39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit
Huntsville, Alabama 20 - 23 Jul 2003

AIAA 2003-4597

Studies on Expander Bleed Cycle Engines for Launchers

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Results of theoretical investigations on expander bleed cycle engines as well as operational experience with the LE-5B rocket motor are described. Based on results of an earlier cycle analysis, an improved engine variant of a high thrust, cryogenic bleed cycle engine is analyzed. Its performance is calculated, and a preliminary engine sizing and mass analysis is carried out. Improvement options as well as growth potential of the cycle is investigated. Special attention is paid to the critical heat-transfer in the thrust chamber region. Finally, an ascent trajectory optimization and an assessment of the complete vehicle are performed to evaluate the engine as part of a future semi reusable launch system.

Nomenclature

F	thrust	N
I_{sp}	specific impulse	s
k	surface roughness	m
l	length	m
l^*	characteristic chamber length	m
$\dot{m}$	massflow	kg/s
M	Mach number	-
NPSP	net positive suction pressure	Pa
p	pressure	bar
P	power	W
Q	total heat flux	W
R	mixture ratio	-
s	wall thickness	m
T	temperature	K
TET	turbine entry temperature	K
u	circumferential velocity	m/s
v	velocity	m/s
V	volume	m³
ε	expansion ratio	-
η	efficiency	-
κ	specific heat ratio	-
ρ	density	kg/m³
Ψ	enthalpy coefficient $2\,\Delta h / u^2$	-

Subscripts

c	chamber
t	throat
sl	sea-level
tot	total
vac	vacuum

1 INTRODUCTION

Recent proposals concerning the introduction of reusable components in space transportation regard the first or booster stages. Such systems are usually known as liquid fly-back booster (LFBB) or a reusable first stage. In the past years, studies on the replacement of the solid rocket boosters (SRB) of the existing Ariane 5 with reusable LFBB have been conducted within the German future launcher technology research program ASTRA. [14, 15] The replacement of solid propellant stages by a reusable booster offers a potential reduction in operation cost and increases mission flexibility.

The SRBs generate about 90% of Ariane 5's thrust at lift-off, engines of substantial thrust levels have to be incorporated into the design of the LFBB in order to produce an adequate substitute of the solid motors. The basic LFBB design of ASTRA incorporates a cluster of three upgraded Vulcain motors [14]. Another interesting option is the development of a completely new engine with regard to reusability requirements.

This paper concentrates on the feasibility of high thrust bleed cycle engines for large reusable boosters. In a bleed cycle, the coolant flow having been heated while passing through the thrust chamber's cooling channels drives the turbines. Subsequently, the turbine exhaust is expelled. Lower cost and higher reliability of the turbines are to be expected due to lower turbine inlet temperatures [4]. In previous DLR studies different engine design options have been investigated [1, 2]. A dual turbopump cycle arrangement is thoroughly investigated in this paper. The available heat and its influence on the cycle will be examined for differently sized regeneratively cooled thrust chambers.

2 OPERATIONAL BLEED CYCLE ENGINES

Presently, the Japanese LE-5A and LE-5B engines are the only flight-proven bleed cycle engines. The LE-5A had been used in the Japanese H2 rocket since 1994. The more cost efficient LE-5B is in operation with the H2A rocket since 2001. The characteristics of LE-5A and LE-5B engines are shown in Table 1.

The LE-5A turbines are driven by the hydrogen gas heated in the chamber and nozzle (Figure 1). The LE-5B turbines are instead driven by the hydrogen gas heated in the chamber only (Figure 2). To increase the heat exchange of the chamber, the LE-5B chamber structure was changed from Nickel double tapered tubes to electrodeposited copper. Consequently the LE-5B turbine drive gas temperature has been successfully reduced. At the same time, the fuel injection temperature of the LE-5B is much lower than that of the LE-5A.

	LE-5A	LE-5B
Vacuum Thrust (kN)	122	137
MR	5.0	5.0
Vacuum Isp (s)	452	447
Combustion Pressure (MPa)	3.98	3.62
Turbine Drive Gas Temp. (K)	589	428
NF (rpm)	52,200	52,100
NO (rpm)	17,400	17,700
Expansion Ratio	130	110
Length (m)	2.67	2.74
Weight (kg)	248	285
Restart Capability	Many Times	Many Times
Throttling Capability	No	60% (option)
Engine Cycle	Nozzle Expander Bleed	Chamber Expander Bleed

Table 1: Characteristics of the Japanese LE-5A and LE-5B engines [9], [10]

The feasibility model of LE-5B, being the earliest model, had no problem of combustion stability. The engineering model chamber of the LE-5B was designed using the measured heat absorption characteristics of the feasibility engine chamber. The test data of the LE-5B engineering model indicated that the combustion efficiency was lower than expected [9]. Therefore, the jet swirling type of the injector element was adopted to improve the combustion efficiency. However, when using the new injector combustion instability occurred in the test. The cause of the instability was considered to be the low fuel injection temperature, as this was so low as to freeze the jet of swirling LOX. The design of the LOX injector element was returned to the original configuration to minimize the risk of instability.

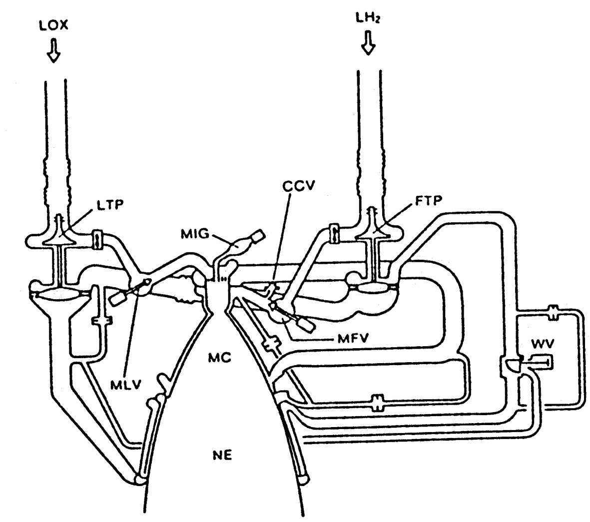

Figure 1: Schematic of Japanese LE-5A engine

In another step, the LE-5B chamber length was increased by 120mm. This provided more heat to the hydrogen. The fuel injection temperature and the FTP turbine inlet temperature were increased. With all these changes, the combustion efficiency was improved by 1%.

Since development completion, the LE-5B powers the second stage of the H2A rocket which has been launched five times successfully so far.

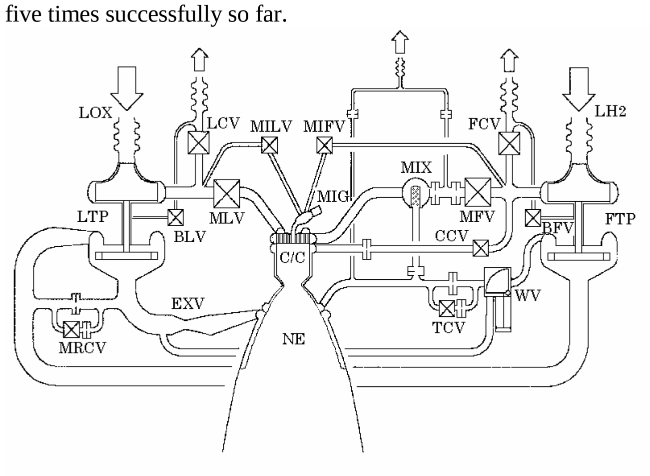

Figure 2: Schematic of Japanese LE-5B engine

3 ANALYSIS METHODS EMPLOYED

3.1 Cycle Analysis Tool

The program used for this paper's cycle analysis, is based on the modular program SEQ [5] of the German Aerospace Center (DLR). In recent years this powerful tool has been constantly 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 turbine mass flow and TET, and finally the desired chamber mixture ratio regulates the engine mixture ratio.

3.2 Preliminary Design and Mass Estimation

The engine preliminary design is performed by thrust chamber and nozzle dimensioning [7], as well as one dimensional turbomachinery conceptual lay-out. The latter is employed in a newly developed computer program and regards important sizing parameters such as stage enthalpy coefficients, (suction) specific speeds, and checks on the required NPSP for pre-selected rotational speeds.

The final estimation of the engine mass is based on structural mechanical methods supplemented by empirical data for several subcomponents [11].

4 ENGINE CYCLE ANALYSIS

4.1 Choice of General Engine Parameters

In the preliminary study extensive variations of the general engine parameters like chamber mixture ratio, nozzle expansion, and chamber pressure have been made to find the most suitable values for the booster application. [1] A variation of the nozzle's expansion ratio and the chamber pressure and the thereby resulting specific impulses is shown in Figure 3.

Engine performance analysis led to the choice of a chamber pressure of 65 bar and an engine mass flow rate of 550 kg/s.

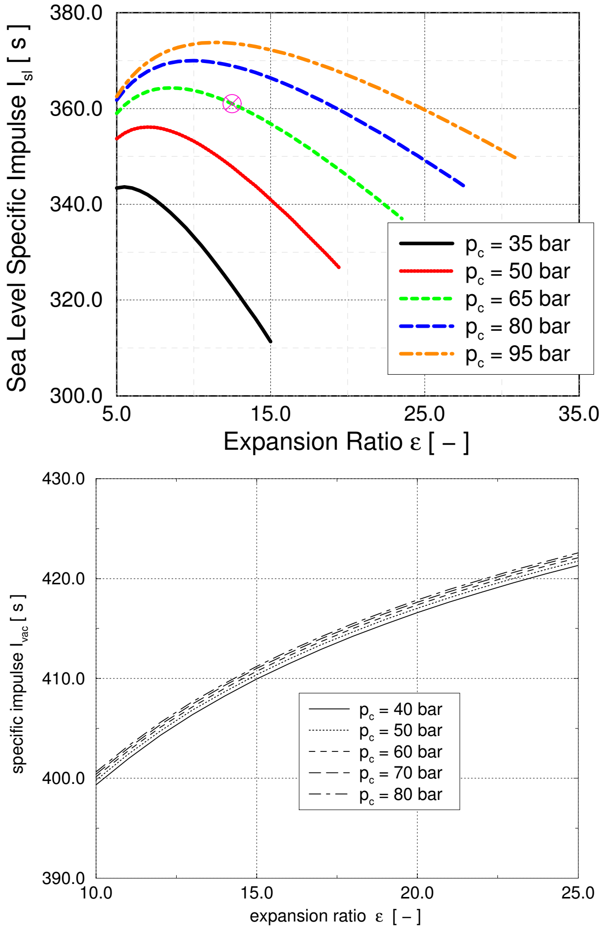

Figure 3: Influence of a variation of the expansion ratio and the chamber pressure on the specific sea level impulse (top) and on the specific vacuum impulse (bottom)

The initial selection of the mixture ratio is based on three considerations. A low chamber mixture ratio leads to a high specific impulse and results in a lower chamber temperature. Figure 4 shows a variation of the chamber mixture ratio and the relative changes of those two aspects. Finally, the system study shows that a lower chamber mixture ratio (leading to a lower engine mixture ratio) increases the launcher's tank volume and thereby the size and mass of the system. Taking all this into account a compromise has to be found.

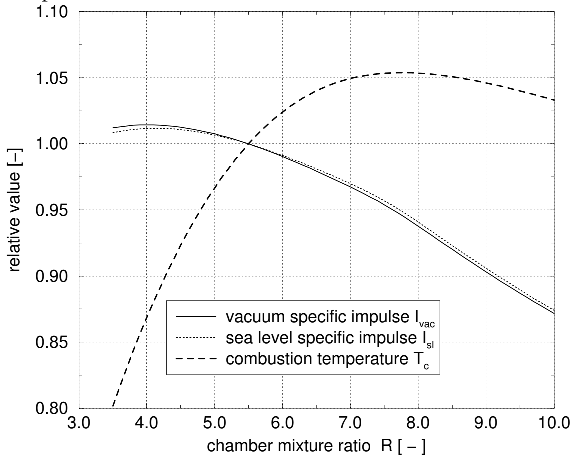

Figure 4: Sensitivity of sea level and vacuum specific impulse and chamber temperature to a change in combustion chamber mixture ratio (normalized to values at R of 5.5)

More data on the preliminary sizing of the large bleed cycle engine (e.g. the relationship between LH2 pump discharge pressure and turbine performance) are presented in reference 1.

4.2 Investigation History of the Chosen Cycle

During the previous DLR analysis [1] it became apparent that the high heat-transfer rate needed by an original cycle with coolant mass flow only used to power the turbines is linked to severe disadvantages. To keep the coolant from overheating during the throat passage a relatively low chamber mixture ratio of 5.5 had to be chosen. Due to the bleed of the fuel, the engine mixture ratio will be even lower than the chamber mixture ratio, which is disadvantageous from a launcher system point of view (see above).

The analyzed cycles of reference 1 do not tolerate a significantly higher engine mixture ratio, since the increasing heat-transfer has adverse effects on the dimensioning of the cooling channels. This is due to the relatively low coolant mass flow, which results in the coolant being superheated. A surge in turbine mass flow will significantly reduce engine performance.

The conclusions of the early DLR studies on expander bleed engines [1] led to the idea of a cycle (variant A) in which the heated fluid is partially used for turbine feeding, while the remainder is injected into the combustion chamber. Such a design allows for higher heat-transfer without 'wasting' too much fluid by expelling it through the turbines. The idea behind this is to flexibly separate the fuel into a highly charged but small mass flow to drive the turbine and a comparatively cool mass flow to enter the thrust chamber.

Slightly improved variants of this lay-out have been more deeply investigated [2, 3]. While the major part of the heated flow is expelled after driving the turbines, a small amount is to be mixed with the main stream of sub-critical hydrogen prior to the injector entrance. The quantity of hot hydrogen redirected into the combustion chamber should be large enough to achieve super-critical conditions of the mixed flow in front of the injector. Moreover, the injection temperature should be high enough to enable stable combustion. As is demonstrated by the operational Japanese LE-5B engine, such flow mixing is achievable (see MIX in Figure 2).

The considered cycle variants B and C of reference 2 assumed that both pumps (fuel and oxidizer) are to be placed on the same shaft and are driven by just one turbine. This resulted in highly demanding conditions for the turbomachinery. Although the obtained data of enthalpy and flow coefficients seemed to be generally acceptable, it has to be stated that such a lay-out is related to serious drawbacks with respect to efficiency, loads, and mass. They are therefore no longer under investigation.

In the cycles analyzed here, the coolant flow is tapped off from the main hydrogen flow behind the impeller. This portion of the stream is boosted to a considerably higher pressure level by a second fuel pump. Although this secondary high pressure impeller is on the same driving shaft, it can not be regarded as a second stage, since the mass flow is about 80 % lower. Figure 5 shows the basic design (variants **D** through **G**). The configuration features a separate turbine and shaft for fuel and oxidizer pump. The two turbines are fed in parallel.

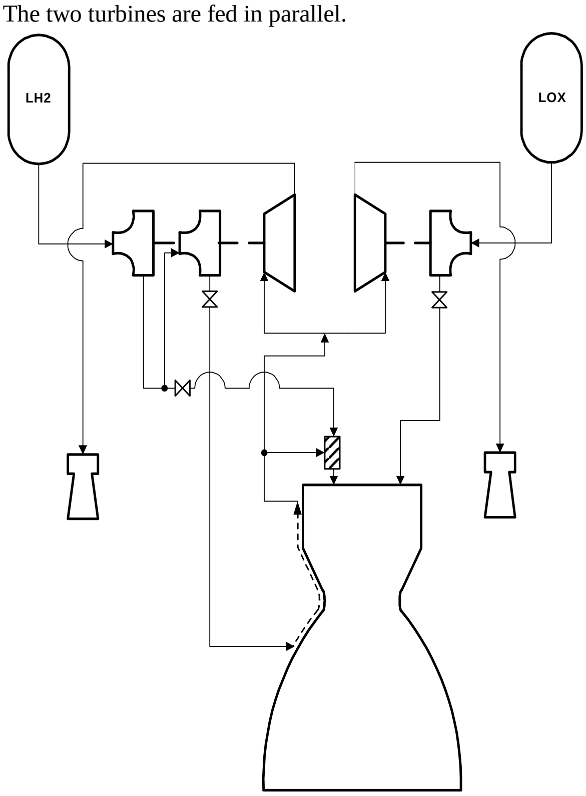

Figure 5: Schematic of cycle variants D through G (dual turbopump with parallel flow feed, secondary hydrogen impeller for coolant flow, supercritical H2-injection)

5 ENGINE ANALYSIS RESULTS

The expander-bleed cycle engine shown in Figure 5 has the DLR internal designation SE-21. A description of both engine variants follows shortly. These cycles differ not only in the assumptions made in the design of the regenerative cooling channels of the thrust chamber, but also in the design philosophy followed. The DLR analysis is based on a simple heat-transfer model for sizing the chamber (version **D**) [1]. EADS has completed more sophisticated chamber design work based on their experience in the European cryogenic engines like VINCI and in the German ASTRA research program. The calculations were made for the latter for a similar high thrust motor, but had to be adapted to the requirements of the present investigation. This design, performed under slightly different sizing constraints, is called version **G**. The intermediate designs E and F are similar to variant D, but with increased engine mixture ratio. They are not described here, but their data can be found in ref. [2, 3].

The influence of unsteady flow conditions and the turbopump's dynamical behavior is not included in this early design investigation.

5.1 High Heat-Transfer Thrust Chamber Design (variant D)

Parametric calculations assuming moderate turbo-machinery efficiency showed that a heat-transfer of at least 90 MW is necessary to sustain the preferred chamber pressure of 65 bar [1, 2]. In order to obtain this heat-transfer while limiting the pressure loss, a large characteristic chamber length of about 4 m has been chosen. In combination with the low nozzle expansion ratio, this has led to an unusual shape for a large cryogenic engine.

As previous DLR studies have shown, truncated ideal nozzles promise a lower divergence angle, while the nozzle mass is about the same as with the thrust optimized parabolic nozzle [12]. Furthermore it seems that parabolic nozzles generate higher side loads due to different flow separation behavior [13]. Thus an ideal contour is chosen for the engines analyzed in this paper. The drawback of this contour is an increased length, compared to the parabolic nozzle.

Figure 6 shows the engine's combustion chamber and nozzle contour generated by the tool NCC [7] in a CAD rendering. The contour design and the heat-transfer analysis of ref. [1] delivered the geometry specifications listed in Table 2.

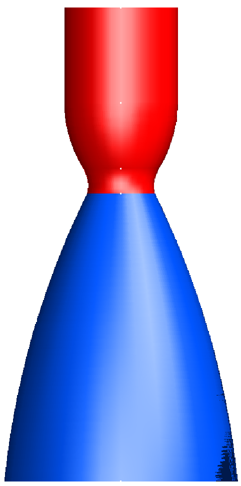

Figure 6: High heat-transfer combustion chamber and nozzle geometry of the proposed expander bleed-cycle engine (CAD rendering)

Chamber diameter	0.985 m
Subsonic chamber length	1.582 m
Char. chamber length l*	4.0 m
Chamber volume	1.029 m³
Throat radius	0.286 m
Exit diameter	2.024 m
Nozzle length	2.548 m

Table 2: DLR Bleed Cycle engine geometry specifications of combustion chamber and nozzle

The computation of the high heat-transfer engine started with the parameters specified above and with assumed overall pump efficiency of 0.7 and turbine effectiveness of 0.4. The corresponding internal engine flow data of [2, 3] is used in a preliminary turbomachinery analysis, which checks for feasibility and for achievable efficiencies as function of specific speed (ratios) based on data obtained from [8, 17, 18]. Incorporating two separate shafts in bleed cycle D allows an increase in the hydrogen turbopump rotation speed w.r.t. the oxygen side. Still being cautious, turbine effectiveness can be raised to 45 % while efficiencies of the oxygen and high pressure fuel pump increase to at least 75 %. On the fuel side a partial admission turbine [18] seems to be an attractive solution.

The rocket engine's nominal mass flow is fixed to 550 kg/s. As can be seen from Table 3 turbine entry temperature is about 500 K, which is advantageous in case of a reusable engine. A pre-selected turbine exit pressure of 0.3 MPa enables supersonic exit conditions of the secondary flow at sea-level.

	SE-21 D
Engine mixture ratio [-]	4.87
Combustion chamber mix. ratio [-]	5.5
H2 Pump power [MW]	16.5
O2 Pump power [MW]	4.3
Turbine mass flow [kg/s]	10.2
TET [K]	506
Turbine exit pressure [MPa]	0.3
Sea level impulse [s]	360.9
Vacuum impulse [s]	407.2
Sea level thrust [kN]	1947
Vacuum thrust [kN]	2196

Table 3: Performance comparison of the high heat-transfer bleed cycle D

The thermodynamic conditions of the high heat-transfer D-cycle are depicted in the flow chart of Figure 7. As can be seen, the H2 temperature at injection reaches 63 K under nominal conditions.

Due to the high circumferential velocity, the hydrogen impellers are assumed to be of Titanium alloy, while an aluminum construction seems to be possible for the oxygen pumps [20]. A total turbomachinery mass of around 500 kg is estimated with a total engine mass at about 3000 kg.

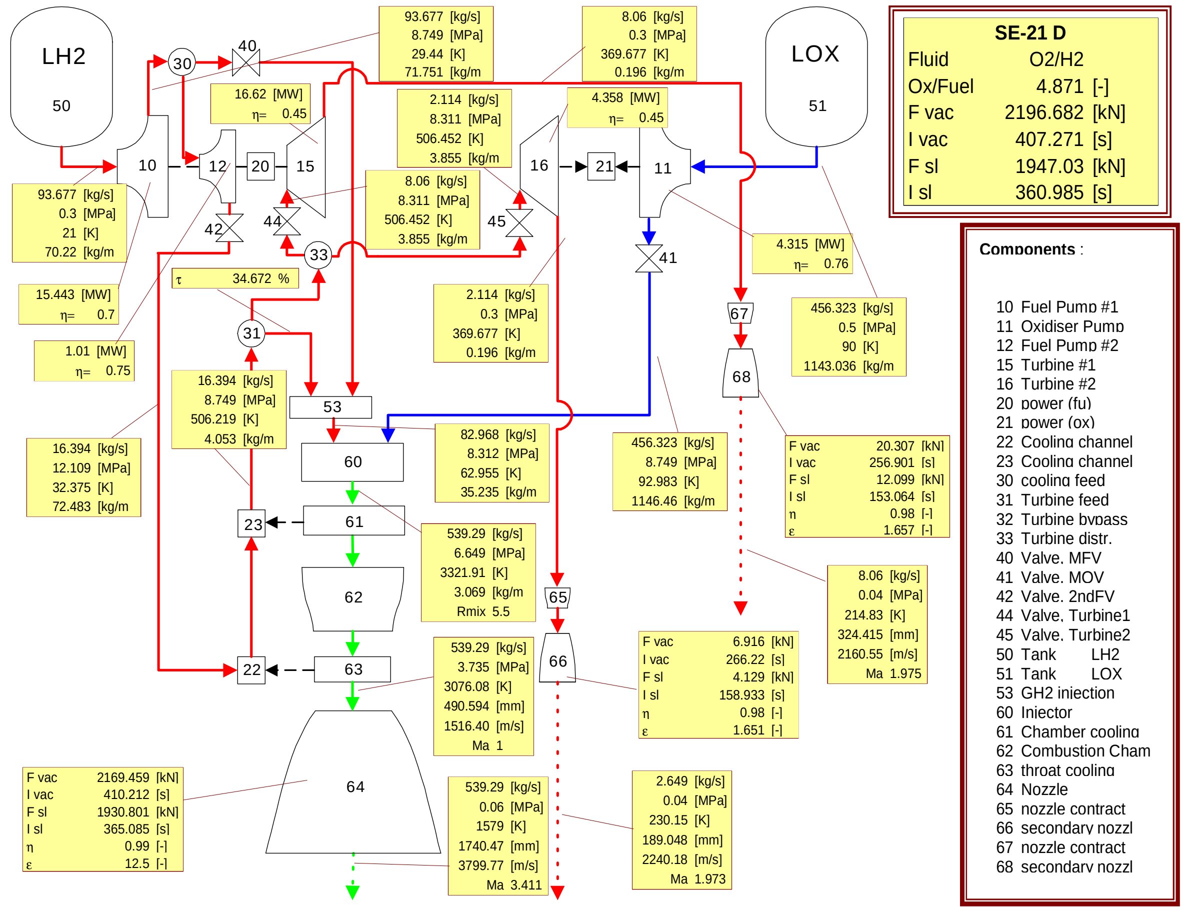

Figure 7: Thermodynamic conditions inside expander bleed cycle engine variant D taking into account iterative estimation of turbopump efficiency

5.2 Restricted Heat-Transfer Thrust Chamber Design (variant G)

At this point of the investigation, the actually achievable heat-transfer from the combustion chamber to the turbine driving fluid is to be checked. Unfortunately, no in-depth analysis has been performed yet for the chamber design of Figure 6. Instead as a baseline, a detailed regenerative channel lay-out of EADS for a similar cycle engine, but for a different application in frame of the ASTRA system study is used. During these system studies the possibility of an expander-bleed cycle as candidate for a future low-cost engine for the Ariane 5 EPC was examined with the aim of evolving it into a reusable version. This intended application requires an increased mixture ratio but reduced mass flow compared to engine variant D.

Several combustion chamber concepts were designed for a range of available bleed mass flow rates and cooling inlet conditions and their characteristics necessary for effective heat-transfer were studied. Most of the general design input data, such as required thrust or engine mixture ratio range, were sufficiently close to those required in this investigation. The bleed mass flow rate used for cooling the combustion chamber varied between 10 kg/s and 15 kg/s, equal to 17.5 and 26 % of the injected fuel mass flow rate. Contour and cooling channel design was based on the technological limits of combustion chamber design and cooling and was geared towards maximizing the total heat-transfer while keeping the pressure losses reasonable.

Under the conditions of Table 4 the available total heat flux, depending on the coolant flow direction, was found to be around 68-70 MW. Since the nozzle expansion ratio of this paper's application for booster engines is less than 15, the application of a regenerative nozzle extension for total heat flux enhancement renders ineffective. The coolant pressure loss depends strongly on the available coolant mass flow. For a coolant flow rate of more than around 20 % of the injected fuel flow rate, the pressure loss is in the range of 30-40 bar, but grows over-proportionally for coolant flow rate percentages below this threshold.

Vacuum thrust	2000	kN
Chamber pressure at throat	90	bar
Injected mass flow rate	460.8	kg/s
Injected mixture ratio	7.05	-
Throat diameter (approx.)	378.5	mm
Injector diameter (approx.)	574	mm
Chamber contraction ratio A_c/A_t	2.3	-
Chamber characteristic length l*	1300	mm
Area ratio @ regenerative exit	5	-
Chamber overall length (approx.)	1000	mm

Table 4: Expander-bleed combustion chamber parameter (of previous ASTRA system study)

These earlier EADS-design-results serve as a reference for scaling the heat-transfer to the booster engine's requirements. The heat-transfer scaling law is derived from proportionality considerations using general Nusselt number correlations for the heat-transfer on the hot-gas side of the combustion chamber [21]. Figure 8 shows the relationship between the chamber pressure and the heat-transfer.

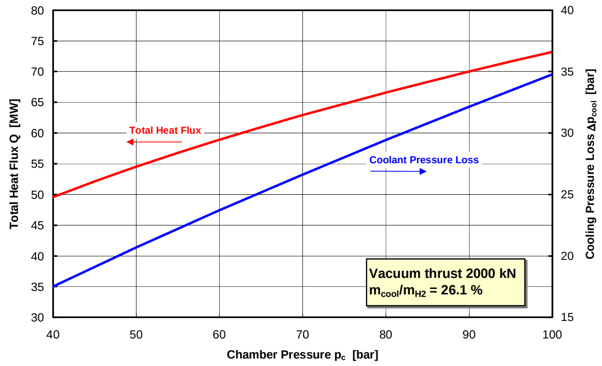

Figure 8: Typical heat flux and coolant pressure loss for expander-bleed combustion chambers

The cooling pressure loss depends on the coolant channel design, which is adapted to the heat-transfer layout. This loss has to be calculated by designing the cooling system for each specific case in order to get reliable results. The scaling of the pressure drop as shown in Figure 8 is restricted to a first-level approximation for preliminary engine cycle calculations.

The characteristic chamber length l* is considerably reduced to only about 1.3 m similar to the VINCI chamber. The nozzle and thrust chamber lay-out is shown in Figure 9 and geometry data are listed in Table 5.

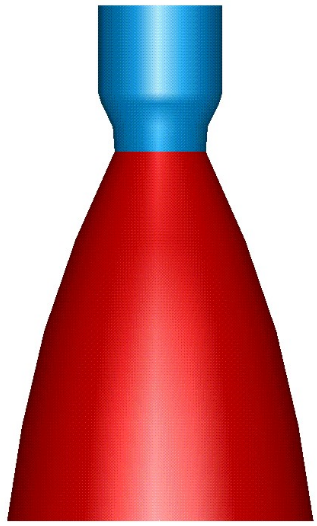

Figure 9: EADS designed combustion chamber and nozzle geometry of the proposed expander bleed-cycle engine (CAD rendering)

Chamber diameter	0.628 m
Subsonic chamber length	0.734 m
Char. chamber length l*	1.3 m
Chamber volume	0.202 m³
Throat radius	0.222 m
Exit diameter	1.577 m
Nozzle length	1.959 m

Table 5: Bleed Cycle engine geometry specifications of combustion chamber and nozzle based on EADS chamber design

The limited heat flow makes a convergent cycle design more difficult, if the constraint of supercritical hydrogen injection is maintained. Some parametrical analyses have been performed to find the sensitivities of hydrogen pre-injector mixing temperature and chamber pressure on required turbine efficiency. This calculation is based on the assumption of a constant pump effectiveness of 70 %. Variation of the temperature from the critical point at 33.14 K up to 55 K shows a necessary increase in η_T by more than 15 % (Figure 10, top). For the purpose of combustion stability, a minimum mixing temperature of 50 K is assumed and the engine chamber pressure is varied with its corresponding heat-transfer and regenerative flow pressure losses. A minimum in turbine efficiency demand can be found close to the SE-21 **D** chamber pressure of 65 bar. (Figure 10, bottom) The assumption of a feasible H2 injection temperature of 50 K is quite optimistic. However, successful operation of the LE-5B engine has been demonstrated at 55 K [9]. Therefore, a design with adequate characteristic chamber length is worth to be tested at these low injection temperature. (See also the remarks on cold hydrogen injection in chapter 5.3 !)

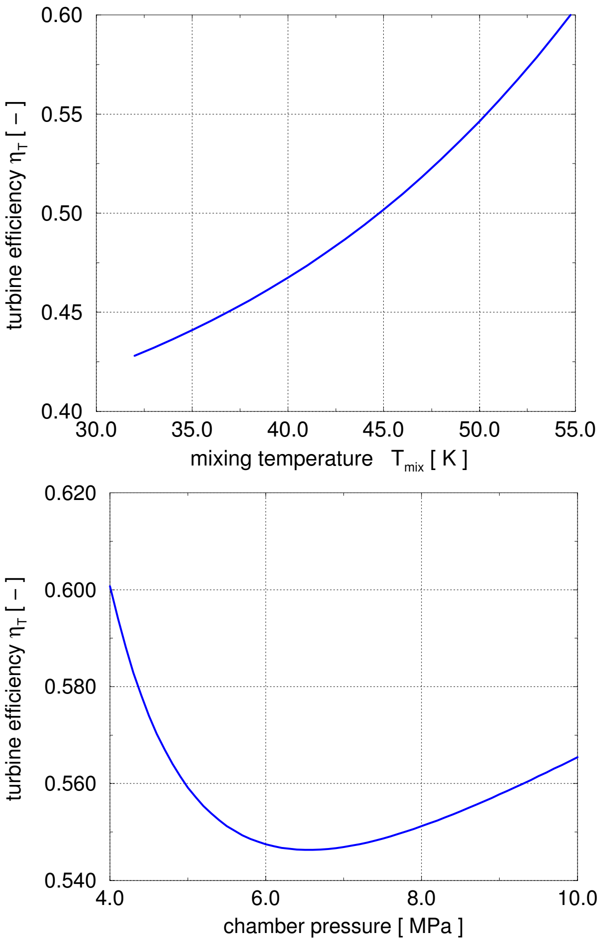

Figure 10: Minimum required turbine efficiencies as a function of pre-injection hydrogen mixing temperature (P_C= 6.5 MPa = const.) and chamber pressure (T_{mix}= 50 K = const.)

It has to be stated that the turbopump, hence turbine efficiency has to surge from 45% of cycle D to at least 55 %. (Figure 10, bottom) This fact poses challenging design issues for the expander bleed cycle with the restricted heat-transfer of cycle G. A preliminary assessment of the turbomachinery based again on the design criteria of [8, 17, 18] indicates that an impulse turbine operation area extents to the required effectiveness range. While pump effectiveness might surge compared to the original assumptions, it will be challenging to reach the required 55 % efficiency of the turbines. Were a partial admission design to be used for the hydrogen turbine, an η of 55% seems achievable. The oxygen turbine reaches at best an efficiency percentage in the low fifties. The situation on the oxygen side will be slightly eased by an increased pump effectiveness of 80 %, which is well within reach due to the favorable specific speed.

Despite all technological effort, the hydrogen combustion chamber injection temperature does not exceed 50 K. For the intended use as a reusable engine, one has to take into account thrust chamber cracks occurring during the operational life. Although this is not a general problem of safety, in the case of the G-variant the H2-injection temperature would drop further, raising serious concerns of combustion stability. Should it not be possible to improve the amount of chamber heat-transfer, the realization of the proposed SE-21G is put into question.

Engine specific impulse of that heat-flow restricted engine (G-version) (Table 6) is below that of SE-21D, which is due to the increased mixture ratio. Note the low TET of 300 K resulting from the reduced heat-flow. Turbo-machinery mass is estimated at 420 kg, while total engine mass is in the range of 2800 kg.

	SE-21 G
Engine mixture ratio [-]	5.85
Combustion chamber mixture ratio [-]	7.0
H2 Pump power [MW]	12.5
O2 Pump power [MW]	3.8
Turbine mass flow [kg/s]	11.3
TET [K]	301
Turbine exit pressure [MPa]	0.4
Sea level impulse [s]	352.6
Vacuum impulse [s]	396.5
Total engine mass flow [kg/s]	471
Sea level thrust [kN]	1628
Vacuum thrust [kN]	1831

Table 6: Performance comparison of the different bleed cycles with different heat-transfer

5.3 Remarks on Expander-Bleed Thrust Chamber Technology Issues

The combustion chamber for the expander-bleed cycle differs in some aspects from chambers developed so far. For that reason, some considerations on feasibility are made from an industrial point-of-view.

Enhanced heat-transfer

All the energy driving the propellant pumps comes from heating a portion of the fuel in the thrust chamber cooling. Therefore, the characteristics of this cooling are of utmost importance for the whole engine system performance. Several technologies like lengthened combustion chamber, ribbed chamber wall, and increased wall roughness are currently under study by EADS in frame of the national ASTRA technology program aiming at increasing the heat-transfer from the combustion products to the chamber wall [22]. However, the increased heat flux needs to be transferred to the coolant effectively, which is usually coupled with an upsurge of the coolant pressure drop. Currently, no technology is readily available which would allow to increase the heat-transfer to the coolant without significantly augmenting pressure loss.

Cold hydrogen injection

Splitting of the hydrogen flow to the main injector and to the chamber cooling, results in hydrogen temperatures in the main injector well below those appearing in current thrust chambers like Vulcain or HM-7. The experience from development testing showed that a reliable ignition becomes a problem for hydrogen temperatures below approximately 70 K. The hydrogen temperature at ignition is usually established by the transient cooling of the chamber structure when the hydrogen begins to pass the coolant channels. This transient heating could be employed by rerouting part of the coolant to the main injector and mixing it with the hydrogen fed directly to the main injector. Besides the ignition problematic the combustion stability is also influenced by the fuel temperature. Colder hydrogen tends to stimulate combustion instabilities. Therefore, mixing part of the heated hydrogen to the injector flow to increase the overall fuel temperature is mandatory. A solution to this issue must provide for temperatures above 70 K and homogeneous mixing. At the same time, it has to provide sufficient mass flow for the turbines to drive the engine cycle and allow regulation of the flow split. The equipment required might cause additional pressure losses. Currently, no straightforward solution to this issue seems to be available.

Large chamber size

Due to the high thrust demand and the low chamber pressure to be expected, the combustion chamber size will significantly exceed the size of the Vulcain chamber. Dimensions given in Table 4 are related to a chamber pressure of 90 bar, which may not be possible to reach with the expander bleed cycle. For lower chamber pressures the chamber dimensions grow even further, as can be seen from Table 2 and Table 5.

The combustion stability needs to be assured carefully in specific test programs. Modifications of the injection and application of acoustic cavities and/or baffles may become necessary. Some experimental experience exists in Europe for such stability-enhancing devices from earlier development programs. But, the large size of the booster engine’s chamber surpasses the existing experience.

6 BOOSTER VEHICLE DESIGN

The viability of the expander bleed cycle engine for boost vehicle applications is demonstrated in a launch vehicle systems analysis. Previous DLR-studies [14, 15] have shown that an Ariane 5 ECB version with a replacement of its powerful solid rocket-motors by reusable liquid propellant engines is able to deliver a heavy payload mass into geostationary transfer orbit (GTO).

The assessed, partially reusable space transportation system consists of a dual booster stage, which is attached to the expendable Ariane 5 core stage (EPC) on an upgraded future technology level. This stage is assumed here to be powered by a single advanced derivative of the Vulcain engine with increased vacuum thrust, and contains about 185000 kg of slightly 'subcooled' propellants. Decreasing the temperature of cryogenic oxygen and hydrogen below their respective boiling points is able to augment the density of the propellants. Although this 'subcooling'-technology is not in use with today's launchers, it seems to be in reach for the next generation of launch vehicles. Studies on future reusable and expendable (e.g. Ariane 2010 [16]) space transportation largely employ 'subcooling' of cryogenic fuels within their system investigations. Oxygen of 75 K has a density of more than 1200 kg/m^3 and hydrogen of 15 K reaches around 76 kg/m^3. The lower vapor pressures of these fuels, increase engine NPSP. If the pump entry pressure is unchanged an elimination of the inducers can be examined. Such a decision however requires a more detailed analysis. A second option might be reducing the tank pressurization in order to save stage mass.

A new cryogenic upper stage (ESC-B) is already in the development phase. It should include a new *closed* expander cycle motor of 180 kN class (VINCI) later in the decade. This study assumes a total propellant load of the upper stage of about 27000 kg.

The common mission of commercial Ariane 5 flights will continue to be operated from Kourou to a 180 km x 35786 km GTO with an inclination of 7 degrees. This orbit data and a double satellite launch including the satellite support structure SPELTRA is assumed as a basis for this research analysis.

Both reusable boosters are attached to the sides of the core stage using the same joints as the solid boosters (Figure 11). Since each expander bleed cycle engine develops a sea level thrust of around 1950 kN, both boosters will carry two such engines each, to replace the 5400 kN sea level thrust EAP's solid rocket motors currently in use. Due to the lower propellant loading, the launcher T/W at lift-off still reaches around 1.3.

This study reuses a basic design of a reusable vehicle, as already described in [14, 15]: A cylindrical fuselage contains three separate tanks (one oxidizer in front and two fuel in the center and rear), two vertical stabilizers on top of the fuselage in V-configuration, and two rocket engines at the aft protected on their lower side by a body-flap. The investigated configuration is similar to the latest LFBB-wing design of the German ASTRA-study, including canards to achieve superior trimmability during the return flight. The first and the second tank are of integral structure, the third is a separate fabrication, mounted on top of the wing-carry-through. In case of the LFBB powered by the expander-bleed engine, an intertank structure between the main LOX and LH2 tanks serves as the attachment point to the EPC. The booster design contains three military air-breathing engines without afterburner in the nose section for the return flight. They are powered by hydrogen. The complete lay-out regards near term technology and avoids comprehensive development programs. The simple geometry intentionally sacrifices some aerodynamic efficiency to structural strength and hence weight reduction.

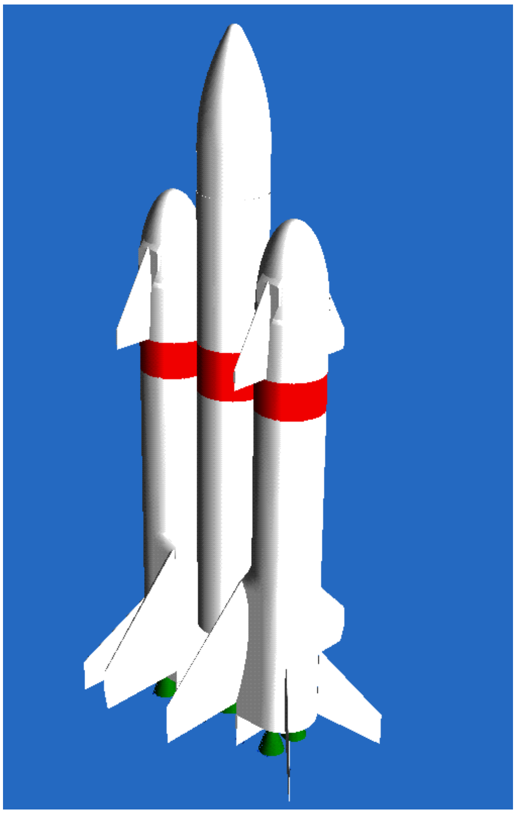

Figure 11: Launcher configuration with two LFBB with two expander bleed cycle engines each

6.1 Size and Mass of the LFBB

The selected bleed cycle engine mounted in the reusable stage is variant SE-21 D as shown in Figure 7. The alternate G-version's thrust is too low to be investigated with this launcher configuration. The low mixture ratio of 4.8 requires large liquid hydrogen tanks. If the same tank diameter as for the EPC is used, the overall length reaches more than 44 m. Some of the stage design parameters are listed in Table 7. Note the structural index of nearly 0.27 reflecting conservative, near-term technology assumptions for this reusable vehicle.

Liquid Fly-back Booster SE-21 D	GTO mission
overall length	44.5 m
fuselage diameter	5.45 m
wing span	21.2 m
	mass [kg]
stage mass empty (incl. marg.):	**47345**
Ascent propellant	**170000**
stage structural index	**0.266**
GLOW stage mass:	**225224**

Table 7: Dimensions and estimated component masses of one twin LFBB with bleed cycle engines (variant D)

Stage separation of the liquid fly-back booster occurs close to 2 km/s. For GTO a payload mass above 12000 kg is achieved from trajectory optimization. This is in the same class as other investigated reusable stages including gas-generator or staged combustion cycles motors. [14, 15]

Since the LFBBs are designed to be reusable they have to return to the launch site in powered flight and by aerodynamic flight maneuvers. The return trajectory is generated with the constraints of a maximum normal load of 3.5 g and a maximum dynamic pressure of 60 kPa. The complete powered return flight is controlled along an optimized flight profile, always directly heading to the launch site. More data on similar ascent and descent trajectories is provided in [1, 14, 15].

7 CONCLUSIONS

A bleed cycle engine lay-out is proposed, which combines low turbine entry temperature with high overall efficiency. The design intends to inject hydrogen at a supercritical state into the combustion chamber by mixing a fraction of the regenerative flow with the main stream. Although already realized in the Japanese LE-5 motor, the viability of the mixing process is to be demonstrated for the size of a high thrust engine. The achievement of a sufficiently high heat-transfer with acceptable pressure losses is found to be highly critical for the realization of the expander bleed cycle. In case a severely restricted amount of heat is available to drive the turbines, the demand on the turbomachinery efficiencies is strongly increased. Consequently, questions of improved heat-transfer, combustion stability for low temperature hydrogen, and mastering of the very large chamber size should be investigated theoretically and experimentally.

The performance analysis demonstrates that bleed cycle engine designs, producing a sea level thrust of nearly 2000 kN at moderate chamber pressures, are promising candidates for future reusable first stages. The basic analysis of the implementation of high thrust bleed cycle engines for fly back booster operations, in support of a future upgraded Ariane 5 core stage, shows that a payload mass of more than 12000 kg for a geostationary transfer orbit can be achieved.

Although bleed cycle engines are more severely restricted in chamber pressure than gas-generator types, and therefore are subject to performance limitations, they are able to lift nearly the same heavy payload to orbit. On the other hand, they may offer reduced complexity and less demanding technology. If the open design questions could be successfully solved, bleed cycle engines can be viewed as an interesting alternative in introducing low-cost reusable rocket technology.

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