Component Analysis of TBCC Propulsion for a Mach 4.5 Supersonic Cruise Airliner

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Abstract

This paper describes the status of the study on component analysis for the different Variable TBCC cycle configurations. The paper investigates different Variable Cycle TBCC configurations and compares them with an advanced turbojet for the generic configuration of a Mach 4.5 supersonic passenger airliner.

One VCE engine variant and the turbojet are preliminarily designed and their mass including air-intake and nozzle is estimated. The air-intake has been preliminarily sized and pressure recovery and mass flow in design and off-design conditions is estimated. The intake's dimension and airflow data have been subsequently delivered for further analyses. Engine weight analysis is also conducted both for existing engines and proposed LAPCAT turbo engines.

Nomenclature

D_2	Diameter of diffuse exit
Isp	(mass) specific impulse
L	Length of subsonic diffuser part
M	Mach number
T	Thrust
W	Weight
m	Mass
sfc	Specific fuel consumption
ρ	Density

Subscripts, Abbreviations

BPR	Bypass ratio (1 for front 2 for rear side)
FEM	Finite Element Method
HP	High pressure
HPC	High pressure compressor
HPT	High pressure turbine
HSCT	High speed civil transport
JP	Jet propellant
LAPCAT	Long-Term Advanced Propulsion Concepts and Technologies
LP	Low pressure
LPC	Low pressure compressor
LPT	Low pressure turbine
OPR	Overall pressure ratio
RTA	Revolutionary Turbine Accelerator
SST	Supersonic transport
TBCC	Turbine based combined cycle
TET	Turbine entry temperature
TSTO	Two stage to orbit
VCE	Variable cycle engine
0,0	See-level static

1. Introduction

The EU sponsored LAPCAT study investigates different types of advanced propulsion systems for supersonic and hypersonic cruise airplanes [1]. One of the most promising reference vehicles is a supersonic airliner with cruise velocity of Mach 4.5 called LAPCAT-M4.

A turbine based combined cycle (TBCC) is the natural propulsion system for high speed passenger transport, because one can exploit the tremendous experience in aircraft jet engines. A combination of turbo-engines with RAMjet and kerosene propellant is foreseen as the LAPCAT-M4 propulsion system. Turbojet as well as Turbofan cycle engines had been in operation with the first generation of supersonic passenger airplanes, the Concorde and the Tupolev Tu-144. However, these engines with a fixed cycle had insufficient efficiencies for operation in the full flight regime and thus limited flight range. The demands of subsonic to supersonic flight at very high speed require adaptations of the thermodynamic cycle in order to improve operational efficiency. Variable Cycle Engines (VCE) might offer a good compromise for such applications. Therefore, VCEs have been under study for more than a decade. The US started investigations already in the 1970ies and recently pushed this technology for military and space functions (RTA, Revolutionary Turbine Accelerator) [2].

The preliminary design of LAPCAT-M4 [3] is based on a critical recalculation of a 1990 NASA Langley and Lockheed study [4] of a 250-passenger, Mach 4 high-speed civil transport with a design range of 6500 nautical miles (12045.8 km). The LAPCAT mission range Brussels to Sydney is highly ambitious and by almost 40 % larger than NASA's 12000 km, which requires a re-design.

The new supersonic cruise airplane has to be considerably enlarged compared to the earlier NASA design to meet its ambitious range requirement. To keep the wing loading in an acceptable range the wing size has been increased to 1600 m^2 (+ 36%). The span grows almost proportionally by 16 %, while the total length reaches 102.78 m which is only slightly longer (+ 8.8 %) than the earlier HSCT proposal.

The general arrangement of the generic airplane geometry is illustrated in Figure 1. Four advanced turbo-RAM-jet engines are mounted in two nacelles on the wing lower surface adjacent to the fuselage. The location of the engine and nacelles is still open for adaptation if required by trim as long as they remain under the wing. The axial-symmetric geometry and the size of the air-intakes, nacelles and nozzles as shown in Figure 1 are not representative of the actual LAPCAT design. A rectangular shape of the air-intake with vertical ramps as in [4] is the preferred design option.

The large, slightly inclined wing might help to achieve a good maximum L/D of 7.8 at a small angle of attack and cruise Mach number 4.5 according to preliminary DLR-analysis. Actually, a high L/D is essential to achieve the ambitious range requirement.

The total take-off mass of the supersonic cruise airplane has been iterated in a two loop approach to the huge value of 720 Mg, which is well beyond any supersonic passenger aircraft built to date. The dry mass is estimated at 184.5 Mg and the structural index is at a for airplanes low 36.8 %. A sample trajectory calculation has been conducted without the engine mass taken into account using the VCE-214 engine configuration as propulsion device. Figure 2 shows variation of calculated fuel consumption per range with time. From the figure, there are two acceleration phases with succeeding quasi-plateau for each case. The former is initial acceleration with subsonic cruise, and

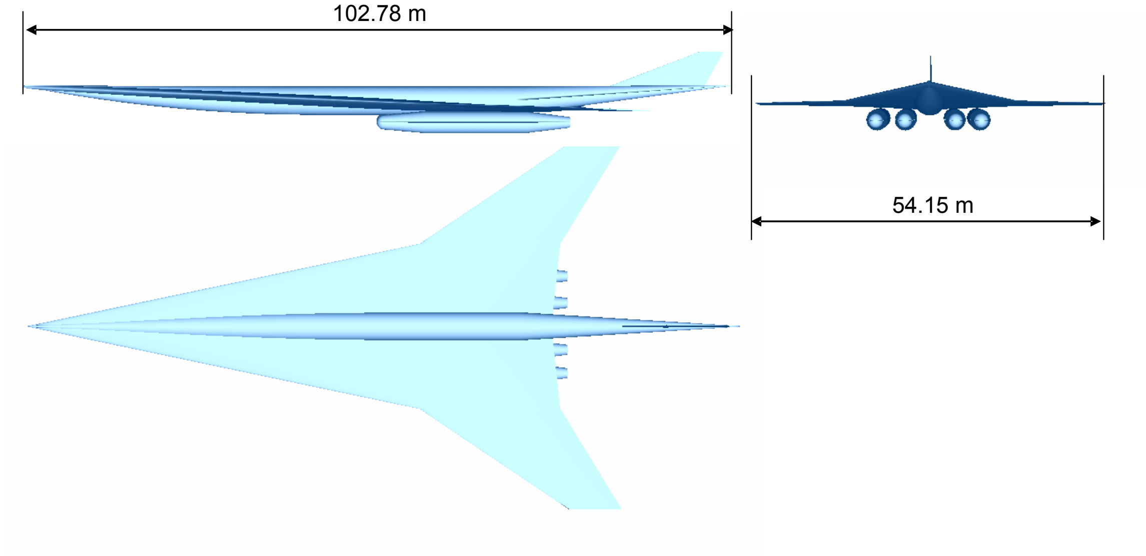

Figure 1: Generic Mach 4.5 supersonic cruise airplane of LAPCAT study (nacelle design not representative)

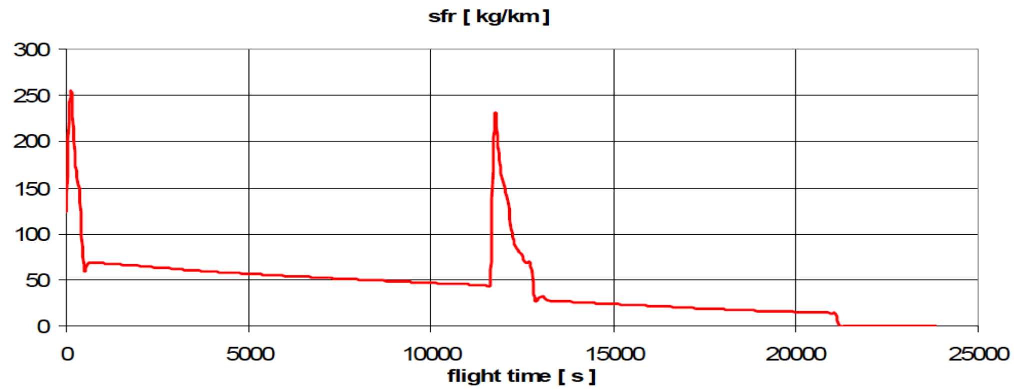

Figure 2: LAPCAT-M4 fuel consumption per range as a function of flight time [3]

the latter is afterburner-lit acceleration with supersonic ram cruise. This former subsonic cruise time is scheduled due to prevent noise problem above densely populated area, and if the noise concern is to be ignored better solution is possible. One important notification is that operation time of turbine engine at supersonic speed is relatively small, compared to the ratio of about half (supersonic cruising time is about half of the flight time) for the Concorde case. Most of the flight time are occupied either subsonic cruise or supersonic ram cruise. According to current data the HSCT would be able to transport about 200 passengers with their luggage. More data on the LAPCAT-M4 airliner design is published in reference [3].

2. Analysis

Construction

Through the TBCC engine analyses, the DLR air-breathing propulsion analysis computer program abp [5] has been utilized. The abp has the capability of analyses on simple cycle description and of geometrical design features both intake and turbo engine including nozzle. The abp is a fairly integrated tool, and specific composition can be separately used. In this meaning, the abp can be sub-divided into three categories; abp(-main) (or simply abp), abpi and abpw.

Intake analysis tool - *abpi*

The abpi is a highly generalized purpose code for two-dimensional supersonic intake and the size and performance are calculated almost automatically. Aerodynamic flow-path in the compression part is calculated using wedge shock relationships. By giving an appropriate loss factors, the leading edge separation shock and other features introducing losses are estimated. The function for the geometrical configuration design automatically generates flow-path which is adequate both in aerodynamic and structural meaning. For weight estimation, a material can be specified for each section, and several limiting values such as maximum allowable displacement are to be specified. To fulfill the geometrical designing, structural features such as materials and allowable deformation rate are to be input. The abpi has a function to generate a transition duct from the rectangular shape to circular shape, but abpi needs information such as offset, length scale and exit area as input parameters.

Therefore, the information for diffuser design in this paper is introduced independently referencing a report by Murakami [6]. Murakami [6] obtained a set of empirical expressions on the effect of geometry on pressure loss through the subsonic diffuser section to configure the preliminary duct design. The expressions obtained have been verified by applying to existing experimental data. Through the analysis, core engine entry section dimension and thus the Mach number are also evaluated in comparison with existing engine data. The isolator section scale is estimated with the expression using the experimental data in [7].

Cycle analysis tool – *abp(-main)*

By using abp(-main), engine performance is calculated for a modular arrangement of components by fast, one dimensional analysis. Therefore a method based on known component efficiencies (e.g. generalized compressor maps), and known turbine, and compressor entry temperatures, as well as aerodynamic relations is adapted for high speed air-breathing engines.

Core engine geometry and weight estimation tool - *abpw*

Abpw is an independent weight and geometry design tool for turbo-engines. The input arrangement such as component numbering and connecting between the related components such as high pressure turbine and compressor are similar to that of abp(-main). The abpw requires the input of a sequential thermodynamic data file derived from the above abp(-main) and other specific parameters to deal with structural features. This approach resembles with [31] in the point that in-detail component analysis will be done passing by simple cycle analysis result. The evaluation of the rotating parts can be done for any selected operating conditions, but is usually performed for the sea level static case. In contrast, the nozzle configuration is usually determined as for the supersonic cruising conditions. Design of the combustor parts (core engine combustor and ram combustor) is also at arbitrary conditions and is done for LAPCAT at maximum sea level flight speed case (M=0.8), where the mass flow and fuel flow take maximum values [32]. General size determination is done by the velocity diagram estimation for rotation parts, spray, mixing, combustion and cooling empirical evaluation for combustor parts, and flow field evaluation for the nozzle. The method for the structural weight analysis is quite straight-forward with a small number of variables. Since the tool only deals with simple velocity flow diagrams, the limitation of the cascade flows such as surge margin can not be estimated, instead the present tool calculates the mean flow characteristics such as loading coefficient, flow coefficient, diffusion factor, generalized velocity diagrams, shaft powers and relevant stresses on rotating parts. These flow and structural parameters are used to convince the design wellness referencing the general trends.

Approach

In this paper, analysis of several existing SST engine components are conducted with the same analysis tool abp to confirm the validity of the present tool and to attain several technical features which are unique for high speed propulsion systems compared to the propulsion systems for conventional and subsonic airliners. Especially on Olympus engine the background of each specific design are looked back and the expected configuration for future high speed propulsion system will be presented. In the LAPCAT engine structural configuration estimation, newly developed materials are aggressively applied to have lighter weight.

3. Analysis of historical SST engine components

Intake analysis

To evaluate the validity of the intake analysis tool abpi and to obtain the general rules in the specific conditions applied to the existing in-flight or model intakes, the following intakes are investigated with abpi:

- Concorde intake
- XB-70 intake
- Japanese HYPR scaled intake
- Japanese S-engine intake for hypersonic pre-cooled turbojet engine propulsion

All the intakes above are rectangular, variable geometry intakes. This configuration corresponds to the expected selection of the intake type to the LAPCAT M4 intake.

Following [8], the supersonic intakes applicable subsonic in-duct configuration, which means excluding SCRam configuration, are classified into three; Mach2+, Mach 3 and Mach 4-6 intakes.

The Mach 2+ class intakes usually deal with external compression intake because of high performance and high stability. An example intake is the Concorde intake. The solution adopted on the Concorde is relevant to external compression, in the sense that it presents no starting problem associated with mixed compression; however, it allows a small amount of internal compression, due to a wide internal boundary bleed slot.

The Mach 3 class intake essentially needs the mixed compression intake configuration, in order to attain a high pressure recovery with an acceptable intake cowl drag. Another particular aspect is the intake capture area of faster than this speed regime is obviously larger than the compressor face front area of the turbo engine, offering an opportunity of multiple-cycle engine arrangements without a nacelle drag penalty. An example intake of this speed regime is the XB-70 intake.

The Mach 4-6 class intake continues to have all the problems of slower-speed-intake configuration, and major problem arises is the operation in terms of various mass flow range matching and of combined cycle installation. For the engine systems of this speed regime, engine cycle will be a combined cycle configuration mostly the combination of turbo engine and ramjet engine. The large flight speed variation and engine cycle transition brings the condition of mismatch between intake swallow provision and engine operation and nozzle cooling requirements. The other important feature is that heat treatment becomes of more importance. There is no existing example, and in this case two experimental intakes, HYPR and S-engine intakes are investigated.

The HYPR intake is a rectangular mixed compression intake with 4 ramps and the engine cruise Mach number is 5. HYPR (Hypersonic Transport Propulsion System Research) project is a Japanese project held between 1989 and 1998 to develop the technologies for the future Mach HST (Hypersonic Transport) airplane propulsion system.

The JAXA S-engine intake is designed at Mach 6 with 7 oblique shock waves and a normal shock wave system. This is to be used for a hypersonic scaled air-breathing engine model whose engine cycle is pre-cooled turbojet engine using liquid hydrogen as the fuel. The hypersonic flow is decelerated through 7 oblique shock waves and finally become to subsonic speed through a weak normal shock wave. The ratio between external compression and internal compression is 2: 8 at the Mach 6 design point. Pressure recovery analysis is performed using oblique shock wave equations and normal shock wave equations.

Figure 3 shows the comparison on pressure recovery data between in-flight or wind tunnel experiments and the present abpi calculations. The figure clearly shows that the abpi results rather match with corresponding experimental data. The S-engine intake result defers from the general trend and MIL-E-5000B8, which is regarded as practical expected value of the intakes, and the present abpi correctly captures even for the case. The primarily reason of relatively small pressure recovery for the S-engine configuration comes from direct scaled down of the actual size intake design, and the problems from low Reynolds number introducing thick boundary layer have some effects. The HYPR intake is also a scaled intake actually, but the test intake is independently designed for each Mach number to achieve the best performance. The simple analysis tool abpi is found to be useful for wide range Mach number range from the result.

The second most important characteristic is mass capture ratio of an intake. The actual experimental data is only available for the Concorde intake and the S-engine intake. The abpi overestimate the mass capture ratio for the whole range of Concorde and underestimates for S-engine. The fact comes from the bleeding schedule and that for abpi is full boundary layer bleeding. In the S-engine configuration, the bleeding is done not only from the throat but also from the small holes on the ramp surface, and the amount of bleeding is determined as small as possible for depleting the harmful amount of boundary layer growth. In the Concorde intake, bleeding schedule is determined by the pressure balance of the intake sections and by the nozzle ejector flow scheduling. It can be said at this moment that the assumption of full boundary layer depletion is adequate for the initial phase of preliminary design. However, the recalculation as the present time may require a bit more robust and precious bleeding expressions. As for hypersonic engine intakes, air partitioning in between core engine and bleed ducts are very important due to wide range of speed operation for these engines. The bleeding air would flow circumferentially through the side duct, utilized as cooling air for after burner and nozzles, finally is thrown to nozzle flow in ejector configuration. The scheduling of the air flow distribution would be rate determining process for the whole engine controlling.

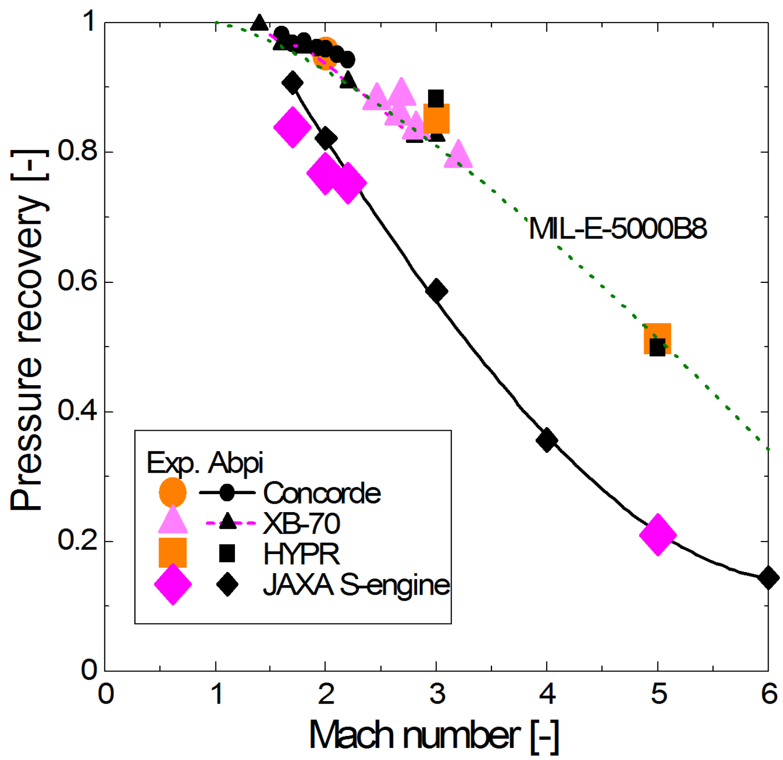

Figure 3: Pressure recovery data for existing intakes (dashed line shows MIL-E-5000B8).

In the structural point of view, an air intake is regarded as a thin-light weight pressure duct. Therefore, the preliminary design of the intake should take into account the pressure distribution, allowable amount of deformation and the supporting struts arranging so as to have smallest weight. The abpi has a function to estimate the structural weight of an intake. One reference empirical weight estimation equation available is provided in [27] as following:

$$W_{\text{intake}} = 4.345\beta + 117.35\eta^{0.294} \text{ [lb]}$$

where

β =(length of duct [ft]* number of inlets)$^{0.5}$* (total inlet capture area [ft^2] / number of inlets)$^{0.3334}$ * (engine inlet pressure [psia])$^{0.6667}$* out of round factor * mach number factor,

η = (total length of lamp [ft]) * (number of inlets)* (total inlet capture area [ft^2] / number of inlets)$^{0.5}$ * temperature correction factor

There is almost no available intake weight data from publications. Therefore, using the available weight estimation data of JAXA S-engine intake (except the subsonic diffuser part) with FEM analysis and accompanying simple structural analysis, the validity of the present abpi analysis is confirmed [9]. Figure 4 shows the comparison of the present abpi estimation of intake weight with available data. There are several plots in variation of intake size, and EFM data and WAATS data are also shown for the actual flight model (large scale) intake. The intake uses carbon-composite materials to keep the weight light. The figure shows that the WAATS data overestimates the result. The present abpi data is corresponding to the FEM and accompanying simple structural analysis data. The WAATS has no input parameter for the variation of material and the empirical expression is taken from mostly military aircrafts of 1950s. Still the WAATS data is used in many preliminary designs, but it is clearly seen that these analysis would not be applicable for future aircrafts.

Figure 5 shows the estimates of intake weight of existing intakes of large SSTs both by abpi and WAATS. In contrary to the previous case, WAATS data is much smaller than the present abpi calculations. The primary difference comes from the diffuser part. The estimated weight of the XB-70 weight seems heavier as actual comparing with total dry mass (93,000 kg), resulting in more than one third of the total aircraft weight for the intake system. Considering the largeness of the airframe, this estimation may be somewhat overestimating. At least, it can be said that the WAATS estimations are clearly underestimate weight for large intakes and the calibrated present abpi give more precise data at least for compression part. The WAATS data are taken mostly integrated intake to the airframe, and the speed range is relatively low (for approximately 1950s technology airplanes as F-411-1, F-101A, A3J, F-106B, F-101B, F8U-3, T38A, F11F-1, F9F-8, A4D-2N, A2F and F84-1[27]). Therefore, subsonic diffuser weight estimation would not be precise for the present cases and it seems the WAATS is only applicable for relatively small Mach number and small scale cases.

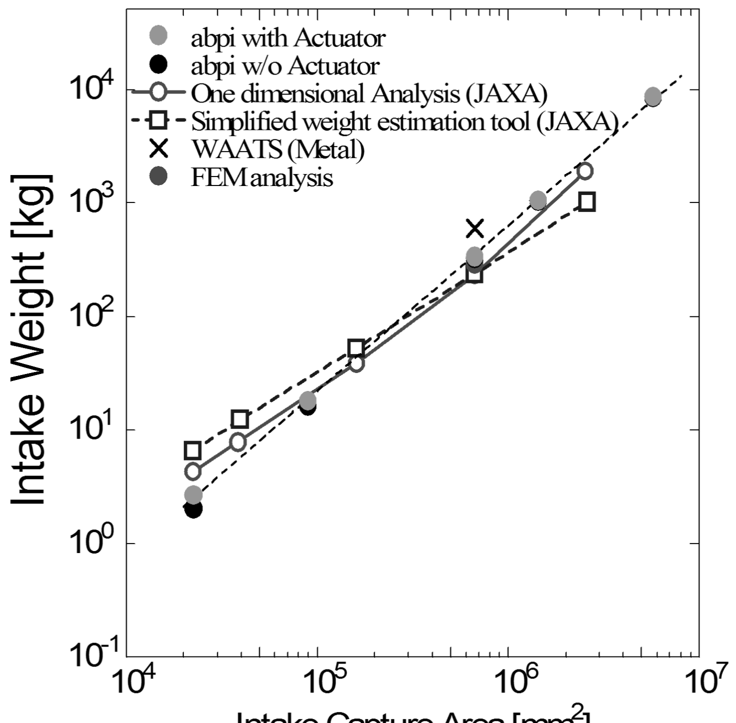

Figure 4: Validation of weight estimation tool by JAXA S-engine intake configuration

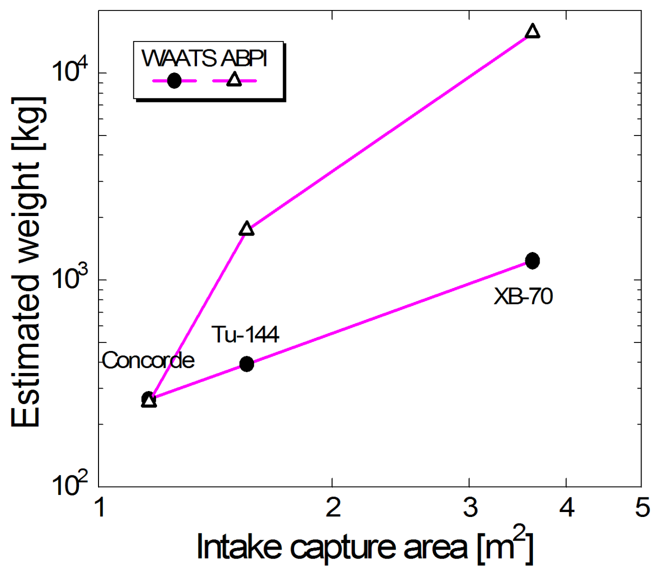

Figure 5: Estimated weight of several existing intakes (abpi and WAATS)

For some intakes, structural arrangement of the intake is available. The structure of the Concorde intake is explained in [35]. The operating temperature of under Mach 2.0 cruise conditions is approximately 400[K] and aluminum alloy is used for the structure. The forward lip, the forward and center duct skins and the intake frames are mostly machined from thick plate material. A typical frame is shown in Figure 6 (left). Structure of the XB-70 intake is described in [36]. Figure 7 shows the materials and the construction of the lower fuselage part of the XB-70, including description on the intake part. The lower fuselage is a hybrid structure consisting of a variety of different construction techniques. Multiple shallow-depth crossbeams are used to form the upper of the unit. The side and lower transverse frames supported the engine access doors and were used to complete the unit. The spars were machined from H-11 steel and used titanium webs. At the side of the fuselage, the honeycomb sandwich wing stub was joined to the H-11 frames with high-strength mechanical fasteners. The skin covering the top and sides of the lower fuselage is 6A1-4V titanium alloy riveted in place. The configuration of the diffuser part is different with usual rectangular intakes, because one intake of the XB-70 involves three engines. On the development phase of XB-70, a one-third-flow intake model combined with one engine was tested. This structure is rather similar to the present abpi calculation condition. Reference [36] displayed that the model of the lower fuselage weighed 210000 pounds and is 75 feet long. In the abpi calculation, similar value was acquired by applying the most steel materials and the 20 times stricter bending allowance (reference value was subscribed by the engine weight beforehand).

Although there is some information about the intake structure of existing engines, the present estimated weight results are still obscure because the amount of deformation allowance in the actual intake is not known, which is very important in the structural weight determination. Therefore, further comparison with obtainable actual data should be made.

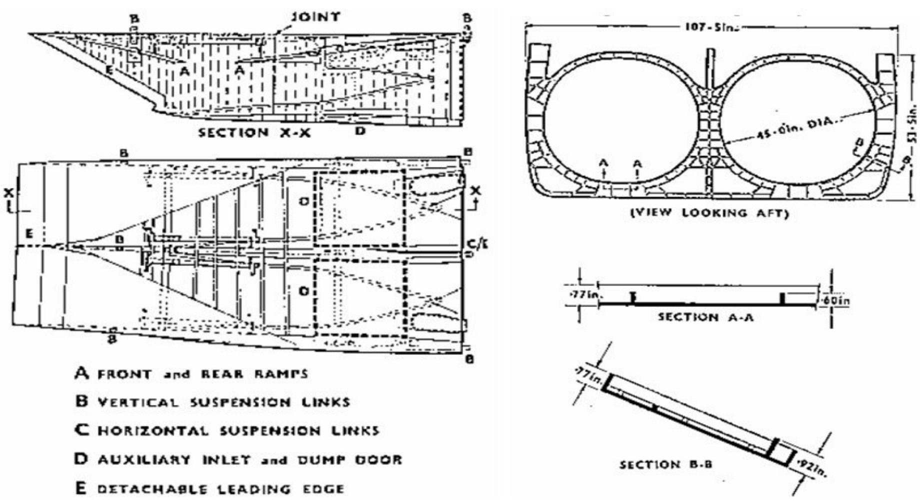

Figure 6: The concorde intake structure [35]

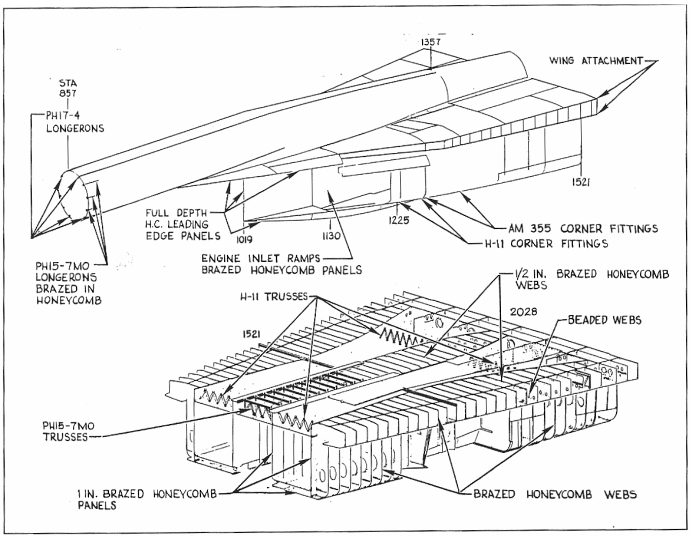

Figure 7: Materials and construction used in the upper and lower intermediate fuselage of XB-70 [36]

Core engine analysis

In the previous report [17], cycle analyses with abp(-main) are conducted for the following existing engines:

- Olympus 593 Mk610 turbojet of Concorde.
- Kuznwetsov NK-144 turbofan of Tu-144
- Kuznwetsov RD-36-51A turbojet of Tu-144
- Samara NK-321 turbofan of Tu-144
- General Electric GE-4, originally foreseen for the cancelled Boeing B2707 project

The five turbo engines shown above are regarded as first generation supersonic aero-engines which were developed in the 1960s. However, they are relatively diverse in their thermodynamic cycle definition. Three turbojets, one in two spools (Olympus 593) and the other two single spool (RD-36, GE-4) layouts with different OPR and TET can be found. The two turbofans include a two-spool and a three-spool design with a significant range in OPR and TET, reflecting different engine technology generations.

As summary of the simple cycle analyses of the first generation SST turbo engines [17], it can be stated that conventional turbofans can only be operated in augmented mode during supersonic cruise flight. The resulting poor efficiency of turbofans significantly reduces their possible flight range.

The Olympus 593 turbojet of Concorde demonstrates a well balanced performance in the whole flight domain. Although this engine was a good compromise at its time, subsonic and take-off performance has to be significantly improved for a viable future SST.

From the above notation, the Olympus 593 engine is found to be a good example to be investigated in terms of more in-detail configuration and mechanics (eg, [9][11][12][13][14][15][16]). From the references available on the engine, a discussion on the supersonic engine configuration will be given below.

The weight and configuration analysis tool abpw is utilized to employ an analysis of the engine. Detail description of the mechanical aspects of the Olympus 593 is seen in [11]. Further characteristics on rotating machine are described in [18] and [19]. After determining the geometrical configuration, abpw provides several characteristics of the performance which are determined only by given geometrical features (impossible to give after simple cycle analysis with a so-called “rubber engine”). For example, power input to the compressors and compressor off-design characteristics [18][19] and compressor load characteristics are confirmed valid. By confirming the available flow characteristics, the engine mechanical configuration is calculated to introduce an estimated weight. The derived value is about 3000kg without secondary nozzle. The secondary nozzle of Olympus engine is a module structure for identical two engines align and has a complex ejector flow path. For other turbojet engines, GE-4 and RD-36, simplified sectional image and some information are available ([20] and [21]). Those engines do not provide abundant information, and some properties are assumed by analogy to the Olympus 593 engine.

The present results are compared with estimation by Taguchi and co-workers [22]. Figure 8 shows the result of weight estimation in comparison with actual reported value. The dotted line shows the estimation of weight variation with change of FAC, which are described as following [22]:

$$W_{engine} = 9.8851 \times FAC \text{ [kg]},$$

where

$$FAC = \overline{m}_a \times \pi^{0.265},$$

and $\overline{m}_a$ and π are corrected mass flow in [kg] and pressure ratio, respectively.

From the figure, the present weight estimation gives relatively close value to the actual value. And also it can be seen that the quite simple (statistical) estimation gives relatively good result and the parameter FAC is a good reference to show the largeness of the engine. The RD-36 engine result by abpw and [22] somewhat overestimates to the actual reported value. The engine structural data is not sufficient (because of the historic trend of Russian engines more titan materials are applied) and has plugged nozzle structure, which can not be calculated precisely in the present tool. By comparing the actual weight data with the empirical expression of [22], GE-4 is found to be properly estimated in the abpw estimation but the absolute weight (compared to the thrust) is heavy as a supersonic engine. The resultant heavy weight is reported to be one of the reasons of the cancellation of the whole program.

Other than the method described in [22], there are several engine weight estimation methods reported. Kobayashi [23] introduced weight estimation expressions to turbo engines and applied them to his integrated optimisation calculation of the TSTO space-plane. The expressions are based on [24] and improved with available model fabrication data for specific purpose to the TSTO engines. Basically, these weight estimations are given in the following expression:

$$W/W_{ref} \propto \left(D/D_{ref}\right)^k ,$$

where W and D are weight and sectional diameter, respectively, and ref means referential value. Because a turbo engine is a pressure vessel which flows high pressure fluid, the factor k would be specified as 2, 3 and 0 for pressure balancing structure, not pressure balanced structure and accessory device, respectively. Therefore actual components give the value between 2 and 3, for example WAATS [27] provides data corresponding to the k value of about 2.5. The approach in [25] is also straight-forward, and has the following expression:

$$W = \sum_n \rho_n V_n .$$

Component density and specific diameter of the components are empirically determined. Most recently, a modified expression is reported based on [25], and the new approach deals with blade lifetime [26]. The actual design of the high temperature and/or high loading components is restricted by the environment of loading. As will be seen in the following, supersonic engine components suffer from relatively severe loading environment, and the design should consider not only the performance such as pressure ratio and turbine inlet temperature, but also inlet temperature and pressure environment and their variation during operation.

Present abpw can be regarded an intermediate level weight estimation tool between the precise structural calculation with FEM and more simple calculations such as shown above. The abpw has merits of fast calculation compared with FEM and needs calibrations for specific purposes. Also, another merits of abpw is that it has an ability to break down the estimated weight component as shown in Figure 9 for Olympus 593 engine. As seen in the figure, compressors and nozzle weighs among the components. However, some verification data is needed in such a kind of analysis. As shown in Figure 8, it is shown that the present abpw is reliable as far as the preliminary design level.

The information on the actual engine test data and the present abp(-main) calculation data on Olympus 593 reveals some important features on mechanical aspects of the supersonic turbo engines. Reference [11] shows and abp(-main) calculation confirmed that throughout engine components suffer from high temperature (840K) during cruise. The Olympus cruises at compressor exit temperatures that the subsonic engine only sees briefly at take-off. The significance is long enough (about half time of the flight) for fully stabilized soak temperatures to be achieved throughout the engine, whereas in the typical subsonic engines only the thin airfoil sections and some of the casings achieve those high temperatures. As a result for supersonic engines, this temperature gradients and consequential stress range are much higher than subsonic engines [11]. Further, the temperature available for cooling of turbine disks and blades with acceptable pressure level is higher than for subsonic engines, limiting the ability of turbine cooling by the provided cooling air [11]. Therefore, as in [26], supersonic engines should consider the operation sequence especially on thermal loading more carefully than conventional subsonic engines.

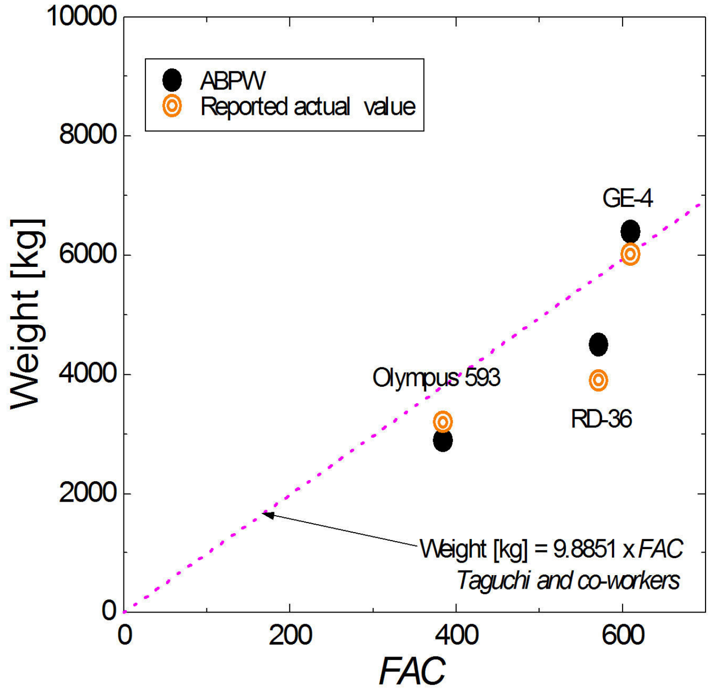

Figure 8: Estimation of turbojet engine weights

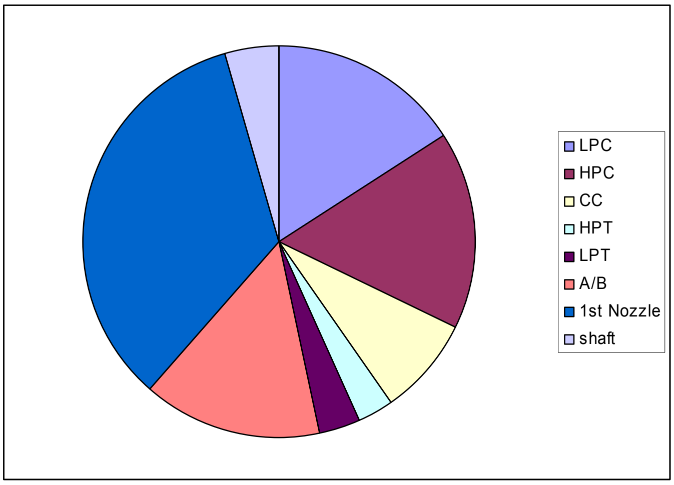

Figure 9: Weight distribution of major components by abpw calculation on Olympus 593

Remarks on nozzle

As mentioned in the intake section, the nozzle operation should be coupled with intake bleeding flow path. The present nozzle configuration is only the simple convergent-divergent nozzle matching with whole operation regime. To have a precise estimation of performance and weight for the nozzle, ejector flow configuration should be taken into account. At this moment, the flow configuration can be incorporated with the information (eg. [28], [29]) and the mass flow distribution pattern strategy is being developed. This revision would be reported in the future.

4. Analysis of LAPCAT M-4 engine components

General description of the LAPCAT M-4 engine

Table 1 lists all propulsion interface data obtained from the vehicle design simulations of LAPCAT. These data are the basic requirements for the subsequent preliminary design of all TBCC propulsion components. A minimum sea-level take-off thrust at Mach 0.3 of 1800 kN has been defined keeping the runway length requirement within acceptable limits.

Table 1: Propulsion system thrust requirements based on LAPCAT-M4 vehicle design process

Trajectory point 1 (take-off):	
Mach:	0.3
Altitude:	0 m
Total installed thrust:	1800 kN
Trajectory point 2 (cruise 1):	
Mach:	4.5
Altitude:	24300 m
Total installed thrust:	640 kN
Trajectory point 3 (cruise 2):	
Mach:	4.5
Altitude:	29100 m
Total installed thrust:	292 kN

Trajectory point 1 is the major sizing criterion for all turbo engines, while the other two trajectory points operate in RAM-mode and are relevant for the air-intake sizing.

An advanced turbojet is regarded as well as three combinations of variable cycle turbofans. A ram burner will be integrated with the convergent divergent nozzle for all cycle variants. This device acts as an afterburner in high altitude acceleration flight and as the single propulsion system beyond the transition Mach number of around 3.5. Cycle temperatures are limited to similar, ambitiously high values. A TET of 1950 K and an afterburner temperature of 2100 K nevertheless seem to be achievable by advanced engines within a development perspective of about 20 years. In the following estimations, materials under research and development are aggressively applied to propose a reasonable thrust to weight engine configuration within the term perspective.

Intake analysis and design

All engines will be fed by the same two-dimensional, variable-geometry, mixed compression intake. Different designs have been investigated. One of the most promising configurations is an up-to six-shock, three-variable ramp mixed compression intake. Attainable pressure recovery has been estimated with the DLR code abpi [5]. An optimized cruise point pressure recovery of around 70% at Mach 4.5 could be possible, as can be seen in Figure 10. This translates into an intake efficiency of 0.974 at cruise. The pressure recovery and efficiency at lower Mach numbers can be significantly raised by a suitable positioning of the second and third ramp angles. External compression with the terminating shock in front of the lip seems to be advantageous for Mach numbers up to approximately 2.5. The curves in Figure 10 include the effect of fore body compression and the bow shock. The cycle analyses of the turbojet and of the variable cycle engines presented in the following section are based on the pressure recovery data from Figure 4. The capture area of 7.5m2 has been checked by abpi analyses in design and off-design conditions on available air-mass flow and seems to be sufficient in the complete flight mission.

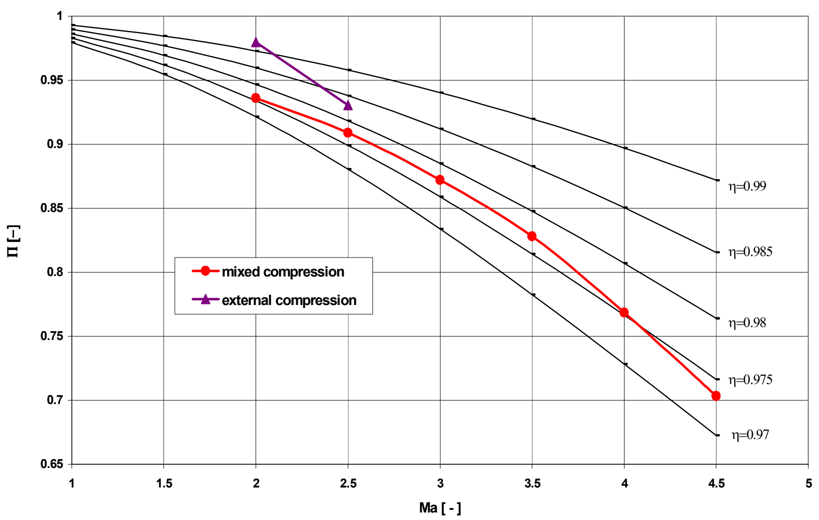

Figure 10: LAPCAT-M4-3R-V3 Air Intake geometry pressure recovery data

Further 2D-CFD analyses on compression part of the intake have been conducted and basic correspondence and several remarks for the treatments have been obtained. The CFD results are available for several in-flight conditions not only the design point but also subsonic cases for which it is hard to handle with simple analytical tools such as abpi. In-detail information on the analysis is described in [17].

A parametric analysis has been conducted by changing the exit diameter and reduced length (duct length divided by exit diameter), and appropriate exit diameter and reduced length of the divergent duct are determined as D_2=1.95 m and L/D2=4.25, respectively. This value of non-dimensional length would be within allowable level in terms of small distortion and results in actual length of 7.5m.

Figure 11 shows the preliminary sized geometry of the LAPCAT-M4-3R-V3 intake. Maintaining the previous capture area of 7.5m2, the total length of the ramps reaches about 9m and the width is 2.5m. Total length of the subsonic diffuser including a movable section with rectangular cross section is about 8.6m and the overall length of the intake with the isolator section is more than 19m.

Geometry data as automatically generated by the abpi tool or provided as input values allow a first mass estimation which is based on a simplified structural mechanical analysis. The material of the intake is mostly selected as an advanced C-SiC composite for high temperature resistance and high strength in an un-cooled environment. Due to the extremely light-weight design, a total intake mass without equipment and actuators of slightly above 5500kg is obtained.

One important feature in the in-detailed design of the intake is actuator design and by-pass flow ducting. As seen in the previous chapter, the present wide Mach number operation requires large variation of by-pass airflow schedule related to ejector nozzle configuration. The present intake analysis assumes the amount of bleeding at throat section as much as needed for full boundary layer depletion. Further improvement of nozzle design in respect of ejector configuration should be combined with intake bleed scheduling.

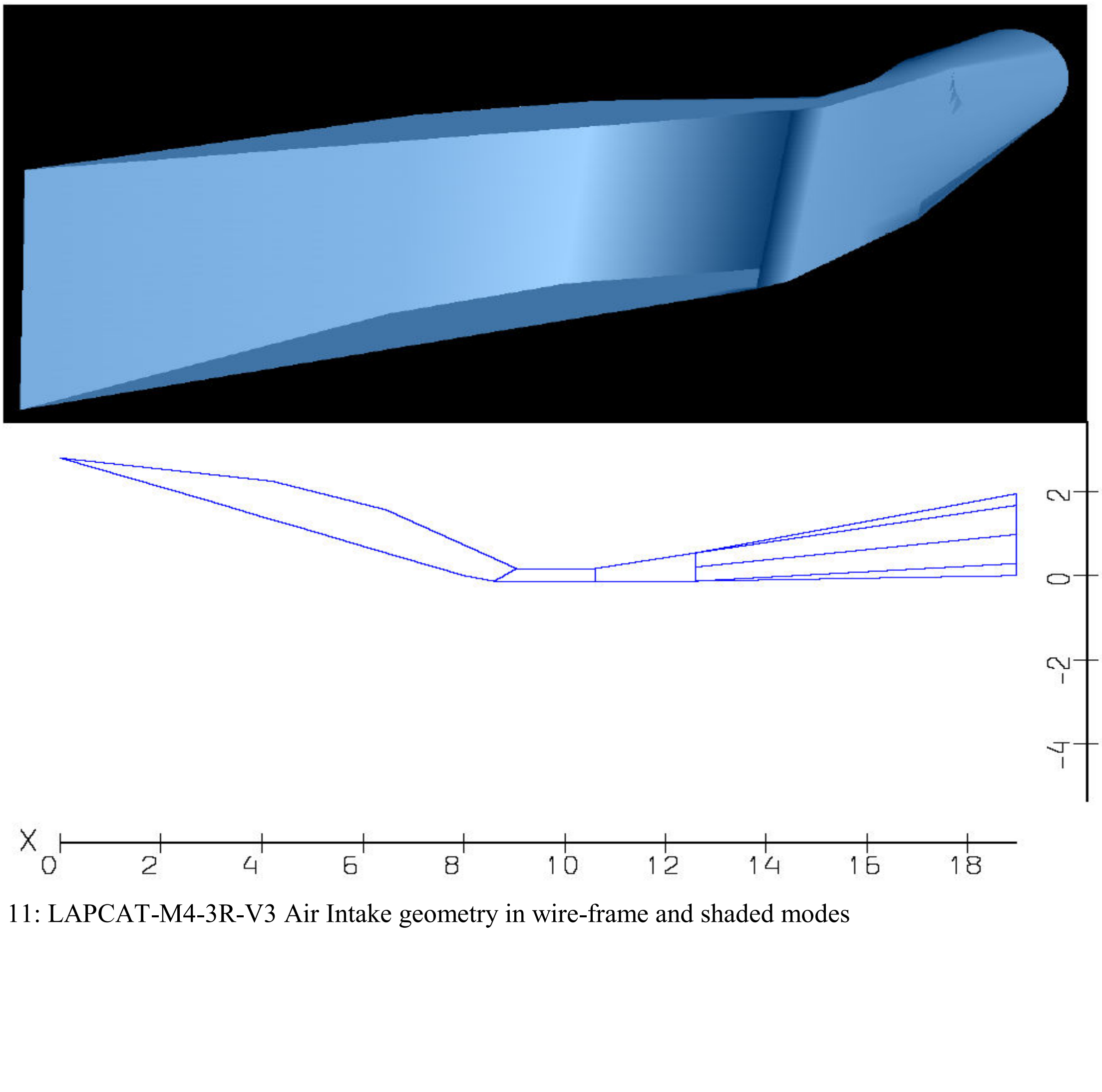

Figure 11: LAPCAT-M4-3R-V3 Air Intake geometry in wire-frame and shaded modes

Core-engine configuration determination

The turbo engines under investigation as candidates are listed with major cycle specification data at SLS conditions in Table 2.

The LAPCAT TJE-6 is a relatively simple and compact design with a six stage core compressor powered by a twin stage HPT. Three different variants of advanced variable cycle engines have been investigated for the LAPCAT supersonic airliner. All are double bypass turbofans principally similar to RTA-a design [2], however adapted to the mission requirements of the present LAPCAT-M4. The major differences of the three concepts are in the number of fan or compressor stages and their OPR, as also summarized in Table 2. A RAM burner will be integrated with the convergent divergent nozzle in all variants. Table 2 also lists the major cycle specification data of the LAPCAT-M4 VCE engines at SLS conditions prior to take off. Note that both of the VABIs are closed and both bypass ratios are consequently set to zero to achieve a high specific thrust. Complicated lay-out of the VCE allows an adaptation of the different compressors more or less independently of each other. Around Mach 2.5 three engines are switching to an operation mode in which the first fan is transitioning to windmilling. At Mach 3 the low pressure spool is in full windmilling, thus temperature and pressure of the incoming flow will not be altered in this component. At Mach 3.5, transition to pure RAMjet is initiated. The core engines of VCE will be closed beyond the Mach 3.5 trajectory point and the complete airflow is directed through the bypass duct to the RAM chamber. Several preliminary investigations in the speed regime are ongoing [30]. One issue would be a thermal environment around 1050K for the windmilling LP-fan during long RAM cruise time could raise problems. The maximum RAM combustion temperature is limited to 2100K as for the afterburner. This high temperature is used during the (short) acceleration phase and can be reduced during the (longer) cruising phase. The lower thermal loads ease somewhat cooling concerns because of no cryogenic fluid applicable to active cooling.

A comprehensive cycle analysis and corresponding flight simulations of LAPCAT-M4 have been conducted in the previous report [30]. As a summary, it is found assuming an identical amount of available fuel and an identical take-off mass, that the achievable range of the four turbo engines has the same ranking. However, these assumptions disregarded any differences in engine dry mass.

Table 2: LAPCAT-M4's different turbojet and VCE double bypass turbofan cycles' specification data at sea-level static conditions according to abp calculation

		TJE-6	**VCE-114**	**VCE-213**	**VCE-214**
OPR	-	15	23.256	14.887	24.0
Π_{Fan}	-	-	1.9 (1 stage)	2.1 (2 stages)	2.1 (2 stages)
Π_{CDF}	-	-	1.8 (1 stage)	1.7 (1 stage)	1.7 (1 stage)
Π_{HPC}	-	15 (6 Stages)	6.8 (4 stages)	4.17 (3 stages)	6.72 (4 stages)
HP-Turbine	-	2 Stage	1 stage	1 stage	1 stage
LP-Turbine	-	-	1 stage	1 stage	1 stage
Bypass ratio (BPR1) λ_1	-	-	0.0	0.0	0.0
Bypass ratio (BPR2) λ_2	-	-	0.0	0.0	0.0
air mass flow	kg/s	450	435	450	450
TET	K	1950	1950	1950	1950
$F_{spec\ 0,0}$, dry	Ns/kg	1116	1129	1109	1126
$F_{0,0}$, dry	kN	502	491	499	506
$sfc_{0,0}$, dry	g/kNs	30.67	28.1	30.7	27.9
$T_{afterburner}$	K	2100	2100	2100	2100
$F_{spec\ 0,0}$, afterburner	Ns/kg	1343	1400	1353	1415
$F_{0,0}$, afterburner	kN	589.4	601	603	627
$sfc_{0,0}$, afterburner	g/kNs	43.84	42	43.7	41.8

As mentioned above, it is quite important to have information about engine mass data to compare the type of engines. The engine configurations which have been evaluated first are the TJE-6 turbojet engine and the VCE-114 double bypass turbofan engine. The very broad range of engine operation from sea level Take-off up to M=3.5 in more than 20000m requires a through investigation of each component under all relevant conditions. The preliminary results show that both of the engines can be accomplished with an inlet diameter of 1.9m^2 and a maximum engine diameter of approximately 2.1 m. The total length of the engine with nozzle is at least about 7 m. Assuming advanced fibre reinforced materials (eg. TMC as described in [33]), the turbojet weight is almost 5 tons class (4600kg) and the variable cycle engine in the 6 tons class (6100kg), both including nozzle. Figure 12 shows comparison of the LAPCAT TJE-6 weight with other weight trends. This figure shows estimated weight data on LAPCAT-TJE-6 both for advanced material case and conventional material case. The case of conventional material results corresponding to the trend described by the curve by Taguchi and co-workers [22] and it apparently results heavy. By applying the advanced light weight materials [33] as much as possible within the allowance of mechanical stresses, it is shown that the weight can become almost half. It should be noted that the application of unrealised materials are investigated with much care, because the long-life reliability and toughness to various environments are still obscure [34]. At least in this moment it can be said that with a simple weight estimation tool which is valid for several existing supersonic engine the LAPCAT TJE-6 engine is derived to weigh almost 6 tons. The first weight estimation of turbojet and one VCE shows that the latter weighs heavier about by 1 ton. Also note that the present estimated nozzle weight is for a variable axy-symmetric convergent-divergent type. A larger and more complicated ejector nozzle with advanced noise suppression features will become heavier. For the comparison among various VCEs the duct flow path and their actuation configuration for the phase transition would be important. Further comparison among the engine concepts should be done with these estimated engine weight derived in the same standpoint.

One difficult feature is the applicability of temperature resistance materials. As mentioned above, the Olympus engine component performance is not such an extreme one compared with recent civil aircrafts, but by accounting its thermal environment the configuration itself was very challenging. The Olympus engine components should have suffered from relatively long-time high temperature gradient conditions, resulting in severe thermal loading and corresponding cycle. The present LAPCAT engine in a whole suffers in similar or severer thermal loading condition, however as seen in Figure 2, the time for the core engines to suffer from the high thermal and mechanical loads is quite limited to the acceleration phase. Most of the time is occupied with subsonic cruise and high speed RAM mode. Still care should be taken because of high temperature windmill condition and long time operation of RAM combustor (with relatively low load).

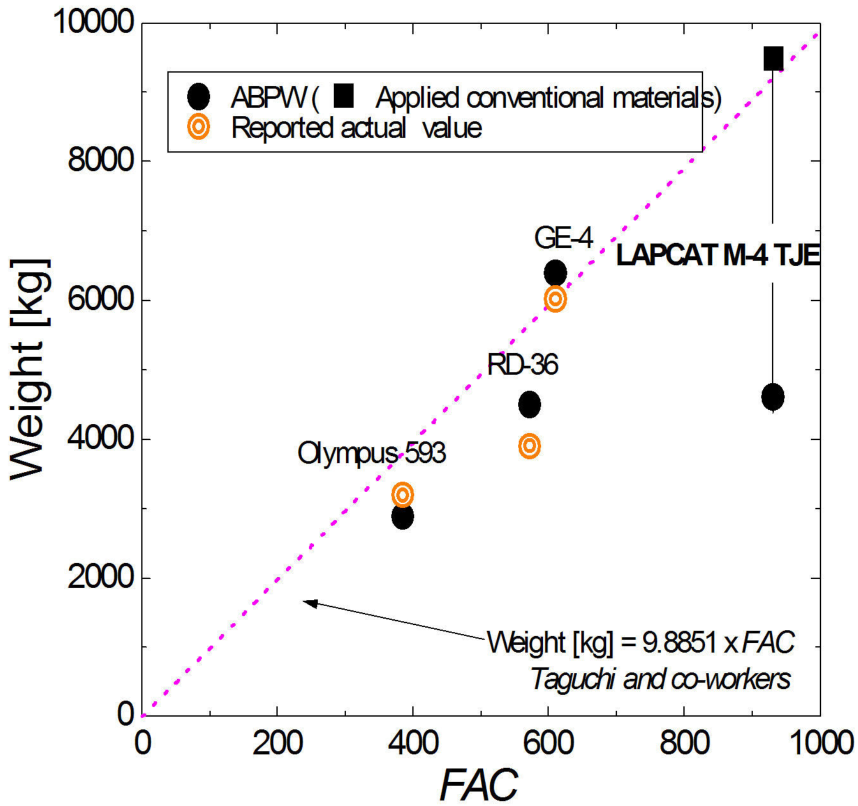

Figure 12: Engine weight estimation of LAPCAT TJE-6 and some existing turbojet engines

5. Conclusion and Future work

A turbine based combined cycle is an attractive propulsion solution for an advanced ultra long-haul supersonic airliner. Within the framework of the present LAPCAT M-4 propulsion which seeks a Mach 4 class conventional fuel turbo-engine feature, component analysis on the already-proposed engine type candidates are carried out. In advance to the analysis, the present analysis tool are applied to several existing engine to assure the validity and to better understand the specific features which sometimes are different from conventional subsonic engines.

A preliminary high performance air-intake design is carried out and shows promising data. Preliminary sizing design of the intake is carried out and the result shows that the total length reaches almost 20 m and assuming advanced high temperature composite materials an intake mass of about 5.5 tons is estimated.

The configuration and weight analysis are conducted for two of the candidate turbo engines (TJE-6 and VCE-114). It is shown for the TJE-6 case that the application of advanced light weight materials can reduce the engine total weight tremendously compared to the conventional material case. The resultant weights are 5tons class and 6tons class including nozzle for TJE-6 and VCE-114, respectively. The configuration analysis shows that both of the engines can be accomplished with an inlet area of $3m^2$, and the total length is about 7 m including nozzle.

The first estimation of the concrete configuration and rough estimation of weight gives much insight for further investigation. As for the future work, more in detail description on nozzle configuration, feasible air partitioning and duct configuration is favoured. All of the core engine types and further more integrated analyses on the total propulsion system will be pursued.

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http://www.estec.esa.int/techresources/lapcat

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