

LAPCAT: High-Speed Propulsion Technology

J. Steelant

ESA-ESTEC

Division of Propulsion and Aerothermodynamics

Keplerlaan 1, 2200AG Noordwijk

Netherlands

Johan.Steelant@esa.int

ABSTRACT

In spring 2005, the EC kicked-off a 3 years long project called LAPCAT: Long-Term Advanced Propulsion Concepts and Technologies to initiate research on propulsion concepts for sustained hypersonic flight. The project, composed of a consortium of 12 partners from industry, research institutions and universities, is coordinated by ESA-ESTEC.

The ambitious mission goal is to reduce travelling time of long-distance flights, e.g. Brussels to Sydney, to about 2 to 4 hours. Therefore advanced propulsion concepts and technologies need to be developed. This requires a new flight regime with Mach numbers ranging from 4 to 8. At these high speeds, classical turbo-jet engines need to be replaced by advanced airbreathing engines. Different combined cycles, i.e. TBCC and RBCC, are evaluated both on system and fundamental research level in combination with a corresponding vehicle system study.

At the writing of this article, the project has passed its midterm review and its achievements up to this point will be highlighted here.

1.0 INTRODUCTION

Tendencies in aeronautics clearly show a continuous increase in air traffic. Based on IATA statistics the international passenger traffic growth increased with 6.8% in February 2006 over 2005 which continues on pace with forecast. Even for long-distance flight such as from Europe to Asia Pacific, IATA's five year forecast shows an annual average growth of 5.9% between 2005 and 2009. This is primarily due to the explosive economic growth in Asia and in particular in China. These long-distance flight taking easily flight times of 16 hours or more to connect two major intercontinental cities, would become more attractive when travel-time would be reduced drastically such that a final destination can be reached within 4 hours or less.

However, with present aircraft and propulsion designs, we're getting close to the optimal design and margins for further improvement are getting smaller. Only drastic changes in aircraft configuration, propulsion concepts and flight velocities are able to achieve these goals.

New aircraft configurations and related propulsion engines presently studied for classical flight Mach numbers around $M=0.9$ look into e.g. blended wing-body configuration for aerodynamic performance and multiple engines mounted on top of the wings close to its trailing edges to improve propulsion efficiency. These interesting developments will decrease further fuel consumption up to 30% but will not enable the shortening of travel times.

New aircraft development seems to be stalled with respect to flight speed, despite the proven technical possibility shown by the supersonic Concorde. Opponents to supersonic transport development always point to the large specific fuel consumption of Concorde which undeniable is roughly twice the value of present commercial aircraft. However, one should not forget that the specific fuel consumption, sfc, obtained for the first turbojet driven aircraft, e.g. Comet in 1951 were only 20% lower. Since then fuel consumption reduction for aero-engines has been drastically driven throughout time by technology e.g. cooling techniques, new alloys, improved thermodynamic cycles by increased pressure ratios and TIT, etc... As the Olympus 593 engine was based on the Olympus design of 1950 for the Canberra and later for the Avro Vulcan in 1956, it is hence impossible to compare its sfc with e.g. the latest Trent's of R&R or the GE90-family when half a century of technology development has not been implemented in these Olympus engines.

Before given an overview of the LAPCAT goals, some basic considerations about supersonic vs. subsonic flight and its potential for evolution will be discussed. Finally some first results obtained so far within the LAPCAT project will be discussed.

NOMENCLATURE

ATR	Advanced Tactical Fighter
CG	Center of Gravity
CP	Center of Pressure
EoS	Equation of State
ER	Equivalence Ratio
HHV	Higher Heating Value [MJ/kg]
HSCT	High Speed Civil Transport
I_{sp}	Specific Impulse [s]
LHV	Lower Heating Value [MJ/kg]
m'	air mass flow [kg/s]
m_f'	fuel mass flow [kg/s]
MTF	Mid Tandem Fan
NGLT	Next Generation Launcher Transportation
OPR	Overall Pressure Ratio
RBCC	Rocket Based Combined Cycle
RTA	Revolutionary Turbine Accelerator
RTD	Research and Technology Development
SST	Supersonic Transport
sfc	Specific Fuel Consumption [kg/s/daN]
TIT	Turbine Inlet Temperature [K]
TBCC	Turbine Based Combined Cycle
VCE	Variable Cycle Engine

Subscripts

a	ambient
cc	combustion chamber
j	jet condition
∞	cruise flight point

2.0 MOTIVATION AND ASSESSMENT

Reducing travel times by going supersonic has only sense on long-distance flights. Range is hence an important figure of merit to evaluate high-speed aircraft concepts. It is strongly dependent on total

available fuel mass and its consumption throughout the itinerary, i.e. from taxiing, speed-up cruise and final descent manoeuvres. Among these different parts, cruise represents a major portion of the needed fuel. The range achieved during cruise can be easily derived from the Bréguet range equation:

$$R = \frac{H}{g} \eta \frac{L}{D} \ln \left[\frac{1}{1 - W_F / W} \right] = \frac{V_\infty}{g \text{ sfc}} \frac{L}{D} \ln \left[\frac{1}{1 - W_F / W} \right] \quad (1)$$

where:

R	Range [m]
H	the fuel energy content [J/kg]: 120 (LHV) and 142 (HHV) MJ/kg for H ₂ , 43.5 (LHV) and 47MJ/kg (HHV) for kerosene, 50.0 (LHV) and 55.5MJ/kg (HHV) for Methane
g	gravity constant [m/s ²]
η	the overall installed engine efficiency
sfc	specific fuel consumption [kg/s/N]
V	flight velocity [m/s]
W	total take-off mass [kg]
W_F	fuel mass [kg]

The range depends linearly on the energy content H in the fuel which can be increased with a factor of 2.7 by switching e.g. from kerosene to hydrogen.

The aerodynamic performance given by L/D in eq. (1) depends primarily on the Mach number and was analyzed by Küchemann [1] who formulated a general empirical relationship referred to as the “L/D barrier”:

$$\left(\frac{L}{D} \right)_{\max} = \frac{4(M_\infty + 3)}{M_\infty} \quad (2)$$

Further studies optimized waverider designs taken into account viscous effects resulted in better L/D ratio resulting in a shifted L/D barrier (Anderson [2]):

$$\left(\frac{L}{D} \right)_{\max, \text{viscous}} = \frac{6(M_\infty + 2)}{M_\infty} \quad (3)$$

For an increasing Mach range the values are decreasing asymptotically to a value of 4 or 6:

Table 1: Aerodynamic L/D barrier and overall installed engine efficiency in function of flight Mach number

M_∞	0.9	2	4	6	8	10
$L/D_{\max, \text{euler}}$	17.3	10	7	6	5.5	5.2
$L/D_{\max, \text{viscous}}$	19.2	12	9	8	7.5	7.2
η	0.25	0.4	0.57	0.67	0.73	0.77

This decrease of aerodynamic performance with increasing Mach number would inherently exclude long-range supersonic flight as it would be economically not viable. However, the overall propulsion efficiency, defined as

$$\eta = \frac{T \cdot V_{\infty}}{m_f' H} = \frac{V_{\infty}}{sfc H} \quad (4)$$

increases with Mach number for turbojets and ramjets as will be explained further. A first approach, suggested by R.G. Thorne according to [1], is given by:

$$\eta = \frac{M_0}{M_0 + 3} \quad (5)$$

To better understand the increase of the overall efficiency η of an aircraft engine, one can split the term thermodynamically into a thermal and propulsion efficiency $\eta = \eta_t \eta_p$. given approximately for a single jet by:

$$\eta_t = \frac{m'(V_j^2 - V_{\infty}^2)}{2m_f' H} \quad \eta_p = \frac{2V_{\infty}}{V_j + V_{\infty}}$$

The thermal efficiency of either compressor or ram-based engines can be approached as a Brayton cycle and hence its efficiency is mainly driven by the combustor temperature T_{cc} to intake temperature T_a ratio. This ratio would be at its optimal point when operating the combustor close to the stoichiometric value. However, for turbojets or turbofans, the rotary turbine components limit this ratio due to material yield strengths to a value of about $T_{cc}/T_a=6$ or $\eta_t = 47\%$.

Typical values for propulsion efficiency of a modern engine at $M_{\infty}=0.85$ is 48% for a turbojet and 77% for a turbofan with a bypass ratio of 6. The overall efficiency in cruise results into values of 20% to 37% and increases above 40% for larger bypass ratios [3].

For ram- and scramjets, the combustion temperature is not limited by rotary components. Hence higher equivalence ratios are easier to reach and $ER=1$ is presently used in scramjet flight experiments. Hence, the thermal efficiency can reach values as high as $\eta_t = 60-70\%$. The propulsion efficiency is clearly better as the jet/flight velocity difference is rather small resulting into a $\eta_p = 70-90\%$ leading to an overall efficiency of $\eta = 42-63\%$. This large η_p implies that a massive intake needs to be foreseen, which can occupy the complete frontal section of the aircraft in order to provide the necessary thrust given by $T = m'(V_j - U_{\infty})$.

As shown above, both factors η and L/D have reverse dependencies on flight Mach number and for a first assessment the combined value $\eta L/D$ can be considered in first order to be constant, i.e. a value of about 3 to 4, at worst only 40% smaller for careful designs. This means that the range is more or less independent of the flight speed and is then only determined by the relative fuel fraction W_F/W and the fuel type.

However, one point of caution should be raised. This general analysis is purely based on the Bréguet range, i.e. cruise flight. However, aircraft's operation and handling largely depends on this performance at take-off and during acceleration. In particular the latter is increasingly more dominant for higher Mach number flights. For example, a practical flight at Mach 5, such as the LAPCAT A2 vehicle, reveals that the acceleration and deceleration entails about 50 minutes each compared to a 2.8h flight. At higher Mach numbers, these phases are becoming even more dominant in the propulsion optimization process not only due to the longer speed-up and slow-down phases, but also to the relatively shorter cruise phase.

Another important point, also highlighted by Cain and Walton [4], is how far one can extend or extrapolate parameters for subsonic transport to supersonic or even hypersonic transport. So far, this first order approximation does not include the effect of changes in structural mass fractions that may be

required to cope with the high heating loads during flight. Also the requirement that the plane needs to operate over a wide range of flight Mach numbers, i.e. from take-off to cruise, will demand for a largely flexible engine, most likely as a variable cycle engine, which sequentially will penalize the concept by a larger engine mass fraction. Related to this is the need for fully integrated design of engine and airframe to obtain a global maximum in efficiency whereas up to now engine and airframe can and are being designed and optimized quite separately.

Despite these concerns, it is worthwhile to assess the performance of a wide range of existing aircraft in the light of the above described logic. Data were mainly obtained from websites of a/c manufacturers or others. Hence, these might not necessarily represent the true values but should rather be taken as indicative.

The specific fuel consumption (sfc) is plotted versus a non-dimensional range defined as R/R_g in fig. 1. R_g is the ultimate anti-nodal point for a final destination, i.e. 20,000km. The Concorde is very competitive in range compared to the aircraft designed in the same period (60's and 70's) e.g. Comet, DC-9,... despite a larger sfc. Ranges for subsonic aircraft designed in the 90's have almost doubled, e.g. MD-11, B767,... This is of course related to the improved aerodynamics, availability of light-weight and high-temperature materials, lower sfc due to higher bypass ratios, higher TIT, etc. Similar improvements can be applied to a successor of the Concorde to improve its sfc or range for the same flight speed: implementation of a bypass ratio of 0.5 to 1 or higher instead of presently none, application of more advanced cooling concepts for turbine blades, use of light-weight heat resistant materials, use of a Variable Cycle Engine rather than using an afterburner or reheat, etc...

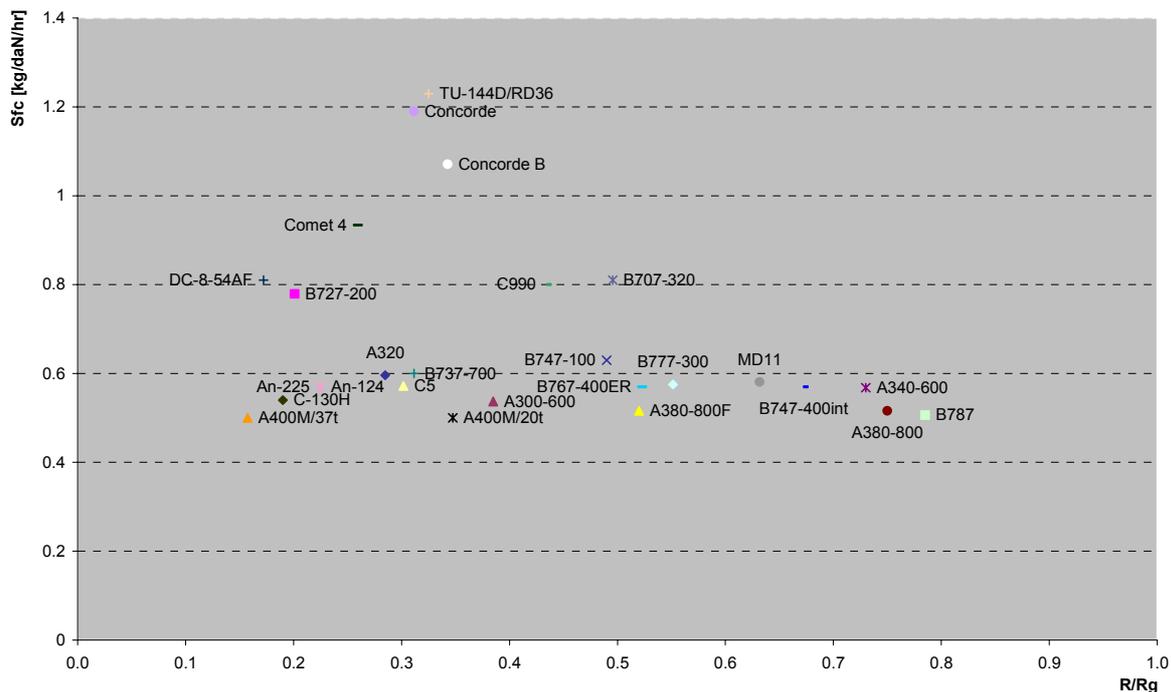


Fig.1: Indicative specific fuel consumption values for various sub- and supersonic aircraft in function of non-dimensional range. A400M is given for two different payloads.

The positive effect of the cruise speed on the overall propulsion efficiency is clearly depicted on fig. 2. Equation (5), represented as a full line, clearly fit the efficiencies of the first designs of their kind for

subsonic, e.g. Comet-4 and supersonic commercial vehicles, e.g. Concorde. The formula (5) seems to bound a lower limit. The earlier mentioned 20% efficiency for turbojet and its evolution towards values above 40% for large bypass turbofans seems to be confirmed by the trend observed for commercial aircraft. With respect to fig. 1, this efficiency doubling results inherently in a doubling of the range. Not surprisingly, efficiencies have in particular improved for the heavily used subsonic aircraft. This is entirely due to massive resources made available to improve this range of aircraft in their aerodynamics' design and engine fuel consumption and not due to intrinsic physical limitations for supersonic transport vehicles.

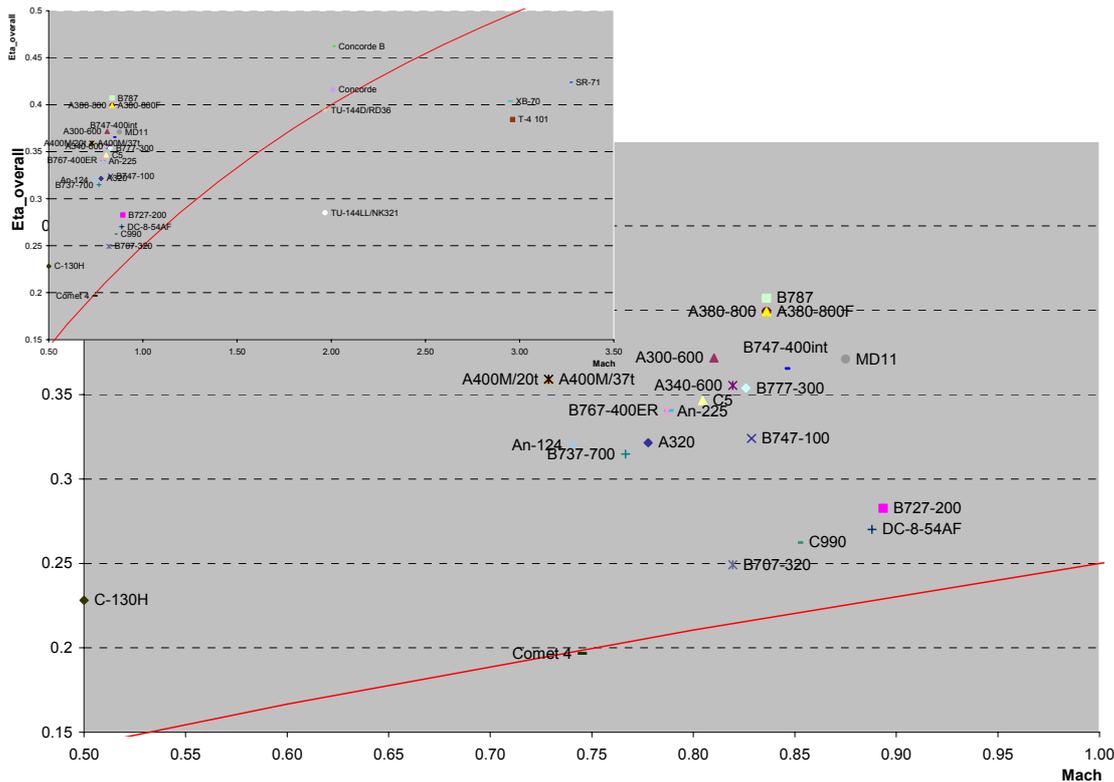


Fig.2: Indicative overall engine efficiencies for various sub- and supersonic aircraft in function of flight Mach number. Eq. (4) is given by full line.

The aircraft indicated below this line are all supersonic transport vehicles. In order to sustain supersonic flight, these aircraft are equipped with engines which require to be run with afterburning, which is of course thermodynamically inefficient (low η_i). Concorde and TU-144D are however equipped with high thrust engines enabling to sustain supersonic flight without afterburner.

Further improvements were in the planning which would have led to a further efficiency increase away from the lower limit given by eq. 4. Indeed, less known to the public is the start of studies to improve the performance of the Concorde four months after the start of scheduled services in 1976. This project should lead to the Concorde B model. Among modifications on aerodynamics, systems, weight, fuel tanks, the modification on the propulsion unit consisted of replacing the low-pressure compressor by a compressor with increased diameter and the low pressure turbine assembly by a two-stage turbine (fig. 3). The installation of a discharge system to increase the margin of air flow through the engine would result in an increase in air flow which reaches 25 % on takeoff and 35 % during approach. The thrust gains obtained at takeoff and at transonic speeds also make it possible to remove the reheat (afterburner) system with its very heavy fuel consumption and significant addition to the noise generated by the powerplant.

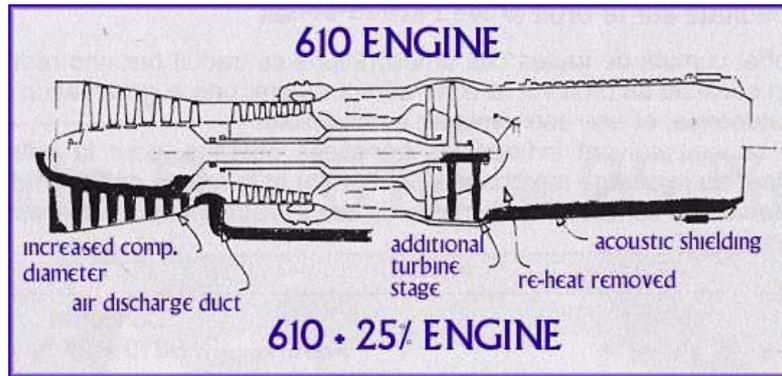


Fig. 3: Suggested improvements (lower part) to the original Olympus (upper part) engine shortly after the Concorde introduction with higher thrust and lower sfc, avoiding the use of an afterburner [5].

Presently, within the military aircraft development, there is an objective to equip their fighters with high-thrust engines to avoid the use of afterburner not only for sustained supersonic flight but also to accelerate through the high-drag transonic flight regime. Examples are the Eurofighter Typhoon and The F-22 raptor. The latter incorporates a pair of new, higher thrust-to-weight turbofan engines, the Pratt & Whitney F119-PW-100, which is designed for efficient supersonic operation without afterburner (called supercruise). The F119 engine develops more than twice the thrust of current engines under supersonic conditions, and more thrust without afterburner than conventional engines with afterburner. Of course, limited data is available related to sfc, bypass ratio, TET, OPR, ... to plot them relatively to the other data.

In [1], a weight breakdown analysis is described for which the total take-off weight W is split into different parts. Items including wings, undercarriage, services and equipment are proportional to the overall weight, i.e. c_1W . Other items are proportional to payload c_2W_p including fuselage weight, furnishings and the payload itself, hence $c_2 > 1$. Finally we have the engine and fuel weight W_E and W_F . This results into:

$$W = c_1W + c_2W_p + W_E + W_F$$

Combined with eq. (1) one can obtain:

$$\frac{W_p}{W} = \frac{1}{c_2} \left[\exp\left(-R \frac{H}{g} \eta \frac{L}{D}\right) - c_1 - \frac{W_E}{W} \right] \quad (6)$$

Evaluating W_E/W from a large range of data (fig. 4), setting this value to 0.05 seems to be a good average. The factors c_1 and c_2 largely depend on the use of state-of-the-art structural materials and are retained here as variable parameters. In fig. 5, the payload fraction W_p/W , i.e. passengers or cargo, for multiple existing aircraft is plotted against the non-dimensional range. These data have been fitted by adapting the structural parameters c_1 and c_2 , along with the propulsion and aerodynamic performance parameter $\eta L/D$ of eq. (6) according to the values given in table 2.

Table 2: Parameter sets used for evaluation of future trends

	$\eta L/D$	c_1	c_2
A	4	0.3	2.25
B	5	0.25	2.00
C	5.5	0.2	1.90
D	6	0.15	1.75
E	3	0.35	2.75

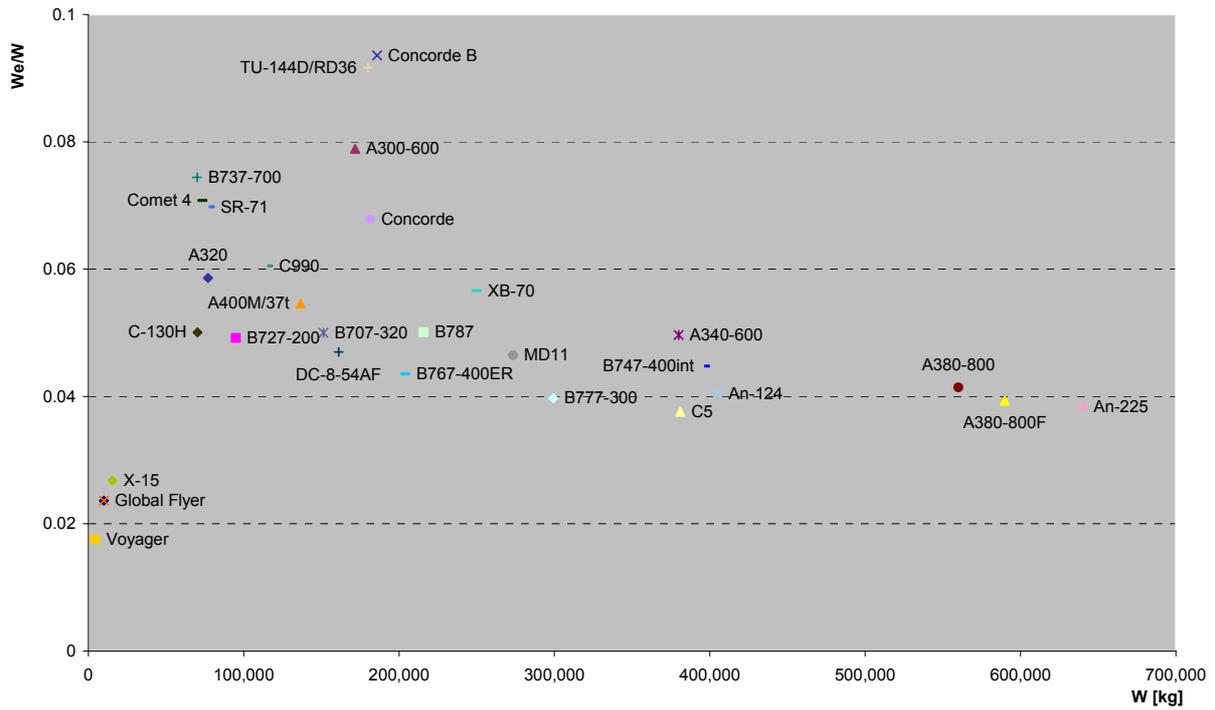


Fig. 4 Indicative engine fraction for various aircraft

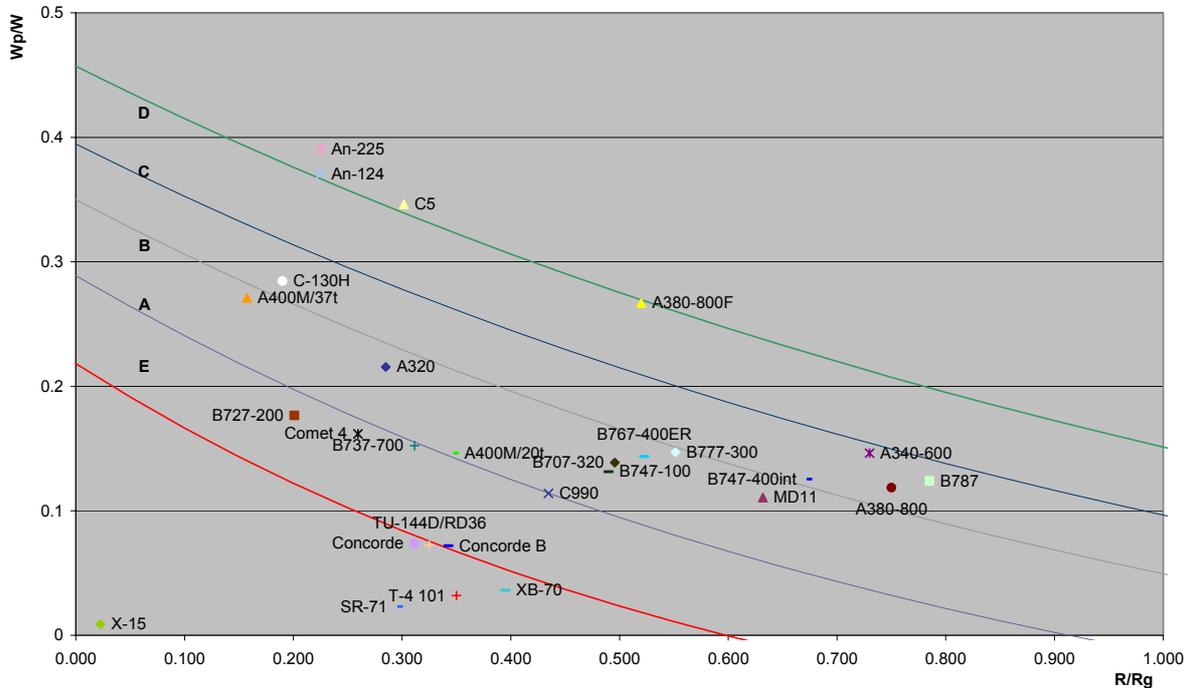


Fig. 5: Indicative payload fraction in function of non-dimensional range for various aircraft. Full lines A to E are based on eq. (6) for structural, propulsion & aerodynamic parameter variation.

Generally speaking, the payload fraction increases from A to D due to the improved structural and propulsive parameters and demonstrate the technological evolution introduced into the newer airplanes. The line E is definitely lower for the supersonic aircraft which is due to the lower $\eta L/D=3$ versus 4 for the subsonic ones achieved in the same era. Small changes on the aerodynamics, as proposed for Concorde B in 1976, combined with efficiencies of recent engines developed for supersonic fighters, indicate that a recent SST might achieve a performance which lies in between lines A and B.

Another important figure of merit for aircraft is seat-kilometre produced which can be expressed as $W_p/W \times R/R_g$. This is shown in figure 6 along with the fittings discussed previously. Also here the tendencies are well represented by the equations with higher values for the more recent airplanes and the lowest for the obsolete commercial SST.

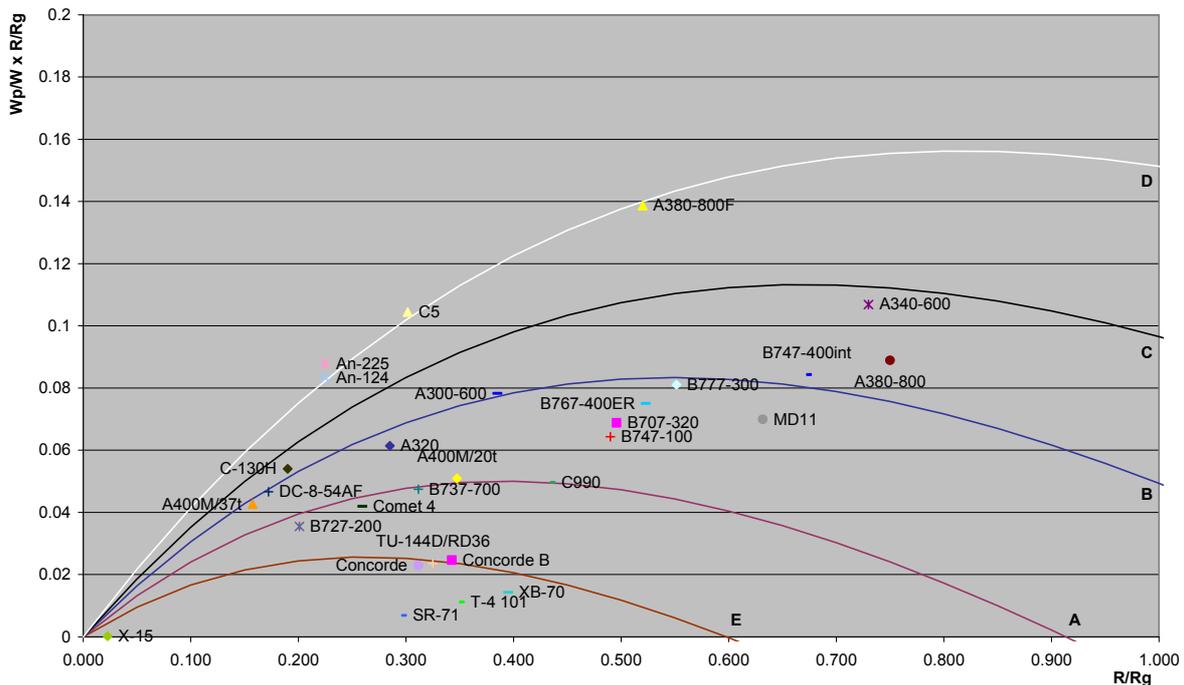


Fig. 6: Seat-km: figure of merit comprising both payload and range for various aircraft

The still remaining parameter to be discussed is the use of hydrogen as fuel. Studies have been performed in Europe (e.g. Cryoplane) and Russia, but little information is available on the aircraft performance. However, making use of the suggested correlations, the influence of hydrogen as a fuel can be easily assessed. In figs. 7 and 8, the previous parameter settings A, D and E have been applied for a hydrogen aircraft, denoted respectively AH2, DH2, EH2. This is a first approximation as the larger required volume for hydrogen storage will induce a higher drag which is not accounted for. The dashed lines clearly indicate that aircraft have a larger potential in range with still an interesting payload capacity, including SST. Aircraft of lower performance, according to A, have now a potential equivalent for the ultimate range to aircraft of type D by switching to hydrogen. This opens up the potential to reach anti-nodal destination with optimum seat-km already for conservatively designed aircraft.

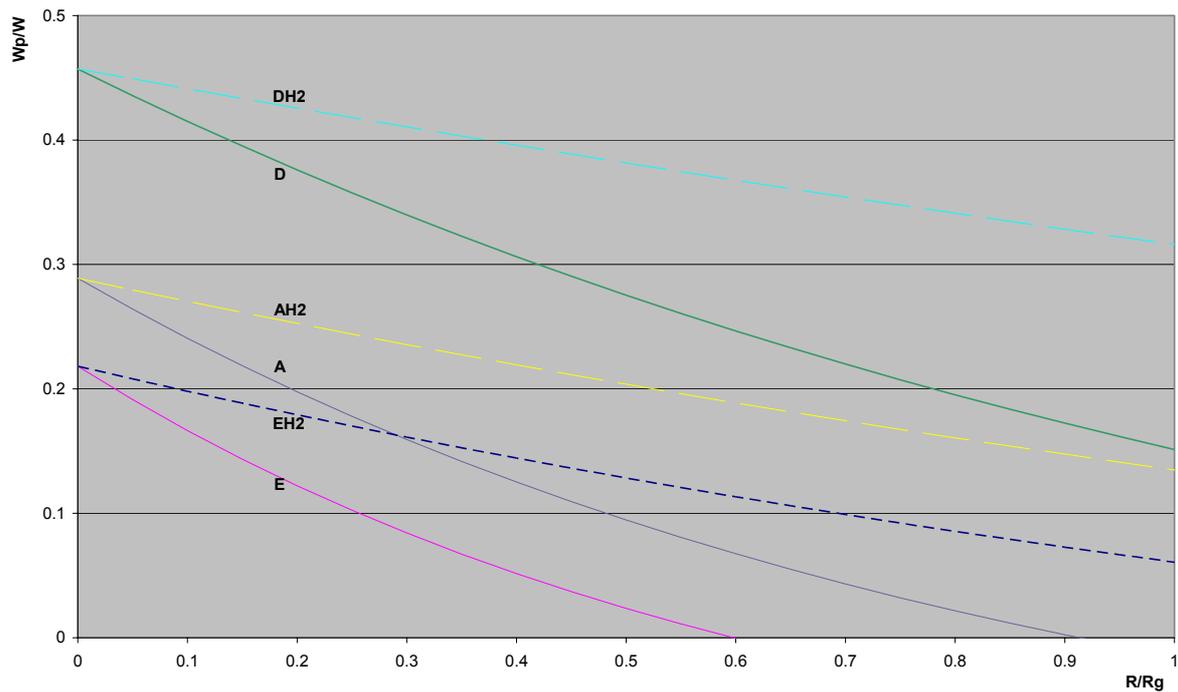


Fig. 7: Payload fraction dependence on fuel type: Kerosene (full), hydrogen (dashed).

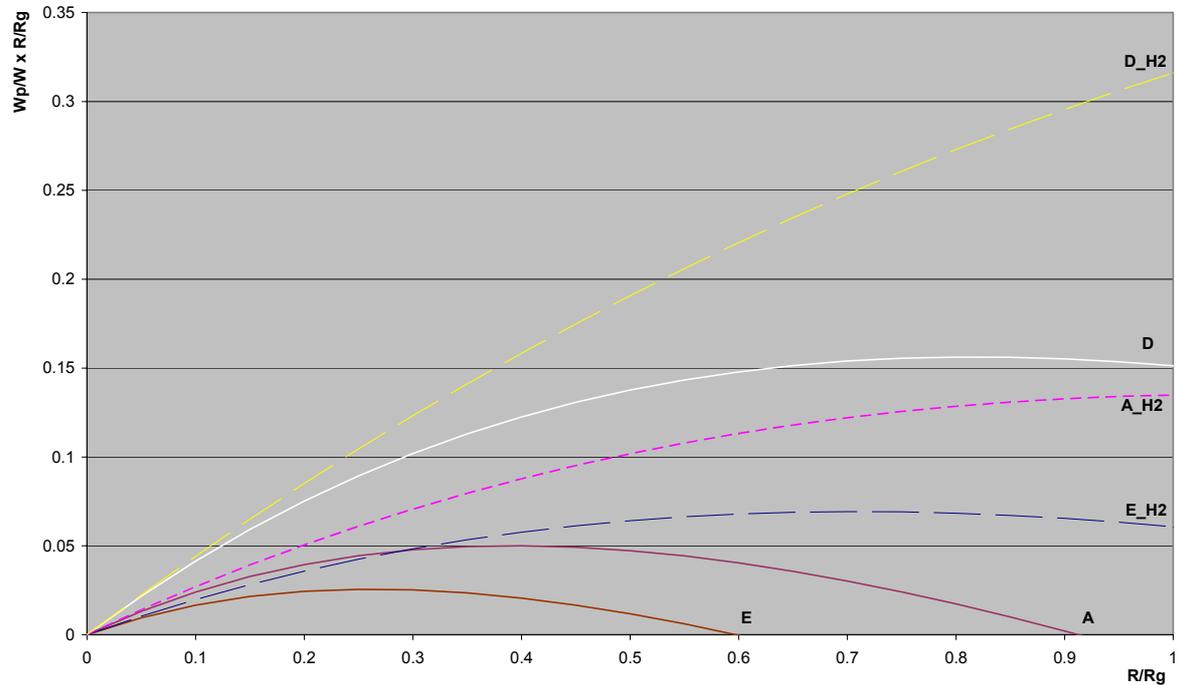


Fig. 8: Seat-km dependence on fuel type: kerosene (full), hydrogen (dashed)

This motivated the LAPCAT-team to tackle the final technological challenge within aviation: can man travel to the other side of the world within a relatively short time of two to four hours?

3.0 LAPCAT: PROJECT DESCRIPTION

3.1 High-Speed Propulsion: Introduction

Two types of air-breathing engines for high-speed aircraft have so far been successfully developed: The turbojet and the ramjet. The turbojet, based on active compression, has reached a high state of development and is widespread within the aeronautical world, mainly due to its accelerated appearance in the mid 40's. A supersonic application of turbojets in commercial aviation is the well-known Olympus 593 engine of Snecma/Rolls-Royce used for the Concorde. Another particular application is the reconnaissance aircraft Blackbird SR-71 with an integrated P&W J58 turbojet which has an afterburner fed in part by air vented from the compressor.

The ramjet, based on ram-compression, has found extensive applications in missiles and some experimental aircrafts. The technological transfer towards commercial aviation has been hampered so far due to the lack of an economical driver for setting up a novel and expensive development cycle with adaptation of or new high-speed wind tunnel facilities and flying test beds. The largest stumbling block, however, is the environmental concern (noise) of supersonic planes.

Moreover, thermal aspects limit the flight Mach number for both airbreathing types as well as the flight duration. Even at moderate pressure ratios, turbojets become impractical beyond Mach 3 when using conventional materials and classical thermodynamic cycles. Ramjets are considered feasible up to Mach 7. Beyond this, the increasing stagnation enthalpy prohibits further acceleration of the jet flow due to endothermic dissociation of the combustion products. The consequence is a high loss of energy in the exhaust due to a large amount of unreacted species.

Precooling the airflow as is done by the Japanese ATREX engine only partly alleviates the thermal limit for turbojets. Thermal limits for ramjets, however, are alleviated by keeping supersonic flow conditions in the combustion chamber (scramjets) with corresponding lower static temperature. Major challenges are the reduction of the higher viscous losses and achieving a high combustion efficiency in spite of lower mixing rates and smaller residence time in the combustion chamber. Scramjets have received a great deal of analytical, computational and experimental attention, especially in the USA. The major drive was in connection with the National Aero-Space-Plane (NASP) project from 1984-1995. Recently an experimental flight project with the X-43A or Hype-X has been concluded in the US. A self-accelerated scramjet flight at $M=7$ took place on the 27th of March 2004 as a first step, followed by a Mach=10 flight on November 16, 2004. In France, the "Prométhée" program focuses on a scramjet cruise vehicle using storable propellants. The University of Queensland (Australia), in collaboration with a number of international partners including DLR (Germany) and QinetiQ (UK) performed flight and ground tests for scramjet engines under the HyShot program. Their supersonic combustion flight test in 2002 lasted for 5 seconds during vertical descent at Mach 7.6. At flight altitudes from approximately 35km down to 23km, hydrogen supersonic combustion data were successfully collected. The US-Australian Hypersonics Research Program, HIFiRE, calls for 10 flight tests over five year in the field of hypersonics. Three flights are dedicated towards a scramjet powered waverider above Mach 8.

Ram-based engines cannot accelerate from zero to operating speed without the presence of an integrated accelerator. A TBCC integrates a gas turbine within the ram-duct, whereas a RBCC is based on an integrated rocket engine. The J58 engine of the SR-71 (USA) and also the proposed engine for the Sänger first stage (Germany) are examples of a TBCC. The Revolutionary Turbine Accelerator (RTA) once under development in the USA for the X-43B scramjet flight test program is expected to self accelerate up to Mach 4+. At Jaxa, the Air Turbo Ramjet of a precooled Expander Cycle (ATREX), designed for operation at Mach 6, has completed static conditions at sea-level. All these concepts vary widely in purpose and timeline and hence in vehicle and propulsion concept, development strategy and technology demonstrator.

3.2 LAPCAT: Objectives

In Europe, continuous effort for basic high-speed airbreathing propulsion research has been made at many institutions. However, these efforts are scattered and strongly specialized. The LAPCAT offers the opportunity to practice the indispensable cooperation on European level and to integrate specialized findings into a system to assess the overall relevance and benefits. During the project, system design tools are developed as well as rules and guidelines for conceptual development of system which have not been in place before. The capability to systematically guide a system development process through interface management and to assess its output will be enhanced.

The baseline mission requirement is to reduce travelling time of long-distance flights, e.g. Brussels to Sydney, in about 2 to 4 hours. This requires a new flight regime with Mach numbers ranging from 4 to 8. At these high speeds, classical turbo-jet engines need to be replaced by advanced airbreathing propulsion concepts and hence related technologies need to be developed.

As objectives, two major directions at conceptual and technological level are considered: ram-compression and active compression. The latter has an upper Mach number limitation but can accelerate a vehicle up to its cruise speed. Ram-compression engines need an additional propulsion system to achieve their minimum working speed. Key objectives are the definition and evaluation of:

- different propulsion cycles and concepts for high-speed flight at Mach 4 to 8 in terms of turbine-based (TBCC) and rocket-based combined cycles (RBCC) in fig. 9
- critical technologies for integrated engine/aircraft performance, mass-efficient turbines and heat exchangers, high-pressure & supersonic combustion experiments and modelling.

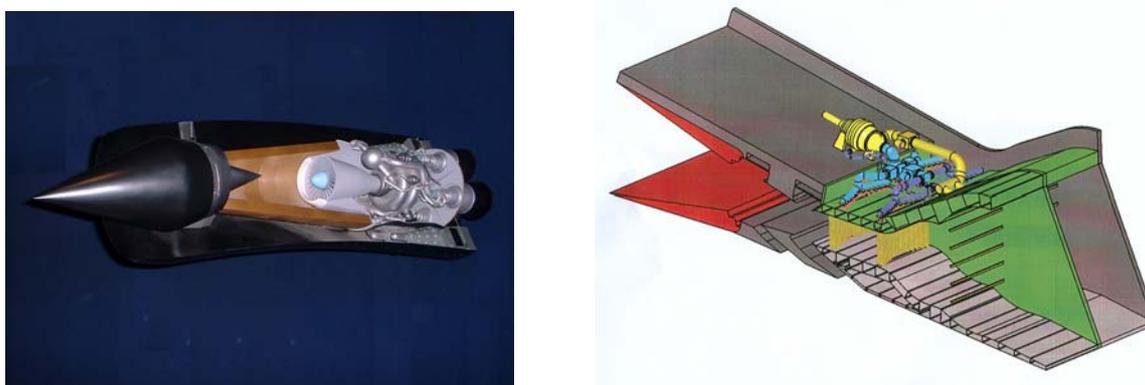


Fig. 9: Turbine Based (Sabre-engine: courtesy REL) versus Rocket Based Combined Cycle (Prepha: courtesy SNECMA).

A sound technological basis will be determined for long-term (20-25 years) to advance innovative propulsion concepts. The most critical RTD-building blocks will be identified employing analytical, numerical and experimental tools to address issues of the following road-map:

- Two airbreathing engines for selected reference vehicle(s) and trajectory point(s).
- Dedicated combustion experiments for supersonic and high-pressure combustion, including potential fuels and its interaction with flow-field turbulence.
- Modelling and validation of combustion physics on the basis of chemical kinetics and fuel spray vaporization models and turbulence affecting the combustion.
- Aerodynamic experiments for major engine components (intakes, nozzles, full engines), and establishment of data for interaction of vehicle and propulsion aerodynamics.

- Evaluation and validation of advanced turbulence models to evaluate unsteady, separated flow regimes and to develop transition models based on intermittency related parameters.
- Performance prediction of contra-rotating turbines and light cryogenic fuel heat exchangers.

The three years lasting project will result into:

- Definition of requirements and operational conditions for high-speed flight at system level
- Dedicated experimental data-base specific to high-speed aerodynamics for supersonic and high-pressure combustion and flow phenomena.
- Setting-up and validating physical models supported by numerical simulation tools to address supersonic and high-pressure combustion, turbulence and transition phenomena.
- Feasibility study of weight performance turbine and heat exchanger components

The team consists of 12 partners out of 6 European countries and is coordinated by the European Space Research and Technology Centre ESTEC-ESA in the Netherlands. This involves four industries EADS-Astrium (D), Reaction Engines (UK), Snecma (F) and Cenaero (B); four research institutions being ESA-ESTEC (NL), DLR (D), CIRA (I) and VKI (B) and finally the universities of Rome (I), Stuttgart (D), Southampton (UK) and Oxford (UK).

4.0 PRELIMINARY RESULTS

In order to specify realistic conditions for both the detailed experimental and numerical campaigns, several vehicle systems studies were performed. Two different designs were conceived for a Mach 4 to 5 range on the basis of turbine based combined cycles, either using hydrogen or kerosene as a fuel. This approach was taken in order to capitalise as far as possible on the vast experience now available from turbojet design and operation. For the Mach 8 range, a single vehicle concept was designed on the basis of a rocket based combined cycle where the airbreathing part is done consecutively by a hydrogen fuelled ejector rocket, ramjet and finally scramjet.

The initial work on RBCC mainly concentrated on deriving first performance figures to be used in the context of vehicle pre-design and analysis. This was accomplished by providing net I_{sp} figures over a wide flight Mach number range such that absolute dimensioning could be avoided in the initial absence of requirements derived from a vehicle concept.

4.1 Vehicle and Propulsion System Studies based on TBCC

The project objective was to examine two turbine based cycle (TBCC) engine concepts for high Mach number (4 – 5) flight in the context of future civilian transportation. The experience accumulated from turbojet design and operation is huge and this should obviously form the basis of the next generation of engines if at all possible.

Using the engine concepts, vehicles were defined and their aerodynamic performance optimised. These were incorporated into trajectory models together with subroutines of the engine performance to derive scaled performance and formed the basis of estimating vehicle size and mass. The nominal mission was to fly economically from Brussels to Sydney in approximately 4 hours.

4.1.1 Hydrogen Precooled Turbo-Ramjet

The first study focused on a precooled Mach 5 engine, named Scimitar, employing a cycle based on the Reaction Engines SABRE spaceplane engine and fuelled by liquid hydrogen. The Scimitar engine must have good subsonic and supersonic performance if it is to be a practical engine for a new generation of

hypersonic aircraft. This would allow it to operate from normal airports and over-fly inhabited regions without the nuisance and political problems which limited Concorde's effectiveness. These characteristics have been successfully incorporated into the Scimitar design (fig. 10) by incorporating a high bypass fan into the bypass duct which encloses the core engine and is otherwise needed to match the intake air capture flow to the engine demanded flow over the supersonic Mach number range. The bypass fan is driven by a hub turbine using flow diverted from the core engine nozzle. The flow then discharges into the bypass and mixes with the bypass flow. More details on the engine and its thermodynamic cycle are given by A. Bond [6].

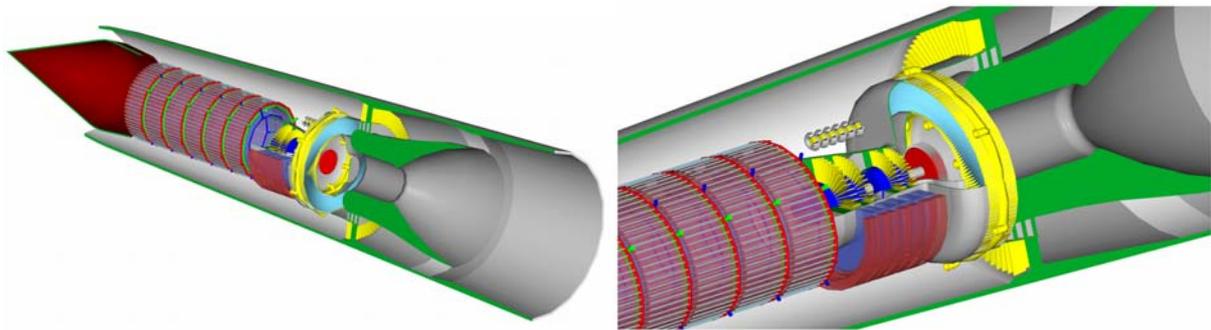


Fig. 10: Scimitar: pre-cooled Turbofan-Ramjet engine for LAPCAT A2 cruiser.

Due to their central role to the concept of the pre-cooled engine two technologies are being addressed at experimental level: a lightweight heat exchanger and contra-rotating turbine.

The air precooler is a metallic tubular matrix with the captured intake air in cross flow over a helium cooled small diameter (1.0mm) tube bank. The Scimitar power cycle depends on maintaining a constant top-cycle helium temperature at around 1000K. The preburner is needed to heat the helium leaving the precooler exit at low flight Mach numbers when the captured air enthalpy, and therefore the helium temperature also, is low. The precooler heat exchanger is a smooth plate Silicon Carbide surface in order to minimise the pressure loss on the combustor gas side. It is intended to design, manufacture and test heat exchanger technology associated with the high temperature part of the engine cycle.

The design of the tubular surface test items has been done and oxidation trials of Inconel 718 tube material has been completed. Strength problems have been identified at high temperature with the metallic materials and further manufacturing contracts have been placed to address these. The SiC technology is in the early stages of development.

It is intended to design, build and test a stator-less contra-rotating model turbine which is aerodynamically similar to the real turbine driving the turbocompressor of the Scimitar engine concept. The real turbine runs on high pressure (200 bar) hot (1000K) helium with a power output of over 60 MW. It is impractical from a cost aspect to carry out aerodynamic testing on representative components on the real fluid at this preliminary design stage. The model will therefore be run on a low pressure high molecular weight gas which can simulate the high pressure helium used in the engine. REL prepared an initial scoping design for the turbocompressor of the real engine for which VKI have optimised the aerodynamics. CENAERO has performed 3-D optimization of the VKI design for Helium flow. 3-D optimization for the Argon scaled model is currently underway.

The test program will begin in 2008 and demonstrate that very efficient ultra-compact turbines are feasible for applications in hypersonic aerospace engines. The Scimitar engine analysis suggests that it can produce

efficient supersonic and subsonic flight and meet the anticipated noise regulations for normal airport operation.

An important side result is the critical role of environmental impacts, specifically NO_x, contrails and Ozone damage. These may prove to be one of the most significant potential obstacles to scheduled hypersonic civil aviation and a literature search showed that they played a large part in the USA's decision in the early 1970s not to pursue a Concorde-like aircraft at that time. The Scimitar engine as configured in the current design would be unacceptable for its NO_x production although some options exist for reducing the NO_x at a performance penalty. Future studies need to include these problem at the outset to avoid naïve results and the expenditure of considerable resources on studying projects which are ultimately not viable. The studies in this area are complete for the current project.

4.1.2 Hydrogen Mach 5 Cruiser

4.1.2.1 Vehicle

The LAPCAT A2 vehicle flying at Mach 5 was carried out by Reaction Engines. The preliminary results of this analysis are encouraging. The vehicle study is complete at initial project study level and indicates that a 400ton, 300 passenger vehicle could achieve antipodal range without marginality. The concept is particularly interesting for this mission requirements as a trajectory optimization allowed to fly almost continuously over sea and avoiding sonic boom impact when flying over land.

The proposed aircraft Configuration A2 is shown in Figures 11-12. The vehicle consists of a slender fuselage with a delta wing carrying 4 engine nacelles positioned at roughly mid length. The vehicle is controlled by active foreplanes in pitch, an all moving fin in yaw and ailerons in roll. This configuration is designed to have good supersonic and subsonic lift/drag ratio and acceptable low speed handling qualities for takeoff and landing. A leading edge sweep angle of 55 deg was chosen as roughly equivalent to the Concorde value and known to be the minimum necessary to generate a stable separated vortex at high AOA. A thickness/chord of 3% was selected as typical of supersonic cruise vehicles. An achievable takeoff wing CL of 0.59 gives a minimum wing area of 900 m² for a takeoff mass of 400 tonnes. A fuselage diameter of 7.5 m was chosen to trade a small increase in drag for a saving in fuselage mass. The resulting fuselage is much longer than existing aircraft at 139m.

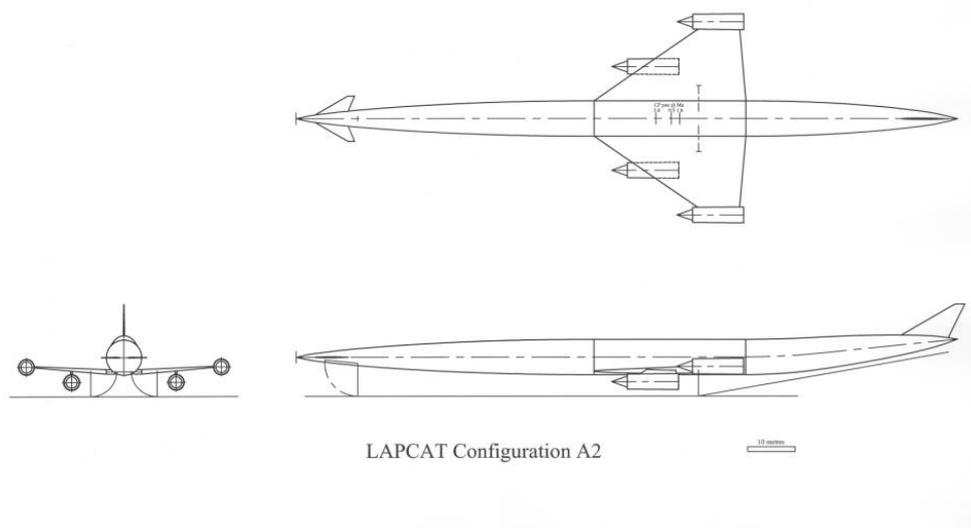


Fig. 11: LAPCAT A2: configuration for a Mach 5 TBCC vehicle based on hydrogen



Fig. 12: Artist's impressions of LAPCAT A2. Left: Mach 5 hydrogen fuelled cruiser for 300 passengers with a MTOW of 400tons; right LAPCAT A2 compared with A380.

The flight deck and passenger compartment (arranged in two decks) occupies a length of about 32 m and is located over the wings on the vehicle centre of gravity. Unlike conventional airliners hydrogen storage within the wings is not feasible since the wing volume is too small. Consequently the liquid hydrogen occupies the remainder of the fuselage volume and is split into two large pressurised tanks either side of the passenger compartment. This permits circular cross section tankage which minimises insulation and pressure vessel mass.

The vehicle has 4 engines (for redundancy), which are mounted in separate axisymmetric nacelles on the wing. Currently two engines are located on the wingtips whilst the remaining two are located within inboard nacelles located under and ahead of the wing leading edge. Physically separating the nacelles reduces the possibility of a mechanical failure in one engine causing damage to the adjacent engine, and also causes less aerodynamic disruption when an engine is shutdown or unstarts. Carrying the engines on the wing gives good CP/CG matching and is structurally efficient, whilst the alternative option of mounting the engines on the fuselage incurs the penalty of a large boundary layer diverter and acoustic fatigue of the fuselage skins. Locating the engines at the rear of the fuselage would pull the vehicle centre of gravity too far back. The wing tip mounted engines generate a large yawing moment if one is shutdown however calculations indicate that the all moving fin is capable of controlling the vehicle since it is acting at a large moment arm.

Normally the engines of supersonic cruise vehicles are mounted underneath the wing towards the trailing edge which permits the designer to capitalise on precompression and allows the nozzle to discharge aft of the wing trailing edge. However at Mach 5 the wing shock wave is at an angle of only 8.9 deg relative to the wing lower surface, which would necessitate moving the nacelle aft until the intake face was behind the wing trailing edge. Therefore this approach is structurally impractical and would scrape the nacelles on the runway during takeoff rotation. Consequently the nacelles are positioned with the intake face ahead of the wing shock wave in relatively freestream conditions, which has the added advantage that no wing boundary layer has to be dealt with. The inboard nacelles are mounted underneath the wing to reduce wing skin acoustic fatigue damage. The main disadvantage of the inboard nacelle location is that the nacelle cross section is introduced ahead of the wing maximum thickness which is counter to normal area ruling practice and will increase transonic wave drag.

4.1.2.2 Routes

Commercial civil transport aircraft, flying subsonically, normally follow a “great circle” route between the airports of departure and arrival, once they are clear of local traffic. However in the case of supersonic aircraft there is the complication of the “sonic boom”, or ground overpressure produced by supersonic

flight. Various tests show broad agreement that an overpressure below 50 Pa is tolerable for regular overflights of populated areas (although there appears to be no safe level which will eliminate complaints). Unfortunately practical civil transports produce overpressures above this level. For example Concorde generated an overpressure of about 93 Pa, which restricted supersonic flight to over water regions only and effectively reduced its commercial viability. Current overpressure estimates for the LAPCAT configuration A2 vehicle suggest that at the start of Mach 5 cruise the overpressure will be about 85 Pa under the ground track, reducing to about 70 Pa at mid cruise. Therefore preliminary route planning for the LAPCAT vehicle has assumed that supersonic flight is only possible over regions of very low population density (oceans and the North and South Poles). The possibility of overflying desert regions (eg: central Australia or the Sahara) has been ignored although this may be feasible following international negotiation. This philosophy will give a “worst case scenario” for the range requirement – in the event that the vehicle will be able to fly over land the distances traveled will obviously be smaller.

The Brussels to Sydney route was adopted as the baseline mission. This was seen as a sensible starting point in that it requires extreme range and would greatly benefit from hypersonic speeds in significantly reducing flight times. Nevertheless, other potential routes of interest are evaluated in table 3. Some of the routes are depicted in the figs. 13 to 16.

Table 3: Approximate Flight Times: 1) Includes 2 hour refuelling stop; 2) Assumes supersonic overflight of the Bering Straits; 3) No wind (jet stream) effects; 4) No "airport flight stacking" (i.e. straight in approaches); 5) Nominal ascent/descent times and distances; 6) Detailed trajectory modelling indicates 4.6 hours due to longer ascent/descent times than initially forecast.

Route	Subsonic aircraft	Mach 5 aircraft
Brussels – Sydney	22.25 hours ¹	3.8 hours ^{2,6}
Brussels – Los Angeles	10.0 hours	2.5 hours ²
Brussels – Tokyo	10.75 hours	2.5 hours ²
Brussels - New York	7.5 hours	1.6 hours
Brussels - Beijing	8.9 hours	4.9 hours
Brussels - Delhi	7.2 hours	5.3 hours
Paris - Kourou	7.9 hours	1.7 hours
Los Angeles - Tokyo	9.75 hours	2.0 hours
Los Angeles - Sydney	13.4 hours	2.6 hours
Los Angeles - Singapore	15.7 hours	3.0 hours
Los Angeles - Delhi	14.3 hours	7.5 hours

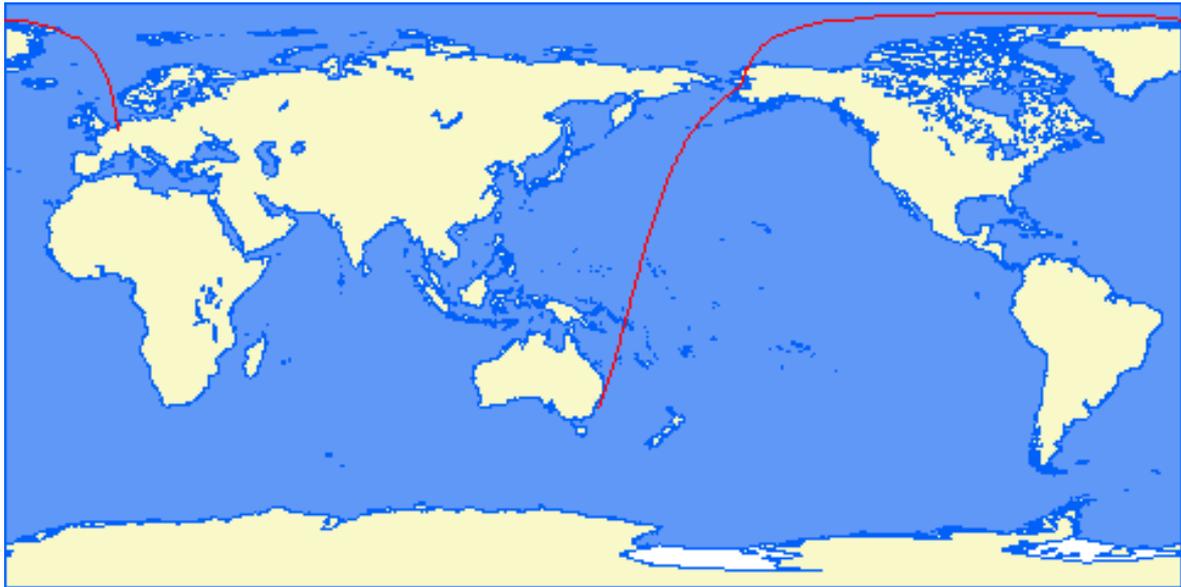


Figure 13: Brussels to Sydney via Bering Strait (18,728 km)



Figure 14: Brussels-Beijing via Nome and Tokyo (14,100 km)

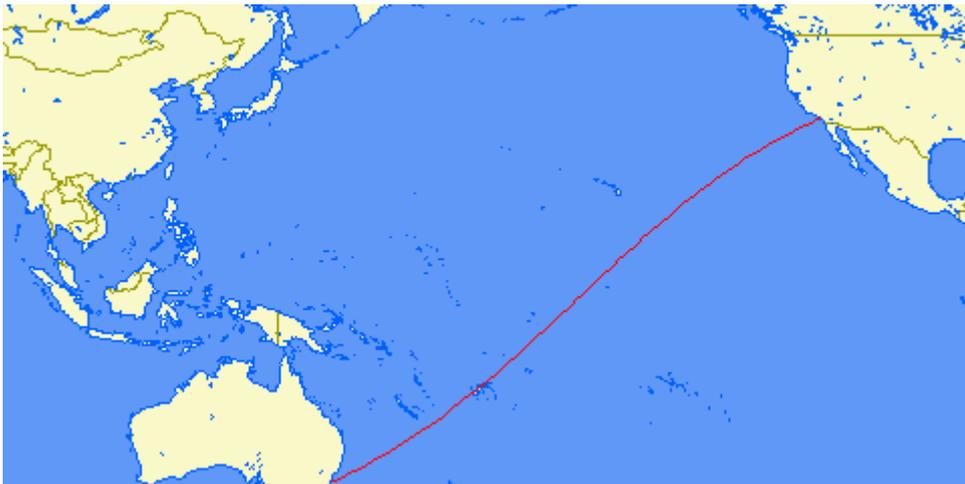


Figure 15: Los Angeles-Sydney (12,071 km)



Figure 16: Los Angeles-Delhi via Singapore (18,256 km)

4.1.2.3 Development plan and economics

To address the relatively high technical risk of this project it is proposed that the development program proceed in a step by step basis in 3 phases, namely Concept Validation (2 years), Technology Demonstration (3 years) and System Development (8 years). At the end of each program stage the project would be reviewed before deciding whether to proceed with the next stage. An arbitrary start date of 2010 has been assumed which implies an Entry Into Service date at the beginning of 2023.

The predicted engine development cost in 2006 prices is 8,147M€ and vehicle development cost 14,454M€ to give a total development cost of 22,601M€. The first vehicle production cost is 979M€. Assuming an 85% learning factor and a total production run of 100 vehicles implies an average vehicle sale price of 639M€ (including full development cost recovery).

The estimated annual operating cost per vehicle is 553.8M€ of which the liquid hydrogen fuel comprises 83%. This assumes hydrogen derived from electrolysis of water however hydrogen derived from steam reforming of hydrocarbons would be about 1/3rd the cost which would roughly halve the annual operating cost.

The A2 vehicle should carry about 148,000 passengers per year assuming 2 flights per day, 90% availability and 75% load factor. This implies an average one way ticket price to Australia of 3940€ in 2006 prices. This compares with a current Business class ticket of around 4060€ and First class ticket of 5075€. Therefore in principle the A2 vehicle could capture all of the current business and first class traffic due to the greatly reduced journey time of 4.5 hours compared to the current 22 hours (assuming subsonic carriers do not drop their prices in competition). The ticket prices would roughly halve (2000€) if the hydrogen is produced by steam reforming. This analysis assumes no hydrogen subsidy, however in reality it is likely that the first generation of hydrogen fuelled aircraft would be subsidised to promote the switch to a more environmentally friendly fuel.

Unlike Concorde the A2 vehicle has exceptional range (both subsonic and supersonic) and is therefore able to service a large number of routes whilst simultaneously avoiding supersonic overflight of populated areas. Its good subsonic performance enables it to service conventional subsonic overland routes thereby increasing its sales potential to airlines.

4.1.3 Kerosene Variable Cycle Engine

An parallel study carried out by DLR-Sart [7,8] focuses kerosene as a fuel in order to explore the performance of this fuel in preference to hydrogen since its supply infrastructure is well established.

In section 2, it was identified that turbofans are most efficient in subsonic flight whereas turbojets are better suitable for supersonic flight as their dry thrust drops far more slowly than that of a fan with increasing vehicle airspeed. A supersonic aircraft would profit of an engine which combines both performances: one for supersonic cruise flight and the other for take-off and landing as well for subsonic cruise over densely populated areas. This demands for a Variable Cycle Engines (VCE) might offer a good compromise for such applications where the specific thrust is low at low altitude flight and subsonic cruise, but is forced high during supersonic cruise and acceleration.

In [8], Sippel made a comprehensive overview of VCE which is repeated here. Recently the US was aggressively pushing this technology for military and space function (RTA, Revolutionary Turbine Accelerator) [9, 10]. The cycles are based on a suitable arrangement of the fan stages and variable by-pass flows. The MTF (Mid Tandem Fan) type VCE engine has been studied in Europe by Rolls Royce and Snecma in the early 1990ies [11].

Based on development work for the GE-21 [12], the YF120 is a variable cycle engine capable of adjusting its bypass ratio to the optimum for a given flight regime [13]. The F-120 can be seen as one of the most advanced jet engines ever flown in the YF-22 and YF-23 ATF test airplanes. Another VCE project has been the Revolutionary Turbine Accelerator (RTA) as the TBCC demonstration project in the NGLT program of NASA (fig. 17). Designated as the GE57 by General Electric (GEAE), the RTA engine represents a unique variable cycle engine where internal flowpath changes allow for high I_{sp} throughout the flight trajectory for an accelerator vehicle [9,10].

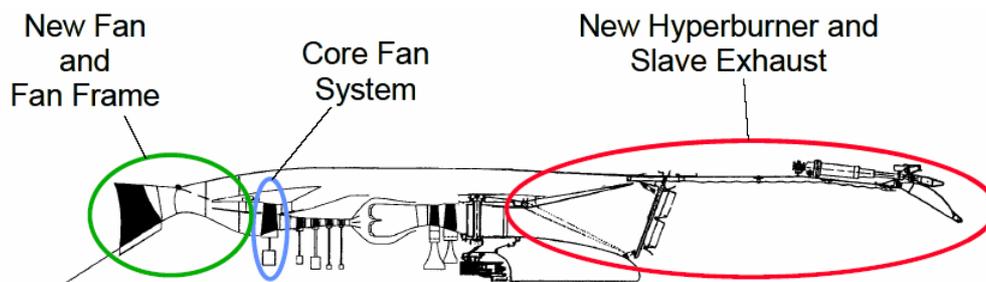


Fig. 17: General Electric RTA-1 (GE-57) cut view [9]

Three different variants of advanced variable cycle engines have been conceptually designed for the LAPCAT supersonic airliner. All are double bypass turbofans principally similar to the RTA-1 design, however adapted to the mission and thrust requirements of the LAPCAT-M4. The major differences of the three concepts are in the number of fan or compressor stages and their OPR. A RAM burner is integrated with the convergent divergent nozzle in all variants. The different concepts are evaluated in depth by Sippel [8]. The discussion below is however limited to the one selected as most optimal for the M=4.5 vehicle.

The LAPCAT VCE-214 is a variable cycle incorporating a two stage fan with large blades mounted on the low pressure spool. The first bypass flow is controlled by the first variable area bypass injector (VABI). This device is followed by the core section, starting again with a single stage fan (core driven fan stage CDFS). After the second VABI the four stage core compressor delivers the remaining air into the combustion chamber. A single stage HPT is followed by a single stage LPT and the bypass flow is again controlled by an aft VABI to achieve mixing with the core flow in the complete flight regime.

The more complicated lay-out of the VCE allows an adaptation of the different compressors more or less independently of each other. This feature can be used at higher supersonic Mach numbers. Around Mach 2.5 these 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 air is no longer altered by this component. The turbofan is operating in double bypass mode with small bypass ratios λ_1 and λ_2 and still at its maximum TET of 1950K and a OPR of 28 without reaching the compressor exit temperature limitation of 1000 K. The first VABI is opened with λ_1 approaching 0.85. The VABI2 is kept slightly open. The core engines of VCE will be closed beyond the Mach 3.5 trajectory point and the complete air-flow is directed through the bypass duct to the RAM chamber. Technical solutions for the practical design of such a TBCC might become very challenging. Some preliminary proposals will be investigated in later LAPCAT work. A thermal environment around 1050 K for the windmilling LP-Fan during the several hours of RAM cruise could raise problems. In case of the turbojet these design challenges might become even tougher and a switch to two separate flow passes might be necessary. The RAM cycle performance is presented in reference [14]. The maximum combustion temperature is limited to 2100 K as for the afterburner. This high value is used during the acceleration phase and can be significantly reduced when reaching the cruise flight. Depending on the remaining airliner weight, the combustion temperature is reduced in steady cruise to values between 1850 K and 1700 K. The lower thermal loads ease somewhat cooling concerns because no cryogenic fluid for active cooling is available. Further optimisation is ongoing related to the engine and the intake design. The latter is performed numerically by CENAERO.

4.1.4 Kerosene Mach 4.5 Cruiser

4.1.4.1 Vehicle

The preliminary design of the LAPCAT-M4 vehicle [14] is based on a critical recalculation of a 1990 NASA Langley and Lockheed study [15] 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 m² (+ 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 18. The LAPCAT-M4 employs similar to the NASA concept a blended wing-body with a modified nose, a highly swept inboard wing panel, and a moderately swept outboard wing panel (see fig. 18). The inboard wing panel is swept

78°, allowing the flow component normal to its leading edge to remain subsonic even at the Mach 4.5 cruise condition. The outboard wing panel is swept 55° but its exact, possibly curved form has not been defined yet. The total wing is inclined with a positive angle of attack of approximately 2.75°. Note that this angle and the wing's airfoil are not optimized yet and might be adapted in the future, if required.

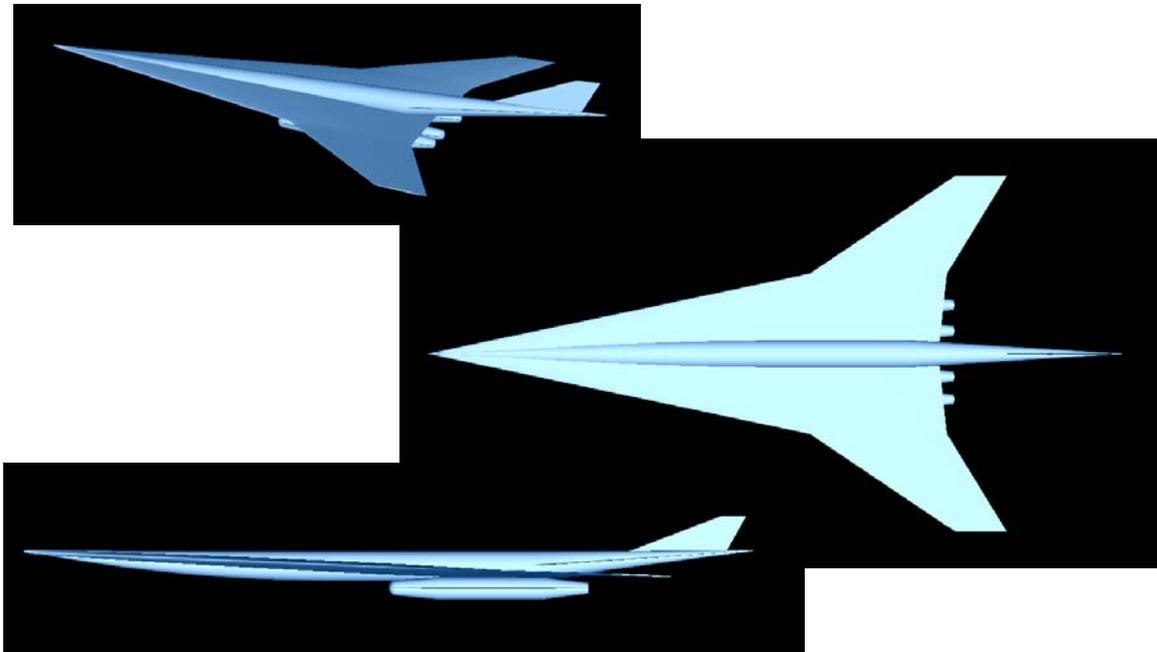


Fig. 18: Preliminary Mach 4.5 kerosene fuelled aircraft for 200 passengers with a MTOW of 720 tons.

The forebody of the concept is slender and elliptical in cross section. The wing-mounted struts of the main landing gear should retract into the engine nacelles and are housed between the inlet ducts as in [15]. The two-wheeled nose gear is mounted on the bulkhead forward of the crew station and retracts forward. The 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 18 are not representative of the actual LAPCAT design. A rectangular shape of the air-intake with vertical ramps as in [15] is the preferred design option.

Fuel is carried in integral wing tanks and in a single aft fuselage tank. It is further assumed that the vehicle uses thermally stabilized jet fuel (TSJF) because the existing airport infrastructure is designed around conventional jet fuel. A single vertical stabilizer is attached to the upper part of the aft fuselage. 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 the first loop to the huge value of 720ton, which is well beyond any supersonic passenger aircraft built to date. The dry mass is estimated at 184.5ton and the structural index is at a for airplanes low 36.8%. 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 [8, 14].

4.1.4.2 Routes

The complete flight trajectory of LAPCAT-M4 from take-off, via ascent, acceleration to the supersonic cruise, descent and landing approach has been simulated using control algorithms described in [14]. The calculation ends after depletion of nominal propellants with descent to sea-level altitude. The vehicle follows almost the orthodrome, however, with some deviations to avoid some densely populated areas, adding about 200km to the shortest great circle route (fig. 19). It's no objective in LAPCAT to prove if the chosen track is actually acceptable for supersonic high altitude flight. Nevertheless, the propulsion system performance should be assessed under realistic operational conditions. Thus, a significant portion of the trajectory (from Brussels in eastern direction up to Volgograd) is assumed to be flown in subsonic cruise.

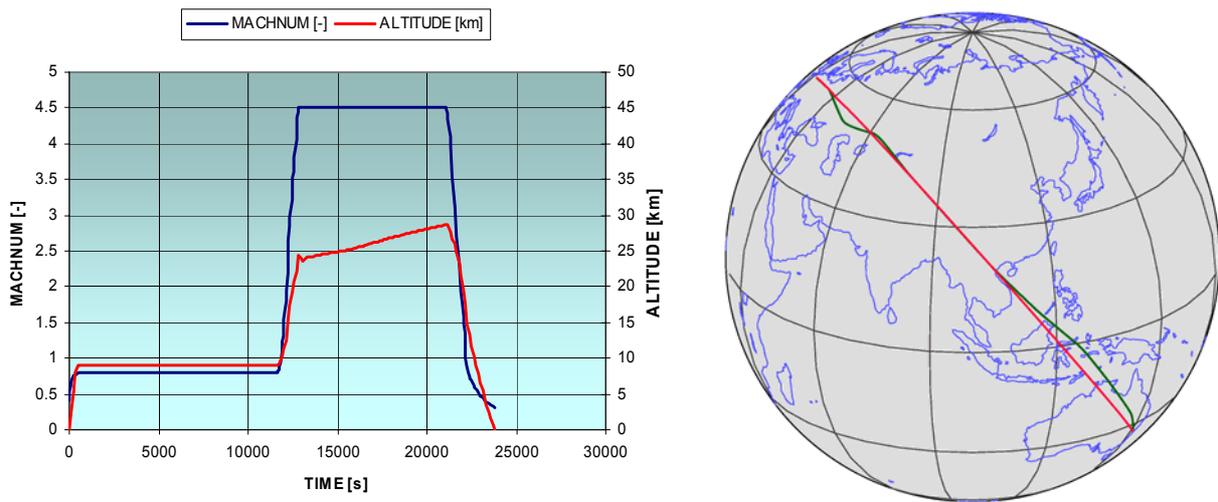


Figure 19: Brussels to Sydney: flight profile and route (16,700 km)

Under these hypotheses LAPCAT-M4 with the available 484.8ton of JP-fuel does not fully reach its final destination Sydney but at least arrives after 15,989 km in central Australia (145.9° E; 26.4° S). Flight time is 23,811s (6.6h). It has not been intended to further enlarge the vehicle to finally fulfil the mission goal. Such work could be the subject of a separate study while LAPCAT focuses on propulsion system research. Simulations show that if a direct acceleration to Mach 4.5 already over the European continent would be acceptable, the HSCT could reach its final destination of Sydney.

Therefore, the obtained results demonstrate the principle feasibility of a TBCC-powered ultra long-haul supersonic airliner. The VCE-214 engine variant with relatively high OPR shows the best performance of all investigated types as long as potentially different engine masses are not taken into account [8].

4.2 Vehicle and Propulsion System Studies based on RBCC

In parallel to TBCC propelled vehicles, Rocket Based Combined Cycles are evaluated for the two vehicle concepts. As the thrust to weight ratios for rockets are far higher (~60-100) than turbojets (~3), they might be a good alternative for the acceleration phase despite their higher sfc. The objectives of rocket based combined cycles are:

- Identification and preliminary definition of system-related parameters for high-speed aircraft and propulsion unit RBCC
- Preliminary design and dimensioning of RBCC engines coupled with vehicle and a reference trajectory

- Evaluation and comparison of RBCC engines
- Trade-off of technical and economic characteristics of RBCC engines
- Comparison of RBCC engine performance with that of TBCC engines

The preliminary design and dimensioning of RBCC engines coupled with vehicle and a reference trajectory was addressed after the first vehicle designs for M4 (kerosene) and M8 (hydrogen) became available. For each of the vehicles, a basic RBCC concept was derived, and tools and rules for dimensioning the RBCC were developed.

4.2.1 Kerosene RBCC engine for Mach 4.5 cruiser

For the kerosene-fuelled RBCC for cruise flight Mach number 4.5, the preferred configuration turned out to be a rotationally symmetric concept with the separated rocket in the center. The ramjet consists of a contoured annulus around the center block which exhausts on a truncated plug nozzle (fig. 20). A preliminary contour approximation was developed to concentrate on HC ramjet combustion on the basis of CFD combustion analysis and to assess the range of combustion efficiencies and characteristic velocities to be expected.

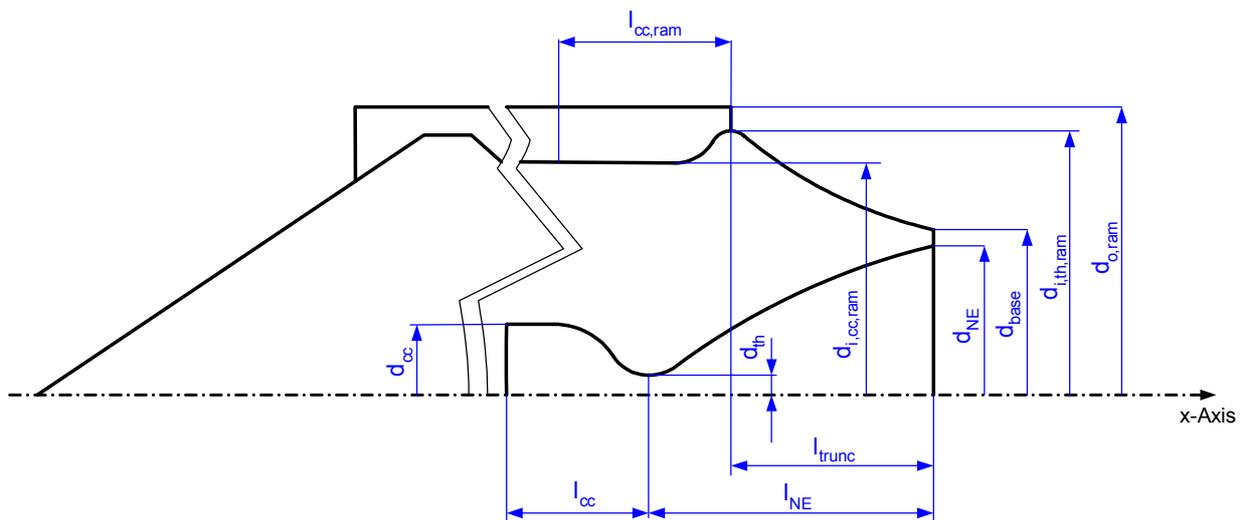


Figure 20: Contour parameters of rotationally symmetric kerosene-fuelled RBCC engine for M4.5 vehicle

The evaluation showed that for the given mission kerosene as fuel was unfeasible, but that the mission can theoretically be achieved using a hydrogen-fuelled RBCC. Also, the performance-sensitive factors have been highlighted and their influence on the net I_{sp} of the RBCC was shown. It turned out that the net I_{sp} performance values of the RBCC are highly dependent on an efficient combustion and nozzle expansion process.

4.2.2 Hydrogen RBCC engine for Mach 8 cruiser

The hydrogen-fuelled RBCC for Mach 8.0 is a planar design with a sophisticated intake system, and rockets integrated into struts. The struts separate short scramjet combustors, for which maximum length and divergence could be determined from basic gas dynamic relations. Also it turned out that it was feasible to design a side-wall injection concept such that the combustor would be sufficiently penetrated with the fuel jet for mixing. The nozzle consists of a single expansion ramp nozzle of the Sanger type and was tentatively demonstrated to be efficient for the proposed vehicle type. After one year the settings for

the basic RBCC cycle input parameters were refined by conducting systematic parametric studies, and by taking into account the latest CFD results about the physics of supersonic combustion from the specialized work packages, allowing a more realistic setting without performance loss at the operational point. Currently, the RBCC engine model for the M8 vehicle is extended to include ramjet combustion and thermal choking to enable the examination of a mixed ramjet-scramjet configuration with different fuel injection positions and side wall struts in the remainder of the system.

The most important issue encountered was the difficulty in compartmentalizing work and developing interfaces between RBCC components as well as between the organizations working on it since an RBCC propulsion system is inherently a highly integrated one. This was addressed by assuming the role of overall engine system responsible via developing guidelines and requirements to component design, and through pursuing decisions on design paths to take within the LAPCAT team.

One important driving parameter was the combustor length, and the drivers' definition for its dimensioning. Regardless of the intricate supersonic mixing and combustion processes the system design team could determine from practical gas dynamic and manufacturing considerations that the scramjet combustion chamber should not exceed a maximum length allowing for a slight divergence to give margin for design issues other than the mixing process. By cooperation with specialized CFD analyses, the assumed model input parameters could be refined in a series of parametric studies to represent more realistic values.

A basic concept for the SERN nozzle and the contouring criteria was defined such that the concept was valid for both the subscale experiments and the fullscale M8 vehicle design. The same applied for the definition of a flexible intake geometry for parametric testing but still representative for the fullscale M8 vehicle. Both intake and nozzle work logic is described below.

The TRL of RBCC propulsion is low and a high degree of uncertainty exists on its actually achievable performance in ejector-rocket and SCRAM-mode. Therefore, an iterative approach in defining the thrust requirements and subsequent calculation of the mission performance has been chosen. All variants studied are based on LH₂ propellant and on LOX as the oxidizer in rocket mode. Hydrocarbon propellants had also been regarded at an early phase but fuel consumption was found tremendously high due to the poor specific impulse in ejector-rocket operation mode. Therefore, all hydrocarbons were dropped quite early in the study as a feasible propellant for LAPCAT-M8. The following paragraphs give a brief overview on the evolution process.

4.2.3 Hydrogen Mach 8 Cruiser

The dimensioning of the propulsion system components allowed the definition of the lower part of the latest generic LAPCAT-M8 airplane geometry as illustrated in Figure 10 through Figure 12. The upper section of the vehicle is dependent on the necessary volume for fuel tanks and the SERN expansion ratio intended to be as far adapted as possible. The vehicle span is influenced by the intake width which is directly proportional to the required thrust. LAPCAT-M8 as a generic airplane is designed as a lifting body with a simple 2D-geometry in the central air-intake part, easing not only the conceptual lay-out but also CFD and experimental investigations (fig. 21). The total length is 101.2 m with a total span of 41.58 m. Its height mounts up to 19.5 m. The outboard region converges rapidly to the "wingtips", so that the leading edge sweep angle is about 82°. The stabilizer located in the tail part of the lifting body and two vertical fins, slightly inclined outboards, are to be used for aerodynamic trim and control. Their respective sizes are not yet designed by flight dynamic and stability requirements.

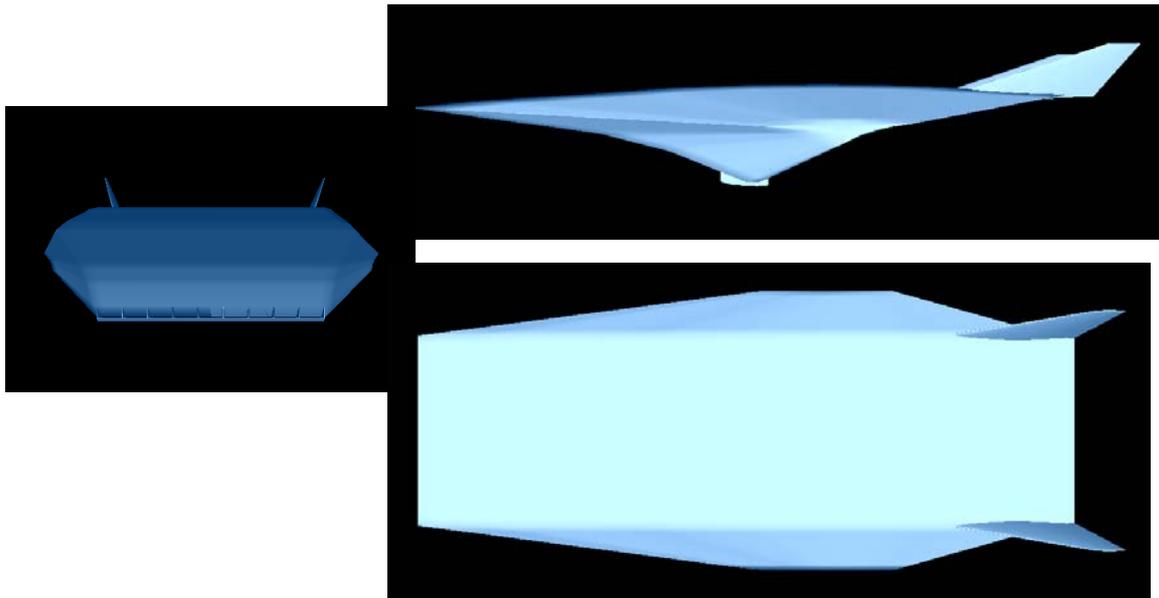


Fig. 21: Preliminary design of a Mach 8 cruiser based on a hydrogen RBCC engine

The engine operational sequence supposes ejector-rocket mode up to $M=2.5$, a pure RAM operation up to $M=5$, and subsequently a transition to SCRAM-jet. In its current configuration based on the best available consistent data the airbreathing hypersonic airliner is not able to reach or come any close the LAPCAT mission requirement. The total amount of required LH2 and LOX-propellant is 669ton. Though using ultra light-weight structural design in high load and very high temperature environment, its empty weight mounted still to 267ton with an incredibly large take-off mass of 944ton.

As the RBCC requires a rocket ejector operation at low Mach number flight, its low performance along with the availability of reliable data, results in very high fuel consumption during the acceleration phase. Its performance is highly critical to overall feasibility. Large scale SCRAM propulsion is yet to be demonstrated. This version of a Mach 8 hypersonic RBCC airliner could reach intercontinental range of up to 9500km. LAPCAT-M8 flight performance calculation should not be interpreted as a proof of its feasibility. Intention is to show “best case” performance and identify critical points.

4.3 Combustion Experiments

Dedicated combustion experiments are clearly needed for both TBCC & RBCC concepts in order to evaluate and check the performance and characteristics at specific conditions for supersonic combustion and high-pressure combustion, to evaluate the performance and achievements for potential fuels, either hydrogen or hydrocarbons (HC) and potential oxidizers (air or liquid oxygen), including the investigation of reaction products and hence the impact on emission requirements, to evaluate the chemical kinetics and its interaction with the flow-field turbulence, its impact on ignition delay and flame stabilization, to evaluate potential injection systems with emphasis on mixing and combustion efficiencies and to set-up a database for modelling and validation purposes in these areas where detailed and dedicated experiments are lacking.

So far, experimental data obtained in supersonic combustion experiments performed in the M11 connected tube facility at DLR Lampoldshausen have been evaluated for differently shaped strut injectors (fig. 22). The main aim was to describe as precise as possible the entry conditions of the flows (pressure, temperature, flow rate, ..) and provide experimental data of physical values which are accessible from CFD solutions such as, wall pressure distributions taken with static pressure measurements, temperature

profiles from CARS measurements and velocity data from PIV. Additionally, qualitative information about the position of shock waves as well as the mixing intensity at some position downstream of the injector were deduced from flow visualisation studies.

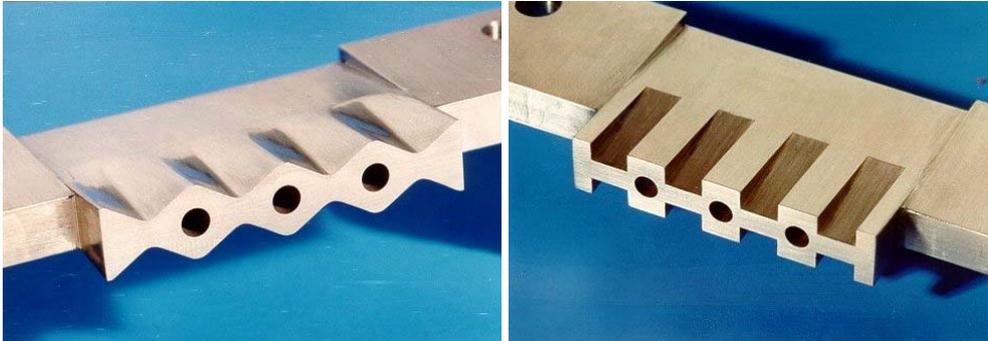


Fig. 22: Different strut injectors tested in M11-facility: left WAVE and right USCER injector

The campaign of testing a complete airbreathing engine in the High Enthalpy Shock Tunnel Göttingen (HEG) is in process. Recent experience gained by designing new operating conditions in HEG have been used to significantly improve the test conditions to be used for the testing in the framework of LAPCAT. Two generic scramjet configurations were selected for the ground based testing in HEG. These two configurations allow comparing the efficiency of two types of fuel injector configurations (perpendicular porthole injection and wall injection using vortex generators). Further, a flexible wind tunnel model with detailed surface instrumentation and optical access was designed and built (fig. 23).

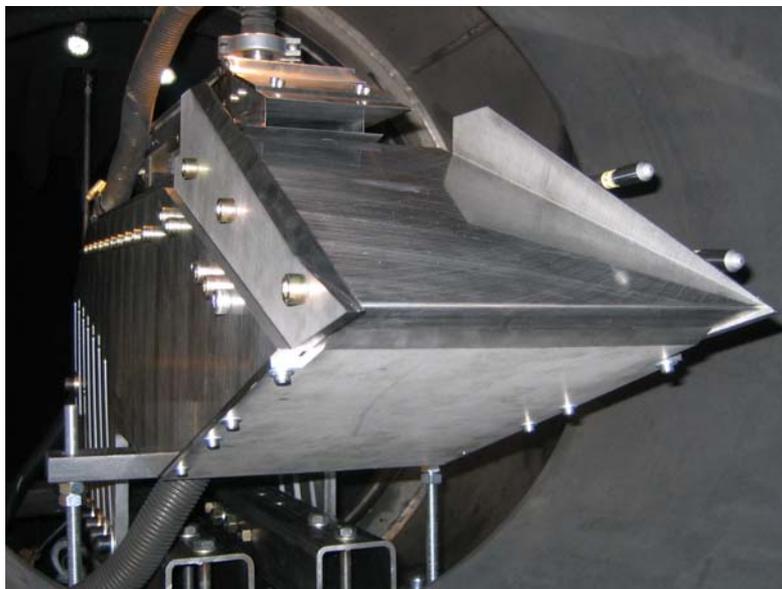


Fig. 23: Full M=8 engine model mounted in the HEG shock tunnel with optical access

In the framework of high pressure combustion experiments with focus on the HC disintegration processes, the ITLR shock tube at the University of Stuttgart (fig. 24) has been equipped with a fast-response fuel injector [16]. An extensive set of "cold" experiments (i.e. with the shock tube not in operation) has been conducted, aimed at characterizing the injection procedure with respect to triggering, delay times, spray duration and fuel injection pressure. The latter was varied over the range 10 - 50 MPa. Subsequently, the

injection system has been tested with the shock tube in operation ("hot" experiments) with timing, spray duration, and fuel injection pressure similar to those employed in the "cold" experiments. As the trigger source, a pressure transducer - located 60 mm from the end wall - is used, which supplies a voltage signal in correspondence of the passage of the incident shock wave. These "hot" experiments were performed for purpose of validating the spray injection technique with respect to reproducibility of the spray, timing of the experiments, and the achievement of steady-state firing conditions. Currently, fluid disintegration experiments are being performed under supercritical and subcritical conditions, employing dodecane (as exemplary hydrocarbon fuel) in argon. The results are visualised by means of flashlight shadowgraphy.

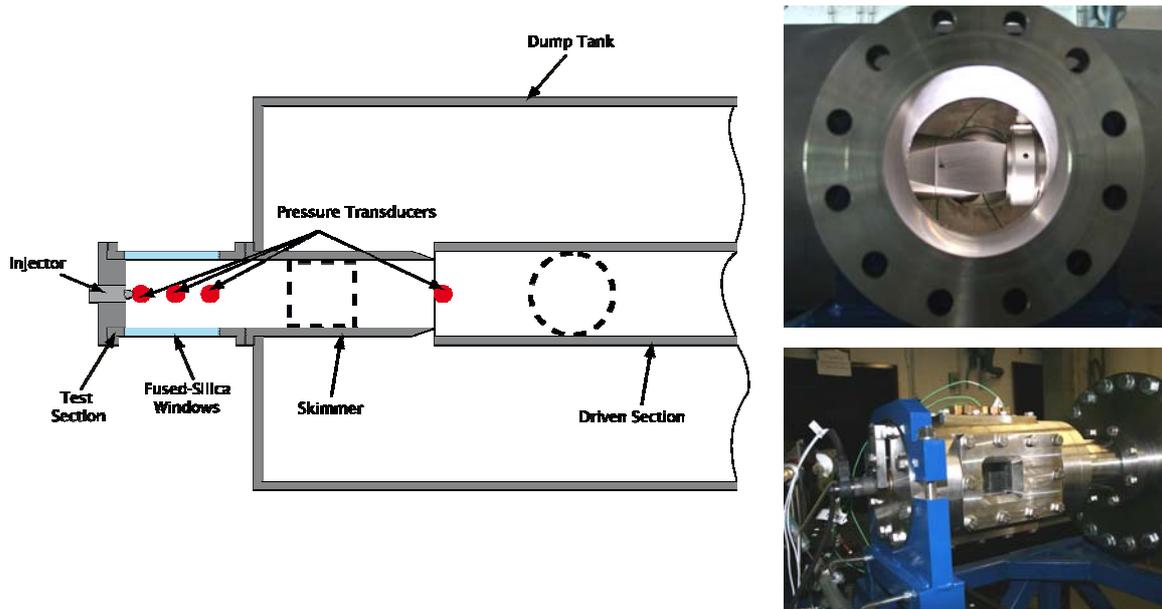


Fig. 24: Shock tube test set-up for investigation of HC disintegration in sub-, trans- and supercritical conditions.

The objective of these experiments is twofold:

- to provide quantitative data on binary fluid disintegration;
- to relate our experimental results (e.g. spray angle, spray penetration length, disintegration regime) to literature data on pure-component fluid disintegration and standard atomization, in order to assess the differences between one-component and binary systems.

It is noteworthy noticing that the analysis of quantitative data and subsequent formulation of empirical correlations for the different break-up regimes has been limited, to date, to one-component systems. This limitation is motivated by the additional difficulties introduced by the occurrence of mixture effects. Due to the enhanced solubility of argon in dodecane at high pressure conditions, the critical pressure of the mixture can be increased considerably above the critical pressure of pure dodecane, depending on the chamber temperature. This leads to some difficulties in analyzing and classifying the experimental results, since the critical point of the binary mixture cannot be known a priori. In order to overcome such difficulties, the development of dedicated software is envisaged, aimed at calculating the fluid-phase equilibrium composition of the binary system dodecane/argon as function of chamber pressure and temperature.

The M3 test facility (fig. 25) at the German Aerospace Center, DLR in Lampoldshausen, designed for the operation with H₂/O₂ (cryogenic or ambient) has been refurbished to allow the use of hydrocarbon fuel such as methane. For this purpose, a new injector head and feed lines have been designed and built.

Moreover, software dedicated for the test bench operation with CH₄ has been developed for the design and the operation of the new hardware. The acceptance tests for operation with LOX/CH₄ have been successfully done. Reliable igniter performance has been demonstrated and first hot runs at representative conditions have been done.

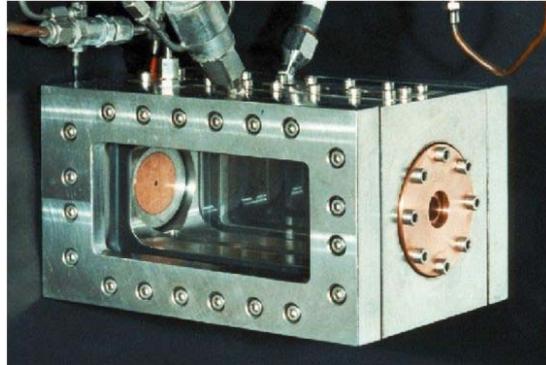


Fig. 25: M3 test facility with optical access

The operation of the laboratory burner provides gas mixtures at representative composition and thermodynamic conditions to develop and test optical diagnostic approaches. Spectroscopy of the flame's emission due to chemi-luminescence and thermal radiation has been analyzed for pressures up to 1 MPa. Quantitative thermometry of the hot gases using CARS-spectroscopy has been started in the CH₄/O₂-flames. Based on the CARS-spectra obtained in the laboratory flames a decision on probe molecules was achieved. For the characterization of the ignition transient and stationary spray combustion software tools are developed to analyze shadowgraphs and high speed recordings of the flame emission.

The intensive literature study and the validation predictions yielded the Leeds mechanism as the most promising candidate mechanism for O₂/CH₄ combustion for further reductions towards a kinetic scheme short enough to become applicable in 3D CFD tools. Within the previous period the mechanism was updated using the recently published data. Additionally, the reactions and reaction rate constants are modified such that they are better suited for our range of parameters which is much broader than the original one. Therefore, the modified mechanism has been extensively and successfully validated by performing verification predictions over the entire range of application using the experimental data base established within the frame of the project. First attempts towards a skeletal mechanism have been performed and the validation predictions are underway but not yet finished.

4.4 Combustion Modelling and Validation

The goal is to investigate physics of supersonic combustion and to develop new tools to enhance numerical simulations. In fact, numerical simulations performed by means of RANS (Reynolds Averaged Numerical Simulations) or LES (Large Eddy Simulations) are fundamental in designing SCRJ combustors; in particular, by focusing on the unsteadiness of the flow, Large Eddy Simulations (LES) can help in understanding how to improve mixing, flame anchoring and combustion efficiency in supersonic reacting flows. A proper reduced kinetic scheme for H₂/air and kerosene/Air is mandatory to correctly reproduce combustion temperature and species in not prohibitive CPU times. In particular, H₂/air kinetics has been examined using different schemes. Development of the RANS codes and of the LES code is proceeding apace.

The extension and validation of the existing axisymmetrical EADS code Rocflam-II towards LOX/HC (Hydrocarbons, i.e. Kerosene and Methane) chemistries has been finished. The necessary fluid data and

reaction schemes have been selected and implemented. The reaction model in Rocflam-II consists of a tabulated equilibrium chemistry with a PPDF (*presumed probability density function*) approach to model turbulent combustion. Code validation has been carried out on available sub- and generic fullscale engines. The code has been further adapted and applied to an Air/Kerosene Ramjet engine. Here, liquid kerosene droplets are injected into a gaseous air surrounding, simulating atmospheric conditions at a sea level altitude of ca. 25 km. The air is already compressed and heated up in an inlet previously, so the combustion process will take place under subsonic conditions.

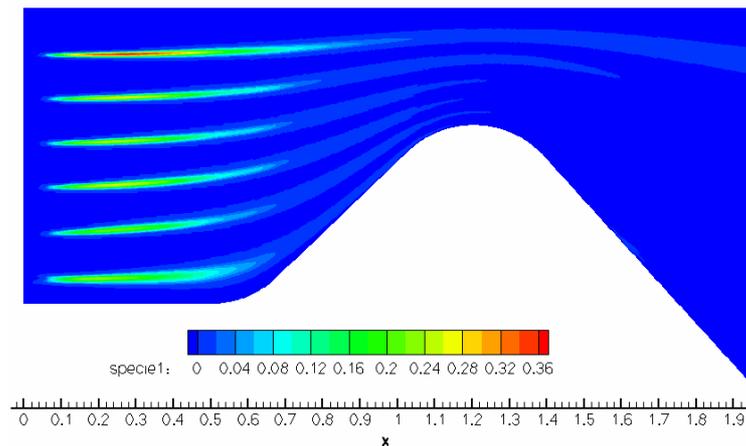


Fig. 26: Ramjet Combustion Modelling for Kerosene-Air: mass fractions contours of Jet-A.

CIRA selects the code C3NS-PB as starting platform, consisting of two different modules: A pressure-based module (C3NS-PB) and a density-based module (C3NS-DB). The pressure-based module is able to describe the flow of a 3D, unsteady, turbulent, chemically reacting mixture of ideal gases. A thermodynamic model able to properly describe propellants injection in high pressure LOx/HC rocket thrust chambers has been developed and implemented.

Two different approaches (compressibility factor formulation and analytical formulation) have been selected: the accuracy and robustness of both approaches have been preliminary assessed through comparison with experimental data available in literature. Furthermore, the compressibility factor formulation has been implemented in the flow solver and a suitable test case has been selected to test the capabilities of the model in typical rocket operating conditions (supercritical injection pressure, transcritical injection temperature): the CFD simulations have shown a good agreement with experimental data (fig. 27). A basic turbulent combustion model (Eddy Dissipation concept) has been also implemented in the code.

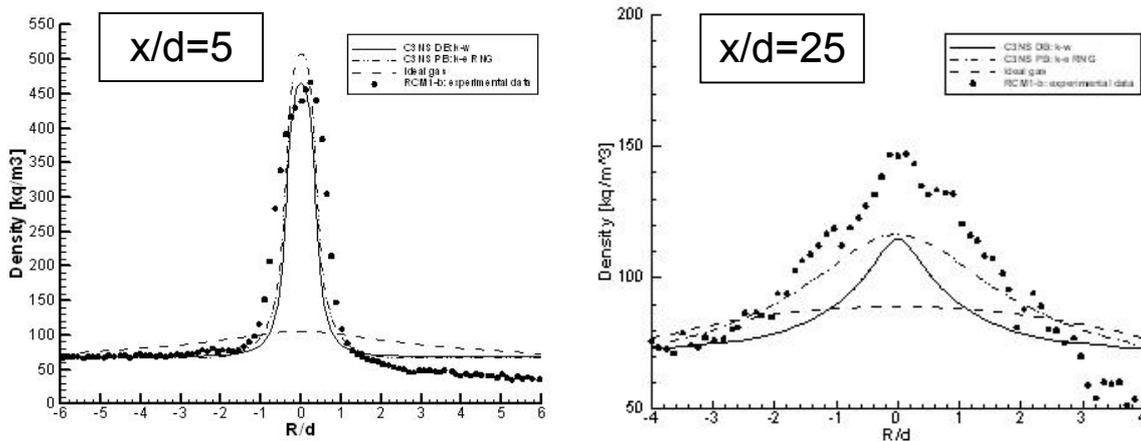


Fig. 27: Influence of thermodynamic (real gas EOS on the basis of compressibility factor) and turbulence modelling: Radial density profiles at different axial stations

In order to simulate supersonic combustion processes in scramjet engines, the following upgrades and developments of the density-based module have been done:

- code extension to a generic multi-component mixture of reacting gases;
- input capabilities have been generalized to allow the input of generic number of species and reactions, with their corresponding physical-chemical properties;
- thermodynamic and transport models have been modified to analyse mixtures of thermally perfect gases;
- a new set of boundary conditions have been introduced in order to simulate mass injections with a specified composition.

A laminar combustion model for a gaseous hydrogen/air mixture has been implemented in the DLR Tau code. It is based on an existing extension of aerothermodynamic description of the re-entry flight of space vehicles. Partition functions for individual species and appropriate mixture rules are used to compute the thermodynamic and transport properties. This offers the possibility of future model extensions (e.g. multi temperature models to account for thermal nonequilibrium). Appropriate Dirichlet and von Neumann boundary conditions for the species mass fractions at fully catalytic and non catalytic walls have been implemented. Transport coefficients are computed for the individual species. Suitable mixture rules are then used to obtain the mixture viscosity, heat conductivity and diffusion coefficients. Basic test cases have been used to verify correct implementation and results for laminar reactive flow. Two reaction rate schemes with different level of complexity are implemented in the code.

The model has been further extended with an assumed PDF method to account for the influence of the turbulent flow in Scramjet combustors on the chemical production rates. The assumed PDF approach has been chosen from many available turbulent combustion models for the following reasons:

- Wide range of applicability for premixed, non-premixed and partially premixed combustion and different Damköhler numbers,
- Limited computational cost, high robustness and straightforward implementation,
- Convergence acceleration techniques can be used because the PDEs for the needed temperature and concentration variances have the same form as the Navier-Stokes equations (contrary to e.g. multi-dimensional PDEs used for transport PDF methods that need to be solved with particle methods).

The complete supersonic combustion model in TAU was validated and tested using experimental benchmark results for a supersonic coaxial burner which produces a lifted flame at typical scramjet flow

conditions (fig. 28). The flame lift-off distance and flame structure could be well reproduced by the numerical investigation of this test case using the DLR Tau-Code. Application to a full-engine scramjet simulation can be found in [17].

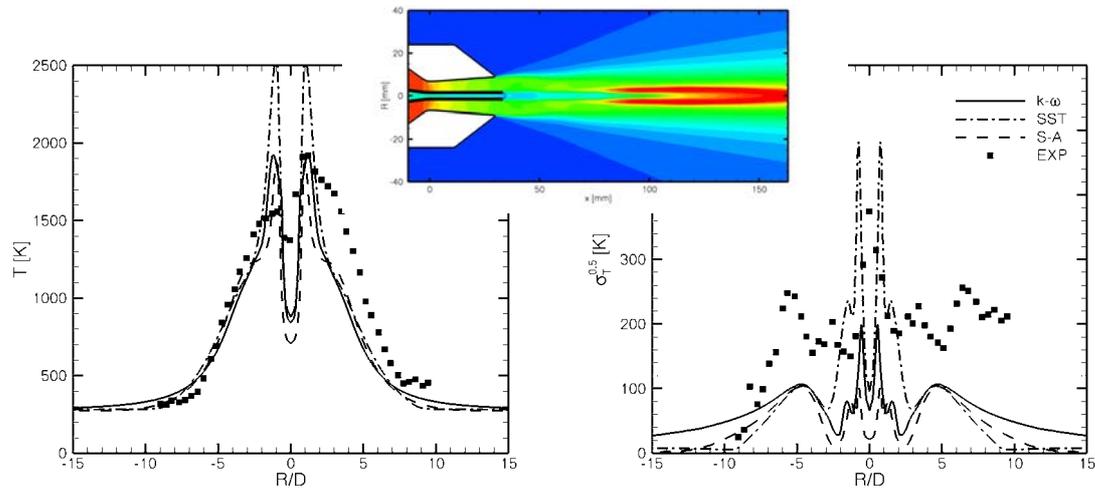


Fig. 28: Radial distribution of temperature (left) and standard deviation of temperature (right) 50 mm downstream of the nozzle exit (middle).

The study concerning the applicability of several hydrogen/air reaction mechanisms for scramjet applications has been completed. Seven detailed (9-species, 19- to 27-step), one reduced (7-step, 7-species) and one global kinetic scheme (in the last 6 month) have been investigated. The basic result is that at critical conditions close to the ignition limit of hydrogen (this corresponds to low flight Mach numbers of a scramjet) only detailed mechanisms are able to accurately predict the ignition delay correctly. However, there are also significant differences between the detailed kinetic schemes and the best suited mechanisms for scramjet simulations are identified.

Next different lobed strut injector concepts have been investigated. In case of axial fuel injection methods for mixing enhancement are required to enable a short combustor length and to reduce the skin friction drag. Possibilities for mixing enhancement may be based on the use of shock waves or on the production of strong streamwise vorticity. The last concept has been investigated numerically using the TASC3D code. The strength and size of the vortices induced may be modified by changing the strut geometry. Different lobed strut injectors are compared in a cold non-reacting mixing study with respect to their mixing efficiencies and losses in total pressure. Aim is to produce vortices which cover large parts of the combustor cross section. A comparison with experimental data for one strut geometry has demonstrated the numerical accuracy of the code. At low flight Mach numbers of a scramjet (7 to 8 as in LAPCAT) detached flames are possible in case of axial fuel injection. Thus the degree and speed of mixing has a strong influence on the ignition delay too. The next step in this investigation will be to extend the studies to hot reactive flows with combustor inlet conditions, corresponding to the LAPCAT flight Mach number.

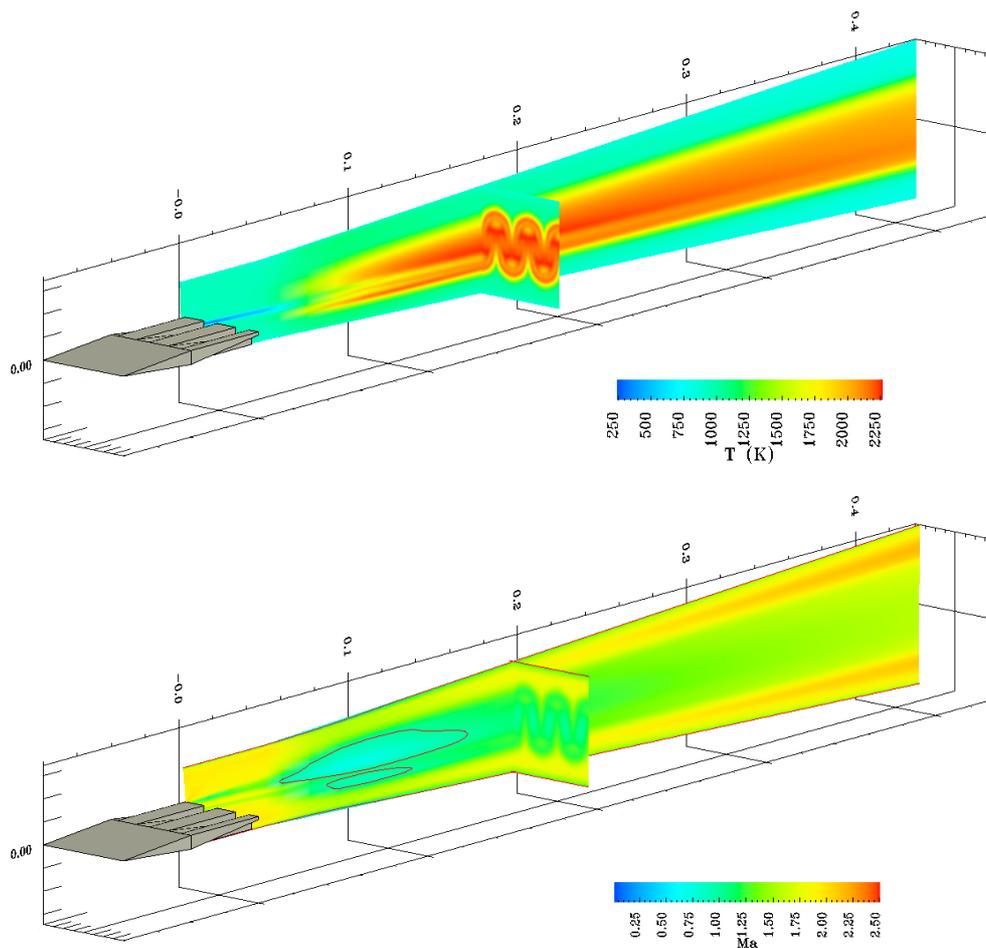


Fig. 29: Flame stabilization of a lifted flame for a lobed strut injector based on a multi-variate assumed PDF closure plotted with contours of temperature and Mach.

In parallel simulations of the HyShot flight experiment have been performed. While the simulation of the inlet and suction region of the engine is completed, the combustor simulations are still work in progress. They are performed using the multi-variate assumed PDF (probability density function) approach and finite-rate chemistry. First simulations demonstrated a strong influence of the chosen inflow and boundary conditions on the flame.

Results obtained by LES simulation (fig. 30) indicate combustion may be made to take place in a short distance (about 1 m or so) by supersonic injection of hydrogen inside the supersonic airstream [18]. The ISCM LES SGS model has been validated using the NASA-Langley data of the supersonic combustion experiment carried on for the NATO RTO, AV-10 a few years ago [19]. In this context, much attention has been put on scaling laws for the mixing and chemistry of supersonic combustion. Although not originally contemplated, scaling laws were examined in order to produce a rough but workable first cut for the combustor length. In fact, combustor length shapes vehicle length, especially that where cross section is maximum, thereby affecting strongly overall weight at take-off.

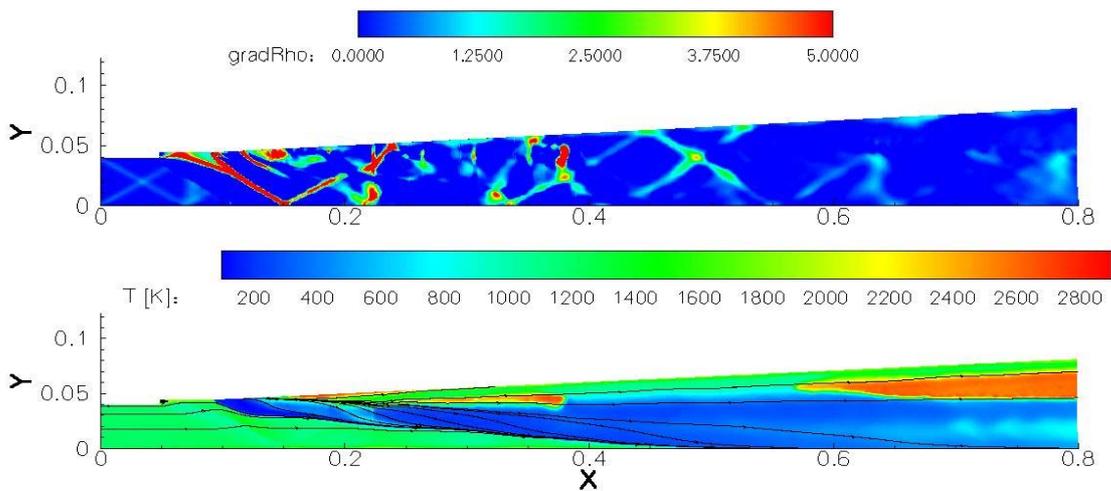


Fig. 30: Density gradient and temperature contour plots obtained with LES-ISCM turbulent combustion model [18].

The conclusions reached so far from theory and simulations indicate that at low air inlet temperature (about 1000 K) combustion length, L , is controlled by the interplay between convection, turbulent mixing and kinetics. As a first approximation, flame anchoring may be predicted according to the Damköhler number scaling $p \cdot L = \text{constant}$. At air T of order 1400 K and higher, kinetics is fast, so that combustion length is dominated by mixing time. Scaling in this regime becomes more complex: in fact, a relationship has been derived that predict flame length and anchoring based on the airstream velocity, angle of fuel injection and fuel jet momentum. This relationship has been tested using the NASA-Langley test case and predicts well the distance from injector where flame is observed to anchor. The actual distance is, in fact very short (of order 1 m). This result allows sizing a combustor realistically. Because the combustor length is short, total vehicle length may now be also determined.

4.5 Design and Aerodynamics of Propulsion Components

For high-speed transportation vehicles powered by air-breathing engines achieving a positive aero-propulsive balance is crucial for the success of the whole system. Along the last 40 years, almost all attempts to fly a scram-jet propelled vehicle have failed due to largely underestimation of the vehicle total drag with respect to the allocated thrust. One of the lessons learned of such experiences is that the vehicle design requires an optimized propulsion airframe integration resulting in an extremely coupled development procedure of the system components, namely the intake, combustion chamber, thrust nozzle and airframe. However, this last statement is not easy to realize since there is no ground facility in the world which allow testing a real sized vehicle under flight conditions including operating engines and furthermore, till today no scaling rules are available at all. According, the only successful flight of a vehicle propelled with a scram-jet, e.g. the X-43, has been done with a vehicle sized to a scale compatible with the size of the ground based facilities used for its design. As like other areas of the hypersonic technology, here the potential of the CFD tools for ground to flight extrapolations is coming on request. To accomplish with that mission CFD tools require dedicate validation experiments, hence specific tasks focus on the carefully design and experimental testing of intakes, nozzles and the interaction with the external flow to be used for CFD validation. Furthermore, being the loss of thrust due to earlier nozzle flow separations and the vehicle total drag the main issues required to be predicted, important efforts are assigned to test advanced turbulence models (fig. 31). Finally, since one of the major potential show-stoppers of intakes is the transition of the incoming boundary layer from laminar to turbulent, specific efforts are allocated to fix from the very beginning of the intake operation turbulent flow.

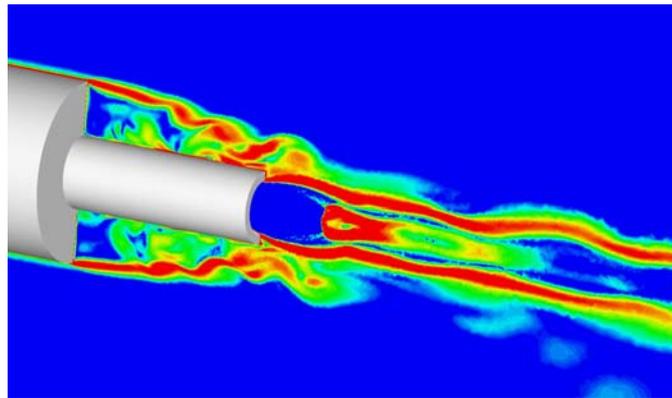


Fig. 31: Base drag estimation by advanced turbulence models: simulation of unsteady turbulent phenomena in base flow regions.

For the definition of the required experimental models it was decided not to use any generic intake or nozzle / base flow but those resulting from the vehicle system parameters studies. Accordingly, two highly flexible wind tunnel models, allowing many configurative variations have been designed and are today under construction: one for the intake (fig. 32) and one for the nozzle flow/external flow interaction problematic (fig. 33). Both models have been designed taking into account the nominal flight conditions resulting from the project system study but also accounting for the facilities capabilities. Several types of CFD turbulence model have been evaluated with respect to geometrical configuration constraints and numerical dissipation. It turns out a clear advantage for the highly developed models even they require large computational resources. The capabilities of different turbulence models for the simulation of unsteady turbulent phenomena have been investigated. The study has shown that the superior results of Detached Eddy Simulation models against Unsteady Reynolds Averaged Navier-Stokes models and the deep insight into the unsteady flow physics are purchased by a significantly higher complexity of the computation.

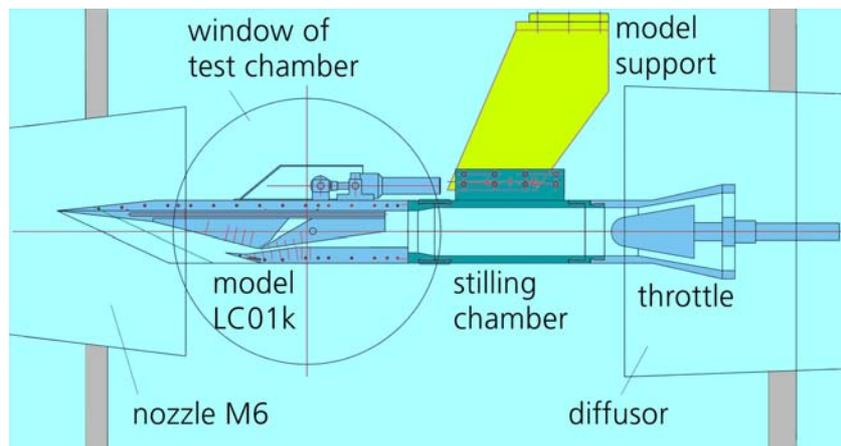


Fig. 32: Intake model without forebody for M8 vehicle.

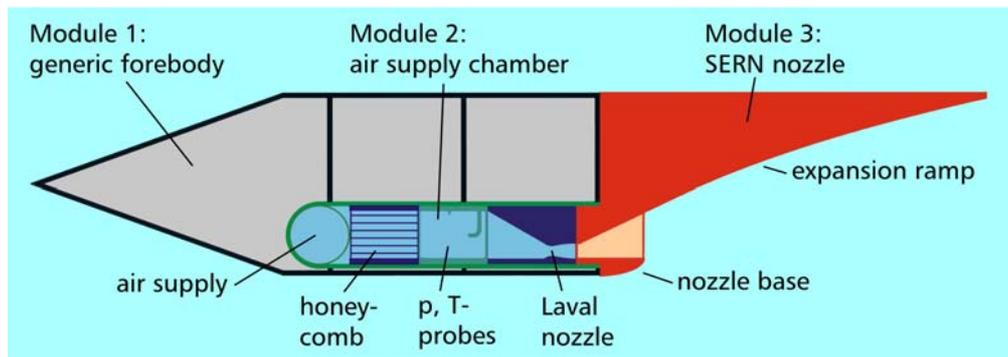


Fig. 33: Concept of a Single Expansion Ramp Nozzle and Base Flow experimental model

Furthermore, large eddy simulation models have been found by comparison with direct numerical simulation (fig. 34), to be accurately enough tools for the prediction of supersonic shock induced laminar-turbulent boundary layer transition. It has been shown that it is possible to have self-sustained transition to turbulence in a shock-induced separation bubble provided the pressure rise over the bubble is high enough. The design of turbulence devices for forcing turbulent flow is underway.

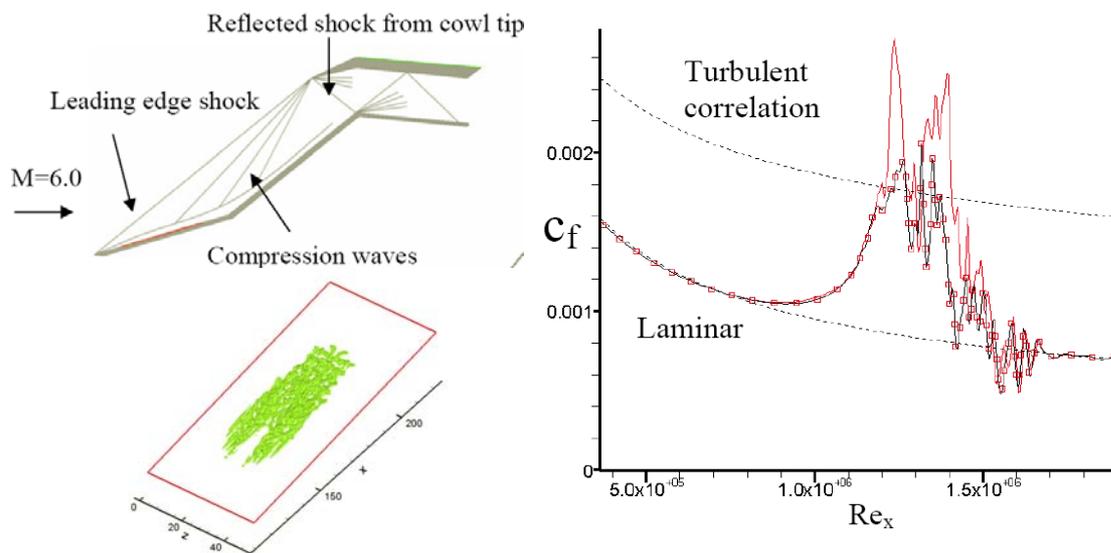


Fig. 34: Skin friction of a passing turbulent spot on an intake ramp by LES for a flight Mach=6; Left: configuration with turbulent spot shape; Right: skin friction induced by spot: legend: symbols LES; full line DNS.

5.0 CONCLUSIONS

Based on general trends in the evolution of aircraft performance and the possible aerodynamic and propulsive achievable efficiencies for high-speed vehicles, there's a potential to achieve antipodal range. LAPCAT wants to (re)-evaluate SST and to go beyond the material's limit imposed for Concorde by integrating light-weight advanced materials allowing speeds 4 to 8 times the speed of sound. Preliminary parametric studies within the Lapcat project have shown so far that Mach 4-5 is achievable and not marginal. However, for the M=8 RBCC propelled vehicle more detailed investigations are needed and ongoing to ascertain its performance.

These vehicle system studies allowed the definition of operational conditions of interest for detailed experimental and numerical work. Windtunnel models are in preparation and will try to reveal or justify some of the used parameters. Also numerical work is well underway and will be soon validated with the newly generated experimental database.

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