Preliminary Component Definition of Reusable Staged-Combustion Rocket Engine

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Full-flow staged combustion cycle rocket engines with a moderate 15 to 17 MPa range in chamber pressure have been selected as the baseline propulsion system of DLR's visionary fully reusable SpaceLiner concept.

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 and a suitable thrust-chamber lay-out. One key-objective is a light-weight, long-life, low-maintenance architecture.

The paper describes the latest status of the SpaceLiner 7 propulsion system:
- **The reference vehicle's preliminary design,**
- **Main propulsion feed- and pressurization system definition and architectural lay-out,**
- **Thrust chamber regenerative circuit,**
- **Pre-design of different turbomachinery and attached preburners,**

Design solutions are traded and compared to other recent high-performance rocket engine developments based on available design data or reassessment.

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

CFRP	Carbon Fiber Reinforced Plastics
FADEC	Full Authority Digital Engine Control
FFSC	Full-Flow Staged Combustion
FRSC	Fuel-Rich Staged Combustion
FTP	Fuel Turbo Pump
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
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
C	chamber
s/l	sea level
vac	vacuum

1   INTRODUCTION

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. The number of launches per year should be strongly raised and hence manufacturing and operating cost of launcher hardware should dramatically shrink.

Ultra-long distance travel from one major business center of the world to another major agglomeration on earth is a huge and mature market. Since the termination of Concorde operation, intercontinental travel is restricted to low-speed, subsonic, elongated multi-hour flight. An interesting alternative to air-breathing hypersonic passenger airliners in the field of future high-speed intercontinental passenger transport vehicles is a rocket-propelled, suborbital craft. Such a new kind of 'space tourism' based on a two stage RLV has been proposed by DLR under the name **SpaceLiner** [1, 2]. Ultra-long-haul distances like Europe – Australia could be flown in 90 minutes. 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 [12].

The SpaceLiner 7 passenger transport is an ideal 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. This approach intends to enable dramatic

savings on development cost and moreover by manufacturing the vehicles on the same production line, also significantly lower hardware cost than would result for a dedicated new lay-out. The satellite launch configuration is described in more detail in [12].

The DLR SpaceLiner is not the only launcher concept designed for high reusability and multiple mission capabilities. In the U.S. the commercial companies SPACE EXPLORATION TECHNOLOGIES CORP. (SpaceX) and Blue Origin are pushing their developments in similar directions: 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. Two rocket engines with ambitious technology goals, Raptor and BE-4, are under development for these applications. Therefore, it is of interest to compare key engine parameters with those of the SLME at the end of this paper.

1.1 SpaceLiner Evolution

First proposed in 2005 [1], the SpaceLiner is under constant development and descriptions of some major updates have been published since then [11, 12]. The European Union's 7th Research Framework Programme has supported several important aspects of multidisciplinary and multinational cooperation in the projects FAST20XX, CHATT, HIKARI, and HYPMOCES.

An important milestone has been reached in 2016 with the successful completion of the Mission Require-ments Review (MRR) initiating the concept's maturing from research to structured development [12].

1.2 SpaceLiner 7 Architecture and Geometry

The general baseline design concept consists of a fully reusable booster and passenger stage arranged in parallel. 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. After a rapid acceleration to its maximum speed the hypersonic transport is gliding for the remaining more than one hour flight to its destination.

The arrangement of the two vehicles at lift-off is presented in Figure 1. 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 confi-guration 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.
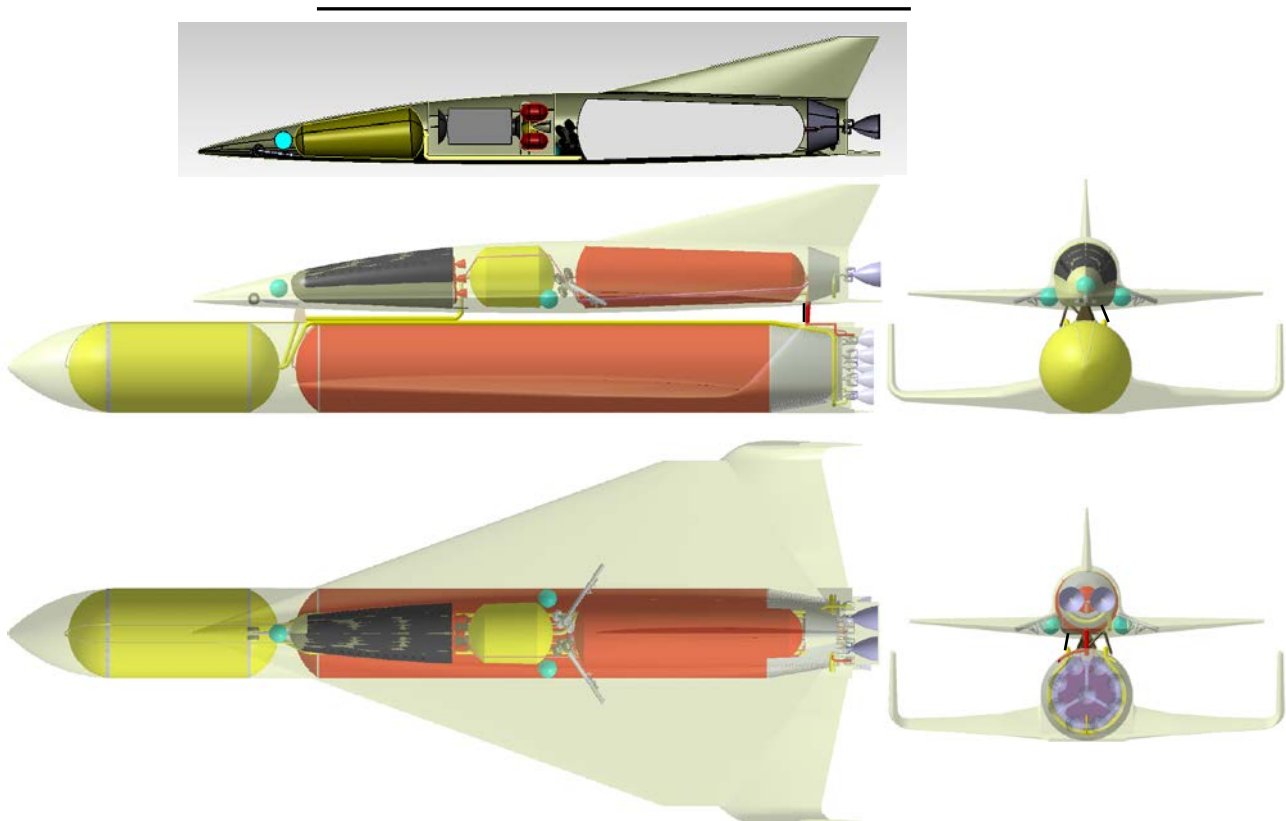

Figure 1: Sketch of SpaceLiner 7 launch configuration with passenger stage (SLP) with its booster stage at bottom position and orbital stage of SLO in insert at top showing the SLME arrangement in the lower right figure

Table 1: 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

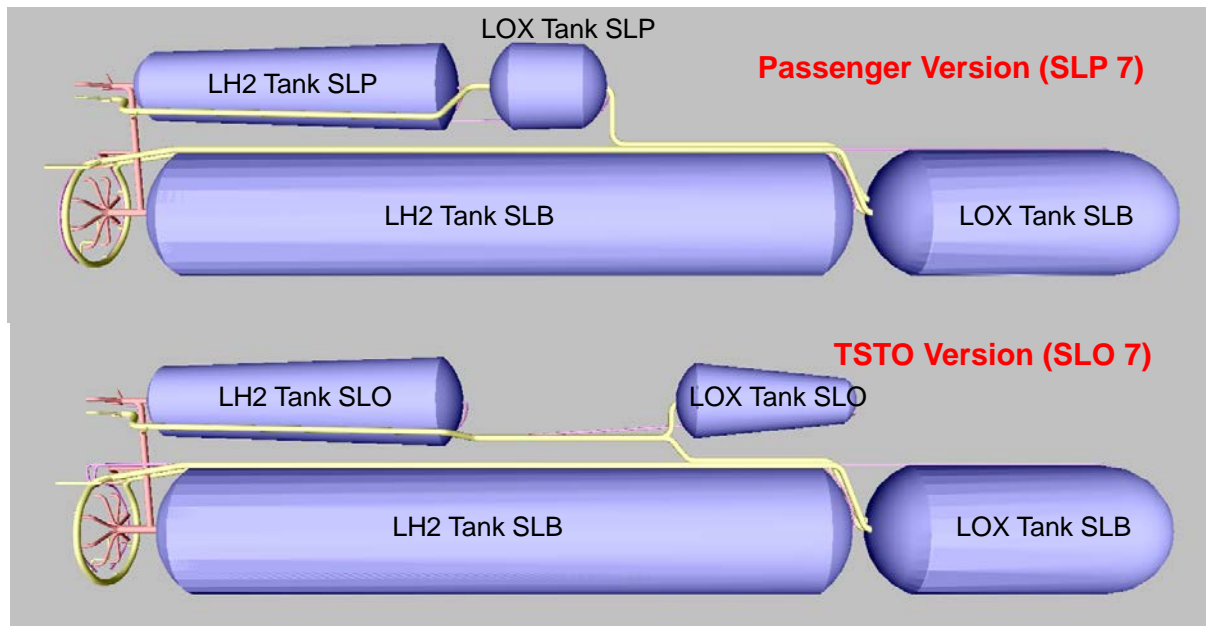| length [m]	span [m]	height [m]	fuselage diameter [m]	wing leading edge angle [deg]	wing pitch angle [deg]	wing dihedral angle [deg]
65.6	33.0	12.1	6.4	70	0.4	2.65

Figure 2: Arrangement of propellant tanks, feed- and pressurization system of SpaceLiner 7 (Passenger version at top, TSTO version for satellite transport at bottom)

2   MAIN PROPULSION SYSTEM

Staged combustion cycle rocket engines around a moderate 16 MPa chamber pressure have been selected as the SpaceLiner main propulsion system called SLME (SpaceLiner Main Engine). Such engine operational values are not overly ambitious and have already been exceeded by existing engines like SSME or RD-0120. However, 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 orbiter engines are 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.

2.1   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 [5].

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 [5, 6]. 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 [9]. This approach should allow avoiding the complexity and cost of additional inert gases like Helium for sealing.

2.2 Propellant feed and tank pressurization system

All main engines of the configuration 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 latter's overall size. The propellant feed- and pressurization system is pre-liminarily designed using the DLR-tool pmp. The recent arrangement of feed- and pressurization lines with the tanks of both stages in the mated configuration is shown in Figure 2 for both versions of the SpaceLiner: the passenger (top) and the orbital transport (bottom).

All LOX-feedlines and the LH2-crossfeed connection are attached on the booster's top outer side close to the upper stage. These lines are subjected to the reentry flow in the relatively cold wake region which is not critical even when undergoing maximum heatflux

around Mach 10. The feedlines of the upper stage are completely internal and ducted underneath the TPS because of the elevated heatloads experienced for much longer duration.

As shown in Figure 2, the major difference between the two variants' propellant system is related to the LOX-tank of the upper stage which is moved forward into the nose section of the SLO. The available internal volume of this tank has to be reduced by 23 m^3 due to space restrictions which affects the stage's tank mixture ratio. Further, an adapted feedline and crossfeed system is needed. The single LOX-feedline is bypassing the satellite cargo-bay while routed through the wing attachment structure. Available space is tight and, therefore, some relatively narrow line bends are required because any blocking of the landing gear box is to be avoided.

Three main options of propellant crossfeed exist:
- Line-to-line
- Tank-to-tank
- Tank-to-buffer-tank

These options have been investigated in the FP7-project CHATT [10] with steady-state flow-simulation along the full powered trajectory and transient simulation of critical phases like engine cut-off or valve closing.

The tank-to-tank crossfeed approach is definitely recommended for the LOX system of the SLP because the LOX-flow is supported by the hydraulic head of the booster tank's forward position. Almost no transient critical issues have been found. If the line-to-line architecture will be used for the LH2-side, the particular focus has to be on the valve timing and control. For the SpaceLiner 7-3 passenger configuration line-to-line crossfeed for LH2 and tank-to-tank crossfeed for the LOX system is the baseline choice [10].

The forward LOX-tank position in the SLO demands a different LOX-crossfeed connection as visible in Figure 2. In case of the SpaceLiner 7 TSTO transport configuration line-to-line crossfeed is the baseline option both for the LH2- and for the LOX-feed system.

Beyond the steady state simulation also the transient behavior in the propellant feed-lines has been analyzed for critical conditions along the powered flight and its preferred actuation sequence has been preliminarily defined [10]. In particular, the process of booster separation is a dimensioning factor for the design of the crossfeed system due to the switch of the propellant supply from the booster to the orbiter tanks [10].

Refined steady state modelling of the propellant feed and tank pressurization system has been performed for both SpaceLiner versions using the latest version of the DLR-code pmp 1.1. The slightly different propellant loading as well as the different acceleration profiles due to adapted throttling have an impact on pressure history inside the lines and the pressurization gas requirements.

Figure 3 shows the NPSP history in the upper stage's feed system of the satellite transport obtained by steady state simulation. The selected position for the

output is exactly at the interface between feedline and engine inlet which is close to the turbopump entry conditions. The low hydraulic head of the hydrogen feed line and the aft position of the LH2 tank results in almost no visible impact of acceleration on available NPSP. Staging at 240 s into flight is clearly visible by a sudden increase in pressure due to the hydraulic head of the at this point completely full upper stage tank. The hydrogen massflow is almost constant and due to the conical tank geometry pressure is slightly decreasing to the end of the mission.
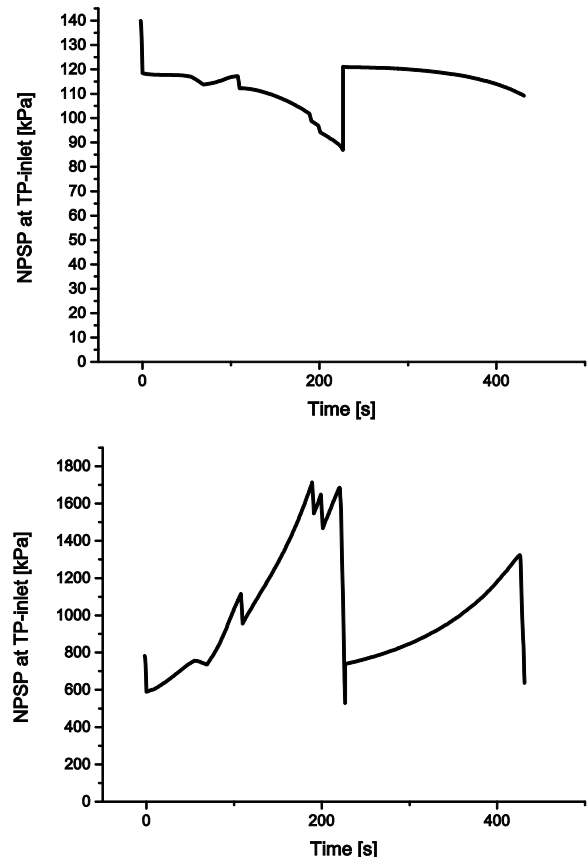

Figure 3: Calculated NPSP at interface feedline-engine inlet (top: LH2, bottom: LOX) of the upper stage of the TSTO configuration

The passenger version has a tank-to-tank crossfeed from the booster stage to the upper stage LOX-tank [10]. Figure 2 shows a different line-to-line crossfeed connection for the TSTO satellite transport version. A tank-to-tank connection has been assessed as unpractical because the feedline would have to be attached to the tank cone instead of the forward bulkhead. The long LOX-feedline continuously generates significant hydrostatic pressure and the vehicles acceleration profile with throttling and staging is clearly visible. Note the unmanned satellite launcher has a larger axial acceleration limit than the passenger version. NPSP at the LOX-feedline-engine inlet interface never falls below 500 kPa (Figure 3). This generous margin might allow for reducing the ullage pressure in the LOX-tank after staging.

Tank pressures are selected that the minimum NPSP requirements in all feedline segments are respected along the full mission; especially those at the engine entry. All LOX-tanks of the SpaceLiner system are pressurized by gaseous oxygen and all hydrogen tanks with gaseous hydrogen. This approach is

selected in order to avoid any excessive use of expensive and rare helium.

The tank pressurization gases are supplied from the SLME and need to be consistent with sufficient pressurization levels in all operational modes while minimizing the bleed demand from the engine. The latest SLME investigations focus on using the cooling fluids of the preburners (see section 2.6.3) for tank pressurization which represents an important design change compared to the previous assumption [9] of tapping the H2-MCC-regenerative circuit and the OTP discharge while using a separate heat exchanger.
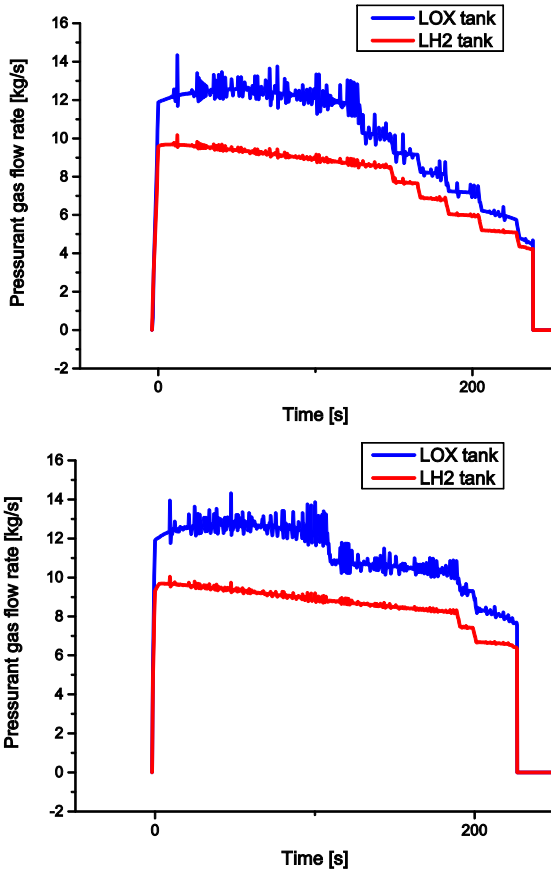

Figure 4: Required pressurant gas flow rate for the booster stage of the SLP (top) and SLO (bottom) configuration

The required total pressurant gas flow rate for the booster stage both for the passenger and satellite missions is shown in Figure 4. Although tank volume and engine NPSP requirements are identical, a notable difference in the profiles is visible due to different throttling requirements. If all nine booster engines are contributing to the tank pressurization, a maximum flow of approximately 1.45 kg/s GOX and 1.1 kg/s GH2 would have to be delivered by each booster engine.

The required total pressurant gas flow rate for the upper stage both for the passenger and satellite missions is shown in Figure 5. During mated flight these engines are fed from the booster. Therefore, the tanks remain filled and additional pressurization gas is not required. At crossfeed disconnect at around 220 s a sudden jump is visible in the simulation results when draining of the upper stage tanks starts. It is worth noting that the maximum GOX flow per engine is

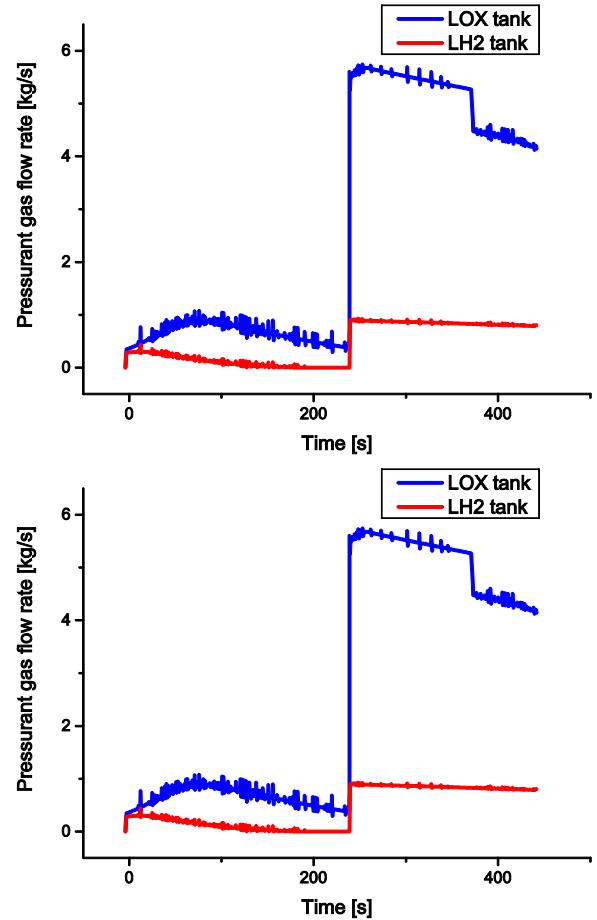between 2.1 and 3 kg/s, higher than for the booster, while the GH2 flow is below 0.5 kg/s.

Figure 5: Required pressurant gas flow rate for the upper stage of the SLP (top) and SLO (bottom) configuration

Maximum similarity between the two versions of the SLME is a key requirement. The slight mismatch in the pressurant gas flow in the current simulation is not a major concern and might be addressed by adjusting the pressure levels inside the tanks. The available NPSP at the feedline engine interface allows such adjustments without violating the minimum required NPSP.

The booster LOX tank ullage pressure can be limited to 1.9 bar because of its forward position always generating a lot of hydrostatic pressure down the line which is beneficial for good NPSP margin. Due to this fortunate situation, the required oxygen gas at booster stage MECO is below 2500 kg. The ullage pressure in the LH2-tank had to be increased to 3.4 bar compared to the previous version to ensure sufficient NPSP in the feedline. Nevertheless, hydrogen gas mass inside the very large 2577 m^3 LH2-tank is no more than 1710 kg at MECO because of hydrogen's low molecular mass.

Currently, ullage pressures in the upper stages have been set to 4.6 bar in the LOX-tank and 2.5 bar in the LH2-tank. A pressure reduction in the LOX-tank and a slight increase in the LH2-tank are options to be investigated in the next design iteration. Residual H2 gas mass at MECO is around 230 kg while GOX is around 860 kg for the SLO and 1000 kg in the larger LOX-tank of the SLP-configuration.

2.3 SLME design requirements

The SpaceLiner 7 take-off thrust requirements of around 2000 kN at sea-level conditions remain unchanged to [9]. 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.

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 require-ments 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.

2.4 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 a preferred design solution for the SpaceLiner. The components and their connections are shown in Figure 6 for the current baseline with FTP split into boost pump driven by separate expander turbine and HPFTP.
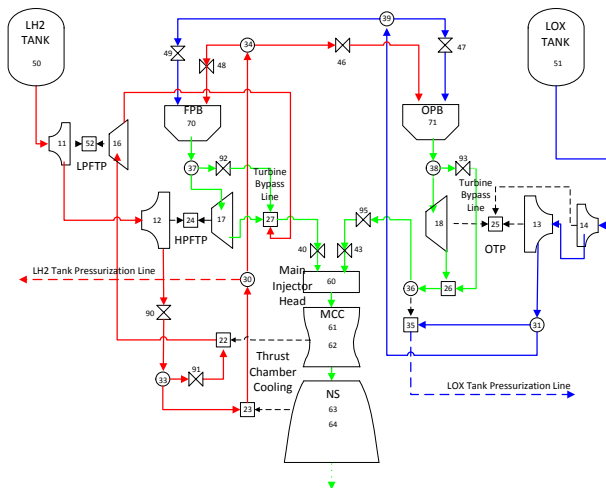

Figure 6: SpaceLiner Main Engine Schematics

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 (FPB) and the oxidizer-rich preburner (OPB) after being pressurised by each turbo pump. After the turbine gas created in each preburner pass through each 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 FTP [5, 6].

2.5 SLME operational domain and performance estimation

The operational domain of the SLME has been refined and extreme points around the nominal operation points have been defined. As already mentioned, the SLME is operating at MR=6.5 during lift-off and is later throttled to MR=5.5 by reducing the LOX-massflow. These modes represent O2 and O3 in Figure 7. O1 is not planned to be used in nominal flight of the SpaceLiner. However, O1 is located in the center of the domain and is also useful in the comparison with other rocket engines because a mixture ratio of 6 is typical for LOX-LH2 engines [16]. The SLME with its calculated performance at O1's moderate 16 MPa chamber pressure is also used in ongoing RLV-system studies at DLR.

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. Maximum LH2 massflow variation within the domain is around 10%.
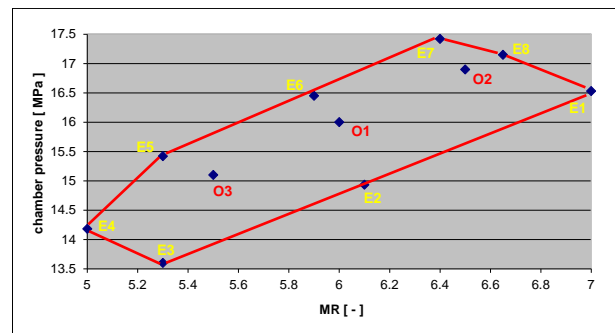

Figure 7: Calculated SLME operational domain

This operational domain is result of preliminary component sizing and dimensioning and engine cycle analyses for each operating point (O1 to O3 and E1 to E8).

One program used for the cycle analysis, lrp2, is based on the modular program SEQ [17] of DLR. Since the 1990ies this powerful tool has been significantly upgraded. The modular aspect of the program allows for a quick rearrangement 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. Certain conditions can be linked to component settings (i.e. the program varies according to user specification the pump exit pressure in order to reach a given chamber pressure). 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.

Another tool has been applied recently for the preliminary analyses of the SLME. The commercially

available program RPA (versions 2.2.3 and 2.3.2) has been used as a second tool for crosscheck of results and for further refinement of component definition (see also section 2.6.1). RPA is capable of predicting the delivered performance of a thrust chamber using semi-empirical relations [18] to obtain performance correction factors, including:

- performance loss due 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 turbomachinery to achieve a required combustion chamber pressure [18]. The full-flow staged-combustion cycle (FFSC) which is the reference mode for the SLME is included in RPA.

The lrp2 program is significantly more flexible in the arrangement of flow paths inside the engine than RPA. 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.

Major SLME engine operation data were presented in [9] based on the preliminary tentative assumption that all turbopump efficiencies are fixed at 70%. Turbopump design parameters were gained by the preliminary turbomachinery analyses [8, 9] and subsequently turbopump efficiencies based on these design parameters have been estimated using empirical data (e.g. from [15], [21]). 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 more realistic engine cycle assumptions and turbopump design requirements.

Table 3 gives an overview about major SLME engine operation data for the nominal operating points as obtained by lrp2 cycle analyses. The type with separately driven LH2 boost pump as shown in Figure 6 is selected for all calculations. Slight deviations of the internal conditions between SLME with small and large nozzle expansion ratio are due to the numerical iteration.

Similar cycle performance analyses were run using RPA. Overall engine performance is found similar with largest deviations in sea-level conditions. The required preburner and hence turbopump discharge pressures are iterated to lower values by RPA than obtained by lrp2 cycle analyses, however, the TET range is found more demanding. The OTP discharge pressure in Table 3 seems to be excessively high, not supported by design needs visible in the flow chart presented in [8, 9].

A critical reassessment of all internal engine functional conditions, pressure drops etc. using different numerical design tools is ongoing. As long as the defined main combustion chamber conditions can be reached, the internal conditions in a staged combustion cycle do not impact the overall engine performance.

Table 3: SpaceLiner Main Engine (SLME) technical data from lrp2 numerical cycle analysis [8]

	Booster SLB			Upper Stage SLP / SLO		
Operation point	O3	O1	O2	O3	O1	O2
Mixture ratio [-]	5.5	6.0	6.5	5.5	6.0	6.5
Chamber pressure [MPa]	15.1	16.0	16.9	15.1	16.0	16.9
Fuel-rich Preburner pressure [MPa]	24.5	25.5	26.6	24.5	25.5	26.6
Oxidizer-rich Preburner pressure [MPa]	24.3	25.2	26.3	24.3	25.2	26.3
Fuel-rich Preburner TET [K]	723	727	735	723	727	735
Oxidizer-rich Preburner TET [K]	763	765	768	763	765	768
HPFTP discharge pressure [MPa]	30.5	31.7	33.0	30.5	31.7	33.0
OTP discharge pressure [MPa]	37.0	39.2	42.2	37.0	39.2	42.2
Mass flow rate in MCC [kg/s]	479	515	553	479	515	553
Expansion ratio [-]	33	33	33	59	59	59
c* [m/s]	2327	2293	2257	2328	2292	2259
c_F [-]	1.851	1.870	1.890	1.900	1.922	1.946
Specific impulse in vacuum [s]	439	437	435	451	449	448
Specific impulse at sea level [s]	387	389	390	357	363	367
Thrust in vacuum per engine [kN]	2061	2206	2356	2116	2268	2425
Thrust at sea level per engine[kN]	1817	1961	2111	1678	1830	1986

2.6 Preliminary subcomponent sizing

The SLME baseline architecture of 2014 [8, 9] 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.

2.6.1 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 [8, 9] do not need to be updated as they still fulfil all requirements.

A preliminary thermal analysis of the SLME on the hot gas side had already been performed using TDK [8, 9]. The program RPA 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. [18, 20] 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. [18, 20] 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 [18, 20] 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 at 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 thrustchamber lifetime.

These thrustchamber cooling design assumptions listed above have been calculated for several operation points using RPA. Figure 8 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.

An exploration of different operating points has been performed in order to check on the feasibility of the regenerative cooling concept in the full operational domain. Figure 9 shows the expected heatflux for the nominal operation points O1, O2, O3 and the extreme points E1, E3, and E7 of Figure 7.
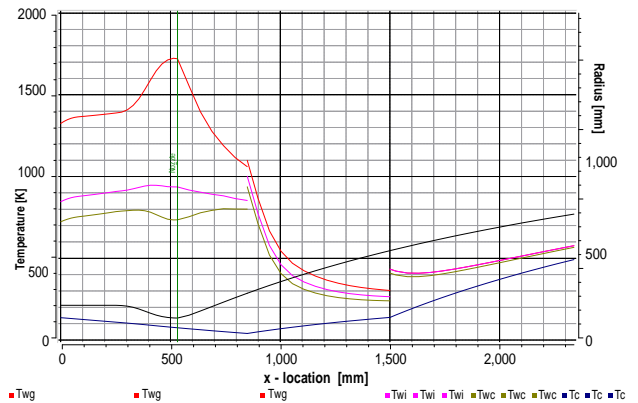

Figure 8: Temperature distribution in SLME thrustchamber at O1 obtained from RPA analysis (*wg* hotgas wall, *i* below TBC, *wc* coolant wall, *w* coolant)

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/m^2. 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.
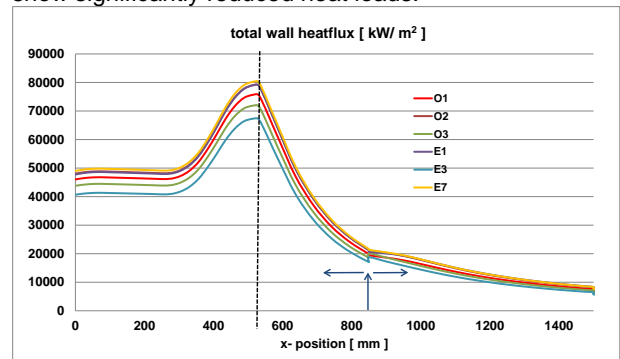

Figure 9: Wall heatflux in regeneratively cooled thrustchamber section of SLME for different operation points obtained from RPA analyses

In the upper stage version of the SLME with expansion ratio 59 the nozzle extension beyond expansion ratio of 33 should be film and radiation cooled. Wall temperatures are expected to be sufficiently low and the main target is the limitation of the engine mass.

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.

2.6.2 Integrated Power Head

An Integrated Power Head (Pre-burner + Turbine + Impeller pump) as it has been used on the SSME (Figure 10) 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.

Figure 11 shows the integration of all major components in the upper section of the SLME and their integration with the combustion chamber injector head. This preliminary layout from end 2012 is maintained.
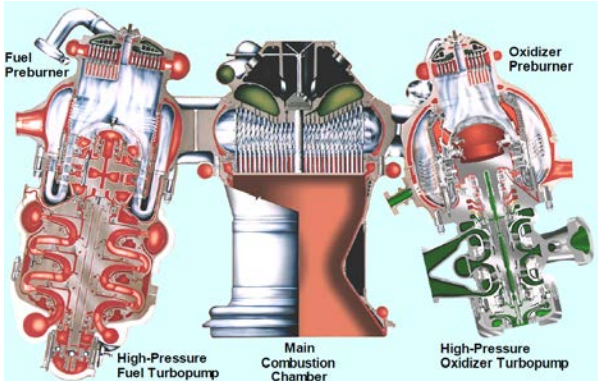

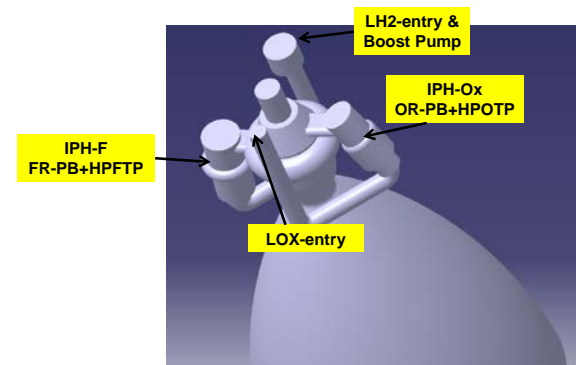Figure 10: Example of SSME Integrated Power Head assembly attached to combustion chamber [19]

Figure 11: SLME simplified CAD geometry showing arrangement of turbomachinery [9]

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

In [22] an FFSC derivative engine of the SSME is proposed which should operate as a highly variable mixture ratio engine, especially for booster applications. The probably most innovative element of this LOX-LH2 engine proposal is an operation mode with chamber mixture ratio of 9.0, beyond stoichiometric combustion. The advantage is a much higher propellant bulk density in the boost phase than usually achievable with LOX-LH2. The engine should be capable of later switching to MR= 6.0 in its sustainer phase. The motivation for this approach is found in avoiding the complications of LOX-hydrocarbon combustion while delivering better performance with increased density propellants. Despite the fact that such an engine is of high interest for an application like the SpaceLiner, the dual mode operation would require a transient crossing of the stoichiometric combustion region with high chamber temperatures. Oxygen-rich hot gas at the engine exhaust coming into contact with the external parts of the vehicle or with ground installations might be another concern. Therefore, the high MR-operation mode is currently not regarded as an option for the SLME.

Other elements of the proposed SSME "Derivative Engine" are of direct interest to be implemented in the SLME design. Figure 12 shows the Integrated Power Head with simplifications and increased robustness compared to the SSME power head (Figure 10).
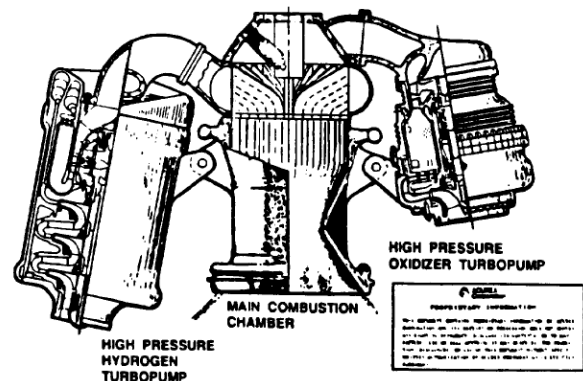

Figure 12: Proposed SSME "Derivative Engine" Integrated Power Head [22]

2.6.3 Preburners

The SLME preburners are attached to each turbopump in the integrated powerhead assembly as visible in Figure 11. 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 100 so that TET is restricted to acceptable values. At each turbine a bypass / tap-off is foreseen for which the flow should be controlled by a hot gas valve in order to allow engine operation in the full operational domain without significantly changing TET or excessively raising pre-burner 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 [5]. 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 [12].

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 section 2.2). 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 is planned. Figure 13 shows a similar design of the proposed SSME "Derivative Engine" but without inducer stage. 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.

2.6.4 Turbomachinery

On the fuel side a boost pump driven by an expander turbine fed from the regenerative circuit is feeding the HPFTP. HPFTP is a 2-stage Impeller pump powered

by a 2-stage reaction turbine. Fuel from the hydrogen feed system enters the LPFTP and is pressurized to 2.5 MPa. The gas pressurized by the LPFTP enters the HPFTP and its pressure is further raised to 32 MPa. The HPFTP turbine is driven by combustion gas from the fuel-rich preburner (FPB). The maximum FTP casing diameter is estimated at less than 500 mm.

On the LOX-side a conventional HPOTP with inducer and single stage impeller on the same shaft is proposed. A single stage turbine is probably sufficient to power the HPOTP. 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. Oxidizer flow from the LOX feed system enters at the Inducer into the OTP and is pressurized to 2.5 MPa. The complete flow then enters into the impeller and is pressurized to around 26 MPa. The OTP turbine

is driven by combustion gas from the oxidizer-rich preburner (OPB). The maximum OTP casing diameter is estimated at less than 350 mm.

More technical data on the preliminary definition of the turbomachinery are provided in references 6, 7 and 8.

2.6.5   Engine Controls

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.
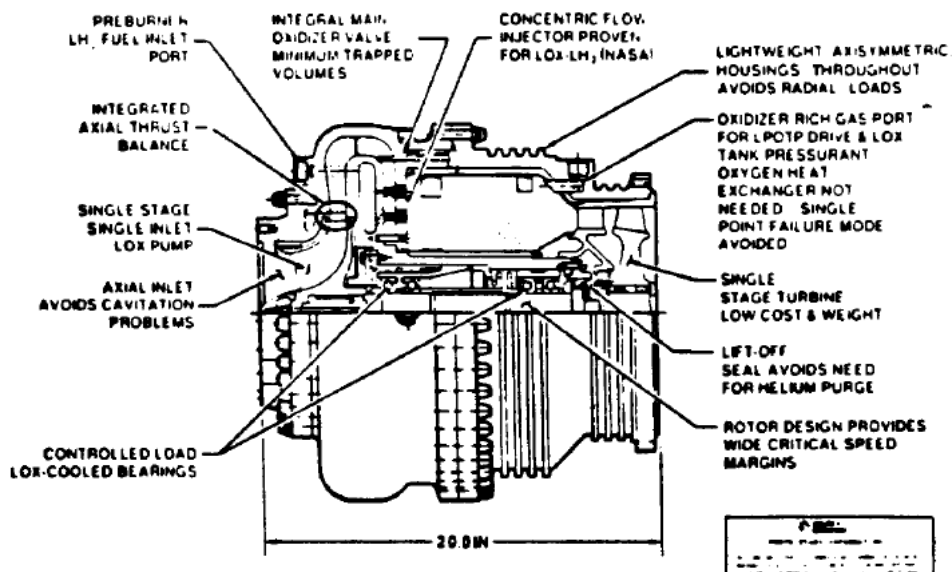

Figure 13: Proposed SSME "Derivative Engine" LOX-rich Integrated Power Head with single-stage HPOTP [22]
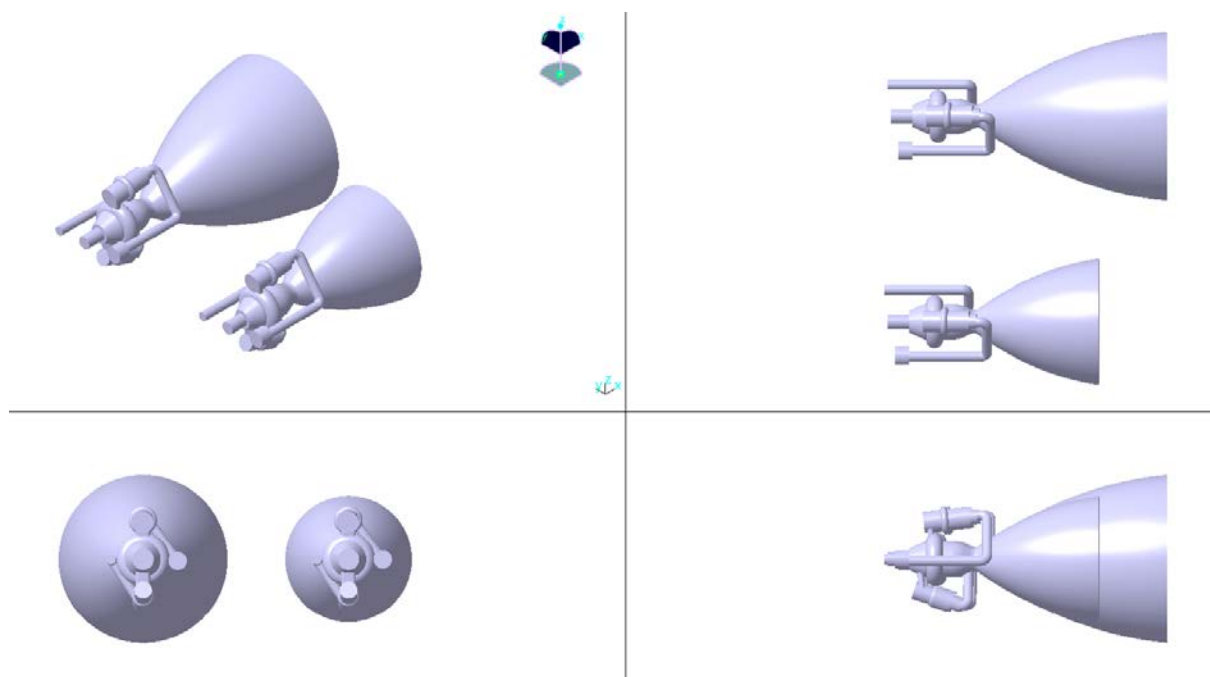

Figure 14: Size comparison of simplified CAD-shapes of SLME with ε=33 and ε=59

2.7 Engine Geometry and Mass

The size of the SLME in the smaller booster 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 and overall arrangement of the engine components is visible in Figure 14.

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.

3 ADVANCED ENGINE DEVELOPMENTS

Currently, some advanced, high-performance rocket engine developments are progressing; mainly in the US and at NPO Energomash in Russia. These activities provide technological benchmarks for the SLME development roadmap intending that the engine is well prepared for a competitive market environment. As the SpaceLiner operation is not to be expected before the late 2030s, the engine design goals should be ambitious, but on the other hand, need to take into account the current TRL in Europe.

An overview on the US booster engine activities has been provided in [23]. The main players in this field are Aerojet Rocketdyne's AR-1, Blue Origin's BE-4 and the SpaceX' Raptor engine. The latter is also designed as FFSC type while the others are staged combustion with oxygen-rich preburners. All engines use LOX-Hydrocarbon propellants, either kerosene or methane.

The Raptor is probably the most prominent one as it has been promoted by Elon Musk, "CEO and Lead Designer, SpaceX" [24, 25]. Originally Raptor was foreseen as an ultra-high performance LOX-LCH4 engine with 3290 kN of vacuum thrust and 30 MPa chamber pressure. An evaluation by DLR using the tools lrp 1.1 and RPA confirmed principal feasibility of the published 334 s (s/l) and 359 s (vac) Isp under the premise that the chamber pressure of 300 bar is achieved. Further assuming highly efficient turbomachinery, pump discharge pressures of the Raptor are expected to be in the range 64 to 68 MPa. Engine mass is estimated at 4200 kg with remarkable agreement between both rocket analysis tools. Thus, T_{vac}/W of Raptor would be close to 80.

Meanwhile, technical requirements of the first generation Raptor have been reduced. In his 2017 IAC speech Musk announced a first generation engine operating at 25 MPa chamber pressure and thrust level of 1700 kN to be included in the fully reusable "BFR" TSTO launcher [25, 26]. DLR recalculation of the smaller Raptor with similar assumptions in RPA-input including a nozzle expansion ratio of 40 indicates minor reduction in vacuum Isp and below 327 s sea-level Isp. Pump discharge pressures of this Raptor variant might be reduced to the still ambitious range of 48 to 52.5 MPa and estimated engine T_{vac}/W could reach beyond 92. Any detailed information for comparison to these calculated data is not yet made available in publications.

Blue Origin's BE-4 is another advanced rocket engine operating in oxygen-rich staged combustion cycle mode using LOX-methane propellants. Applications of this 2400 kN booster type engine are the company's New Glenn and potentially the first stage of the ULA Vulcan. Detailed technical information on the BE-4 is hardly available.

The MCC pressure has been announced to be relatively low, even below the SLME, at 13.44 MPa [27] and nozzle expansion ratio is approximately 28 according to a figure visible in [28]. Estimated Isp performance of the BE-4 based on DLR RPA calculations is below 310 s (s/l) and 350 s (vac).

4 CONCLUSION

The DLR proposed reusable winged rocket Space-Liner for very high-speed intercontinental passenger transport is in Phase A structured development after successful completion of its MRR. Assuming advanced but not exotic technologies, a vertically launched rocket powered two stage space vehicle is able to transport about 50 passengers over distances of up to 17000 km in about 1.5 hours or transport heavy payloads to LEO in a technically similar unmanned configuration.

A full-flow staged combustion cycle around a moderate 16 MPa chamber pressure has been selected for the SpaceLiner Main Engine (SLME). The engine operational domain is defined by numerical analyses and crosschecked by different tools. A 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 selected as the baseline.

This paper presents a preliminary definition of the architecture and size of major engine subsystems such as thrustchamber, preburners and turbomachinery (combined in an Integrated Power Head assembly) based on the reference cycle. 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

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