2013-g-20

Preliminary Design Study of Main Rocket Engine for SpaceLiner High-Speed Passenger Transportation Concept

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The revolutionary ultrafast passenger transportation system SpaceLiner is under investigation at DLR in the EU-funded study Future high-Altitude high-Speed Transport 20XX. SpaceLiner's configuration is being amended continuously, and SpaceLiner7 is the brand new version at the point of April in 2013. SpaceLiner7 is two staged reusable launch vehicle with liquid rocket engines. SpaceLiner Main Engine (SLME) is required to have high performance for the total system to be feasible, and also to be easy on the environment for frequent launches. Therefore staged combustion cycle (SC) rocket engine with liquid hydrogen and liquid oxygen (LH2/LOX) is accounted to be promising for SLME. The engine cycle analysis and the component predesign of SLME are performed with DLR developed codes and NASA developed Two-Dimensional Kinetic Thrust Chamber Analysis Computer Program (TDK). They show SLME's feasibility and subject to be researched in the future.

Key Words: SpaceLiner, Rocket Engine, Staged Combustion Cycle

1. Background

1.1. SpaceLiner

An interesting alternative to air-breathing hypersonic passenger airliners in the field of high-speed intercontinental passenger transport vehicles might be a rocket-propelled, suborbital craft. Such a new kind of 'space tourism' based on a two staged reusable launch vehicle has been proposed by DLR under the name SpaceLiner [1], [2]. Ultra long-haul distances like Europe and Australia could be flown in 90 minutes. Another interesting intercontinental destination between Europe and North-West America or between North-West America and East Asia could be reduced to flight times of about one hour.

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

The environmental impact of the liquid hydrogen and liquid oxygen (LH2 and LOX) propelled SpaceLiner is relatively benign. 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 airplane 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.

Different configurations in terms of propellant combinations, staging, aerodynamics 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 Fig. 1. These configuration studies support the definition of the next reference configuration dubbed "SpaceLiner 7".

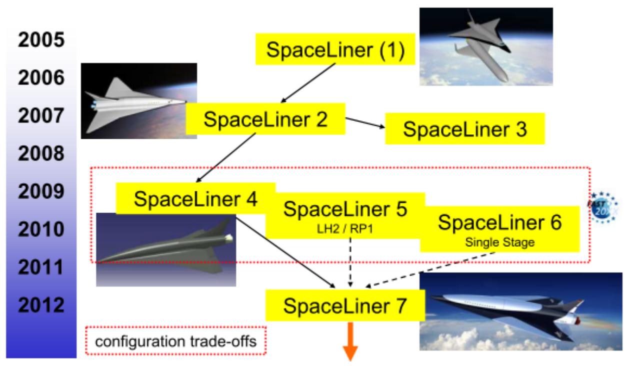

Fig. 1. Evolution of the SpaceLiner concept

1.2. SpaceLiner Main Engine

SpaceLiner Main Engine (SLME) is required to have both of high performance and safety for passengers. Therefore, staged combustion cycle (SC) rocket engines with a moderate 16MPa chamber pressure have been selected as the baseline propulsion system. SC rocket engine is able to make enough performance required for SLME and have potential of improving safety by amending the cycle detail. The engine performance data are not overly ambitious and have already exceeded by existing engines like Space Shuttle Main Engine (SSME) or LE-7A. The nozzle skirt (NS) expansion ratio (ER)s 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.

A mixture ratio (MR) of 6 is a typical selection in a high performance LH2/LOX rocket engine and has been used for all the SpaceLiner variants up to SpaceLiner 6. However, the optimum engine mixture ratio is always mission-dependent. Further, adaptation of the MR during flight might improve performance with better specific impulse (Isp) and improved thrust level. Fig. 2 shows the impact of MR variation on the Isp difference compared to the reference point of 6.0, based on a simple rocket engine cycle model of full flow staged combustion cycle (FFSC) engine. The vacuum Isp at lower MR is higher than at higher MR, and the sea level Isp at lower MR is lower than at higher MR. The sensitivity is stronger for larger nozzle (ER of 59) of the orbiter engines with a difference of 14s in sea level operation at MR of 5.0. The corresponding thrust level is changing by up to more than 20% of nominal thrust. If engine operating points are switched at the right flight condition, significant performance improvements of SpaceLiner configuration are possible.

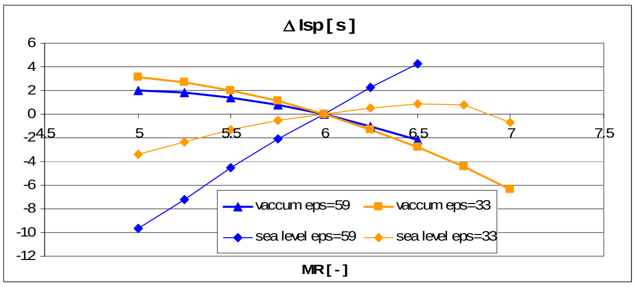

Fig. 2. Potential Isp impact of MR variation

The best mixture ratio of the SpaceLiner main propulsion system along its mission has been defined by system analyses optimizing the full nominal trajectory optimization under the consideration of all relevant mission constraints and objectives is performed for the SpaceLiner4 using the AeroSpace Trajectory Optimization Software (ASTOS) [3]. 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 2.5g acceleration and 5.5 afterwards) with relatively short transits in between is found most promising. This approach allows for a significant propellant saving on the booster, 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 [4].

This paper's study is based on the configuration of the vehicle definition named SpaceLiner 7-1. (shown in Fig.3.) The design of SpaceLiner 7-1 indicates the required performance of SLME as Table 1. The SLME design in this paper aims for these values.

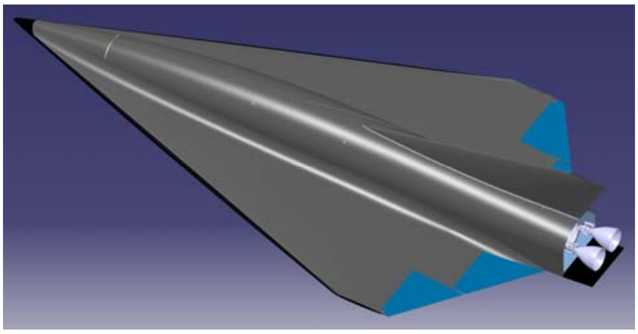

Fig. 3. Latest configuration of SpaceLiner 7-1 orbiter

Table 1. SLME Performance Requirement

	Booster			Orbiter		
MR	5.5	6	6.5	5.5	6	6.5
Thrust in vacuum [kN]	2060	2200	2350	2110	2260	2420
Thrust at sea level [kN]	1810	1960	2110	1670	1820	1980
Isp in vacuum [s]	438	436	434	450	448	447
Isp at sea level [s]	386	388	389	356	362	367

1.2. Engine Cycle of SLME

Fuel rich staged combustion cycle (FRSC) engines with a moderate chamber pressure were selected for the two SpaceLiner stages already in the early designs [5]. These SC 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. Therefore some alternatives in SC are considered.

One alternative is about gas to drive each turbo pump's turbine. Fig. 4 shows simple engine schematics of FFSC and FRSC. FRSC is the engine cycle using only fuel rich gas generated by fuel rich preburner (FPB) to drive all turbo pumps' turbines. On the other hand, in FFSC, fuel rich gas by FPB is used to drive fuel turbo pump (FTP)'s turbine and oxygen rich gas by oxygen rich preburner (OPB) is used to work oxygen turbo pump (OTP)'s turbine.

FFSC has two advantages against FRSC. One is that required temperature and pressure for turbine gas is able to be decreased by using more turbine gas mass flow. The other is elimination of criticality that fuel and oxygen would be mixed in OTP. That allows avoiding the complexity of turbo pump sealing design and reducing cost of additional inert gases like helium for sealing. Disadvantage of FFSC is that engine cycle becomes more complex and that we have less experience of development, even though there are precedents of RD-270 and Integrated Powerhead Demonstration by USAF and NASA.

The other alternative is allocation of turbo pump. Especially allocation of FTP relates to regenerative cooling performance, so some consideration is necessary. Although the engine cycle with 2 FTPs is more complicated than with 1 FTP, it is expected to make FTP discharge pressure lower. That is because 2 FTP cycle enables preburner line to be divided from regenerative cooling line on main combustion chamber (MCC), which makes much pressure loss.

We are concerned to evaluate feasibility and safety for these alternative engine cycles.

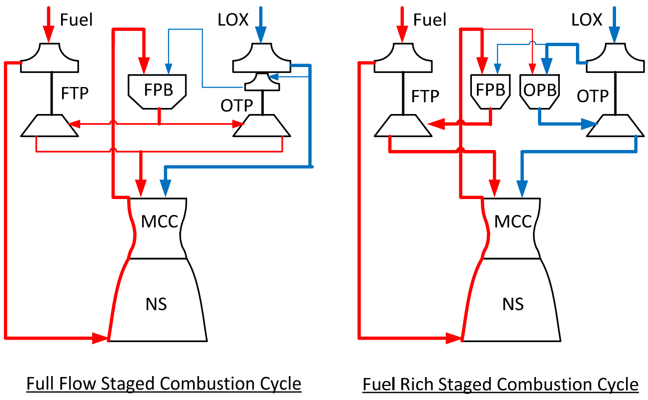

Fig. 4. Engine schematics of FFSC and FRSC

2. Design Approach

2.1. Engine Cycle Analysis

Engine cycle analyses are performed with several analysis codes in order to define the requirement for the components of SLME and to evaluate the feasibility and the potential safety of the alternative engine cycle. Fig. 5 illustrates a flowchart of the engine cycle analysis.

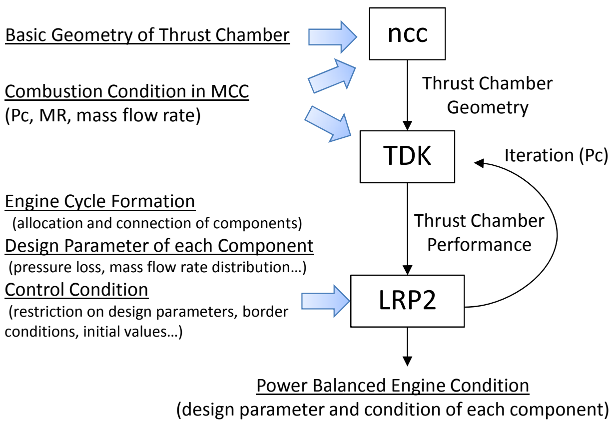

Fig. 5. flowchart of the engine cycle analysis

At first, thrust chamber geometry including throat diameter is calculated with DLR code "ncc". This calculation is on the basis of the designed combustion condition (MR, combustion pressure, fuel flow rate, combustion efficiency and so on) and geometry parameters (compression ratio of chamber, ER, characteristic chamber length, angle of contour and so on).

Second, thrust chamber performance including Isp and heat flux in regenerative cooling part are calculated with NASA code Two-Dimensional Kinetic Thrust Chamber Analysis Computer Program (TDK) [6]. The input for TDK is thrust chamber geometry taken by ncc, combustion condition newly set, and the distribution of wall temperature, which is set on the basis of SSME chamber wall temperature.

At last, power balanced engine condition including each component performance is calculated with DLR code Liquid Rocket Propulsion 2 (LPR2). That input is thrust chamber performance by TDK, engine cycle formation (information of allocation and connection of components), design parameter of each component (pressure loss, mass flow rate distribution, and so on), and control condition (restriction on design parameters, border conditions, initial values, and so on). In calculating with LRP2, the mixture ratios of FPB and OPB are controlled to be 0.7 and 130 respectively so that turbine entry temperature (TET) would be restricted to around 780K. This restriction is set with the aim of increasing the life span of turbine blades. The flow resistance of each component is set using examples from already existent designs of SSME [7] and LE-7A [8].

The iteration of main chamber combustion pressure between TDK and LRP2 is necessary in calculating the design point except basis design point MR 6. That is because TDK need combustion pressure value as input, but main chamber combustion pressure except basic design point MR 6 is calculated by LRP2 analysis at the downstream of TDK.

2.2. Components Design

The designs of main components in SLME are performed with DLR tool. The geometry of the thrust chamber including MCC and NS is calculated as described in engine cycle analysis. The turbo machinery design is performed with DLR tool Liquid Rocket Propulsion for Mass (LRP-MASS). Fig. 6 describes the flowchart of the turbo pump design. Input into LRP-MASS consists of basic requirement and condition for turbo pump by LRP2 analysis, turbo pump type (reaction turbine or impulse turbine, allocation of inducer and impeller), number of stages, rotation of shaft, fluid velocity in entering each part, and geometry conditions such as diameter ratio. LRP-MASS outputs detailed geometry information of each part (case, rotor, stator, turbine, turbine ring, shaft and so on) and some design parameters (specific speed, specific speed, head coefficient, inlet flow coefficient and so on), in keeping basic formations for turbine or pump. These outputs are checked with typical design values from the turbo pump design standard [9], [10], and alteration of input is repeated until the proper output is gained.

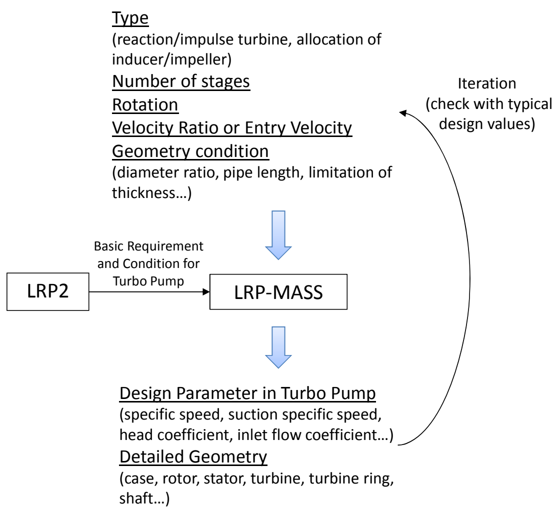

Fig. 6. flowchart of turbo pump design

3. Results and discussion

3.1. Comparison of FFSC and FRSC

Fig. 7 and Fig. 8 show the engine cycle schematic of FFSC and FRSC respectively. The analysis for comparison is performed only in common ER 33, same as SLME for booster. That is because difference of SLME for booster and SLME for orbiter is basically only NS geometry and the influence caused by such difference for comparison of two cycles is considered to be little. The calculation is performed in MR 5.5, 6.0, and 6.5.

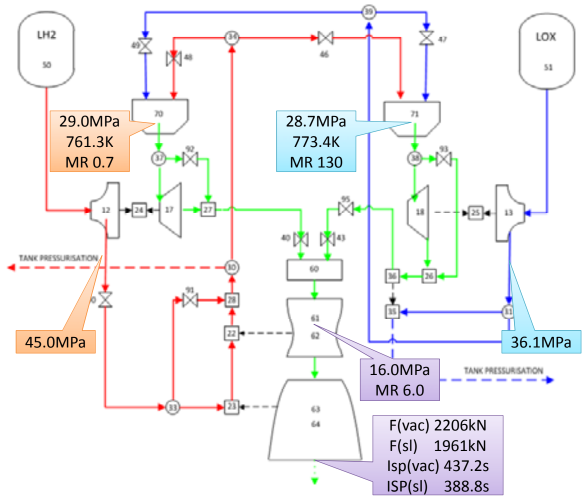

Fig. 7. Results of engine cycle analysis for FFSC (MR=6.0)

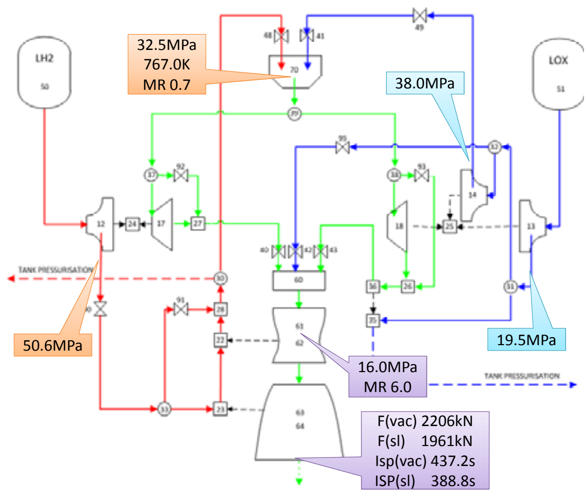

Fig. 8. Results of engine cycle analysis for FRSC (MR=6.0)

All turbo pumps are set as simply devices pressurizing each fluid and their detail configuration are not taken into account in this comparison, while turbine efficiency and pump efficiency are set to be 70% all. The distribution to bypass line is set in the upstream of the turbine of each turbo pump. That is in order to conditions are to reduce the variation of turbo pump design point. The turbo pump power is getting higher as MR is higher, so turbine bypass ratio increases in high MR. Turbine bypass ratio is set almost zero in MR 6.5. Combustion condition, thrust chamber geometry, turbine bypass ratio and pressure loss rate of each component are common in both analyses.

The engine characteristics by analysis are listed in Table 2. One of important indexes to evaluate safety of an engine cycle is turbo pump discharge pressure. That is because it is maximum pressure in each line of engine cycle. In FFSC, FTP discharge pressure is from 42.9MPa to 45.8MPa in the range of MR 5.5 to 6.0. Those are lower than in FRSC by 5MPa to 7MPa. OTP discharge pressure in FFSC is from 35.5MPa to 36.4MPa in same range, and they are lower than split pump discharge pressure in FRSC by about 2MPa. The difference in FTP discharge pressure is large and not negligible. On the other hand, the difference in OTP discharge pressure is not so much. Those differences resulted from the difference of turbine gas mass flow rate. Since all mass flow of fuel and oxygen is used as turbine gas in FFSC, necessary turbine gas pressure in FFSC is lower than in FRSC. However, while only about 9% of oxygen is high pressurized by the split pump (No.14 component in Fig. 8) in FRSC, all oxygen need to be high pressurized in FFSC. That fact partly reduces the advantage of FFSC in OTP discharge pressure.

The both turbo pump discharge pressure is still lower in FFSC, and there are some other advantages of eliminating the critical failure mode of fuel and oxygen mixing in OTP and avoiding the complex sealing, so FFSC is considered a preferred design solution for the SpaceLiner.

Table 2. Engine Data Comparison of FFSC and FRSC

	FFSC			FRSC		
Mixture Ratio [-]	5.5	6.0	6.5	5.5	6.0	6.5
Main Chamber Pressure [MPa]	15.1	16.0	16.9	15.1	16.0	16.9
Fuel-rich Preburner Rressure [MPa]	27.9	29.0	29.4	32.3	32.5	32.8
Oxidizer-rich Preburner Pressure [MPa]	27.6	28.7	29.1	-	-	-
FTP TET [K]	756	761	764	764	767	770
OTP TET [K]	772	773	774	-	-	-
FTP Discharge Pressure [MPa]	42.9	45.0	45.8	50.2	50.6	51.0
OTP Discharge Pressure [MPa]	35.5	36.1	36.4	18.5 (main) 37.8 (split)	19.5 (main) 38.0 (split)	20.7 (main) 38.4 (split)
Specific Impulse in vacuum [s]	438.8	437.2	434.8	438.8	437.2	434.8
Specific Impulse at sea level [s]	387.0	388.8	389.7	387.0	388.8	389.7
Thrust in vacuum [kN]	2064	2206	2361	2064	2206	2361
Thrust at sea level [kN]	1820	1961	2116	1820	1961	2116

3.2. Comparison of 1 FTP FFSC and 2 FTPs FFSC

Fig. 9 and Fig. 10 illustrate the engine cycle schematic of 1 FTP FFSC and 2 FTPs analyzed for comparison.

In 1 FTP FFSC, FTP has an inducer to produce first head rise of fuel from feed line and an impeller to increase fuel pressure still. Fuel discharged from FTP enter regenerative cooling part of NS and MCC, and is injected into two preburners. FTP's turbine is worked by turbine gas from FPB.

In 2 FTPs FFSC, FTP is designed to be divided into a low pressure fuel turbo pump (LPFTP) and a high pressure fuel turbo pump (HPFTP). LPFTP has only an inducer to produce head rise. The turbine of LPFTP works by hot fuel gas from the regenerated cooling part on MCC. This system enables fuel line to divide into MCC regenerated cooling line with much pressure loss (from 6MPa to 8MPa) and preburner line, and that makes HPFTP discharge pressure lower. This system is considered to be partial expander cycle and same as SSME. It seems that the benefit of large turbine gas in SC may be reduced. However larger turbine gas is supplied in FFSC than FRSC, so the effect is considered to be limited. Actually the mass flow rate into LPFTP turbine line (MCC regenerative cooling line) is about 22% of entire fuel mass flow and 3% of all turbine gas. HPFTP has an impeller to raise fuel head in downstream of LPFTP. The turbine works by fuel rich combustion gas from FPB. The both turbine gases of LPFTP and HPFTP enter into MCC after mixture.

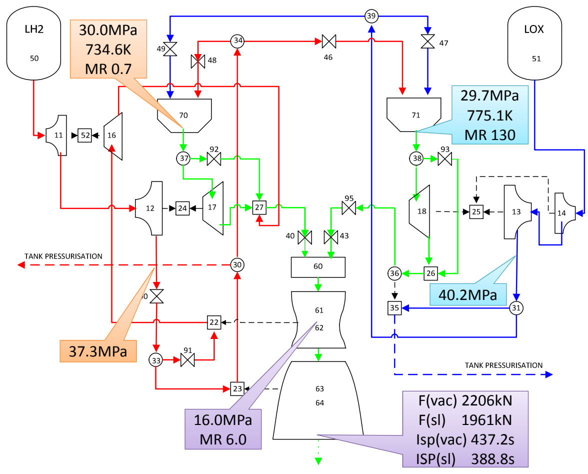

Fig. 9. Results of engine cycle analysis for 2 FTPs FFSC (MR=6.0)

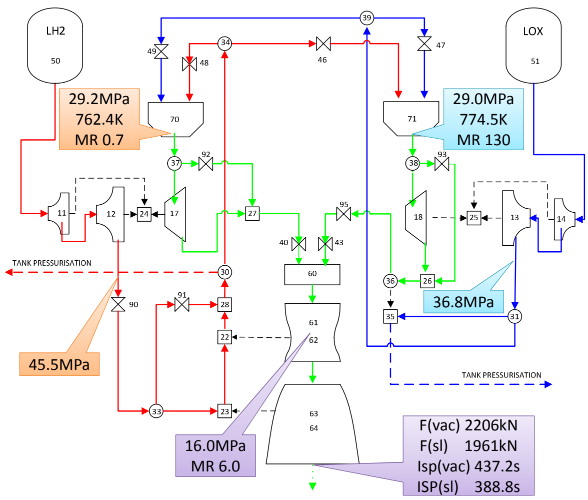

Fig. 10. Results of engine cycle analysis for 1 FTP FFSC (MR=6.0)

In focusing on turbo pump discharge pressure again, HPFTP discharge pressure in 2 FTPs FFSC is 37.3MPa and that is lower by 8.2MPa than 45.5MPa in 1 FTP FFSC. It is considered that dividing FTP works efficiently. OTP discharge pressure in 2 FTPs FFSC is 40.2MPa and that is higher by 3.4MPa than 38.9MPa in 1 FTP FFSC. That is reason why whole turbine gas flow rate is lower in 2 FTPs FFSC and higher turbine gas pressure is necessary while all LOX is used as turbine gas similarly with 1 FTP FFSC. As 2 FTPs FFSC and 1 FTP FFSC have advantages against each other, in the point of view that engine maximum pressure should be as low as possible, 2 FTPs FFSC are preferable to SLME.

3.3. Design Point of SLME

Table 3 shows SLME design characteristics for the booster and the orbiter of SpaceLiner 7-1. The engine cycle is 2 FTPs FFSC. Difference of SLME for the booster and for the orbiter is basically only geometry of NS. However that difference makes the performance of regenerative cooling and slight difference of engine design point entirely. While LH2 regenerated cooling for the booster engine is in all area of NS, one for the orbiter engine is only in a part of NS ranged to ER 33. Additionally NS for the orbiter has smaller nozzle entry angle, so surface of regenerative area in NS for the orbiter is smaller than for the booster. Therefore, regenerative cooling heat transfer rate in NS for orbiter is lower than for booster, and that makes turbine gas temperature lower, turbine gas and turbo pump discharge pressure higher. As this difference is not so much (about 0.5MPa in HPFTP discharge pressure), it seems to make design point coincident with the orbiter and the booster by adjusting the regenerative cooling area. However this difference may be only little compare to scattering of real hard ware, so it had better contain matter simple as same area is applied to regenerative cooling at present.

The engine performance at MR 5.5, 6.0 and 6.5 satisfy the requirement by SpaceLiner7-1 system, achieving other some engine design aims of lower turbo pump discharge pressure, lower TET, and similar design point of booster engine and orbiter engine except NS.

Table 3. Engine characteristics of SLME

	Booster			Orbiter		
Mixture Ratio [-]	5.5	6.0	6.5	5.5	6.0	6.5
Main 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
Oxidizer-rich Preburner Pressure [MPa]	29.1	29.7	30.5	29.2	29.9	30.7
FTP TET [K]	732	735	738	720	722	724
OTP 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]	2014	2154	2299	2067	2216	2366
c_F [-]	1.807	1.826	1.843	1.856	1.878	1.898
Specific Impulse in vacuum [s]	438.8	437.2	434.8	450.6	449.4	447.6
Specific Impulse at sea level [s]	386.9	388.8	389.7	357.4	362.5	366.6
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

3.4. Thrust Chamber Pre-design

The geometry of thrust chamber is designed with DLR tool ncc. Internal contour of the thrust chamber is illustrated in Fig. 11 and geometric characteristics are shown in Table 4. The booster engine and the orbiter engine have same geometry in the part of MCC including the throat area, but not same in the part of NS. NS for orbiter has not only larger ER but also smaller nozzle entry angle, so that total length of NS would be not so long. As the result of calculation, the thrust chamber total length of booster is 2.7m and one of orbiter is 3.6m. The orbiter engines works also in booster accelerating phase, extendible nozzle is possible to make total propulsion performance better. That is subjected in the future.

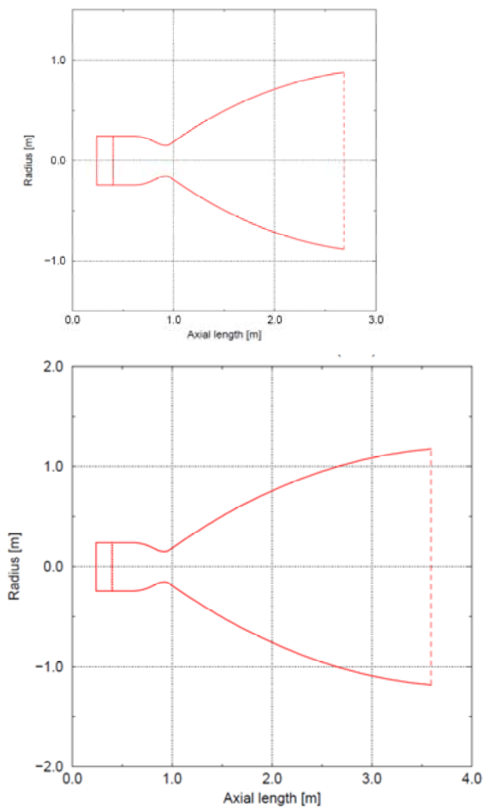

Fig. 11. Internal thrust chamber contour of SLME
(Top: Booster, Bottom: Orbiter)

LH2 regenerative cooling and film cooling are applied, for thrust chamber cooling in booster engine. Regenerative cooling works in all area of thrust chamber with the two passes. One pass chills chamber including the throat area, and the other pass chills the nozzle area. LH2 for the film cooling is supplied from the part of chamber regenerative cooling, enter into the side of injector plate and chills chamber wall. On the other hands, LH2 regenerative cooling, film cooling, and radiation cooling are applied for thrust chamber cooling of the orbiter engine. Regenerated cooling is used in the chamber wall and the part of nozzle wall ranged to ER 33. LH2 regenerated cooling has two passes as the booster engine. LH2 for film cooling is also same as the booster engine. Radiation cooling is applied in the part of nozzle ranged from ER 33 to 59.

The coaxial injector is selected as other oxygen-hydrogen engines. Even though injector element design for gas hydrogen and gas oxygen is not so many, full scale engine combustion test and computational fluid dynamics analysis for axial type injector of gas oxygen are reported in [11] and [12]. The design of injector elements will be done henceforth and it is necessary to make an elementary or subscale test for evaluation of the performance and combustion stability of injector in developing phase of SLME.

Table 4. Thrust chamber geometric characteristics of SLME

	Booster	Orbiter
Chamber		
Contraction ratio	2.5	2.5
Characteristic chamber length [m]	1.1	1.1
Upstream contour angle [°]	25	25
Chamber volume [m3]	0.081	0.081
Throat radius [m]	0.153	0.153
Nozzle		
Expansion ratio	33	59
Nozzle entry angle [°]	35.6	35
Nozzle exit angle [°]	8	5
Exit diameter [m]	1.76	2.35
Total length [m]	2.7	3.6

3.4. Turbo Machinery Pre-sizing

The geometry and design characteristics of SLME turbo machinery are calculated by LPR-MASS. Since SLME turbo machinery design for booster and orbiter is not so much different, the analysis is performed in booster conditions. The part of them is shown in Table 5. Number of stage in turbo machinery is reduced as possible as it can be, in order to make the each turbo pump design simple and reliable. As a result, OTP consists of an inducer, single impeller and single turbine, LPFTP has an inducer and single turbine, and then HPFTP has two staged impeller and two staged turbine. Turbine type of all turbo pumps is reaction turbine for high efficiency with small size. Interface conditions in inducer inlet of OTP and LPFTP is set by nominal. LH2 is assumed to be supplied with 0.21MPa and LOX is with 0.69MPa in this design point. Actually they are varying in some range as SpaceLiner flight sequence, so further off design study is necessary in the near future.

Table 5. Turbo machinery pre-sizing of SLME

	OTP	LPFTP	HPFTP	
Inducer				
Specific speed [$(m/s^2)^{3/4}$]	6840	3479	No Inducer	
Head coefficient [-]	0.247	0.278		
Inlet flow coefficient [-]	0.054	0.183		
Impeller				
Number of stage	1st	No Impeller	1st	2nd
Specific speed [$(m/s^2)^{3/4}$]	1923		505	
Head coefficient [-]	1.022		1.105	1.105
Inlet flow coefficient [-]	0.105		0.122	0.108
Turbine				
Type	Reaction	Reaction	Reaction	
Number of stage	1st	1st	1st	2nd
Head coefficient [-]	3.330	4.160	2.541	2.541
Inlet flow coefficient [-]	0.701	0.345	0.830	0.634
Shaft				
Power [MW]	21.58	3.38	46.30	
rotation [1/min]	24000	28000	32000	

At the present moment, the parameters of specific speed, head coefficient, inlet flow coefficient and so on are in the range which the design of past existent rocket engines or standards shows as proper. New values of turbo machinery efficiency are estimated by the result, so one more cycle analysis.

4. Conclusion and subjects for further study

The preliminary design of the main propulsion system SLME for the revolutionary ultrafast passenger transportation system SpaceLiner has been practiced with some tools of DLR and NASA. Tradeoff studies on the engine cycle are performed for evaluation of design safety and feasibility. One of them is the comparison between FFSC and FLSC, and the other is between 2 FTPs FFSC and 1 FTP FFSC. By the reason why maximum pressure in engine is much lower, 2 FTPs FFSC is concluded to be preferable to SLME. The adjusted engine cycle makes enough performance for Spaceliner7-1 vehicle's system requirement and achieved other some engine design aims for passenger safety. They are lower turbo pump discharge pressure, lower TET, and similar design point of booster engine and orbiter engine except NS.

The primary image on the basis of configuration by these results is shown in Fig. 12. HPFTP, OTP, FPB and OPB are attached on MCC such as SSME power-head so that allocation of pipes in engine would be as simple as possible. LPFTP is at the interface with fuel feed line for keeping enough head pressure.

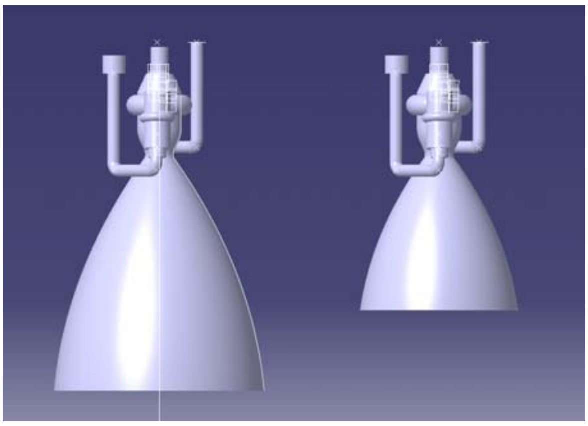

Fig. 12. SLME configuration image (Left: Orbiter, Right: Booster)

Some subjects for further study are there. First is off nominal design. The estimation of design point in some design parameter varying is necessary for robust and reliable design. Especially turbo machinery design parameter and interface conditions at the inducer inlet have an impact on turbo pump design and entire engine cycle design.

Second subject is possibility of amendment for the engine cycle or engine component design. OTP discharge pressure perhaps can be much lower by bringing a part of LOX pressurized in OTP to MCC directly. Applying of extendible nozzle to the orbiter will improve engine performance to SpaceLiner flight plan with Isp increased. At last, some development risks peculiar to FFSC should be evaluated sufficiently. One of them is combustion stability with gas hydrogen and gas oxygen. Additionally, even though this is not described in this paper, consideration to the material for oxygen rich hot gas from OPB is essential in applying FFSC.

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