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Space Propulsion 2014, Session 30 - ST - Future Liquid Stages & Engines (3)

Staged Combustion Cycle Rocket Engine Subsystem Definition for Future Advanced Passenger Transport

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DLR's launcher systems analysis division is investigating since a couple of years a visionary, extremely fast passenger transportation concept based on rocket propulsion. Thanks to the multi-national collaboration, the technical lay-out of the SpaceLiner has now matured to Phase A conceptual design level.

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. The expansion ratios of the engines are adapted to their respective optimums required by the stages; while the mass flow, turbo-machinery, and combustion chamber are assumed to remain identical.

The paper describes the SpaceLiner 7 propulsion system:

- **The reference vehicle's preliminary design,**
- **Main propulsion system definition and architectural lay-out,**
- **Thrust chamber geometries,**
- **Pre-design of different turbomachinery and attached preburners,**
- **Advanced ceramic material fuel- and oxidizer-rich pre-burners and injectors as an alternative to increase lifetime of components.**

The presented work is including preliminary sizing on component level and first mass estimation data.

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
CMC	Ceramic Matrix Composite
FFSC	Full-Flow Staged Combustion
FRSC	Fuel-Rich Staged Combustion
FTP	Fuel Turbo Pump
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
MR	mixture ratio
NPSP	Net Positive Suction Pressure
OTP	Oxidizer Turbo Pump
SLME	SpaceLiner Main Engine
SSME	Space Shuttle Main Engine
TET	Turbine Entry Temperature
TRL	Technology Readiness Level
C	chamber
vac	vacuum

1 INTRODUCTION

A strategic vision of DLR which ultimately has the potential to enable sustainable low-cost space transportation to orbit is under technical evaluation since a couple of years. The number of launches per year should be strongly raised and hence manufacturing and operating cost of launcher hardware should dramatically shrink. The obvious challenge of the vision is to identify the very application creating this new, large-size market.

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 might be 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]. Ultra long-haul distances like Europe – Australia could be flown in 90 minutes. Another interesting intercontinental destination between Europe and North-West America could be reduced to flight times of slightly more than one hour.

Ultra-fast transportation far in excess of supersonic and even potential hypersonic airplanes is definitely a fundamental new application for launch vehicles. By no more than partially tapping the huge intercontinental travel and tourism market, production rates of RLVs and their rocket engines could increase hundredfold which is

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out of reach for all other known earth-orbit space transportation. The fast intercontinental travel space tourism, not only attracting the leisure market, would, as a byproduct, also enable to considerably reduce the cost of space transportation to orbit.

Figure 1: The SpaceLiner vision of a rocket-propelled intercontinental passenger transport could push spaceflight further than any other credible scenario

2 TECHNICAL OVERVIEW OF THE SPACELINER CONCEPT

2.1 Configuration Evolution

First proposed in 2005 [1], the SpaceLiner is under constant development and descriptions of some major updates have been published since then [2, 3, 4]. The European Union's 7th Research Framework Programme has supported several important aspects of multidisciplinary and multinational cooperation in the projects FAST20XX, CHATT [6], HIKARI and HYPMOCES.

Different configurations in terms of propellant combinations, staging, aerodynamic shapes, and structural architectures have been analyzed. A subsequent configuration numbering has been established for all those types investigated in sufficient level of detail. The genealogy of the different SpaceLiner versions is shown in Figure 2. The box is marking the configuration trade-offs performed in FAST20XX in 2009/10.

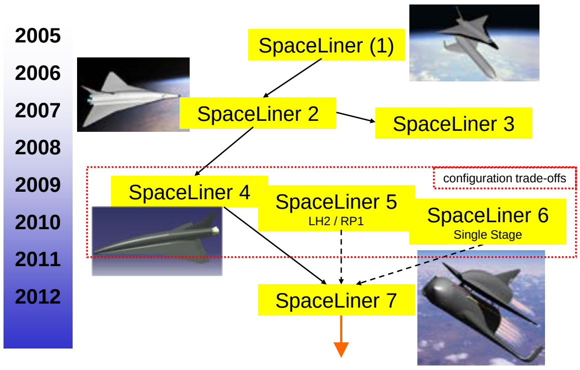


Figure 2: Evolution of the SpaceLiner concept

At the end of 2012 with conclusion of FAST20XX the SpaceLiner 7 reached a consolidated technical status.

2.2 Latest SpaceLiner 7 Configuration

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 (also called orbiter) 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 current arrangement of the two vehicles at lift-off is presented in Figure 3. Stage attachments are following a classical tripod design. The axial thrust of the booster is introduced through the forward attachment from booster intertank into the nose gear connection structure of the orbiter. The aft attachment takes all side and maneuvering loads. Major geometrical data of the SpaceLiner7 stages are provided in Table 1 and Table 2.

Total dry mass of the SpaceLiner 7 launch configuration is estimated at 310.9 Mg with a total propellant loading of 1520 Mg resulting in 1838.7 Mg GLOW incl. passengers & payload. Latest improvements of the SpaceLiner 7-3 configuration allow for a slight reduction in masses.

Table 1: Geometrical data of SpaceLiner 7-2 booster stage

length [m]	span [m]	height [m]	fuselage diameter [m]	wing leading edge angles [deg]	wing pitch angle [deg]	wing dihedral angle [deg]
83.5	36.0	8.7	8.6	82/61/43	3.5	0

Table 2: Geometrical data of SpaceLiner 7-1 passenger 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

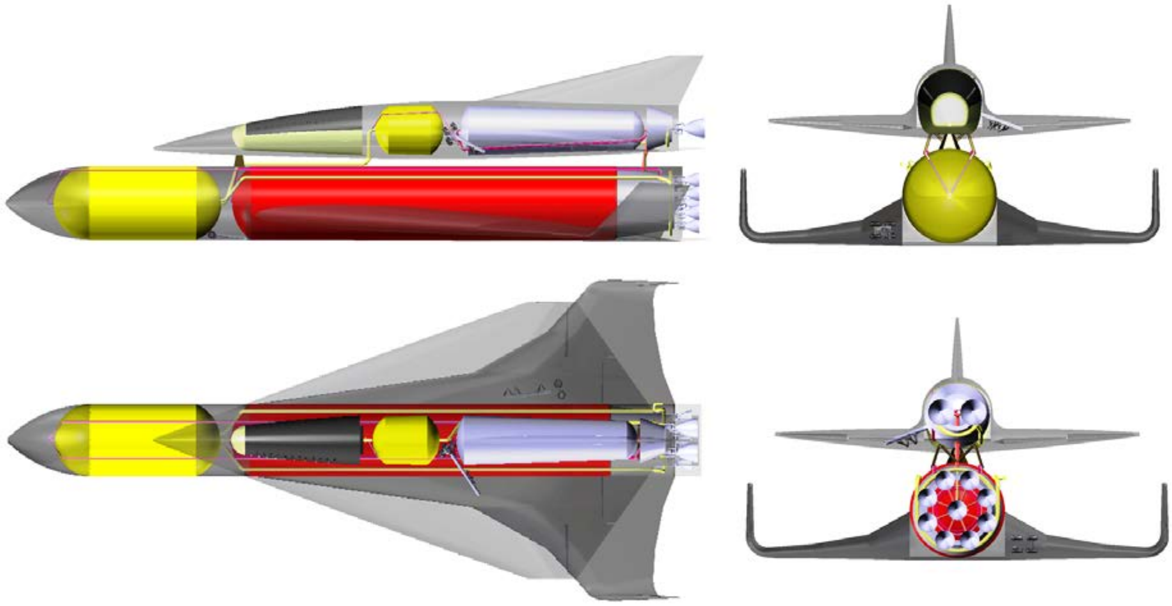


Figure 3: Sketch of SpaceLiner 7 launch configuration with passenger stage on top and booster stage at bottom position showing the SLME arrangement in the lower right figure

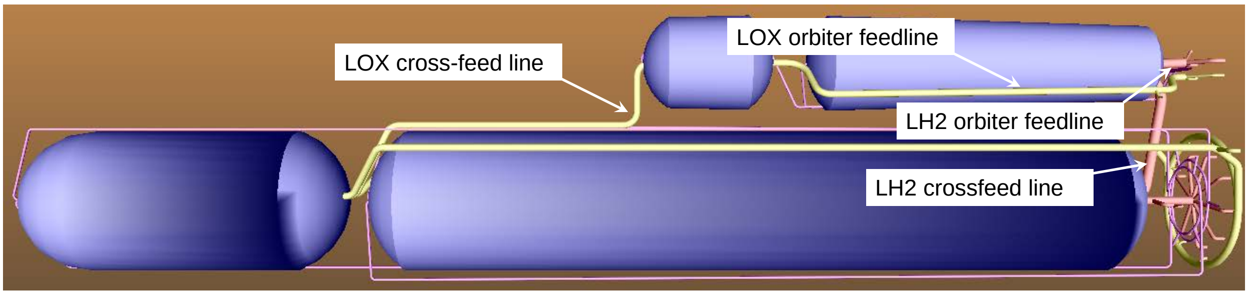


Figure 4: Arrangement of propellant tanks, feed- and pressurization system of SpaceLiner 7

3 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 data 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 assumed to remain identical in the baseline configuration.

3.1 Previous Engine Analyses

The best mixture ratio of the SpaceLiner main propulsion system along its 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) is 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, 7]. 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. This approach should allow avoiding the complexity and cost of additional inert gases like Helium for sealing.

3.2 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. No crossfeed system for a configuration like the SpaceLiner has ever been built and therefore investigations have been performed to determine how such a system could be implemented and how complexity issues can be addressed.

Three main options of crossfeed exist:

- Line-to-line
- Tank-to-tank
- Tank-to-buffer-tank

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

The propellant feed- and pressurization system is preliminarily designed using the DLR-tool pmp. A preliminary arrangement of feed- and pressurization lines with the tanks of both stages in the mated configuration is shown in Figure 4.

Figure 5 shows the interesting pressure history inside the passenger stage's LOX-feed system obtained by steady simulation. A tank to tank crossfeed from the booster LOX-tank, positioned more than 25 m forward, generates significant hydrostatic pressure, indirectly forcing ullage pressure in the upper stage tank up to more than 8 bar. Further downstream in the feedline pressure values can exceed 16 bar. The effects of throttling and staging are clearly visible in Figure 5. NPSP at the LOX-turbopump entry is generous and might allow for reducing the ullage pressure after staging.

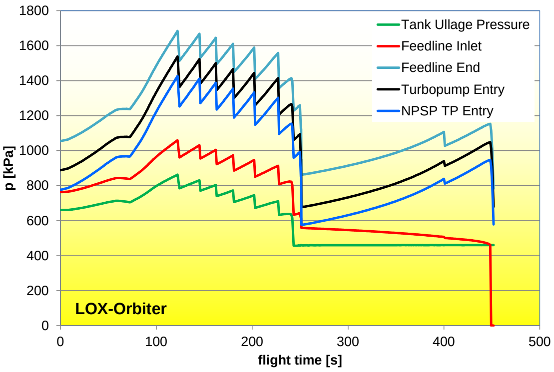


Figure 5: Pressure history at certain stations inside the orbiter LOX-tank feed system

After the steady state simulation also the transient behavior in the propellant feed-system has been analyzd along the powered flight and its preliminary design has been defined [10].

The LOX-tanks are pressurized by gaseous oxygen and the hydrogen tanks with gaseous hydrogen. This approach is selected in order to avoid any excessive use of expensive and rare helium. The fuel tank pressurization gas is supplied from the SLME after leaving the regenerative circuit while the oxidizer tank pressurization gas is bled from the oxidizer line behind the OTP discharge and then heated-up in a heat exchanger.

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. The booster LOX tank pressure can be limited to 2.1 bar because of its forward position always generating a lot of hydrostatic pressure down the line which is beneficial for good NPSP. Due to this fortunate situation, the required oxygen gas at stage booster MECO is below 3000 kg. The hydrogen gas mass inside the very large 2632 m^3 LH2-tank is no more than 1400 kg because of hydrogen's low molecular mass.

3.3 SLME design requirements

During the SpaceLiner 7 vehicle definition the need of approximately 30% more thrust than for the earlier engine described in [5] has been identified. This translates to a vacuum thrust of up to 2350 kN and sea-level thrust of 2100 kN for the booster engine and 2400 kN, 2000 kN respectively for the passenger stage. All these values are given at a mixture ratio of 6.5 with a nominal operational MR-range requirement from 6.5 to 5.5.

The SpaceLiner's ascent reference mission requirements define 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 should be 25 missions.
NPSP_min: 70 kPa for LH2-boost pump, 230 kPa for LOX-inducer pump

3.4 SLME Functional Architecture

A Full-Flow Staged Combustion Cycle (FFSC) with a fuel-rich preburner gas turbine driving the LH2-pump and an oxidizer-rich preburner gas turbine driving the LOX-pump is a preferred design solution for the SpaceLiner. The components and their connections are shown in Figure 6 for two design variants: Single FTP with inducer or FTP split into boost pump and HPFTP.

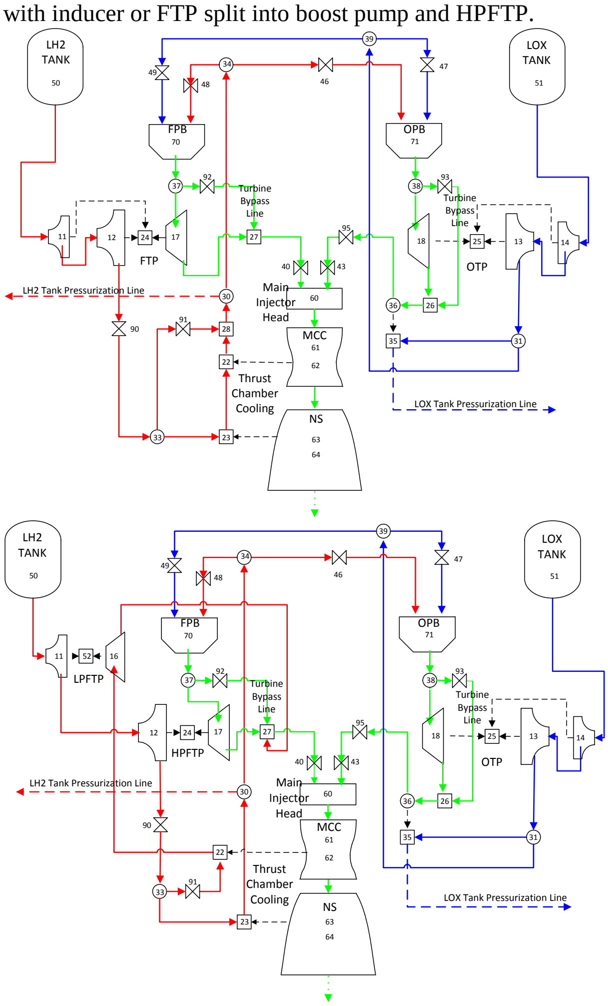


Figure 6: SpaceLiner Main Engine Schematics (Top: 1 FTP type, Bottom: 2 FTPs type)

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 work on 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 made with hydrogen fuel after being discharged by the FTP [5, 7].

3.5 Performance estimation by steady-state analyses

The program used for the updated cycle analysis, lrp2, is based on the modular program SEQ [12] 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.

An engine cycle analysis for two different FTP designs as shown in Figure 6 has been traded. The first one has two separate FTPs, a boost pump followed by high pressure fuel turbo pump (HPFTP), and the other one is of single shaft FTP type. The former is a more complicated engine system than the latter, but the FTP discharge pressure can be kept at lower values. This can be explained by the fact that for the separate FTPs type the significant pressure loss of 6 - 8 MPa in the regenerative cooling of the combustion chamber is not influencing the entry conditions of the preburner. The necessary FTP discharge pressure in case of the single FTP type is 45.5 MPa, while for the two separate FTP type it is only 37.3 MPa [7, 9]. The two FTPs type with separate boost pump is selected as the baseline.

Table 3 gives an overview about major SLME engine operation data in the nominal MR-range as obtained by cycle analyses. These data, based on the preliminary assumption of constant 70% efficiency for all turbomachinery, are used for the preliminary sizing of the major SLME subcomponents. Slight deviations of the internal conditions between Booster and Passenger Stage SLME are due to the numerical iteration.

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

	Booster			Passenger Stage		
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]	29.4	30.0	30.8	29.5	30.2	31.0
Oxidizer-rich Preburner pressure [MPa]	29.1	29.7	30.5	29.2	29.9	30.7
Fuel-rich Preburner TET [K]	732	735	738	720	722	724
Oxidizer-rich Preburner TET [K]	773	775	778	772	774	777
HPFTP discharge pressure [MPa]	36.5	37.3	38.3	36.7	37.5	38.5
OTP discharge pressure [MPa]	38.1	40.2	42.4	38.5	40.7	42.9
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

3.6 Preliminary subcomponent sizing

An Integrated Power Head (Pre-burner + Turbine + Impeller pump) as it has been used on the SSME (Figure 7) 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 8 shows the integration of all major components in the upper section of the engine and their integration with the combustion chamber injector head.

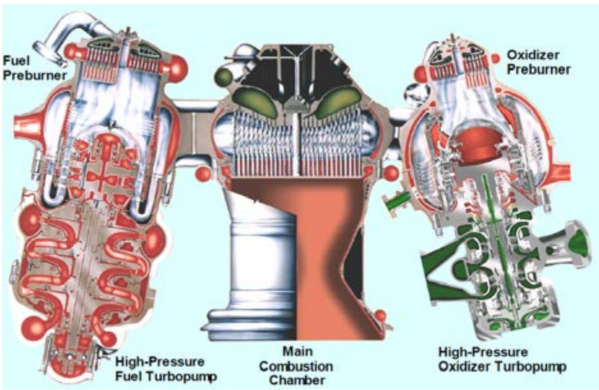


Figure 7: Example of SSME Integrated Power Head assembly attached to combustion chamber [13]

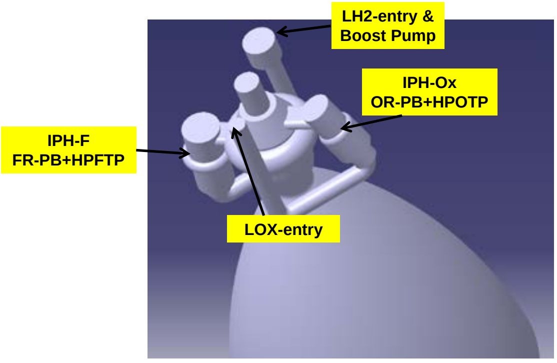


Figure 8: SLME simplified CAD geometry showing arrangement of turbomachinery

3.6.1 Thrustchamber

The geometry of the thrustchamber including chamber and nozzle is 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 has not only 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 are plotted in Figure 9.

In the booster engine H2 regenerative cooling and film cooling are combined for the thrust chamber. H2 regenerative cooling is used for the complete thrust chamber surface with two separate passes. One pass chills the chamber including the throat area and the other pass chills the nozzle area. Fuel for film cooling is supplied from the side of the injector plate chilling the chamber wall. At the orbiter engine the nozzle extension beyond expansion ratio of 33 should be radiation cooled.

The effects of the thrustchamber boundary layers and the different cooling methods are calculated with the NASA code TDK. The boundary layer's effect on I_{sp} is 3 - 4 s. The heat transfer rate into the regenerative cooling circuit is estimated, depending on the MR and hence chamber pressure, between 40 and 50 MW.

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.

3.6.2 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 37 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.

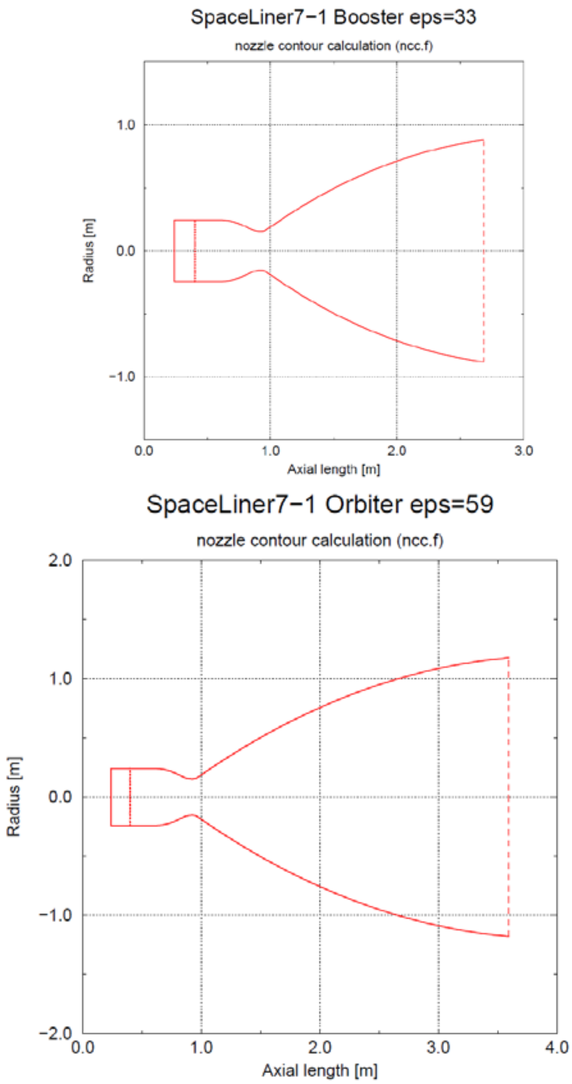


Figure 9: Internal thrustchamber contours of SpaceLiner 7 main engine (ε=59 bottom, ε=33 top)

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 7 MPa. The complete flow then enters into the impeller and is pressurized to 40 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 7, 8 and 9.

3.6.3 Preburners

Two Pre-Burners are attached to each turbo-pump similar to the SSME design (Figure 7), so that they should become compact and light. The mixture ratios of

FPB and OPB are controlled to be 0.7 and 130 so that TET is restricted to around 770 K. At each turbine a bypass line is foreseen for which the flow should be controlled by a hot gas valve in order to allow engine operation in the mixture ratio range from 5.5 to 6.5 without 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].

In case of coaxial injector elements, 60 elements are assumed for the FPB and 50 elements for the OPB.

3.7 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 10.

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.

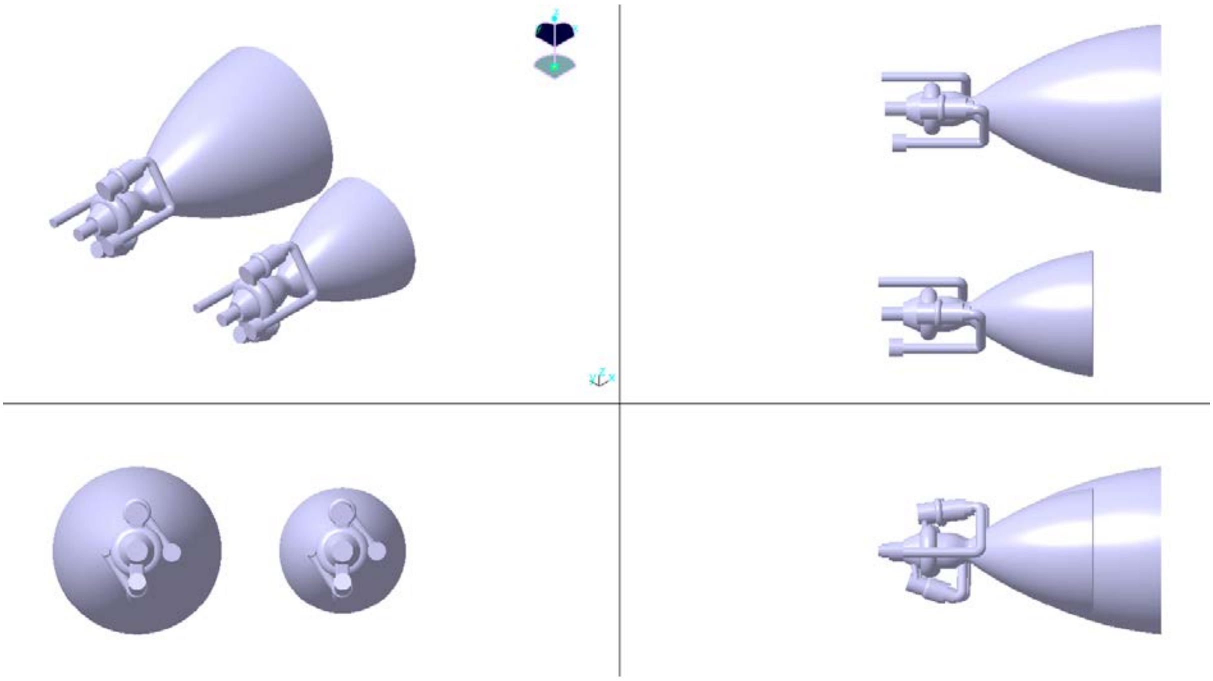
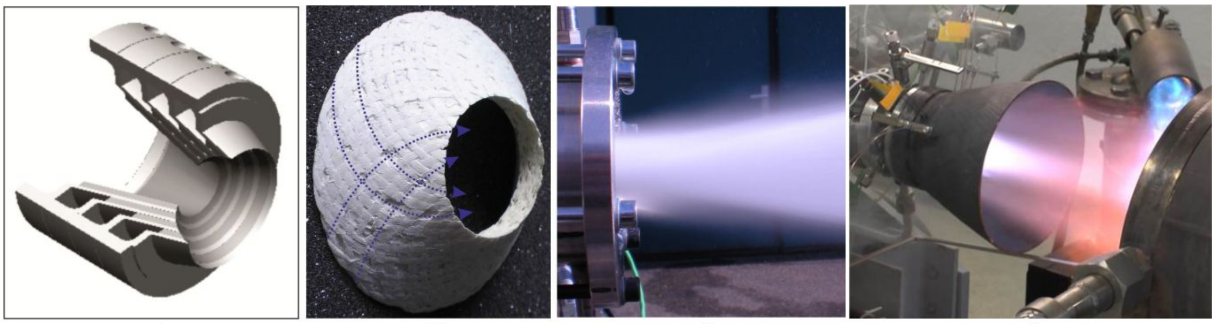


Figure 10: Size comparison of simplified CAD-shapes of SLME with ε=33 and ε=59



a) b) c) d)

Figure 11: Porous CMC injector technology. Design principle (a), highly permeable oxide CMC element (b), flow check (c), 30 bar hot-run of an integrated CMC thrust chamber assembly including the ceramic injector (d)

3.8 Alternative CMC Preburner Technology

Since more than a decade DLR invested in the development of innovative high performance rocket thrust chambers based on transpiration cooled CMC liners. The key technology consists in the structurally de-coupled stack of inner CMC rings combined with a low strain and light weight CFRP housing (Figure 12). Combined with a light weight CFRP housing a widely load-de-coupled thrust chamber structure assembly could be developed, promising substantial innovation concerning higher reliability, longer lifetime, cost and weight reduction.

Significant progress could be achieved by the technology feasibility demonstration in tests up to combustion chamber pressures of 90 bar and hot run durations up to 120 seconds at DLR's technology test benches P8 and P6.1 [14, 15]. LOX / LH2 or LOX / GH2 (100 K) propellant combinations have been used. Damage free operation with 50 mm cylindrical CMC element subscale demonstrators could be proven with overall coolant mass flow ratios of 7% in 60 bar tests [16, 17, 18].

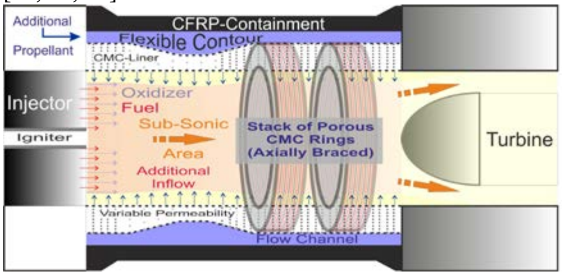


Figure 12: Design principle of a potential ox-rich CMC preburner

Apart from standard coaxial injectors an innovative CMC injector concept (Figure 11, a) has been proposed recently with the goal of high thermo-mechanical compatibility, throttling capability, improved spray forming and the capability of improved hot as well as deep cold injection behavior. High injector flow rates (up to 900 g/s with water and GN2) using highly permeable CMC injector elements could be verified in initial cold flow tests through a 30 mm face-plate (Figure 11, b and c). This injector concept has proven its principal functionality in the latest test campaign end of 2013 in 30 bar firing tests at the P6.1 test bench (Figure 11, d). In these experiments with a conical injector 250 g/s LOX injection were applied.

The CMC permeability can be adjusted both by the application of definable inherent material porosity and specifically implemented channel patterns, which can be created in-situ during the material manufacturing process. Consequently the injection mass flow can be spread to a wide range of mass flow amount, only by defining specific material parameters. Complex manufacturing of multiple metallic injector elements would no longer be necessary.

Apart from main chamber applications this ceramic composite structure approach is also suited for the preburners of a staged combustion cycle engine. The advantage of homogeneous propellant injection through the porous wall component promises system improvements and simplifications (Figure 12). A flexible selection of contour shapes is possible for the optimization of the coolant flow through the ceramic liner.

For ox-rich preburners highly porous oxide CMC derivatives are suitable as oxygen-inert inner liner materials. The permeability can be given by typical Darcy-Forchheimer coefficients k_d and k_f. A favorable oxide material candidate is the WPS material [19, 20, 21] (Figure 13, top). Its natural open porosity ranges at about 30 %. The measured k_d ($\approx 20 \cdot 10^{-14}$) and k_f ($\approx 4 \cdot 10^{-7}$) values allow for high permeability.

In case the permeability has to be further increased, the implementation of additional fine channel patterns at the injector material can be applied as the preferred manufacturing procedure (Figure 13, bottom).

Figure 13: WPS oxide CMC ring segment of a 50 mm model combustor (top), additionally implemented injector channel patterns (bottom)

Considering the low material densities of CMCs (2 ÷ 3 g/cm^3) and CFRP (1.8 g/cm^3) a weight reduction potential compared to continuous metallic systems seems to be not unrealistic. The exact mass saving potential depends on which structural parts (excluding the combustion chamber itself) the application of fiber reinforced materials is of functional advantage. The low thermal expansion behavior of ceramics promises higher lifetime and reliability. De-coupled structural manufacturing could allow for significant cost reduction potential.

3.9 Updated Cycle Analyses with Refined Database

The analyses in section 3.5 were performed under the tentative assumption that all turbo pump efficiencies are fixed at 70%. Once turbo pump design parameters are gained by the preliminary turbomachinery analyses of section 3.6.2, new turbo pump efficiencies based on the design parameters can be estimated using empirical data (e.g. from [11], [22]). These efficiencies obtained from graphs collecting 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.

The analyses in [9] show that SLME turbopump efficiencies have all the potential to be improved beyond the previously assumed 70%. Therefore, the necessary turbo pump powers would all decrease and that would also reduce the necessary pump discharge pressures and hence preburner combustion pressure. Further, the TET could be reduced too. Those changes in internal thermodynamic conditions are preferable for improved engine life time. As long as the defined main combustion chamber conditions can be reached, the turbomachinery efficiencies in a staged combustion cycle are not impacting the overall engine performance.

An example of the SLME internal conditions under the assumption of ambitious but in principle feasible turbomachinery efficiencies is presented for the nominal MR of the booster engine in Figure 14.

Further, the analyses in section 3.5 were performed under a preliminary assumption on the flow rate for tank pressurization. The study on the SpaceLiner tank pressurization system has since progressed, and the necessary flow rates have consolidated. Therefore, cycle analyses with these new flow rates have been performed. As the necessary total flow rate increases slightly, the necessary power of each turbo pump also increases a little bit. However, the overall influence on the SLME parameters is small [9].

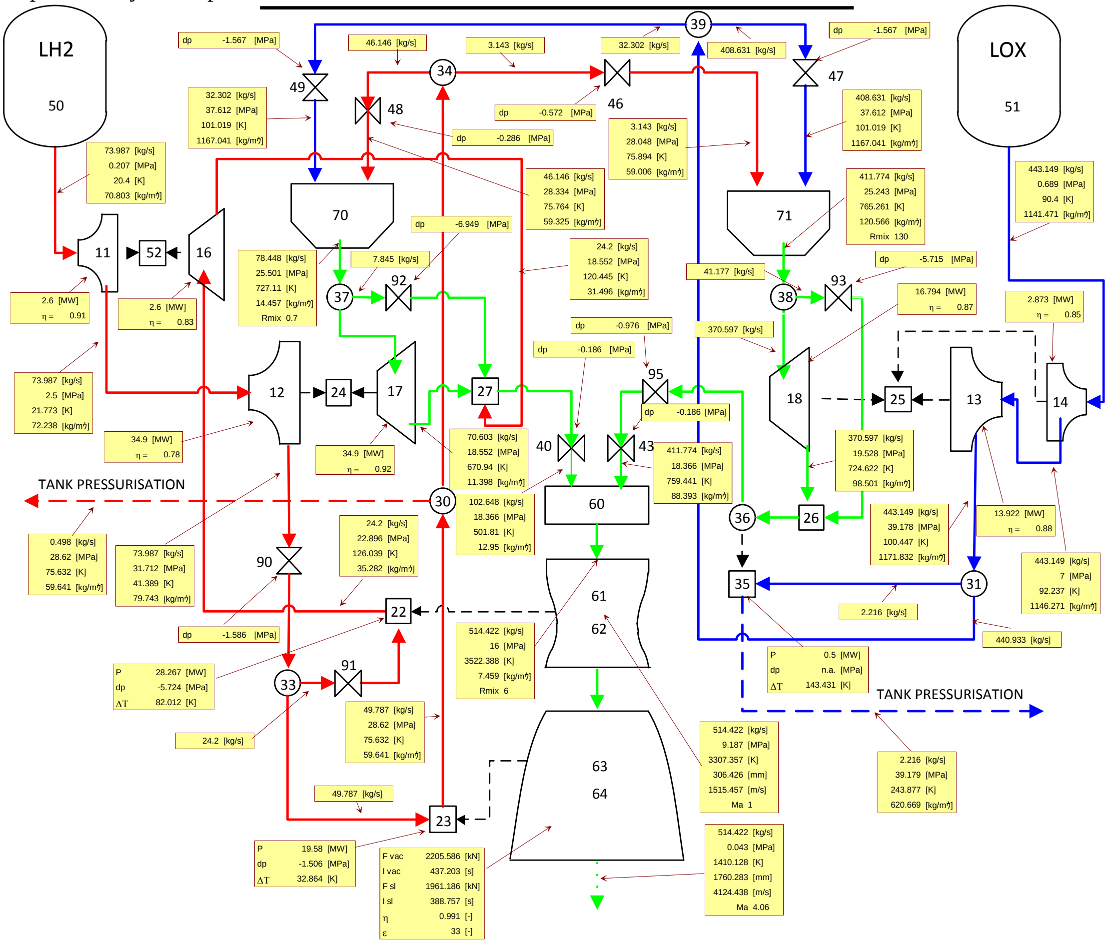


Figure 14: Calculated internal engine conditions with updated turbopump efficiencies (ε=33, MR=6.0)

4 CONCLUSION

The DLR proposed reusable winged rocket SpaceLiner for very high-speed intercontinental passenger transport is constantly maturing in its conceptual design. 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.

A full-flow staged combustion cycle around a moderate 16 MPa chamber pressure has been selected for the SpaceLiner Main Engine (SLME). The engine mixture ratio is variable in the range 5.5 to 6.5 along the ascent flight and hence able to boost average Isp-performance. Different subcycles have been investigated and 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.

The paper presents a preliminary definition of the architecture and size of major engine subsystems like thrustchamber, preburners and turbomachinery (combined in an Integrated Power Head assembly) based on the reference cycle. The preliminary turbopump sizing allows for an empirically based estimation of turbomachinery efficiencies. Analyses show that the SLME turbopump efficiencies have all the potential to be improved beyond the previously assumed 70%. This would result in changes of internal thermodynamic conditions, preferable for improved engine life time.

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

An innovative ceramic material preburner alternative is explained and related technology development work at DLR is outlined.

5 ACKNOWLEDGEMENTS

Part of this work was performed within the 'Future High-Altitude High-Speed Transport 20XX' project investigating high-speed transport. FAST20XX, co-ordinated by ESA-ESTEC, is supported by the EU within the 7th Framework Programme Theme7 Transport, Contract no.: ACP8-GA-2009-233816.
Further information on FAST20XX can be found on http://www.esa.int/fast20xx.

Part of this work was performed within the '**C**ryogenic **H**ypersonic **A**dvanced **T**ank **T**echnologies' project investigating tank technologies for high-speed transport. CHATT, coordinated by DLR-SART, is supported by the EU within the 7th Framework Programme Theme 7 Transport, Contract no.: ACP1-GA-2011-285117. Further information on CHATT can be found on http://www.chatt.aero

The authors gratefully acknowledge the contributions of Ms. Carola Bauer, Ms. Carina Ludwig, Ms. Mona Carlsen, Mr. Ryoma Yamashiro, Mr. David Gerson, and Mr. Francesco Cremaschi to the preliminary design of the SpaceLiner propulsion system.

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