Systematic Studies on Reusable Staged-Combustion Rocket Engine SLME for European Applications

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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 since several years. The baseline propulsion system for the reusable rocket-based multiple-mission launcher concept SpaceLiner is also used as a reference for closed cycle LOX-LH2-engines in several studies of future European RLV. A summary of these potential future applications and interesting regions of nozzle expansion ratios for 1st stage RLV or 2nd stages are presented.

After completing the Mission Requirements Review, the SpaceLiner Main Engine (SLME) component definition is progressing at Phase A conceptual design level. Refinements are focusing on the turbo-machinery designed as an integrated power-head. The commercial AxSTREAM® software tool for turbomachinery analyses has been implemented. Consolidated size, mass, and performance data are available by this analysis and are integrated in the engine model. Further, the fuel-rich and the oxidizer-rich pre-burner have been investigated in a first series of numerical CFD-simulations.

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
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
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 [8, 12, 13] 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 [e.g. 1, 2, 3] summarized in the following section.

2 ENGINE FOR RLV CONCEPTS

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. As the applications are different also the most appropriate nozzle expansion will differ. In Figure 1 the calculated Isp in vacuum and sea-level conditions of SLME is plotted in the huge, hypothetical supersonic nozzle area range from 10 to 120. The latter might be of interest to be integrated in upper stages which are only operational in vacuum and when this engine should be a derivative of the RLV-first stage. Such engine similarity with mainly adaptation of the nozzle allows for reduced development costs and might permit the reusable engine to be expended after certain number of missions on the RLV.

As common assumption the chamber pressure is kept at 16 MPa and mixture ratio (MR) at 6.0, which is consistent with the SLME reference operation point O1 (see section 3.5). Subsonic area ratio is fixed at 2.5 and fractional length of the conventional parabolic type nozzle is kept at 0.8. Such values might be adapted to specific optima in later stage refinements but are sufficiently appropriate for conceptual vehicle definitions.

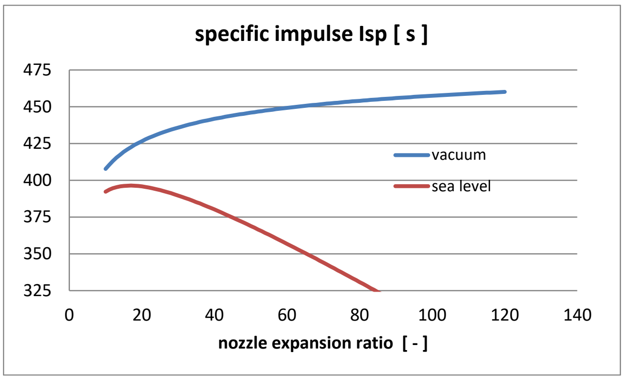


Figure 1: SLME Isp-performance as function of nozzle expansion ratio

Note the small maximum in the sea-level Isp curve showing the nozzle would be underexpanding even on ground at area ratios below 17 which from performance perspective is not interesting.

2.1 ENTRAIN Generic RLV-Study VTHL, VTVL

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. These designed as TSTO for a GTO-reference mission turned out to be feasible, however, reaching significant size of up to 80 m length [1, 2].

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, a launch system using LOX/LH2 and staged combustion engines in both stages was designed. The respective rocket engine was scaled based on the SLME, however with some specific features distinct to the baseline SLME.

Since the total take-off mass of the RLV systems is much lower than that of the SpaceLiner (compare around 400 tons to 1830 tons of SpaceLiner, see section 2.5), 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 thrust of one SLME in the 1st stage is 561 kN at sea-level and about 610 kN at vacuum conditions (expansion ratio $\epsilon = 23$). The 2nd stage thrust is 665 kN ($\epsilon = 120$). Furthermore, the 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 of the returning first booster stage. Such deep-throttling requirement for landing is new compared to the SpaceLiner application.

For the VTHL variant without supersonic retro-burn requirement the nozzle expansion ratio ϵ is set to 35.

2.2 VTHL RLV-C4

Approaching or even exceeding the payload performance expected for Ariane 6 in GTO or Lunar exploration missions would require extremely tall launcher configurations in case of tandem-staged TSTO with reusable first stage. Therefore, for this class of RLV a parallel stage arrangement is preferable: a winged stage is connected to an expendable upper segment with potentially various internal architectures. A payload of approximately 14 t GTO-class with multiple payload capability can be achieved by a 3-stage architecture while still remaining at relatively compact size [3]. Less demanding missions to different LEO can be served as TSTO.

The RLV's propellant loading in this architecture is limited to 370 t despite the powerful performance which makes a relatively compact lay-out possible. The 2nd expendable stage is defined as an H150 and becomes even more compact than the H155 core stage of the classical Ariane 5G. An important design constraint is the requirement of using similar engines in the reusable stage and the expendable second stage, however, with adapted nozzles. The upper stage for high performance missions, mainly GTO-injection, is selected as H14 for all concepts and is placed under the large fairing. Vinci [17] is the engine choice in the 3rd stage. Total lift-off mass for the GTO-mission is estimated at 667 tons.

The architecture definition of the 3STO launcher is visible in Figure 2 at left. Note the winged RLV-stage powered by 4 SLME with expansion ratio 33 and the expendable stage with the H150 and a single SLME with $\varepsilon = 59$ partially protected by the aft skirt. The full launcher is accelerated at lift-off only by the RLV-engines operating at MR= 6.5 and the second stage engine is ignited after stage separation and operated with MR of 5.5. Both rocket motors are exactly similar to the SpaceLiner booster (SLB) and upper stages (SLO/SLP) and any mission optimization in thrust level and expansion ratio has not yet been attempted.

2.3 VTVL

Beyond the winged VTHL-concepts, VTVL options of similar size are studied as a potential alternative shown in Figure 2 at right. The VTVL concept is derived of the RLVC4-III-B winged concept from section 2.2. The first stage is stripped of all hardware specific for VTHLs, namely wings, fins and flaps, rudders and landing gear. Instead, landing legs and grid fins are added.

A certain amount of the propellant in the 1st stage tanks is kept for performing the two re-entry maneuvers, the re-entry-burn and the landing burn. The whole launcher is sized to deliver the same 14 tons of payload into GTO, as the VTHLs. The system is iterated for target payload mass and minimal landing propellant mass.

As presented in [3], the main difference to the VTHL system is the increase in propellant mass. The dry mass is less compared to the VTHL system due to the fewer hardware that is needed while propellant loading has to increase by 10% and hence also length of the RLV-stage by approximately 3 m. In total, the GLOM of the VTVL system is at 676 t including the 14 t payload, leading to a total difference of only 10 t more for the VTVL system.

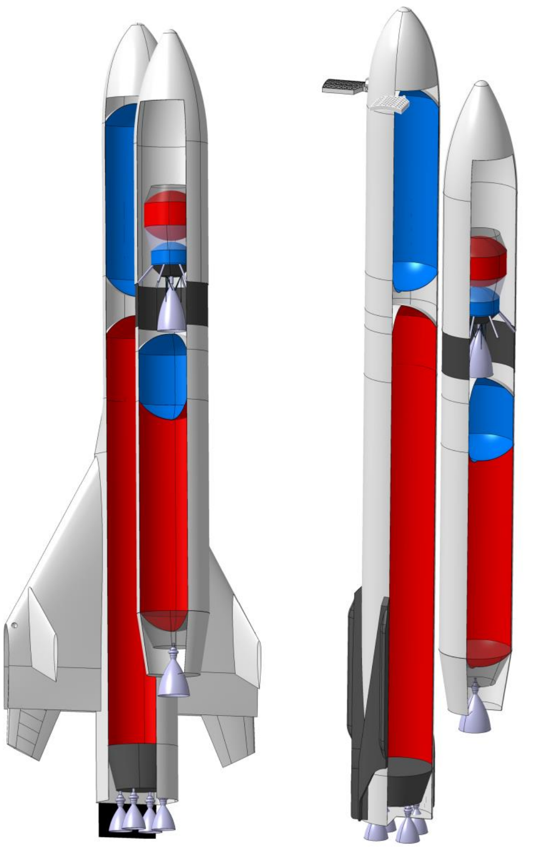


Figure 2: Launcher architecture sketch of 3STO configurations VTHL RLVC4-III-B (left) and VTVL (right) [3] both with SLME in 1st and 2nd stage

In order to achieve a controlled soft landing of the returning booster stage, the thrust-to-weight (T/W) ratio at landing should be close to 1. SpaceX' Falcon 9 is landing with a T/W ratio of around 1.2, as studies at DLR have shown [4]. A higher T/W decreases the duration and subsequently the losses of the landing burn, but requires a fast-responding engine throttle and actuator control. For the VTVL stage in Figure 2 at right, the T/W at landing can be as low as 1.05 with one SLME firing and 35% throttle, which was assumed to be the lower threshold for the engine in this context. With 50% throttle the T/W would be at a value of 1.48. Such low throttling levels have not yet been investigated for the SLME (see section 3.5) and these conditions are to be analyzed in future work.

2.4 HTHL

The ESA-funded system study RESOLVE on future TSTO-options with reusable first stage selected the SLME as baseline LOX-LH2 staged combustion engine for vertical lift-off types as well as for horizontal take-off (HTHL). The expendable upper stages are to be powered by SLME with large expansion ratio nozzles.

The rocket engine key data of the horizontally-sled-launched HTHL1011 configuration are optimized by global vehicle iterations. The thus obtained nozzle expansion ratios for the RLV-stage are 32.2 and 115.5 for the expendable upper stage. Figure 3 shows two of in total 4 SLME arranged in the aft base. Note the relatively small size of the engines in comparison with the large dimensions of the vehicle. A launcher T/W below 1 is sufficient for horizontal take-off configurations.

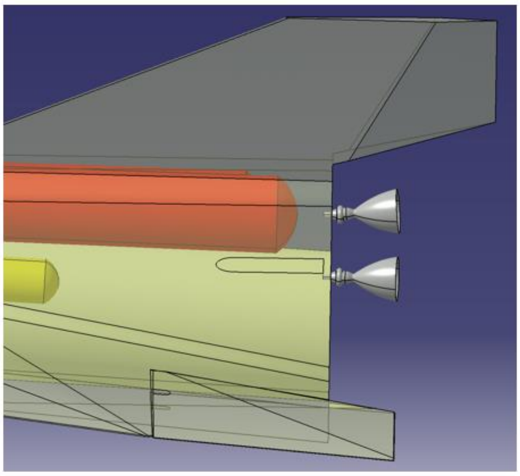


Figure 3: Aft section of HTHL TSTO configuration with SLME arrangement

Estimated Gross Lift-Off Mass of the launcher including target payload and complete propellant loading of all stages is calculated at above 761 t. No specific throttling requirements have been identified for the SLME in this application.

2.5 SpaceLiner

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. For example, Europe – Australia could be flown in 90 minutes. [5, 6, 7] Other interesting intercontinental destinations between e.g. East-Asia and Europe or the Trans-Pacific-route to North-West America could be reduced to flight times of slightly more than one hour [14, 16]. 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 [14].

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 [15].

The SpaceLiner 7 passenger transport is also technical basis for a two-stage fully reusable satellite launch vehicle. The baseline design of the orbital launcher remains unchanged to the passenger version with a fully reusable booster and passenger stage arranged in parallel and the external shapes will be very similar. The satellite launch configuration is described in more detail in [14].

2.5.1 SpaceLiner 7 Architecture and Geometry

The general baseline design concept consists of a fully reusable booster and passenger stage arranged in parallel at lift-off as presented in Figure 4. 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 [13].

External shapes of passenger and orbital configuration with satellite payload are almost identical. The internal arrangement of the upper stage is adapted to the specific mission with the forward passenger cabin replaced by a central cargo bay and adequately placed LOX-tank. The main dimensions of the 7-3 booster configuration are listed in Table 1 while major geometry data of the SpaceLiner 7-3 passenger or orbiter stage are summarized in Table 2.

Total dry mass of the SpaceLiner 7 launch configuration is estimated at 301 Mg (satellite version) and 327 Mg (passenger version) with a total propellant loading of 1467 Mg or 1502 Mg. The resulting GLOWs are 1807 Mg (satellite version) and 1832 Mg (passenger version) either incl. passengers or payload and expendable upper stage.

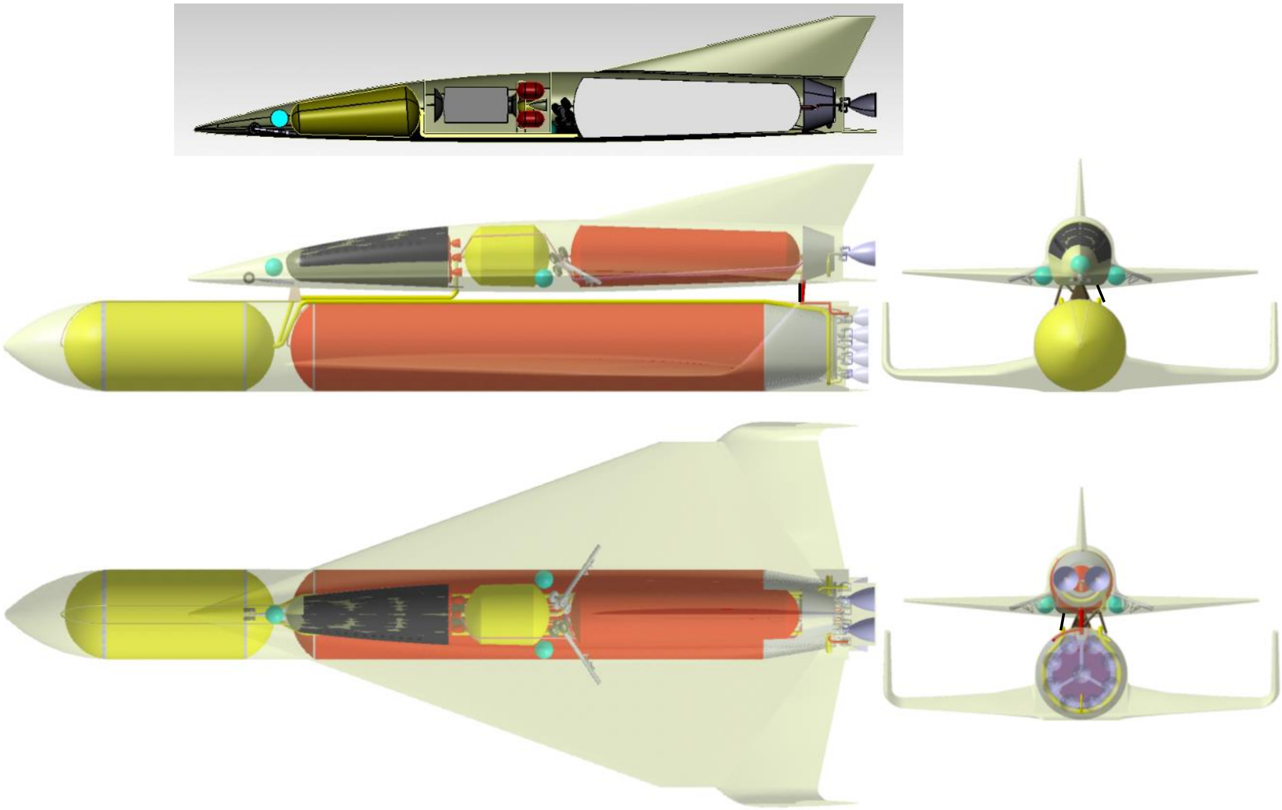


Figure 4: Sketch of SpaceLiner 7 launch configuration with passenger stage (SLP) with its booster stage at bottom position and orbital stage of SLO in insert at top showing the SLME arrangement in the lower right figure

Table 1: Geometrical data of SpaceLiner 7-3 booster stage

length [m]	span [m]	height [m]	fuselage diameter [m]	wing leading edge angles [deg]	wing pitch angle [deg]	wing dihedral angle [deg]
82.3	36.0	8.7	8.6	82/61/43	3.5	0

Table 2: Geometrical data of SpaceLiner 7-3 passenger / orbiter stage

length [m]	span [m]	height [m]	fuselage diameter [m]	wing leading edge angle [deg]	wing pitch angle [deg]	wing dihedral angle [deg]
65.6	33.0	12.1	6.4	70	0.4	2.65

3 MAIN PROPULSION SYSTEM

Staged combustion cycle rocket engines around a moderate 16 MPa chamber pressure are not overly ambitious and have already been exceeded by operational values of existing engines like SSME or RD-0120. The target of 16 MPa is also a good compromise between European expertise and required performance of future launcher applications. The intended demonstrator SCORE-D [18] was the latest ESA-funded design and experimental work on maturing closed cycle rocket engines. The design chamber pressure would have been in the 16 MPa-class. Further, the ambitious goal of a passenger rocket is to considerably enhance reliability and reusability of the engines beyond the current state of the art.

The expansion ratios of the 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 sections 2.1 through 2.4) the reusable booster engines might perform a final mission with additional nozzle extension when approaching its design life-time.

This section is describing cycle conditions and preliminary sizing of components for the two SpaceLiner variants of the SLME with nozzle expansion ratios 33 and 59. 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 Engine 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 [8].

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 [8, 9]. 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 is a preferred design solution for the SpaceLiner [12]. This approach should allow avoiding the complexity and cost of additional inert gases like Helium for sealing.

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 [13] together with the engine's calculated operational domain.

3.2 SLME design requirements

The SpaceLiner 7 take-off thrust requirement per engine of around 2000 kN at sea-level conditions remains unchanged to [12, 13]. 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. A deep-throttling requirement down to 35% of sea-level thrust (≈740 kN) would be required only for the vertical landing of VTVL-concepts.

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.

The SpaceLiner's ascent reference mission requirements define the engine cycle times per flight:

- 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

The average engine life-time is targeting 25 missions 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 rocket engine.

The next generation of partially reusable launchers will see on the RLV-stages similar operation times and conditions 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.

3.3 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. The components and their connections are shown in Figure 5 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.

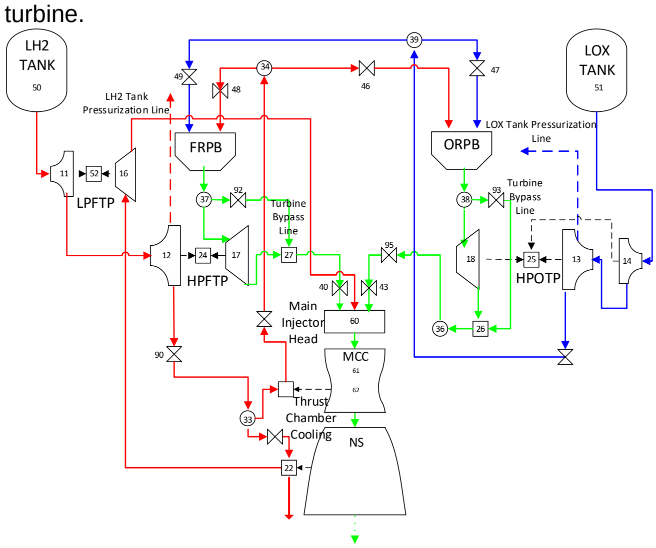


Figure 5: SLME internal flow schematics

In a Full-Flow Staged Combustion Cycle, two preburners 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 [8, 9].

3.4 Preliminary subcomponent sizing

The SLME baseline architecture [11, 12, 13] remains unchanged. Subcomponent sizing and definition is progressing at Phase A conceptual design level. Refinements are focusing on the turbomachinery designed as an integrated power-head and a suitable regeneratively cooled thrust-chamber lay-out. 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 as published in [13].

3.4.1 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 [11, 12] do not need to be updated as they still fulfil all requirements.

A preliminary thermal analysis of the SLME on the hotgas side had already been performed using TDK [11, 12]. The program RPA [21] 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. [21, 22] 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. [21, 22] Test cases of the SSME and the Aestus are in good agreement with experimental data for heatflux and wall temperatures. Only the area close to the propellant injectors shows a systematic overprediction of heatflux on the wall [21, 22] which likely is due to the in reality not yet fully completed combustion process there.

The RPA program is used for preliminary analyses of the SLME thrustchamber and regenerative cooling circuit. 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 side of the injector plate 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 thrust-chamber lifetime.

These thrustchamber cooling design assumptions listed above have been calculated for several operation points using RPA. Figure 6 shows temperatures along the chamber wall and of the H2-cooling fluid. The maximum thrustchamber wall temperatures below the TBC remain at less than 1000 K in this simulation. Note that only convective type cooling is considered in this RPA calculation. The effect of film cooling should allow for a further reduction in hotgas side temperatures.

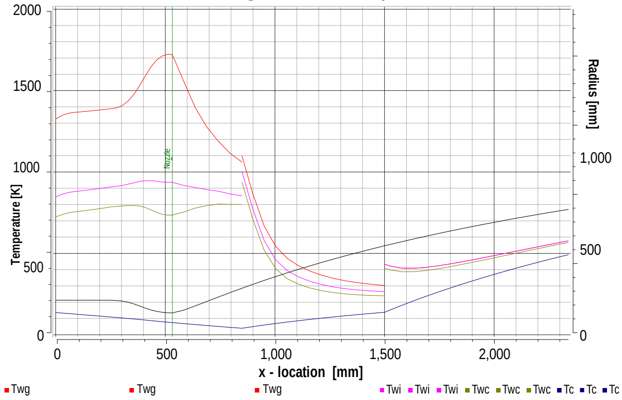


Figure 6: Temperature distribution in SLME-33 thrustchamber at O1 obtained from RPA analysis (*wg* hotgas wall, *i* below TBC, *wc* coolant wall, *w* coolant)

An exploration of different operating points (see description in following section 3.5 and Figure 17) has been performed in order to check on the feasibility of the regenerative cooling concept in the full operational domain. Figure 7 shows the expected heatflux for the nominal operation points O1, O2, O3 and the extreme points E1, E3, and E7 of Figure 17.

At the axial station 850 mm the regenerative cooling fluid is induced. Depending on the engine mixture ratio, the actually available H2-flow for cooling is changing which is considered in the simulations and flow distribution between throat section and nozzle extension is slightly adapted. Maximum fluxes are observed at the nozzle throat for the high loading conditions O2, E1 and E7 reaching beyond 80 MW/m². The maximum expected combustion temperature at MR=7 (E1) is close to 3700 K. The low mixture ratio operation including the nominal operating point O3 show significantly reduced heat loads.

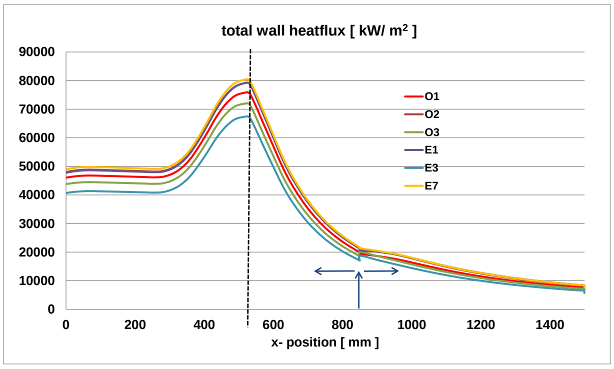


Figure 7: Wall heatflux in regeneratively cooled thrustchamber section of SLME for different operation points obtained from RPA analyses

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. Beyond expansion ratio of 16.6 (reference x-position 1500 mm) a combination of film cooling by small H_2-bleed and radiation cooling is applied. Figure 8 demonstrates that wall temperatures at the nozzle exit would not exceed 650 K according to the simple 1D-calculations. If confirmed by refined analyses for all nominal operating points, the cooling flow schematic of both SLME-variants could be very similar. The main targets of simplifying manufacturing and limiting of the engine mass can be supported.

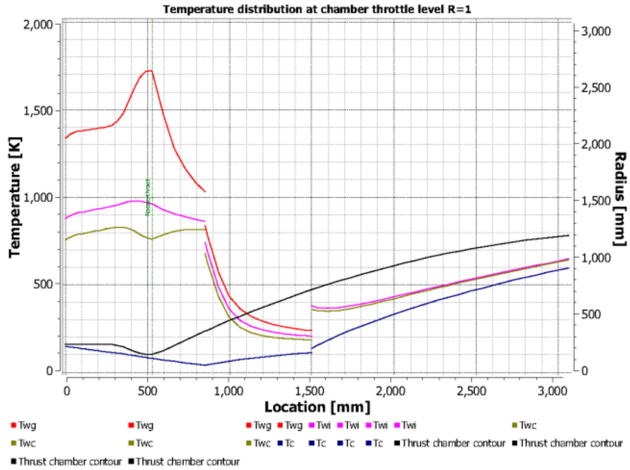


Figure 8: Temperature distribution in SLME-59 thrustchamber at O1 obtained from RPA analysis (*wg* hotgas wall, *i* below TBC, *wc* coolant wall, *w* coolant)

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 9) and [24].

3.4.2 Integrated Power Head

An Integrated Power Head (Pre-burner + Turbine + Impeller pump) as it has been used on the SSME (Figure 9) is also the preferred design solution for the SLME. The reduced length of high-pressure hot-gas lines should enable significant mass saving and a compact and clean lay-out.

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. The 34 years old studies comprise several innovative ideas which are worth revisiting today for potential application in an efficient reusable RLV engine.

In [28] an FFSC derivative engine of the SSME is proposed which should operate as a highly variable mixture ratio engine, especially for booster applications. Reference [13] explains why this approach is not selected for the SLME.

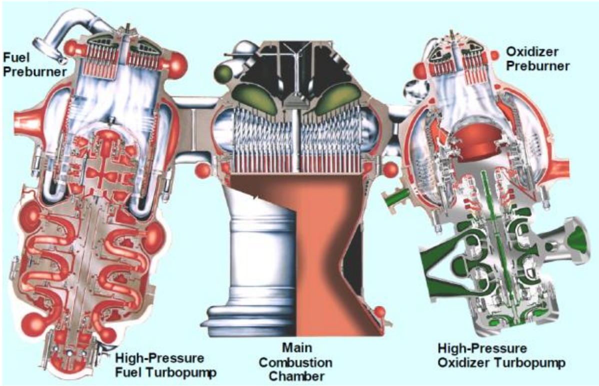


Figure 9: Example of SSME Integrated Power Head assembly attached to combustion chamber [24]

However, other elements of the proposed SSME "Derivative Engine" are of direct interest to be implemented in the SLME design, for example the very compact annular ox-rich preburner [13, 28].

Figure 10 shows 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. The preliminary layout from [12, 13] is maintained but now also considering the preliminary sizing of the regenerative cooling (section 3.4.1) and of the turbopumps (following section 3.4.4). The view of the SLME-33 from above in Figure 10 shows on the left the hydrogen fuel supply and on the right the oxygen flow side.

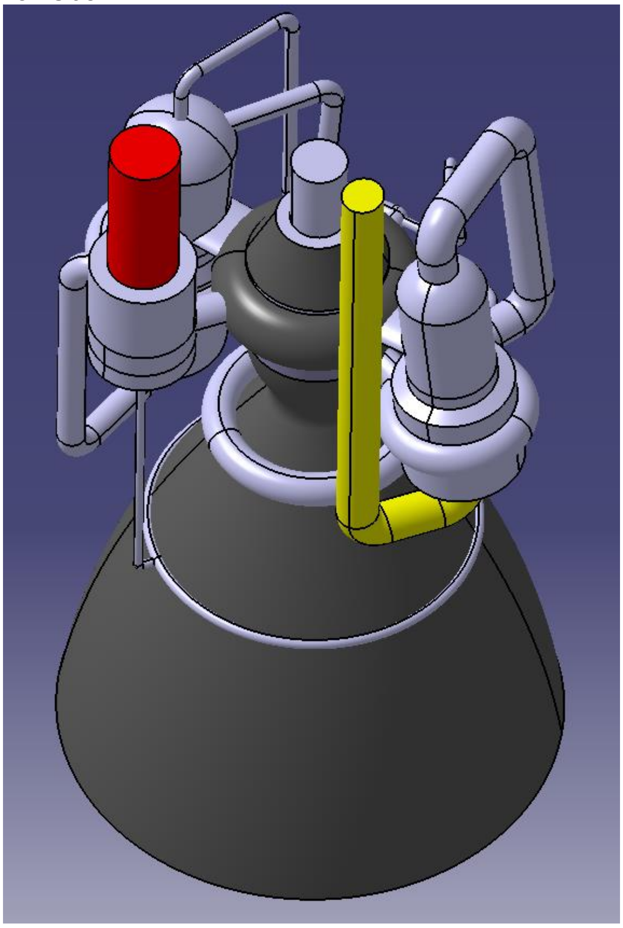


Figure 10: SLME simplified CAD geometry showing arrangement of turbomachinery

3.4.3 Preburners

The SLME preburners are attached to each turbo-pump in the integrated powerhead assembly as visible in Figure 10. 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. At each turbine a bypass / tap-off could be an option for which the flow should be controlled by a hotgas valve in order to allow engine operation in the full operational domain without significantly changing TET or 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 emergency by increasing TET beyond the limitation [8]. 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 [14].

Both preburners' external walls are actively cooled by their respective predominant fluids. The cooling fluid is heated up and subsequently used as pressurization gas for the tanks (see ref. [13]). Additional heat exchangers would not be required in such a design. It remains to be analysed in future work if pressurant gas dynamic control needs and preburner cooling requirements do match in all relevant flight conditions.

A very compact lay-out of the IPH-Ox with annular preburner around the shaft connecting turbine and impeller/inducer based on a similar design of the proposed SSME "Derivative Engine" [28] is considered [13]. More than 80% of the high-pressure oxygen is directly fed into the oxygen-rich preburner, about 15% are directed to the hydrogen-rich preburner and a small portion is used for LOX-tank pressurization.

The combustion in the preburners is a highly complex process involving multiple reaction elements interacting with each other at high pressures and temperatures. Understanding this process is crucial for the preburner design process, since the heat flux and temperature distribution throughout the preburner are important parameters influencing the preburner design.

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 geometry of a potential SLME preburner. The commercial CFD solver ANSYS CFX was used for the analyses. First, the combustion and transport models were validated using the Penn-State test case as described in various papers and can thus serve as benchmark. Once the models were sufficiently set up, a 2D (axially symmetric) geometry of the fuel-rich preburner with one coaxial injector element was analyzed. Different design parameters, such as the impulse ratio between the oxidizer and fuel streams and the mixture fraction were parametrically studied [26].

Figure 11 shows the result of one of these test cases which converged well and featured a stable combustion. Note that the temperature at the outlet is quite evenly distributed which can be contributed to the fact that there is sufficient mixing between the cold, unburnt fuel and the combustion products. For the oxygen-rich preburner, the chosen modelling approach was insufficient to achieve convergence and viable results yet. Here, alternative injector geometries, such as porous injectors [12] or a different coaxial injector geometry, shall be the focus of future work.

It is important to highlight the preliminary nature of the CFD simulation results as they should serve as a first step towards understanding the combustion and flow processes within the preburner, to identify which design parameters might have major influences, and which challenges in preburner design are to be expected for the future SLME design consolidation. The CFD models are planned to be further extended and adapted to support an efficient design process.

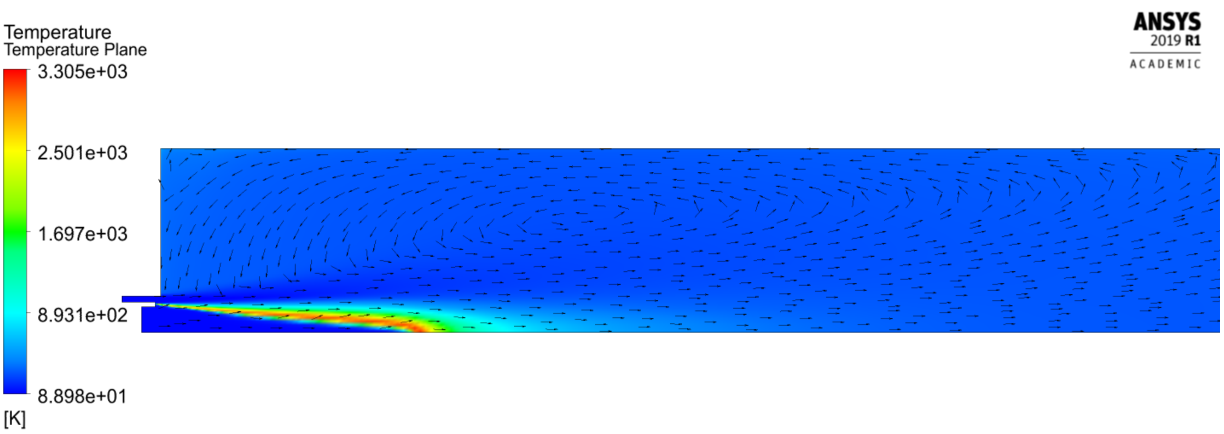


Figure 11: Temperature distribution in a simplified 2D fuel-rich preburner with one coaxial injector

3.4.4 Turbomachinery

The first preliminary definition of the turbomachinery lay-out has been described in references 9 through 13. 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 full-flow staged combustion cycle no LOX-split pump is necessary for raising discharge pressure to the fuel-rich preburner level as with the SSME [24].

A refined update of the SLME turbomachinery design was performed [27] 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.

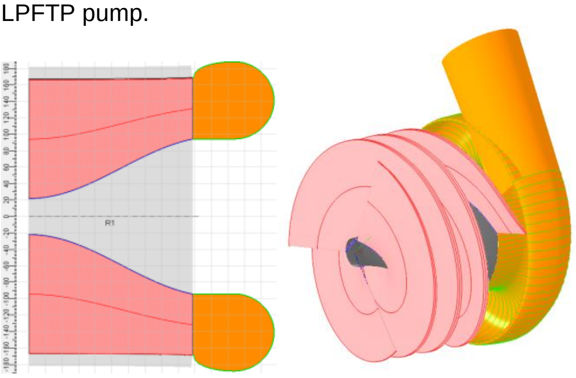
According to the pneumo-hydraulic scheme shown in Figure 5, the following turbomachinery components have been pre-designed: LPFTP pump and turbine, HPFTP pump and turbine and HPOTP pump and turbine. The thermodynamic parameters used for the turbomachines design correspond to the demanding operational point O2 and the SLME cycle design conditions of 2019, mostly similar to those presented in [13].

AxSTREAM® turbomachine internal efficiency accounts for flow path quality, tip leakage losses and disk friction losses without considering mechanical losses and shaft-seal leakage losses. Hydraulic efficiency shows the quality of the flow path design without considering tip/shroud leakage losses and without considering the quality of the seals.

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 pressure of 0.15 MPa and is pressurized to 1.5 MPa. The maximal casing diameter of the LPFTP is preliminarily set to 400 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 12 shows the design meridional (a) and 3D view (b) of the LPFTP pump.



(a) **(b)**

Figure 12: SLME LPFTP pump preliminary design [27]

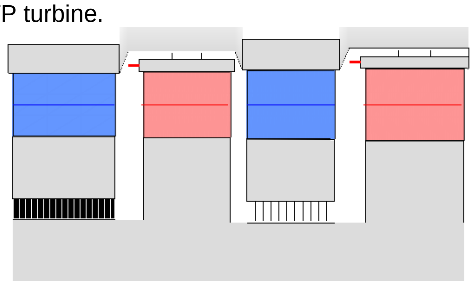
Performance parameters of the LPFTP are presented in Table 3.

Table 3: SLME LPFTP pump preliminary design performance [27]

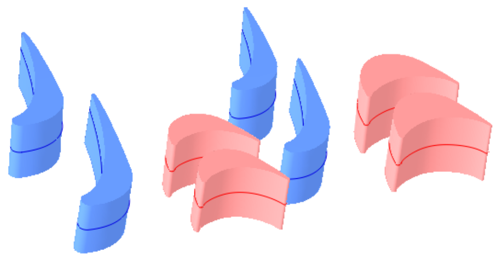
Mass flow rate [kg/s]	74.2
Power [kW]	1785.1
Total-to-total efficiency [-]	0.79
Pressure ratio [-]	10
Axial length [mm]	300
Maximal diameter [mm]	380

The turbine is fed from the regenerative circuit with hydrogen gas to drive for 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 turbopump, 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 respect to blades for the turbine flowpath, it is preferable to use prismatic ones as the change of thermodynamic parameters along the relatively short blade height is negligible. Figure 13 shows the design meridional (a) and 3D view (b) of the LPFTP turbine.



(a)



(b)

Figure 13: SLME LPFTP turbine preliminary design [27]

Main performance and dimensions data are given in Table 4.

Table 4: SLME LPFTP turbine preliminary design performance [27]

Mass flow rate [kg/s]	9.474
Power [kW]	1877
Total-to-total efficiency [-]	0.729
Axial length [mm]	80
Maximal diameter [mm]	377

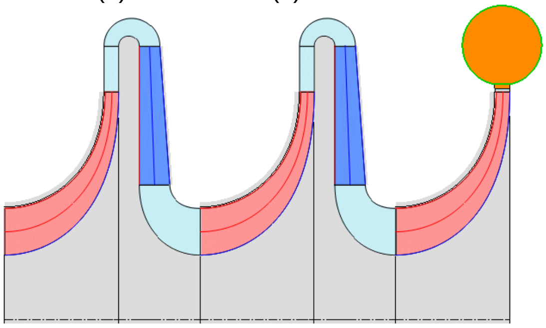
HPFTP

The pressurized flow after the LPFTP enters the HPFTP pump. The HPFTP outlet pressure at 2019 conditions was at 31.5 MPa. Several configurations of the HPFTP pump were considered varying the inducer, the vane diffuser presence, and the number of stages. The above parameters were selected specifically because of their impact on the cavitation margin, the efficiency, and the dimensions of the designed pumps. Additionally, a casing diameter constraint had been set at 500 mm.

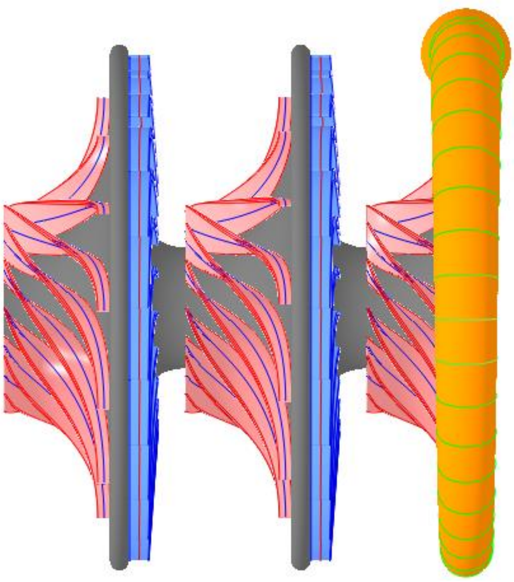
During the preliminary performance estimation, no cavitation was observed on the first impeller wheel of any configuration. For this reason, pump configurations without inducer are preferable due to lower axial dimensions and therefore lower mass.

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 pump meridional (a) and 3D view (b) of the HPFTP.



(a)



(b)

Figure 14: SLME HPFTP pump preliminary design [27]

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 5.

Table 5: SLME HPFTP pump preliminary design performance [27]

Mass flow rate [kg/s]	74.2
Power [kW]	39510
Total-to-total efficiency [-]	0.795
Pressure ratio [-]	21
Axial length [mm]	386
Maximal diameter [mm]	480

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 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. The flow path 3D view is presented on Figure 15.

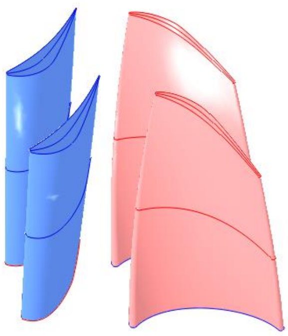


Figure 15: SLME HPFTP turbine preliminary design [27]

Table 6 represents the performance data of the HPFTP turbine.

Table 6: SLME HPFTP turbine preliminary design performance [27]

Mass flow rate [kg/s]	142
Power [kW]	41767
Total-to-total efficiency [-]	0.92
Axial length [mm]	60
Maximal diameter [mm]	400

It should be noted that additional strength calculations are required for the designed turbine as the preliminary estimation shows that the allowable stress values have been exceeded.

HPOTP

Oxygen is fed from the tank directly to the HPOTP with a pressure 0.5 MPa and is pressurized to 26.2 MPa according to 2019 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. Maximal casing diameter of the HPOTP was tried not to exceed 350 mm.

Preliminary design of both pumps shows that exceeding the external diameter target can't be avoided. The design with single stage impeller is selected. The design of the HPOTP is presented on Figure 16.

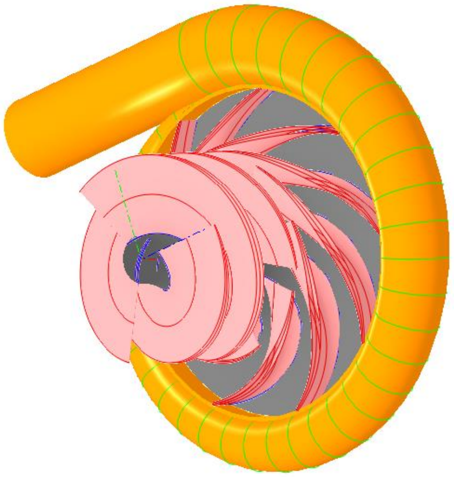


Figure 16: SLME HPOTP pump preliminary design [27]

Table 7 represents performance data of the HPOTP-pump.

Table 7: SLME HPOTP pump preliminary design performance [27]

Mass flow rate [kg/s]	482.3
Power [kW]	12393
Total-to-total efficiency [-]	0.85
Pressure ratio [-]	52
Axial length [mm]	170
Maximal diameter [mm]	435

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 and the maximum casing diameter does not exceed the target constraints.

Performance data of the HPOTP turbine is presented in the Table 8

Table 8: SLME HPOTP turbine preliminary design performance [27]

Mass flow rate [kg/s]	404
Power [kW]	13480
Total-to-total efficiency [-]	0.91
Axial length [mm]	58
Maximal diameter [mm]	320

3.4.5 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.

The HMS 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.

3.5 SLME operational domain and performance estimation

A computer program used in the early phases of the SLME cycle analysis is lrp2, based on the modular program SEQ [20] 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 in the previous publications [8, 9, 10, 11, 12, 13].

The commercially available program RPA [21] (initially version 2.2.3 and now 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 3.4.1). RPA is capable of predicting the delivered performance of a thrust chamber using semi-empirical relations [21] 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 [21]. The full-flow staged-combustion cycle (FFSC) which is the reference mode for the SLME is included in RPA. At the Space Propulsion 2018 conference a new, more flexible and powerful version 3 of RPA was announced [23] 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.x. However, this complicates the user input and slows convergence of lrp2. RPA offers more sophisticated performance estimation methods and can be operated by graphical user interface or by scripts. In the preliminary definition of the SLME both numerical tools are useful and complement each other. All recent updates of SLME-analyses are based on RPA 2.3.2 calculations.

The operational domain of the SLME has been further refined and some of the extreme points around the nominal operation conditions have been repositioned when compared to [13]. 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. Some concepts simply use the reference point O1 with 16 MPa chamber pressure while others implement the increased thrust O2 in the booster stage and O3 with the lower thrust but improved vacuum Isp in the second stage. As already described in section 3.2, in the SpaceLiner-application the SLME is required of switching the mode in flight: operating at MR=6.5 (O2) during lift-off and later throttled to MR=5.5 (O3) by reducing the LOX-massflow.

The newly calculated SLME operational domain is shown in Figure 17. 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. Note in Figure 17 that the higher loaded extreme points E1, E7, E8 have all been raised in massflow and hence chamber pressure compared to the domain presented in [13]. Thus, the SLME extends now 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 [24] or RD-0120.

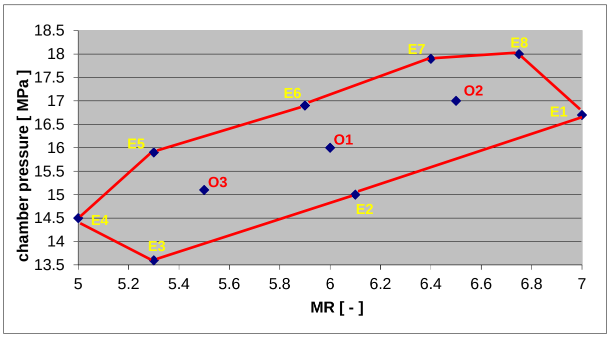


Figure 17: Calculated SLME operational domain

Previous SLME-analyses were based on preliminary turbomachinery analyses [11, 12] and subsequently turbopump efficiencies have been estimated using empirical data (e.g. from [19], [25]). These efficiencies obtained from graphs containing data from existing rocket turbopumps in dependency of specific speed and (volume) flow or speed ratio parameters indicate principal feasibility of such efficiency under similar design considerations. Although these data are not a final proof of actually achievable efficiency, they help in the definition of plausible engine cycle assumptions and turbopump design requirements.

The AxSTREAM® analyses of all turbopumps run by SoftInWay in 2019 [27] and with major results presented in the previous section 3.4.4 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 new cycle analyses but at lower values because not all losses considered (see definition in section 3.4.4). Further reiterations to fully converge on internal turbopump efficiencies in pre-sizing of turbomachinery and cycle calculations are planned in the next steps.

Table 9 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 5 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 750 K and 776 K ox-rich and 740 K to 780 K fuel-rich. The required preburner and hence turbopump discharge pressures are recalculated by RPA assuming increased pressure drops in lines, valves and injectors. Both preburners operate at similar pressure levels up to 23 MPa. However, the pumps are different: the OTP discharge reaches up to 29.3 MPa at O2 while HPFTP goes beyond 34 MPa because the complete hydrogen is directed first through the regenerative cooling before reaching the FRPB.

Table 9: 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]	23.3		24.88		22.14	
Oxidizer-rich Preburner pressure [MPa]	23.06		24.45		21.78	
Fuel-rich Preburner TET [K]	780		780		740	
Oxidizer-rich Preburner TET [K]	760.52		771.6		750.5	
HPFTP discharge pressure [MPa]	32.568		34.15		31.411	
OTP discharge pressure [MPa]	27.625		29.303		25.842	
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	386.13	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

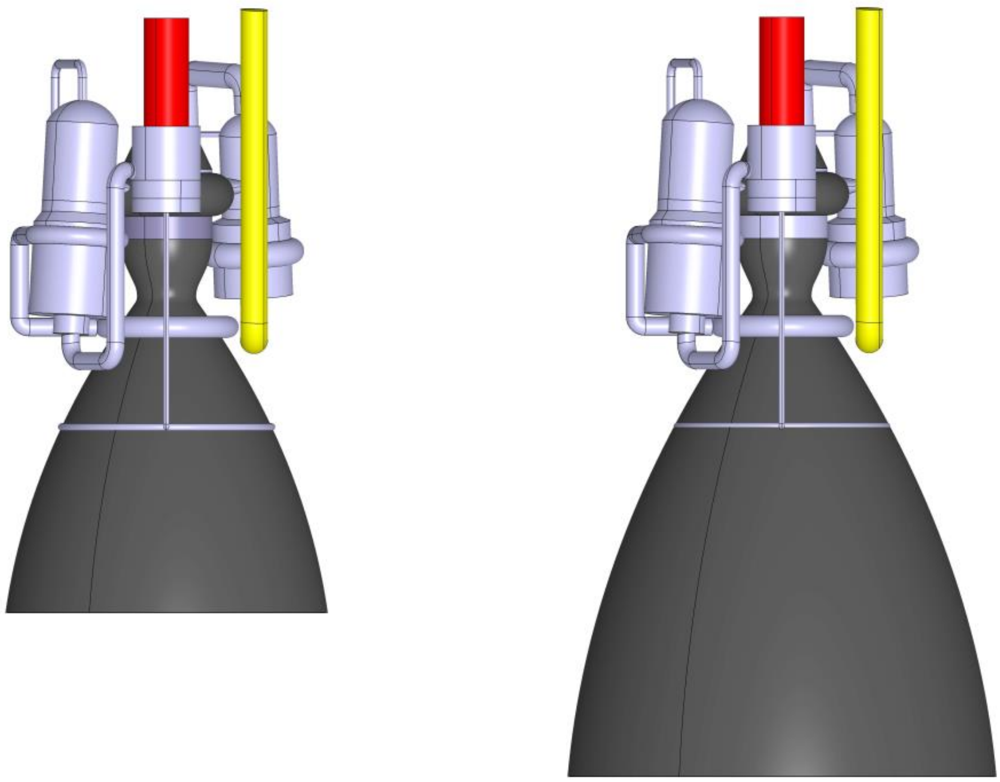


Figure 18: Size comparison of SLME with ε=33 (left) and ε=59 (right) as simplified CAD-models

The new SLME cycle analyses using RPA shows engine performance remaining overall very similar to the data previously published in [13] based on lrp2-calculations. Vacuum Isp-data of both tools are very close with relative difference less than 0.5%. At sea-level conditions the largest deviations could reach up to -1.5%. A similar situation was already observed in [13] and is not dependent on the refined engine analyses.

3.6 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 2981 mm. The larger passenger stage SLME has a maximum diameter of 2370 mm and overall length of 3893 mm. A size comparison of the two variants with a preliminary arrangement of the engine components is visible in Figure 18.

The engine masses are estimated at 3375 kg with the large nozzle for the passenger stage and at 3096 kg for the booster stage. These values are equivalent to vacuum T/W at MR=6.0 of 68.5 and 72.6.

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.

The combustion process in the preburners has been numerically studied in a simplified rotational-symmetric model. While for the fuel-rich burner numerical convergence and stable combustion could be achieved, the ox-rich burner will require further analyses.

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 are estimated at 3375 kg with large nozzle for the passenger stage and at 3096 kg for the booster stage.

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, and Mr. Francesco Cremaschi to the preliminary design of the SLME and the SpaceLiner propulsion system.

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