FEASIBILITY OF HIGH THRUST BLEED CYCLE ENGINES FOR REUSABLE BOOSTER APPLICATIONS

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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 with special regard to the critical heat-transfer in the thrust chamber region. Finally, an ascent trajectory optimization and a discussion of the complete vehicle are performed to evaluate the engine as part of a future semi reusable launch system.

Nomenclature

c* \text{characteristic velocity} m/s
C_F \text{thrust coefficient} -
c_p \text{specific heat (const. pressure)} J/(kg K)
F \text{thrust} N
g \text{gravitational acceleration} g = 9.8067 \text{m/s}^2
h \text{specific enthalpy} m^2/s^2
I \text{specific impulse} s
I' \text{characteristic chamber length} m
\dot{m} \text{mass flow} kg/s
M \text{Mach number} -
\text{NPSP} \text{net positive suction pressure} Pa
P \text{pressure} bar / MPa
\dot{Q} \text{heat flux} W/m^2
Q \text{total heat flux} W
r \text{radius (of curvature)} m
R \text{mixture ratio} -
T \text{temperature} K
TET \text{turbine entry temperature} K
u \text{circumferential velocity} m/s
v \text{velocity} m/s
V \text{volume} m^3

\epsilon \text{expansion ratio} -
\eta \text{efficiency} -
\kappa \text{specific heat ratio} -
\Pi \text{pressure ratio (p_{out}/p_{in})} -
\rho \text{density} kg/m^3
\Psi \text{coolant massflow ratio (coolant/total fuel)} -
\psi \text{enthalpy coefficient } 2
\Delta h/\dot{u}^2

\text{Subscripts}
M \text{Mach number} -
NPSP \text{net positive suction pressure} Pa
p \text{pressure} bar / MPa
P \text{power} W
q \text{heat flux} W/m^2
Q \text{total heat flux} W
r \text{radius (of curvature)} m
R \text{mixture ratio} -
T \text{temperature} K
TET \text{turbine entry temperature} K
u \text{circumferential velocity} m/s
v \text{velocity} m/s
V \text{volume} m^3

\text{INTRODUCTION}

Recent proposals concerning the introduction of reusable components in space transportation regard the first or booster stages. Such systems are usually called a 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. [13, 14] The abolition of solid propellant stages offers a potential reduction in operation cost, if replaced by a reusable booster, and increases mission flexibility.
Since 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 [13]. Another interesting option would be to develop a completely new engine with regard to reusability requirements.

This paper concentrates on the feasibility of high thrust bleed cycle engines able to power the LFBBs. In a bleed cycle the coolant flow is used to drive the turbines, after it has been heated while passing through the thrust chamber's cooling channels. Finally the turbine exhaust is expelled. Lower cost and higher reliability of the turbines are to be expected due to lower turbine inlet temperatures [3]. In previous DLR studies different engine design options had been investigated [1, 2]. A dual turbopump cycle arrangement is thoroughly investigated here, and the regenerative heat-transfer will be adopted in the course of a more detailed analysis.

Until now the Japanese LE-5 A and B engines are the one and only bleed cycle engines which ever reached hardware status. The LE-5A had successfully been used in the Japanese H2 launch vehicle since 1994. The more cost efficient LE-5B is in operation with the new H2A rocket since 2001. Thrust range is between 122 kN (LE-5 A) and 137 kN (LE-5 B). An important difference between the two designs is the considerably reduced TET of the LE-5B engine. [8]

**ANALYSIS METHODS EMPLOYED**

**Cycle Analysis Tool**

The program used for the cycle analysis presented in this paper, is based on the modular program SEQ [4] 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 components, which are passed by the fluid. Certain conditions can be linked to component settings (i.e. the program is told to change 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-mathematics subroutine [5].

For the cycles analyzed in this paper the following dependencies have been selected: The demand of the chamber pressure governs the pump discharge pressure. The pump power determines the turbine power and thereby the turbine mass flow and TET. The desired chamber mixture ratio regulates the engine mixture ratio.

**Preliminary Design and Mass Estimation**

The engine preliminary design is performed by thrust chamber and nozzle dimensioning [6], as well as one dimensional turbo-machinery lay-out. The latter 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 [9], [10].

**ENGINE CYCLE ANALYSIS**

**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 1.

Engine performance analysis led to the choice of a chamber pressure of 65 bar and an engine mass flow rate of 550 kg/s.
The initial selection of a mixture ratio of 5.5 is based on three considerations. Firstly a low chamber mixture ratio leads to a high specific impulse and secondly it creates a lower chamber temperature. Figure 2 shows a variation of the chamber mixture ratio and the relative changes of those two aspects. Thirdly the system study shows, that a lower chamber mixture ratio (which causes 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.

More data on the relationship between LH2 pump discharge pressure and turbine performance of the investigated bleed cycle are presented in [1].

**Description of the Investigated Cycles**

During the previous DLR analysis [1] it became apparent that the high heat-transfer rate needed by the original cycle with fixed mass flow distributions 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. Considering the whole system, higher mixture ratios have the advantage, to alter the mass distribution from the very low density LH2 to LOX. This decreases the vehicle length and mass.

The analyzed cycles of [1] would not tolerate a significantly higher engine mixture ratio, since the increasing heat-transfer had 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 first DLR studies on expander bleed engines [1] led to the idea of the cycle shown in Figure 3. 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.

![Figure 3: Schematic of basic bleed cycle variant A with two different H2-injection modes (sub-critical and super-critical)](image)

A slight alteration of the above lay-out is more deeply investigated. A small amount of the heated flow is to be mixed with the main stream of sub-critical hydrogen in front of the injector. 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. However, a mixing device or a sufficiently long feed line will be required to produce from the two streams a flow at homogenous super-critical conditions. It is demonstrated by the operational Japanese LE-5 A and B engines that such flow mixing is achievable, at least in principle. [8]

The simplest of the considered cycles (variant 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 pumps and turbines. The considerable difference in suction specific speed of cryogenic hydrogen and oxygen required finding a compromise in rotation speed. 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.

In all analyzed cycles here, the coolant flow is tapped off from the main hydrogen flow behind the impeller. Then 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 4 shows the basic design (variant D) of the considered cycles. It uses two fuel pumps, and it features a separate turbine and shaft for fuel and oxidizer pump. The two turbines are fed in parallel.
ENGINE ANALYSIS RESULTS

Two different engine variants will be described, which vary in the assumptions and design philosophy of the regenerative cooling channels of the thrust chamber. The DLR analysis is based on a simple heat-transfer model for sizing the chamber. [1] Astrium has completed much more sophisticated chamber design work based on their experience in the European cryogenic engines like VINCI and research in the German ASTRA program. The latter calculations were produced for a similar high thrust motor, but had to be scaled to the requirements of the present investigation.

Further decreasing the temperature of cryogenic oxygen and hydrogen is able to improve 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 [15]) space transportation largely employ 'subcooling' of cryogenic fuels within their system investigations. Oxygen of 75 K has a density of about more than 1200 kg/m³ and hydrogen of 15 K reaches around 76 kg/m³. The lower vapor pressure 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.

High Heat-Transfer Thrust Chamber Design

Parametric calculations assuming moderate turbomachinery performance showed that to sustain the preferred chamber pressure of 65 bar a heat-transfer of at least 90 MW is necessary. [1, 2] In order to obtain this heat-transfer, while creating a reasonably moderate pressure loss a large characteristic chamber length of around 4 m had been chosen. In combination with the low nozzle expansion ratio, this 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 [11]. Furthermore it seems to be evident, that parabolic nozzles generate higher side loads due to different flow separation behavior [12].
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 5 shows the engines combustion chamber and nozzle contour generated by the tool NCC [6] in a CAD rendering. The contour design and the heat-transfer analysis delivered the geometry specifications listed in Table 1.

![Figure 5: High heat-transfer combustion chamber and nozzle geometry of the proposed expander bleed-cycle engine (CAD rendering)](image)

<table>
<thead>
<tr>
<th>Figure 5: High heat-transfer combustion chamber and nozzle geometry of the proposed expander bleed-cycle engine (CAD rendering)</th>
</tr>
</thead>
</table>

Incorporating two separate shafts in bleed cycle D allows increasing the hydrogen turbopump rotation speed, reducing the machine's diameter. Although a second turbine and shaft has to be added, the overall mass calculated in this preliminary design was found quite similar between the cycles C and D. Engine mass is calculated at about 3000 kg. [2]

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

The computation of the high heat-transfer engine is performed with the parameters specified above and the assumption of an overall pump efficiency of 0.7 and a turbine effectiveness of 0.4. The rocket engine's nominal mass flow is fixed to 550 kg/s. As can be seen from Table 2 turbine entry temperature remains below 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.

| Engine mixture ratio [-] | 4.8 |
| Combustion chamber mix. ratio [-] | 5.5 |
| H2 Pump power [MW] | 15.15 |
| O2 Pump power [MW] | 4.3 |
| Turbine mass flow [kg/s] | 10.9 |
| TET [K] | 498 |
| Turbine exit press. [MPa] | 0.3 |
| Sea level impulse [s] | 360.3 |
| Vacuum impulse [s] | 406.6 |
| Sea level thrust [kN] | 1943 |
| Vacuum thrust [kN] | 2193 |

<table>
<thead>
<tr>
<th>Table 2: Performance comparison of the high heat-transfer bleed cycle D, 'subcooled' conditions at pump entry</th>
</tr>
</thead>
</table>

The thermodynamic conditions of the two shafts D-cycle with 'subcooled' propellants at the pump entry are depicted in the flow chart of Figure 6.
Figure 6: Thermodynamic conditions inside expander bleed cycle engine variant D, 'subcooled' propellants at inlet

**Change in Combustion Chamber Mixture Ratio**

An engine mixture ratio below 5.0 in a first stage or booster application is related to some drawbacks in the vehicle design. The low density of liquid hydrogen requires excessively large tanks. This is even more disadvantageous for reusable vehicles, which have to withstand an atmospheric reentry and return flight to the launch site. Therefore, some options are addressed to increase the combustion chamber mixture ratio.

In a first step two sub-variants of cycle D were defined in [2] which increased this value from 5.5 to 6.0: The first (E) holds the ratio of cooling mass flow to total fuel flow constant. Due to the increase in heat-transfer, the cooling channel pressure loss surges, and TET reaches 691 K. The second option (F) raised the portion of coolant flow, which in combination with a different chamber design allows for much lower TET and pressure losses. If a high TET close to 700 K is acceptable, the engine mixture ratio can grow up to 5.4, which is similar to many gas generator engines like Vulcain. The combination with a moderate TET allows for R= 5.2 but reduces the specific impulse by about 2 s.

At this point of the investigation, it has to be verified if the assumed high heat-transfer from the combustion chamber to the turbine driving fluid is actually achievable. A detailed regenerative channel lay-out of Astrium for a similar cycle engine, but different application in frame of the ASTRA system study, is used as a baseline. During these system studies the possibility of an expander-bleed cycle as candidate for a future low-cost engine for the Ariane 5 EPC with the aim of evolving it into a reusable version was examined.

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 like required thrust or engine mixture ratio range were similar 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 were based on the technological limits of combustion chamber design and cooling and geared towards maximizing the total heat-transfer while keeping the pressure losses reasonable.
<table>
<thead>
<tr>
<th>Parameter</th>
<th>Symbol</th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Vacuum thrust</td>
<td>$F_{vac}$</td>
<td>2000</td>
<td>kN</td>
</tr>
<tr>
<td>Chamber pressure at throat</td>
<td>$p_{c,th}$</td>
<td>90</td>
<td>bar</td>
</tr>
<tr>
<td>Injected mass flow rate</td>
<td>$m_{cc}$</td>
<td>460.8</td>
<td>kg/s</td>
</tr>
<tr>
<td>Injected mixture ratio</td>
<td>$(O/F)_{cc}$</td>
<td>7.05</td>
<td>-</td>
</tr>
<tr>
<td>Throat diameter</td>
<td>$d_{th}$</td>
<td>approx. 378.5</td>
<td>mm</td>
</tr>
<tr>
<td>Injector diameter</td>
<td>$d_{i}$</td>
<td>approx. 574</td>
<td>mm</td>
</tr>
<tr>
<td>Chamber contraction ratio</td>
<td>$A_{c}/A_{t}$</td>
<td>2.3</td>
<td>-</td>
</tr>
<tr>
<td>Chamber characteristic length</td>
<td>$L$</td>
<td>1300</td>
<td>mm</td>
</tr>
<tr>
<td>Regenerative exit @ area ratio</td>
<td>$\varepsilon_{regen}$</td>
<td>5</td>
<td>-</td>
</tr>
<tr>
<td>Chamber overall length</td>
<td>$L_{cc,tot}$</td>
<td>approx. 1000</td>
<td>m</td>
</tr>
</tbody>
</table>

Table 3: Expander-bleed combustion chamber parameter (of previous ASTRA system study)

Under the conditions of Table 3 the available total heat flux was found around 68-70 MW, depending on the coolant flow direction. 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 turns out in the range of 30-40 bar, but grows over-proportionally for coolant flow rate percentages below this threshold.

These earlier Astrium-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 [18]. Figure 7 shows the relationship between the chamber pressure and the heat-transfer.

Figure 7: 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. It 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 7 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. The nozzle and thrust chamber lay-out is shown in Figure 8 and geometry data are listed in Table 4.
The data of 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. Variation of the temperature from the critical point at 33.14 K up to 55 K shows a necessary increase in $\eta_T$ by more than 15 % (Figure 9, left). 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 original chamber pressure of 65 bar. (Figure 9, right)

Figure 9: Minimum required turbine efficiencies as a function of pre-injection hydrogen mixing temperature ($P_C= 6.5\, \text{MPa} = \text{const.}$) and chamber pressure ($T_{\text{mix}}= 50\, \text{K} = \text{const.}$)

It has to be stated that the turbopump, hence turbine efficiency has to surge from 40% of cycle D to at least 55 %. Therefore, the considerably lower heat-transfer of this analysis poses challenging design issues for the expander bleed cycle. A preliminary assessment of the turbomachinery based on the design criteria of [16, 17] indicates that an impulse turbine extents to the required effectiveness range. In case of the oxidizer turbine, a partial admission design can be considered, if necessary. Despite all technological effort, the comfortable margins of cycle D fade, if it is not possible to improve the amount of chamber heat-transfer.

Engine specific impulse of cycle G is below that of D (Table 5), which is due to the increased mixture ratio. Note the low TET of less than 300 K resulting from the reduced heat-flow.
<table>
<thead>
<tr>
<th></th>
<th>Cycle D</th>
<th>Cycle E</th>
<th>Cycle F</th>
<th>Cycle G</th>
</tr>
</thead>
<tbody>
<tr>
<td>Engine mixture ratio [-]</td>
<td>4.8</td>
<td>5.4</td>
<td>5.2</td>
<td>5.9</td>
</tr>
<tr>
<td>Combustion chamber mixture ratio [-]</td>
<td>5.5</td>
<td>6.0</td>
<td>6.0</td>
<td>7.0</td>
</tr>
<tr>
<td>H2 Pump power [MW]</td>
<td>15.15</td>
<td>15.7</td>
<td>14.8</td>
<td>10.9</td>
</tr>
<tr>
<td>O2 Pump power [MW]</td>
<td>4.3</td>
<td>4.38</td>
<td>4.35</td>
<td>3.8</td>
</tr>
<tr>
<td>Turbine mass flow [kg/s]</td>
<td>10.9</td>
<td>8.2</td>
<td>11.57</td>
<td>10.42</td>
</tr>
<tr>
<td>TET [K]</td>
<td>498</td>
<td>691.7</td>
<td>463</td>
<td>297</td>
</tr>
<tr>
<td>Turbine exit pressure [MPa]</td>
<td>0.3</td>
<td>0.3</td>
<td>0.3</td>
<td>0.4</td>
</tr>
<tr>
<td>Sea level impulse [s]</td>
<td>360.3</td>
<td>358.5</td>
<td>356.7</td>
<td>352.8</td>
</tr>
<tr>
<td>Vacuum impulse [s]</td>
<td>406.6</td>
<td>404.1</td>
<td>402.3</td>
<td>396.6</td>
</tr>
<tr>
<td>Total engine mass flow [kg/s]</td>
<td>550</td>
<td>550</td>
<td>550</td>
<td>471</td>
</tr>
<tr>
<td>Sea level thrust [kN]</td>
<td>1943</td>
<td>1934</td>
<td>1924</td>
<td>1630</td>
</tr>
<tr>
<td>Vacuum thrust [kN]</td>
<td>2193</td>
<td>2180</td>
<td>2170</td>
<td>1832</td>
</tr>
</tbody>
</table>

Table 5: Performance comparison of the different bleed cycles with different heat-transfer and 'subcooled' conditions at pump entry

**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 from an industrial point-of-view are given.

**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 Astrium in frame of the national ASTRA technology program aiming at increasing the heat-transfer from the combustion products to the chamber wall [19]. 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 significant pressure drop increase.

**Cold hydrogen injection**

Splitting 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 3 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 1 and Table 4. 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.
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 [13, 14] have shown, that an Ariane 5 ECB version with a replacement of its powerful solid rocket-motors by reusable, existing liquid propellant engines, is able to deliver almost the same payload into geostationary transfer orbit (GTO).

The regarded, 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 powered by a single advanced derivative of the Vulcain engine with increased vacuum thrust, and contains about 185000 kg of slightly 'subcooled' propellants. 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) by 2006. 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 SPERTLA 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. 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.

Figure 10: Launcher configuration with two LFBB with two expander bleed cycle engines each

This study reuses a basic design of a reusable vehicle (Figure 10), as already described in [14]: 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 updated 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. 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.

**Size and Mass of the LFBB**

The selected bleed cycle engine mounted in the reusable stage is variant D as shown in Figure 6. 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 6. Note the structural index of nearly 0.27 reflecting conservative, near-term technology assumptions for this reusable vehicle.

<table>
<thead>
<tr>
<th>Liquid Fly-back Booster SE-21 D</th>
<th>GTO mission</th>
</tr>
</thead>
<tbody>
<tr>
<td>overall length</td>
<td>44.5 m</td>
</tr>
<tr>
<td>fuselage diameter</td>
<td>5.45 m</td>
</tr>
<tr>
<td>wing span</td>
<td>21.2 m</td>
</tr>
<tr>
<td>stage mass empty (incl. marg.)</td>
<td>47345 kg</td>
</tr>
<tr>
<td>Ascent propellant</td>
<td>170000 kg</td>
</tr>
<tr>
<td>stage structural index</td>
<td>0.266</td>
</tr>
<tr>
<td>GLOW stage mass</td>
<td>225224 kg</td>
</tr>
</tbody>
</table>

Table 6: 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. The corresponding achieved payload mass for GTO from trajectory optimization is above 12000 kg. This is in the same class as other investigated reusable stages including gas-generator or staged combustion cycles motors. [13, 14]

Since the LFBB 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, 13, 14].

**CONCLUSIONS**

A bleed cycle engine lay-out is proposed, which combines low turbine entry temperature with high overall efficiency. It is intended 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. It is found highly critical for the realization of the expander bleed cycle to achieve a sufficient heat-transfer with acceptable pressure losses. The considerably lower amount of heat available of a more detailed chamber analysis compared to the original assumptions strongly increased the demand on the turbomachinery efficiencies. Selection of preliminary turbine architecture should be addressed in future studies. Further, questions of improved heat-transfer, combustion stability for low temperature hydrogen, and of mastering 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 motor technology.
REFERENCES


Further updated information concerning the SART space transportation concepts is available at: [http://www.dlr.de/SART](http://www.dlr.de/SART)