1990ies this powerful tool had been significantly upgraded. The modular aspect of the program allows for a quick re-arrangement of the engine components, specifically the turbine and pumps assembly. After selection and suitable arrangement of the components in an input file, the program calculates the fluid properties sequentially according to the specific thermodynamic processes in the components, through which the fluid flows. Each constraint yields a nonlinear equation. This results in a system of nonlinear equations (or rather dependencies) which is solved by an external numerical subroutine. The lrp2-tool has been used as reference for SLME cycle analyses and performance estimation up to 2018 [9, 10, 11, 12, 13, 14].

The commercially available program RPA [30] Version 2.3.2 has been subsequently used for the preliminary analyses of the SLME and for further refinement of component definition (see also section 3.5.1). RPA is capable of predicting the delivered performance of a thrust chamber using semi-empirical relations [30] to obtain performance correction factors, including:

- performance loss due to finite-rate kinetics,
- divergence loss,
- performance loss due to finite-area combustor,
- performance change due to nozzle flow separation.

Those factors are relevant for the SLME design. The RPA engine cycle analysis module is capable of analyzing the operational characteristics of engine configurations, performing a power balance of the turbo-machinery to achieve a required combustion chamber pressure [30]. The full-flow staged-combustion cycle (FFSC) which is the reference mode for the SLME is included in RPA. In 2018 a new, more flexible and powerful version 3 of RPA was announced [32] but might not yet be released.

The lrp2 program is significantly more flexible in the arrangement of flow paths inside the engine than RPA 2.3.2. However, this complicates the user input and slows convergence of lrp2. During the recent activities in SLME de-risking, the lack of expander turbine modeling in RPA in combination with fixed flow paths turned out as a problem because the LPFTP boost pump could not be correctly modelled. This has an impact on the calculated conditions in the FR-preburner which requires a shift in mixture ratio to keep TET-targets. Some internal flow conditions are hence inaccurate, influencing the preliminary sizing of components. Using AxSTREAM® corrected data in certain operating points. Despite the detected problems, the overall effect on the engine assessment is relatively small. RPA offers more sophisticated performance estimation methods for nozzle expansion than other tools. In the preliminary definition of the SLME different numerical tools are useful and complement each other. The recent updates of SLME-analyses are based on RPA 2.3.2 calculations.

3.6.2 Operational domain and performance

The operational domain of the SLME has been further refined and some of the extreme points around the nominal operation conditions have been slightly repositioned when compared to [15]. The different RLV-concepts using the SLME as its reference engine in the design studies summarized in section 2 are all functioning at one or several of the nominal operations points O1, O2 and O3.

The newly calculated SLME operational domain is shown in Figure 21. The extreme points of the domain (E1 to E8) define the ultimate safe operation limits of the SLME with all its subcomponents. The MR-range extends from 5 to 7 and is realized mainly by adjusting the LOX-flow (up to ± 18%). Maximum LH2 massflow variation within the domain is less than ± 8%. This is a preliminary design decision which could be somehow adjusted based on future analyses. The SLME extends up to a maximum chamber pressure of 18 MPa. This value is in no way excessive compared to preceding LOX-LH2 engine developments as the SSME [33], RD-0120 or Raptor (2).

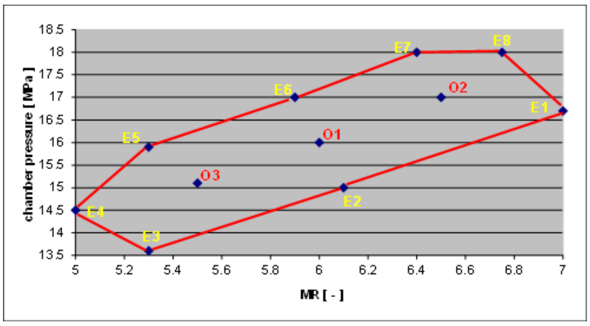

Figure 21: Calculated SLME operational domain

The AxSTREAM® analyses of all turbopumps run by SoftInWay and with major results presented in the previous section 3.5.4 and latest line pressure loss estimation have been used for a refined engine model applied to renewed RPA2.3.2-cycle calculations. The estimated efficiencies obtained by the preliminary AxSTREAM® design are influencing the assumptions in the cycle analyses.

Table 12 gives an overview about major SLME internal operation and engine performance data for the three nominal operating points as obtained by RPA cycle analyses. The SLME flow scheme as shown in Figure 3 is the reference for all these cycle calculations. Performance data are presented for the two different nozzle expansion ratios of the SpaceLiner: 33 and 59.

Preburner combustion temperatures or TET in the full domain are kept in a relatively small range between 740 K to 780 K. The required preburner and hence turbopump discharge pressures are recalculated by RPA assuming latest, partially increased pressure drops in lines, valves and injectors. This impacts mainly the O2 path because of the applied engine control logic and related valves placement. OTP-discharge pressure is raised by up to 3 MPa in comparison with [15]. Both preburners operate at similar pressure levels up to 25.9 MPa. However, the pumps are different: the OTP discharge reaches now up to 32.16 MPa at point O2 while HPFTP goes to 35.8 MPa because the complete hydrogen is directed first through the regenerative cooling before reaching the FRPB.

Table 12: SpaceLiner Main Engine (SLME) technical data from RPA2.3 numerical cycle analysis

Operation point	**O1**	**O1**	**O2**	**O2**	**O3**	**O3**
Mixture ratio [-]	6		6.5		5.5	
Chamber pressure [MPa]	16		16.95		15.1	
Fuel-rich Preburner pressure [MPa]	23.9		25.7		22.6	
Oxidizer-rich Preburner pressure [MPa]	23.9		25.9		22.66	
Fuel-rich Preburner TET [K]	760		770		760	
Oxidizer-rich Preburner TET [K]	760		771.4		760	
HPFTP discharge pressure [MPa]	33.4		35.8		32.2	
OTP discharge pressure [MPa]	29.5		32.16		28.2	
Mass flow rate in MCC [kg/s]	513.5		555		477.65	
c* [m/s]	2310.28		2271.34		2343.45	
Expansion ratio [-]	**33**	**59**	**33**	**59**	**33**	**59**
c_F in vacuum [-]	1.8546	1.9057	1.8712	1.9255	1.8374	1.8855
c_F at sea level [-]	1.6381	1.5187	1.6671	1.561	1.6081	1.4755
Specific impulse in vacuum [s]	436.9	448.95	433.39	445.97	439	450.56
Specific impulse at sea level [s]	385.9	357.77	386.1	361.5	384.2	352.6
Thrust in vacuum per engine [kN]	2200	2260.68	2356	2427.28	2056.7	2110.49
Thrust at sea level per engine[kN]	1943	1801.55	2111	1967.32	1800	1651.56

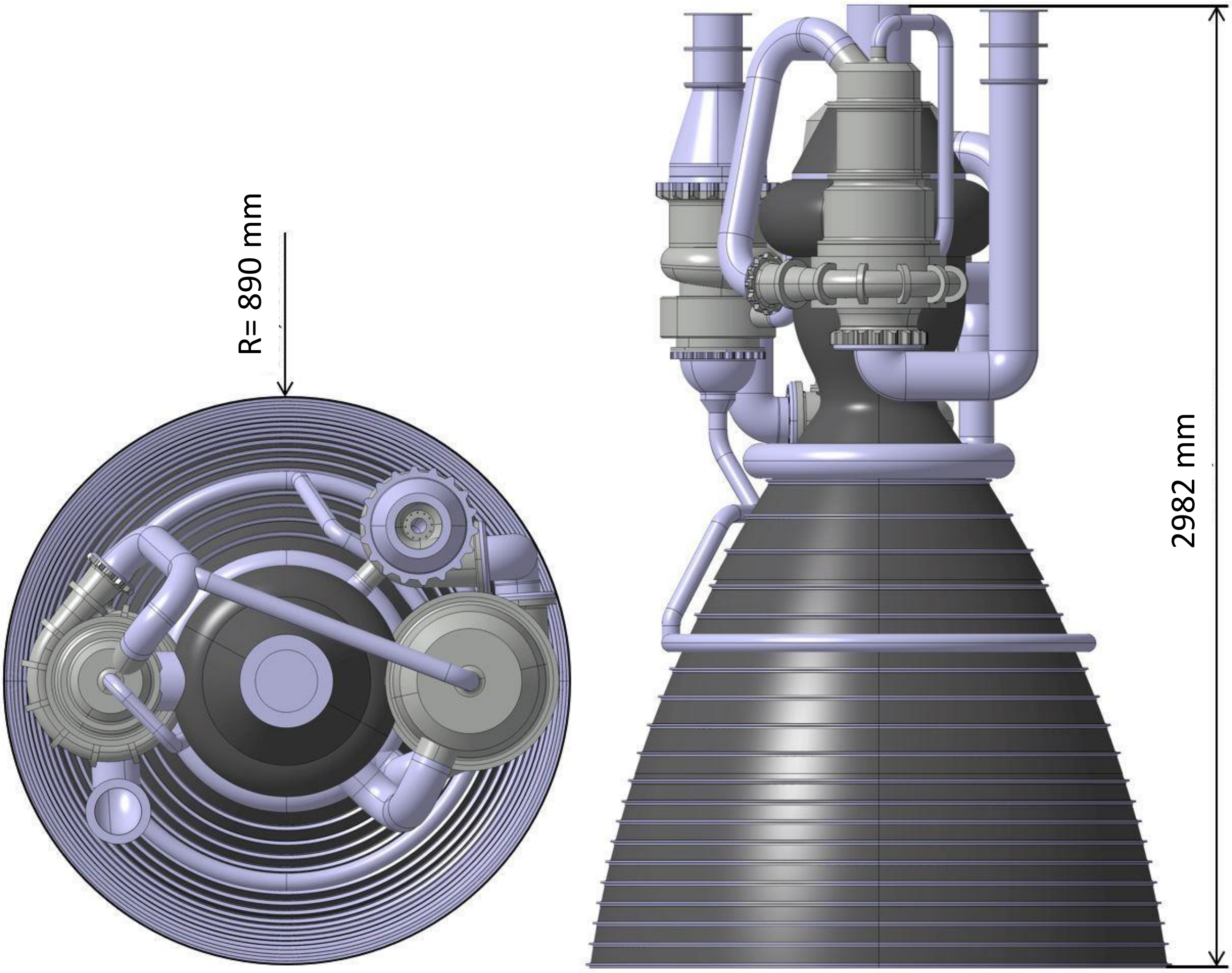

Figure 22: Overall size of SLME with ε=33 as simplified CAD-model

3.7 Engine Geometry and Mass

The size of the SLME in the smaller booster-type configuration is a maximum diameter of 1800 mm and overall length of 2982 mm. The larger upper stage SLME has a maximum diameter of 2370 mm and overall length of 3893 mm. The latest concept of the SLME engine with expansion ratio 33 is visible in Figure 22.

The ongoing de-risk study not only refined the preliminary component sizing but also enabled an update of the engine mass assessment. The engine masses are estimated at 3500 kg with the large nozzle and at 3218 kg for the booster stage nozzle with expansion ratio 33. This is an increase of roughly 4% compared to the previous versions [15]. These values are equivalent to vacuum T/W at MR=6.0 of 65.9 and 69.7. Some optimization potential exists, mainly with elimination of high-pressure LOX-lines by introducing an advanced annular ox-rich powerhead.

4 CONCLUSION

A full-flow staged combustion cycle around a moderate 16 MPa chamber pressure has been selected for the SpaceLiner Main Engine (SLME). Beyond its original application, the SLME has been successfully implemented by DLR in several studies as RLV main propulsion. The engine can serve as a realistic baseline for the next generation of European staged-combustion cycle LOX-LH2 rocket engines.

The engine operational domain is redefined by numerical analyses. The design with separate boost- and high-pressure pump on the LH2 side and a single-shaft for inducer and impeller on the LOX side is maintained as the baseline and supported by preliminary turbopump sizing.

An ongoing de-risk study for ESA refines engine component definition with focus on the turbopumps and validates a mechanical architecture with pre-sizing of fluid lines.

Advanced innovative design solutions are under investigations which should enable reliability for the entire 25 missions design life and low-cost manufacturing and maintenance. The SLME masses in the 2200 kN thrust class are estimated at 3500 kg with large nozzle and at 3218 kg for typical booster stages.

The SLME is now one of the most sophisticated closed-cycle LOX-LH2 engine concepts in Europe and ready for critical component validation.

5 ACKNOWLEDGEMENTS

The authors gratefully acknowledge the contributions of Ms. Carola Bauer, Ms. Carina Ludwig, Ms. Mona Carlsen, Mr. Jascha Wilken, Mr. Jan-René Haferkamp, Mr. Ryoma Yamashiro, Mr. David Gerson, Mr. Andreas Brückner, Mr. Francesco Cremaschi and Mr. Vincent Friesen to the preliminary design of the SLME and the SpaceLiner propulsion system.

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