Staged Combustion Cycle Rocket Engine Design Trade-Offs for Future Advanced Passenger Transport

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Staged combustion cycle rocket engines with a moderate nominal 16 MPa chamber pressure have been selected as the baseline propulsion system for the visionary intercontinental passenger transport SpaceLiner.

Several technical engine design trade-offs are run by numerical simulations and results are presented including:
- **Fuel rich vs. Full-flow cycle**
- **Useful operational domain in MR**
- **Regenerative cooling options of thrust chamber**

The engine operational domain is evaluated on launcher system level. The applicability of current European experimental staged combustion technology development (e.g. SCORE-D) on the SpaceLiner propulsion development is assessed.

Subscripts, Abbreviations

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	-

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
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

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, 2]. 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 about one hour.

The functionality of rocket propulsion is a proven technology since decades and their performance characteristics are well known. Furthermore, a rocket powered RLV-concept like the SpaceLiner is highly attractive because the flight durations are two to three times lower than those of even the most advanced air-breathing systems. It is to be recognized that travel times of any airliner are not identical to flight times. Additional times for commuting to the airport hub which offers the long-distance flight, check-in, security-check and those to be accounted for waiting and transfer are to be considered. A preliminary estimation of the expected travel time for a SpaceLiner passenger shows approximately 5 to 6 hours for ultra-long distances. This result corresponds to a reduction in the actual time needed for travelling between at least 75 % and 80 % compared to conventional subsonic airliner operation.

The general baseline design concept consists of a fully reusable booster and a separate passenger stage (or "orbiter") arranged in parallel. All engines (up to 9 on the booster and 2 on the orbiter) should work from lift-off until MECO. A propellant crossfeed from the booster to the orbiter is foreseen up to separation to reduce the overall size of the orbiter stage.

Figure 1: The SpaceLiner vision of a rocket-propelled intercontinental passenger transport, shown here in an early artist's impression of the new configuration 7, could push spaceflight further than any other credible scenario (© iDS, HAW Hamburg, 2011)

The environmental impact of the LOX-LH2 propelled SpaceLiner is relatively benign and seems to be much less critical than for airbreathing concepts. The rocket concept is releasing even less exhaust gases into the atmosphere than today's commercial airliners because the engines do not burn the air. Most of the flight trajectory is at a much higher altitude than for the airbreathing vehicles considerably reducing the noise impact on ground. Nevertheless, the launch has to most likely be performed off-shore or in remote, unpopulated areas due to expected noise at lift-off. Consequently decoupling of the launch and landing site will create some logistical challenges.

Staged combustion cycle rocket engines with a moderate 16 MPa chamber pressure have been selected as the baseline propulsion system. The engine performance 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.

Investigations of the SpaceLiner concept are mainly supported by internal DLR research funding. The EU-funded FAST20XX (Future high-Altitude high-Speed Transport 20XX) project allows for the inclusion of other European partners and also supports some DLR work. The multinational collaborative research project FAST20XX aims at providing a sound technological foundation for the industrial introduction of advanced high-altitude high-speed transportation in the medium term and in the longer term (SpaceLiner application).

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.

These configuration studies support the definition of the next reference configuration dubbed "SpaceLiner7". The interim research configurations 3, 4, 5, and 6 have been iteratively sized with careful scaling of the reference mass break-down, preliminary aerodynamic sizing and trajectory optimization. An overview on these configurations can be found in [1].

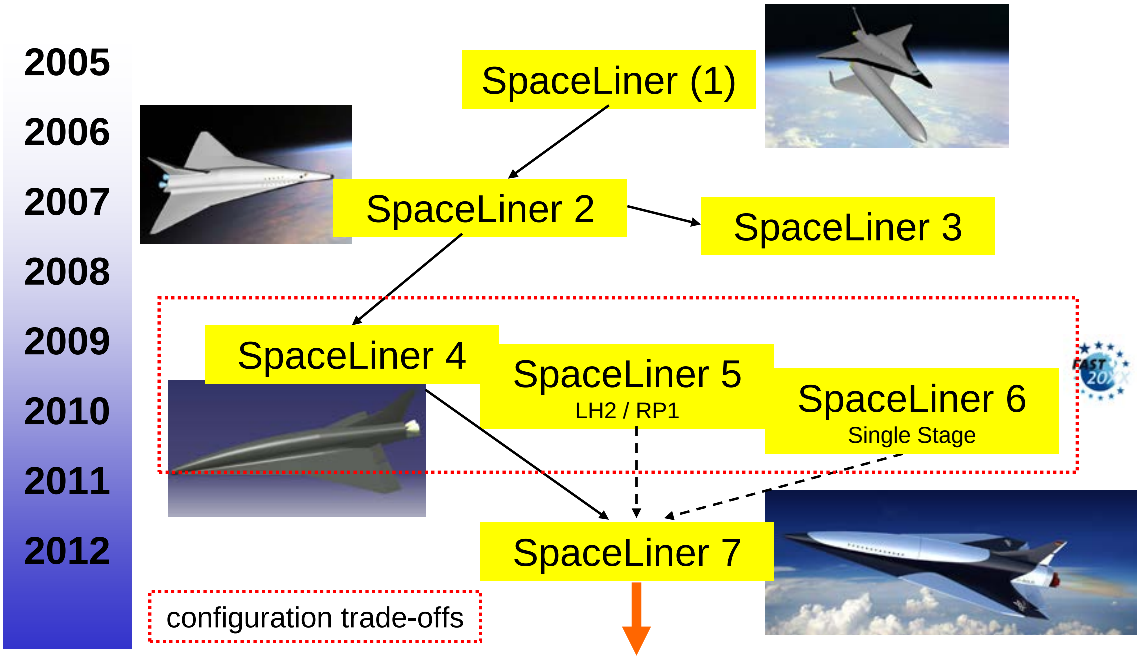

Figure 2: Evolution of the SpaceLiner concept

2 Preliminary definition of engine performance

Fuel rich staged combustion cycle engines with a moderate chamber pressure, approximately 1700 kN thrust in vacuum were selected for the propulsion system of the two SpaceLiner stages already in the early designs [3]. These engine performance 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.

All engines should work from lift-off until MECO. A propellant crossfeed from the booster to the orbiter is foreseen up to separation to reduce the overall size of the orbiter stage. The expansion ratios of the booster and orbiter engines have been adapted to their respective optimums starting with SpaceLiner version 2, while mass flow, turbo-machinery, and combustion chamber remain identical. These engine characteristics are listed in Table 1.

Table 1: Engine data of SpaceLiner versions 2, 3, 4, (5 and 6 only Orbiter)

	Booster	Orbiter
Mixture ratio [-]	6:1	6:1
Chamber pressure [MPa]	16	16
Mass flow per engine [kg/s]	384.5	384.5
Expansion ratio [-]	33	59
Specific impulse in vacuum [s]	437.6	448
Specific impulse at sea level [s]	388.4	360.4
Thrust in vacuum per engine [kN]	1650.6	1689.8
Thrust at sea level per engine[kN]	1465.0	1359.4

The more detailed subsystem definition and lay-out of the latest SpaceLiner 7 configuration required also a reinvestigation of the main propulsion system.

2.1 Definition of nominal engine operation mixture ratio range

A mixture ratio of 6 is a typical selection in a high performance LOX-LH2 rocket engine and has already been used for the SSME. Thus, an MR of 6 has been a good starting value used for all the SpaceLiner variants up to SL6. However, the optimum engine mixture ratio is always mission-dependent. Further, adaptation of the MR during flight might improve performance with better I_{sp} and improved thrust levels.

The reference SpaceLiner 2/4 engine (Table 1) has been investigated by simplified cycle analyses in its notional operational domain wrt. the variation of:

- Engine Mixture Ratio (MR) in the range 5 to 7
- Corresponding chamber pressure p_C in the range 14.5 MPa to 17.5 MPa assuming almost constant hydrogen mass flow for all engine MR values

Other parameters of the SpaceLiner engine have been assumed during these cycle analyses to be remaining constant, notably:

- thrust chamber geometries
- nozzle expansion ratios
- pre-burner MRs and internal line pressure losses
- turbomachinery and combustion efficiencies

Figure 3 shows the impact of the MR variation on the engine performance based on a simple rocket engine cycle model of a full-flow staged combustion engine. The booster engines with expansion ratio of 33 are represented by orange color while the orbiter engines with increased expansion of 59 are shown in blue. The optimum vacuum I_{sp} is found for both engines at its lower boundary of 5.5 and maximum sea-level I_{sp} is at 6.5 or higher. The engine's thrust is almost linearly increasing with MR due to increased p_C and hence mass flow. The decreasing slope from 6.75 to 7 is caused by a limitation in maximum chamber pressure.

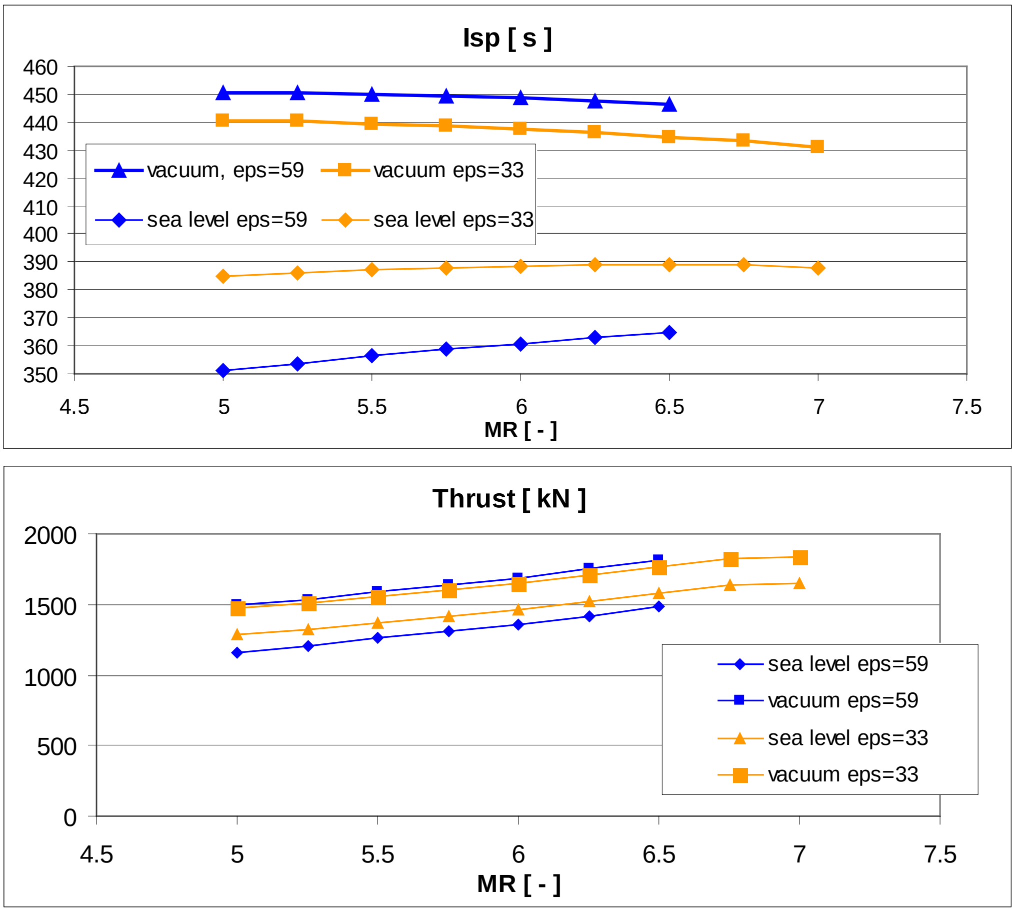

Figure 3: Preliminary engine data of the full investigated MR range (I_{sp}, top, thrust bottom)

Figure 4 demonstrates the impact of MR variation on the I_{sp} difference compared to the reference point of 6.0. As to be expected, at lower mixture ratio the vacuum I_{sp} is increasing as is sea-level I_{sp} with increased MR. The sensitivity is stronger for the larger nozzles of the orbiter engines with a maximum difference of 14 s in sea-level operation. The corresponding thrust level is changing by up to more than 350 kN per engine: 20% of nominal thrust. Significant performance improvements of the SpaceLiner configuration are possible if engine operating points are switched at the right flight condition.

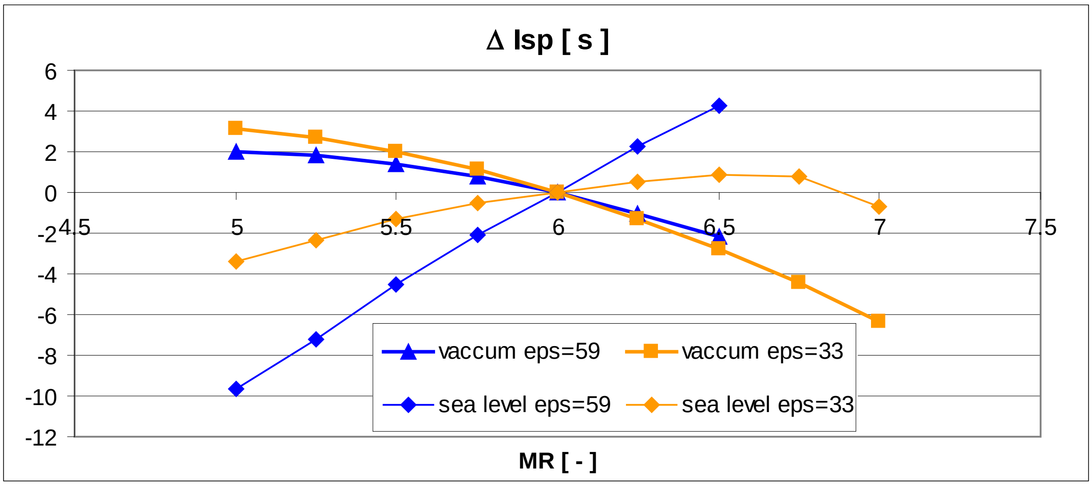

Figure 4: Potential Isp impact of MR variation

However, also a non negligible impact of MR variations on the required turbomachinery power has to be stated, if p_C is also changing as for the SpaceLiner engine. The minimum to maximum power range is found by the cycle analyses at +36% for LH2 and +90% for the LOX turbopumps. Therefore, for further analyses MR has been limited in the range 5.5 to 6.5 to keep subcomponent (e.g. turbomachinery, pre-burner) demands within reasonable boundaries. This choice is a consequence of the ambitious engine reusability requirements of the SpaceLiner.

Engine analyses as presented in Figure 4 already show that it is beneficial running the engines at lift-off with a higher mixture ratio and switching later into flight to the lower mixture ratio. The MR-switch would also serve as a first step in throttling when it is anyhow required to keep acceleration levels for the passengers at a comfortable maximum of 2.5 g. However, the best rate at which the engine oxidizer and fuel mass flows are to be adapted for an optimum SpaceLiner ascent are dependent on the vehicle's acceleration and climb through the atmosphere. Therefore, they can't be calculated by cycle analyses alone but require a trajectory optimization.

The best mixture ratio of the SpaceLiner main propulsion system along its mission has been defined by system analyses optimizing the full nominal trajectory. A trajectory optimization under the consideration of all relevant mission constraints and objectives is performed for the SpaceLiner4 using the *AeroSpace Trajectory Optimisation Software* ASTOS [4]. The same SpaceLiner4 (configuration 2010 [1]) data (booster and orbiter mass and aerodynamics models) have been inserted in the ASTOS Model Browser as these have already been used in a previous ASTOS trajectory which established a new hypersonic gliding reference mission [2]. The main propulsion data are, however, replaced by thrust and I_{sp} as function of MR. The MR is then used as an optimization control by the solver. Several constraints are present in the mission which is composed of different phases: vertical takeoff, pitch over, gravity turn during the combined booster and orbiter burn, optimized pitch and yaw control during the orbiter only burn following separation, deceleration glide ("cruise") phase till 40 km altitude, and final descent till ground.

Several MR-optimization options have been investigated with ASTOS always aiming for a minimization of (booster) propellant mass. Nominal engine MR control at two engine operation points (6.5 from lift-off until reaching the 2.5 g acceleration and 5.5 afterwards) with relatively short transients in between is found most promising. Figure 5 compares the actual calculated engine I_{sp} at each trajectory point for the reference configuration (red lines) with the MR optimized I_{sp} (black lines). The booster engines are represented by continuous lines, whereas the orbiter engine by dashed lines. As already shown in Figure 4, the increased mixture ratio enables a better low-altitude I_{sp} than the reference MR=6 case. Almost exactly at the trajectory point when the I_{sp} curve for adaptable MR would dive below the reference I_{sp} the optimizer switches to lower mixture ratio and hence better vacuum I_{sp}. (In Figure 5 visible approximately at 80 s to 90 s after lift-off; depending on the stage.) This approach allows for a significant propellant saving on the booster of 38 Mg, a reduction of 5% compared to the reference configuration without MR adaptation. That result is readily understandable because the specific impulse during the mission is superior by a few seconds to the reference case with fixed engine mixture.

After completion of the trajectory optimization, the obtained mission average tank MR in orbiter and booster remains very close to 6.0. It is interesting to note that after booster separation the orbiter engines in the best optimized case switch back to high MR (high thrust) to switch again after 100 seconds to low MR (high vacuum I_{sp}). Therefore, the orbiter tank MR remains closely around 6, allowing a more compact stage design than lower tank MR and still having the vehicle's best performance.

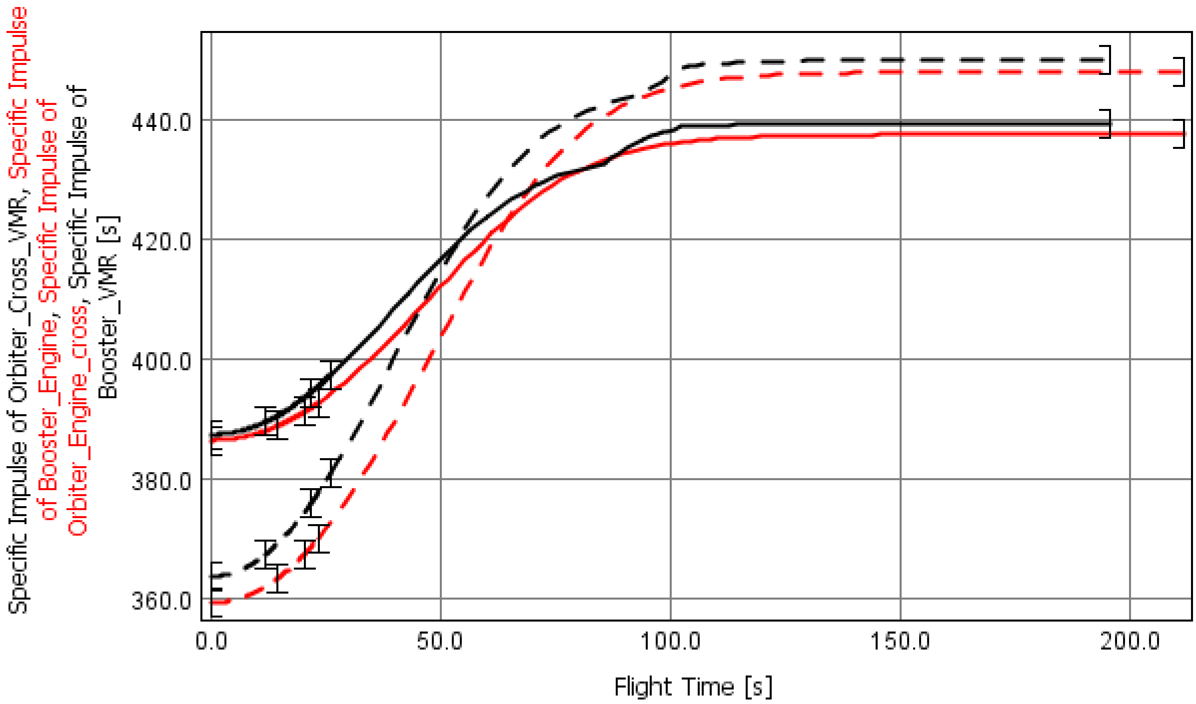

Figure 5: Actual engine I_{sp} along SpaceLiner4 joint booster-orbiter ascent in case of fixed (red) and adaptive (black) MR

2.2 Pre-definition of nominal engine operational domain

Based on the previously described results of the mixture ratio variation and its impact on vehicle performance, a preliminary operational domain of the SpaceLiner main propulsion system can be established (Figure 6). These data are used as guidelines for the more sophisticated cycle analyses of the following section 3.

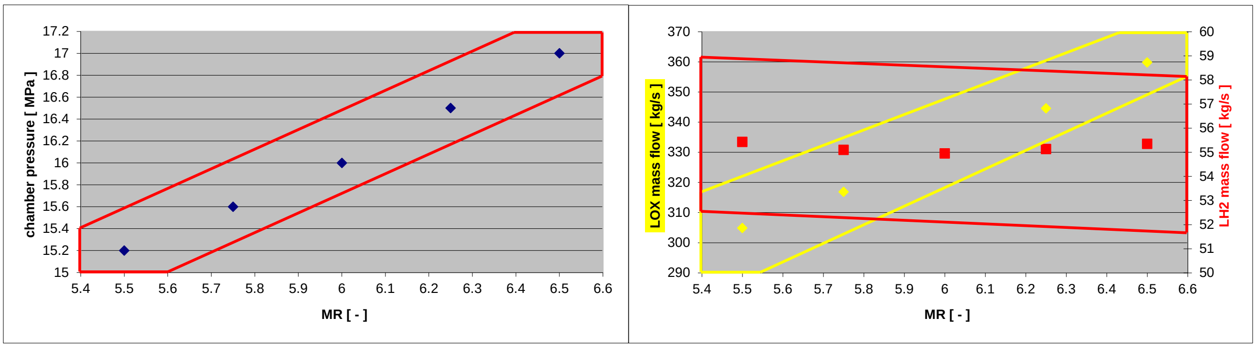

Figure 6: Preliminary operational domain of SpaceLiner main engine

3 Cycle analyses of SpaceLiner Main Engine (SLME)

3.1 Cycle Analysis Tool and Boundary Assumptions

The program used for the improved cycle analysis, lrp2, is based on the modular program SEQ [5] of the German Aerospace Center (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 [6].

The following dependencies have been selected for the cycles analyzed in this paper: The chamber pressure governs the pump discharge pressure, the pump power determines the turbine power and thereby the required turbine pressure drop as TET and pre-burner MR are fixed. The selected chamber mixture ratio of the staged combustion cycles also represents the engine mixture ratio. The engine cycle analysis of SLME consists of two steps as to the required engine condition. First, the chamber pressure 16 MPa and the engine mixture ratio 6.0 are set as the nominal engine condition, and the basic design point of each component in SLME is calculated by lrp2. Second, as some parameters (i.e. throat radius) being fixed, the mass flow rates of some lines (the engine feed lines, the turbine bypass lines and the preburner feed lines) are adjusted to make required chamber conditions.

Nozzle performance and heat exchange rate in the regenerative cooling part are calculated using TDK [7] with the nozzle contour preliminarily defined for the Spaceliner2 with the DLR tool ncc. In order to achieve a good performance, the supersonic section of the nozzle is different for both expansion ratios (Figure 7) while the combustion chamber and injector head should be similar. In case of the booster engine with ε=33 regenerative cooling of the complete nozzle is assumed. The exact cooling concept of the larger orbiter nozzle will be subject to future trade-studies. Approximately 40 MW heat will be transferred in nominal conditions to the hydrogen flow assuming an SSME-like internal film cooling. The flow resistance of each component is set using examples from already existing designs of SSME [8] and LE-7A [9].

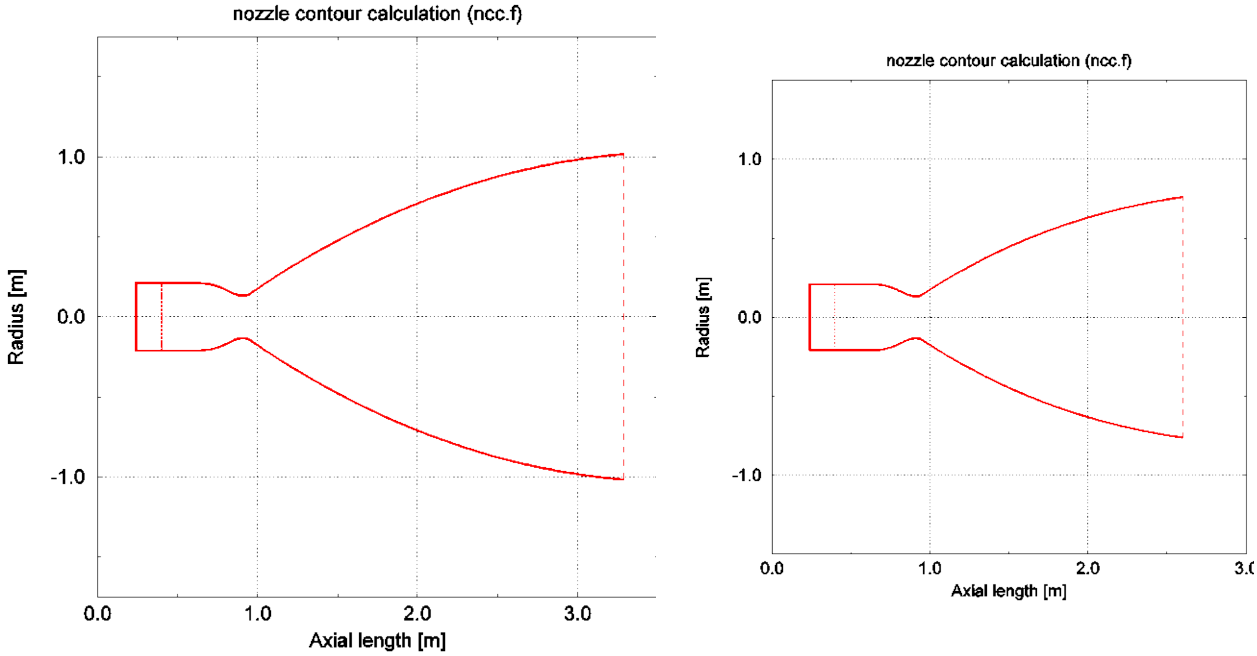

Figure 7: Internal thrustchamber contour of SpaceLiner main engine (ε=59 at left, ε=33 at right)

Although two types of staged combustion cycle (one full-flow and the other fuel-rich) are considered for the SLME cycle analyses, the method of analysis is basically the same. As a basic assumption all turbo-pump and turbine efficiencies are set to 70%. In the next steps a pre-design of the turbo-machinery is intended which will allow for a better estimation of achievable efficiencies.

3.2 Full-Flow Staged Combustion Cycle

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. This approach should allow avoiding the complexity and cost of additional inert gases like Helium for sealing.

In a Full-Flow Staged Combustion Cycle (FFSC), 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 of the turbo pump (FPB to Fuel Turbo Pump (FTP) and OPB to Oxidizer

Turbo Pump (OTP), 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. The fuel tank pressurization gas is supplied from the fuel line after leaving the re-generative 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.

Figure 8 shows the result of an engine cycle analysis in nominal MR=6.0 operation and some major internal engine conditions. The efficiency of each turbine and pump is fixed at 0.7 for the present, and the mixture ratios of FPB and OPB are controlled to be 0.7 and 130 so that TET is restricted to around 770 K. Note the bypass at each turbine 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. Total regenerative heat transfer calculated by TDK increases from 34 MW to 44 MW in the operational MR-domain from 5.5 to 6.5. The highest pressure in the engine system is found at the FTP discharge reaching more than 42 MPa in nominal operation and up to 46.6 MPa at MR=5.5.

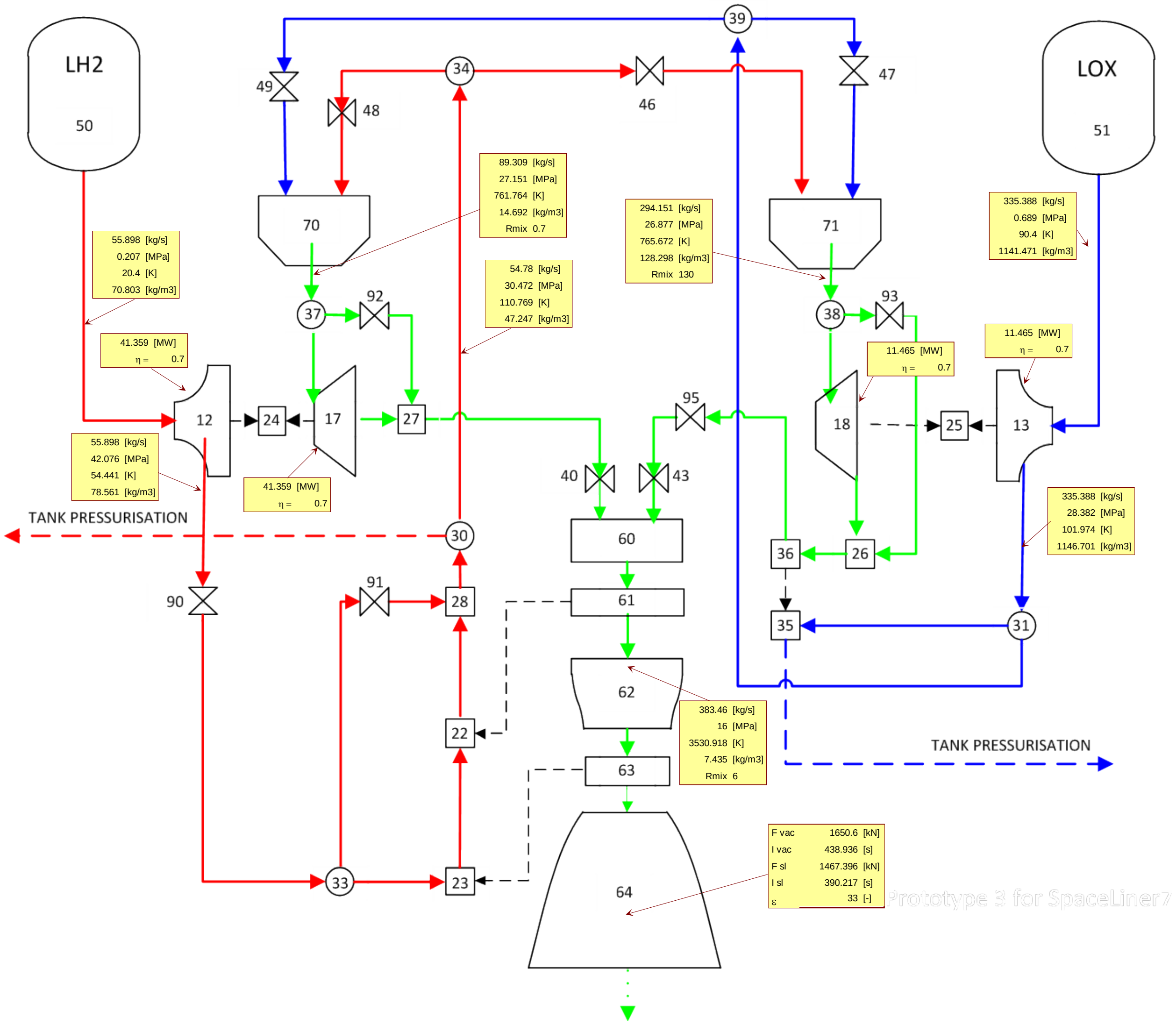

Figure 8: Engine Cycle Analysis of Full-Flow Staged Combustion Cycle (ε=33, MR=6.0)

3.3 Fuel-Rich Staged Combustion Cycle

In the investigated Fuel-Rich Staged Combustion Cycle (FRSC) a single fuel-rich preburner (PB) generates turbine gas which is then split feeding two separate turbo pumps (Figure 9). All of the fuel, except the flow rate for the LH2-tank pressurisation, and approximately 12 % of the oxidizer, are fed into the preburner after being pressurized by each turbo pump (Fuel from FTP pump, Oxidizer from OTP split pump). The hot gas created in the preburner is distributed into the FTP turbine and the OTP turbine. After leaving the turbines the complete fuel rich gas is injected into the main combustion chamber (MCC) together with the LOX flow coming directly from the OTP main pump. Such a lay-out would require a different injector head then for the FFSC. Regenerative cooling of the MCC and the nozzle is achieved with fuel after being pressurized by the FTP. The fuel tank pressurization gas is supplied from the fuel line 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 in a similar way as for the FFSC.

Figure 9 shows the result of an engine cycle analysis at MR=6.0. In this analysis, the efficiency of each turbine and pump is fixed at 0.7 for the present, and the mixture ratio of the PB is controlled to be 0.7 with TET not exceeding 770 K. The FTP discharge pressure is somewhat higher than for the FFSC with almost 45.0 MPa at nominal MR=6 operation.

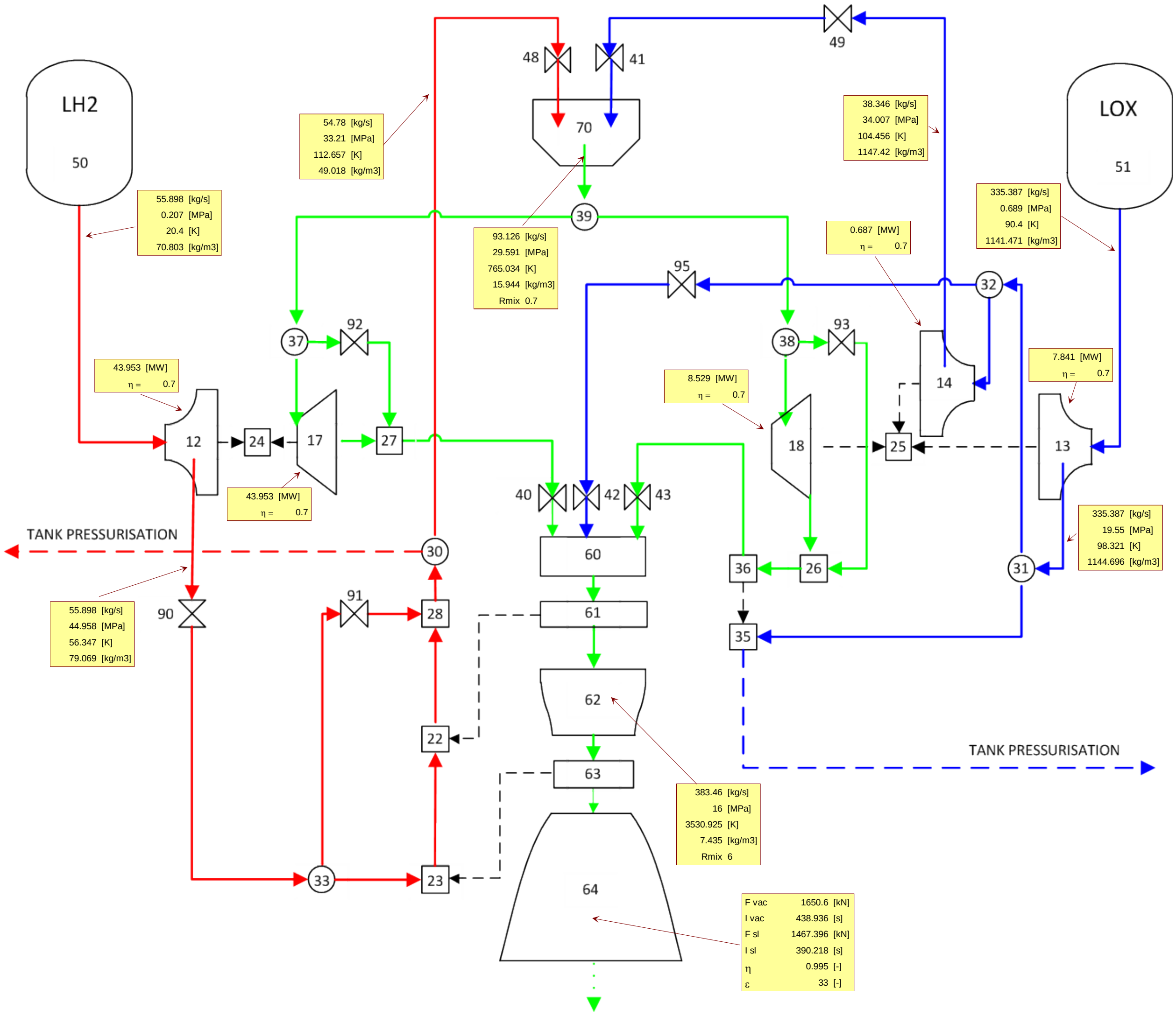

Figure 9: Engine Cycle Analysis of Fuel-Rich Staged Combustion Cycle (ε=33, MR=6.0)

3.4 Comparison of Cycle Analyses

Figure 10 shows the turbo pump power in each engine cycle. OTP power in the FRSC is lower than in the FFSC, because only a part of the oxygen mass enters the preburners in FRSC via a split-pump while all of the oxygen mass is to be raised to the high preburner pressure in the FFSC. FTP power in the FRSC is higher than in the FFSC, because the mass flow of the turbine gas in the FFSC is less on the FTP side than in the FRSC and hence a higher pressure at turbine entry is necessary in fuel-rich cycle.

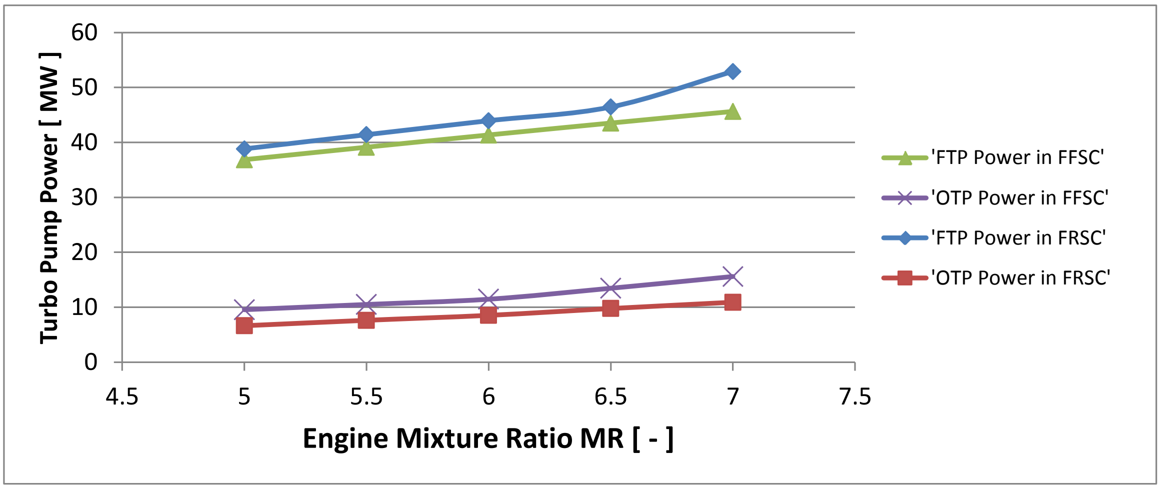

Figure 10: Development of Turbo Pump Power in Full-Flow Staged Combustion Cycle (FFSC) and Fuel-Rich Staged Combustion (FRSC) Cycle as Function of MR

Major engine characteristics obtained by cycle analyses and TDK simulations are listed in Table 2. Overall engine performance of the Full-flow and the Fuel-rich cycles are identical at the same mixture and expansion ratios as to be expected for similar main combustion chamber conditions. The only potential differences could be caused by a variation in combustion efficiency due to the different injector head or by nozzle boundary-layer conditions influenced from potentially different wall heat-transfer requirements. However, the evaluation of such factors impacting c* and c_F and hence I_{sp} and thrust are beyond the current status of the component design.

In comparison to the simplified cycle analyses data of Table 1 a slight improvement in engine I_{sp} seems to be achievable according to the latest calculations. Mass flow and hence thrust based on the thrust chamber geometry of Figure 7 will be slightly lower because boundary-layer effects are now considered. In general, the latest SLME performance data of Table 2 confirm the previous analyses.

Table 2: Engine data of SpaceLiner engine lrp2 cycle analysis at nominal MR= 6 operation point

	FFSC	FRSC
Nominal Mixture ratio [-]	6:1	6:1
Chamber pressure [MPa]	16	16
Fuel-rich Preburner pressure [MPa]	27.2	29.6
Oxidizer-rich Preburner pressure [MPa]	26.9	-
Fuel-rich Preburner TET [K]	762	765
Oxidizer-rich Preburner TET [K]	766	-
FTP discharge pressure [MPa]	42.1	45.0
OTP discharge pressure [MPa]	28.4	19.6 (main) 34.0 (split)
Mass flow per engine [kg/s]	383	383
Expansion ratio [-]	33	33
c* [m/s]	2342	2342
c_F [-]	1.836	1.836
Specific impulse in vacuum [s]	438	438
Specific impulse at sea level [s]	390	390
Thrust in vacuum per engine [kN]	1648	1648
Thrust at sea level per engine[kN]	1467	1467

4 Conclusion and Outlook

The presented variations of different staged combustion cycle design options and obtained analysis data of the tools lrp2 and TDK confirm the SLME engine cycle's system feasibility. A nominal engine operation domain switching from MR 6.5 at lift-off to 5.5 at high altitude flight allows for significantly improved flight performance. The cycle simulations prove that this domain can be operated with TET kept below 780 K. Internal engine conditions obtained from the calculations at different operation points are sufficient for the preliminary sizing of major subcomponents like the thrustchamber, preburner, injector heads, valves, and turbomachinery. Such pre-dimensioning of components will soon start for a thrust-enhanced version of the

SpaceLiner main engine because the latest SpaceLiner 7 vehicle definition indicates the need of approximately 30% more thrust. Such an engine in the 2150 kN class is comparable to the SSME in size and hence scaling of the machineries to similar flow rates should not create a major risk.

The European large-scale engine demonstrator program SCORE-D aims at designing an FRSC lay-out relatively close to the preliminary definition of the SLME with single fuel-rich perburner. The SCORE-D, for which the SRR has been completed in 2011, is not far away from technologies required in an SLME based on a fuel-rich preburner cycle. Currently component tests are running and the successful development and test stand operation of the SCORE-D would also be an important step in the preparation of the SpaceLiner main propulsion system. Operational issues and a more challenging turbo pump design, however, let appear the full-flow configuration to be the more promising engine variant of the future. Necessary technology demonstrations are to be initiated on e.g. preburner and turbine running on oxygen-rich hot gas.

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Further information on FAST20XX can be found on http://www.esa.int/fast20xx.

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