Ramjet and Dual Mode Operation

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ABSTRACT

During last twenty years, a large effort has been undertaken in Europe, and particularly in France, to improve knowledge on hypersonic air breathing propulsion, to acquire a first know-how for components design and to develop needed technologies.

On this scientific and technology basis, two families of possible application can be imagined for high-speed air breathing propulsion: reusable space launcher and military systems.

By combining the high-speed air breathing propulsion with a conventional rocket engine (combined cycle or combined propulsion system), it should be possible to improve the average installed specific impulse along the ascent trajectory and then make possible more performing launchers and, hopefully, a fully reusable one. A lot of system studies have been performed on that subject within the framework of different and consecutive programs. Nevertheless, these studies never clearly concluded if a space launcher could take advantage of using a combined propulsion system or not. Further works are under progress to design more efficient vehicle concepts taking all advantages of the air breathing operation while preserving the mechanical efficiency of the vehicle airframe.

Different possible military applications can be proposed:

- *tactical missile when penetration is the key factor or when pure speed is necessary against critical time targets,*
- *high speed reconnaissance drone with improved mission safety and response time capability,*
- *global range rapid intervention system based on a large body airplane, equipped with high-speed very long range drones and missiles controlled by an on-board analysis/command team.*

Considering required technology level and development risk for these both applications, it appears clearly that military application could be developed more rapidly.

Development of operational application, civilian or military, of the hypersonic air breathing propulsion depends of two key points: development of needed technologies for the fuel-cooled structure of the propulsion system, capability to predict with a reasonable accuracy and to optimize the aero-propulsive balance (or generalized thrust-minus-drag balance).

The most part of the technology development effort can be led with available ground test facilities and classical numerical simulation (thermal, mechanics …).

On the contrary, before any operational application, it is mandatory to flight demonstrate the capability to predict the aero-propulsive balance, providing sufficient margins to start a costly development program.

Falempin, F. (2008) Ramjet and Dual Mode Operation. In *Advances on Propulsion Technology for High-Speed Aircraft* (pp. 7-1 – 7-36). Educational Notes RTO-EN-AVT-150, Paper 7. Neuilly-sur-Seine, France: RTO. Available from: http://www.rto.nato.int.

R&T effort led today in France should allow better estimating advantages provided by high-speed air breathing propulsion and building a coherent development capability including methodology, facilities and adapted technologies.

On the basis of already existing and expected results of current R&T effort, some system studies have been re-started and provide very promising results for the space launcher application.

INTRODUCTION

The ramjet/scramjet concept constitutes the main air breathing propulsion system which can be used in a very large flight Mach number range up to Mach 10/12.

During last twenty years, a large effort has been undertaken in Europe, and particularly in France, to improve knowledge on hypersonic air breathing propulsion, acquire a first know-how for components design and develop needed technologies.

On this scientific and technology basis, two families of possible application can be imagined for high-speed air breathing propulsion and will be discussed hereafter.

But, prior to the development of such operational applications, it is mandatory to solve two key technology issues which are the accurate prediction of the aero-propulsive balance of an air breathing vehicle flying at high Mach number and the development of high-temperature structures for the combustion chamber, able to withstand the very severe environment generated by the heat release process while ensuring reliability and limited mass.

RAMJET/SCRAMJET PRINCIPLE

In a ramjet engine, the initial compression is directly operated inside the air inlet which, by slowing the flow, raises the pressure without any compressor, so that there is also no need for a turbine. This turns to be a very simple system, avoiding all kind or turbo machinery, and associated limitations. Nevertheless, such process becomes really efficient only when the natural compression provided by the inlet is sufficiently high, i.e. approximately to Mach 1,5/2. Therefore, every ramjet-based system needs some additional propulsion for initial acceleration. As soon as the starting point is reached, the ramjet engine is able to provide performance which improves up to Mach number 3.5/4, due to the higher temperature and pressure obtained in the combustion chamber, providing better combustion conditions.

In a conventional ramjet, the airflow is slowed from supersonic speed down to subsonic speed (Mach ~0,3) through the shock system created by the forebody and the compression ramps of the air inlet, simultaneously raising the air temperature. Low speed and high temperature provide very favourable conditions for injecting, mixing, and burning the fuel.

However, the shock system is also a source of entropy losses, which increase with the strength of the shocks, in direct connection with the increase of the upstream Mach number (Minf). These pressure losses reduce compression efficiency. In the same time, the temperature level becomes very high at the entrance of the combustion chamber, causing two related problems:

- internal structures are exposed to high thermal loads, even before the combustor,
- heat addition to an already hot air stream becomes less efficient.

The decrease of efficiency of ramjet performance starts around Mach 5 conditions, so that its potential operation is very limited above Mach 6 or 7.

To overcome this limit, a good solution lies in decelerating the upstream flow but still maintaining supersonic conditions (Mach 2 or 3 for example), thus limiting the pressure losses, allowing an efficient heat release by combustion and lowering the thermal loads on combustor walls. Considering that the residence time at such supersonic speed is something like one millisecond, the problem is to organise efficient injection, mixing, ignition and heat release before the fuel can escape non-burnt to the nozzle. If so, we obtain a supersonic combustion regime, and the engine is called a supersonic combustion ramjet, or scramjet.

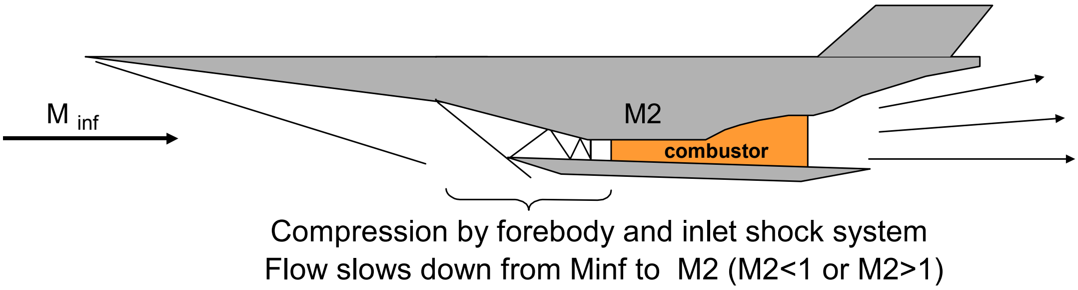

Figure 1. Scheme of ramjet/scramjet (DMR) system

A Dual Mode Ramjet (DMR) is a ramjet engine which can be operated in both subsonic and supersonic combustion mode. DMR operation can be obtained using a fixed geometry if the overall Mach number range is not too wide (i.e. Mach 4 to 8). Extension of this Mach number range, and particularly towards lower Mach numbers, implies variable geometry for the air inlet and/or the combustion chamber. Nevertheless, different solution can be envisaged in order to obtain satisfactory operation of a DMR in the range Mach 2 to 12 within a single integrated engine.

Performances achievable by the ramjet/scramjet in term of specific impulse are illustrated by figure 2 in the case of hydrogen fuel.

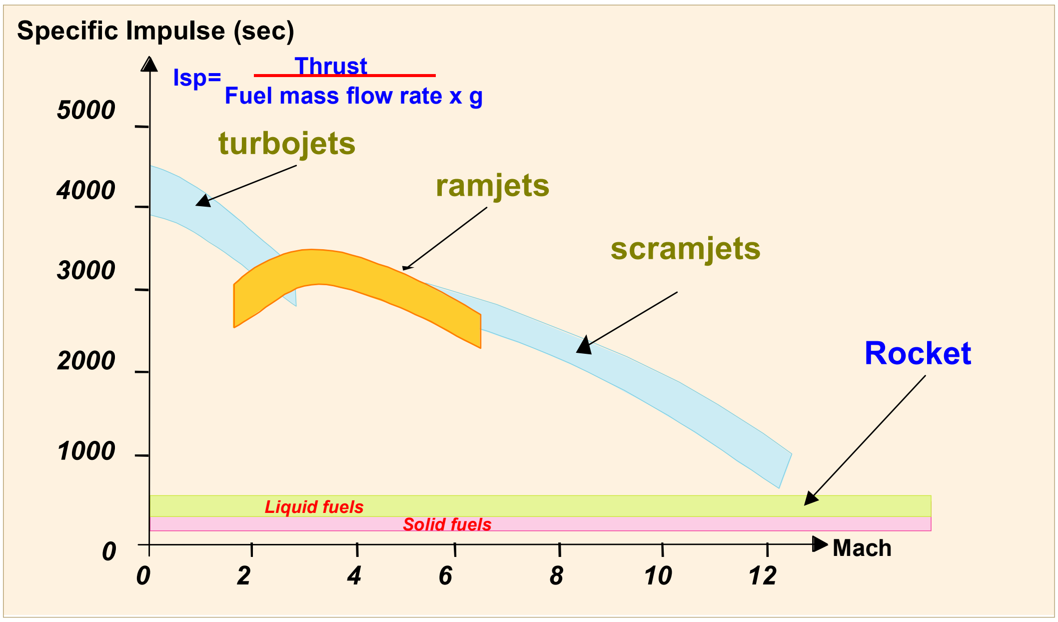

Figure 2. Achievable performance with ramjet/scramjet

POSSIBLE APPLICATION

Considering the previous elements, two families of operational application can be imagined for high-speed air breathing propulsion:

- combined air breathing/rocket propulsion for space launcher,
- military systems, mainly missiles and drone.

Space launcher application

By combining the high-speed air breathing propulsion with a conventional rocket engine (combined cycle or combined propulsion system), it should be possible to improve the average installed specific impulse along the ascent trajectory and then make possible more performing launchers and, hopefully, a fully reusable one.

A lot of system studies have been performed in France on that subject within the framework of different and consecutive programs (Ref [1]). Nevertheless, these studies never clearly concluded if a space launcher could take advantage of using a combined propulsion system or not.

As a matter of fact, past studies were performed sometimes by different teams with different tools and hypothesis, sometimes for particular purpose. For example, the purpose of system studies led in the framework of the National PREPHA program (Ref [2]) was not to assess the feasibility of a fully reusable Single-Stage-To-Orbit (SSTO) space launcher but only to determine some general technical requirements for the study of a scramjet system.

By another way, these studies used systematically a very conservative approach in term of vehicle airframe configuration and it is doubtable that the best trade-off between air breathing propulsion mode needs and the mandatory low dry mass for the vehicle and its propulsion system was obtained with the considered vehicle concepts.

What could be the individual impress or opinion one can have (Ref [3]), it has to be noticed that a large worldwide effort is still under progress for developing the high-speed air breathing propulsion technology in USA, in Japan, in Australia, in Russia, in India, In China but also in France (Ref [4] to [10]).

In that context, a brief review of some of the main design issues of a future space launcher using combined propulsion leads to propose a focused approach for further new system studies which could take into account the progress made these last years in the related technologies.

SSTO

A large part of the past system studies were focused on SSTO application. As a matter of fact, it is clear that the ultimate goal must be the development of a SSTO to finally make the access to space a daily routine with corresponding low cost and, then, to develop new unexpected markets.

It is generally accepted that a fully pure rocket powered SSTO is not feasible with an achievable dry mass. By comparison, studies performed during the PREPHA program led to the conclusion that the combined propulsion could largely improve the feasibility of a SSTO if the air breathing mode can be efficiently used on a very large flight Mach number range (i.e. from Mach 1.5/2 to Mach 10/12) (Ref [11]).

Nevertheless, for such application, the payload mass is a very limited part of the total take-off mass and the remaining uncertainties related to air breathing mode performance and to achievable vehicle dry mass are of the same order of magnitude, making impossible to conclude on the real feasibility of such a SSTO launcher.

By another way, due to the extreme sensitivity of the payload mass for a SSTO launcher, the development of an operational vehicle integrating a completely new and very complex propulsion system would correspond with an unacceptable development risk level.

TSTO

On the contrary, it seems to be relatively easy to develop a Two-Stages-To-Orbit (TSTO) launcher with an air breathing first stage.

Remark : it would be also possible to place the air breathing mode on the second stage. But in this case, a large part of the problems related to the SSTO would remain (for example : heavy propulsion system placed into orbit, atmospheric re-entry of an air breathing vehicle) and would combine with the difficulty one can encounter for the flight back to the launching pad of a rocket powered first stage.

First studies led in France were considering different kinds of combined propulsion systems, for the first stage, with an air breathing mode limited to Mach 6/6.5. They showed that a combined propulsion system was feasible but they also showed that this propulsion system did not improve the overall performance: the better average specific impulse being compensated by the increase in dry mass. Moreover, the pure rocket second stage remained not so easy to develop.

Other studies were performed to assess the interest of an extended air breathing mode by considering a TSTO with a first stage operating up to Mach 10/12 by using a scramjet mode. Obviously, such a solution largely eases the development of the second stage. But, it corresponds with a very complex first stage vehicle.

For all the previous studies, the staging was very close to the end of the air breathing mode. Some complementary studies showed that it would be very interesting, from the point of view of payload mass/gross weight take-off to fix the staging Mach number largely beyond the end of the air breathing mode (Ref [12]). Nevertheless, that would correspond with an even more complex first stage vehicle.

In any case, one can assume that a TSTO would be feasible with pure rocket mode on the two stages (even if it can be necessary to add a limited speed air breathing system to allow a direct and safe flight back to the launching pad for the first stage). Then, even if the combined propulsion can increase the performance in term of payload mass/total take-off mass, it would not avoid, and if fact it would reinforce, the difficulties related to the development and the operation of two complex vehicles. In these conditions, the development of a completely new propulsion system cannot make sense.

Near Earth Orbit

On the base of previous discussion, it appears that further system studies should address the concept that could be called "Near Earth Orbit”" (NEO) (Ref [13]).

Indeed, the use of a very limited expendable upper stage just avoiding to really place into orbit the vehicle can largely increase the payload mass (Fig. 3). For example, the generic mission for the SSTO studied within the PREPHA program was to reach a circular orbit at 500 km. 8 metric tons of propellants were needed to circularize the vehicle orbit and, then to de-orbit. In the case of a NEO, the most part of this mass could be complementary payload improving the performance or could be considered as design margin reducing the development risk.

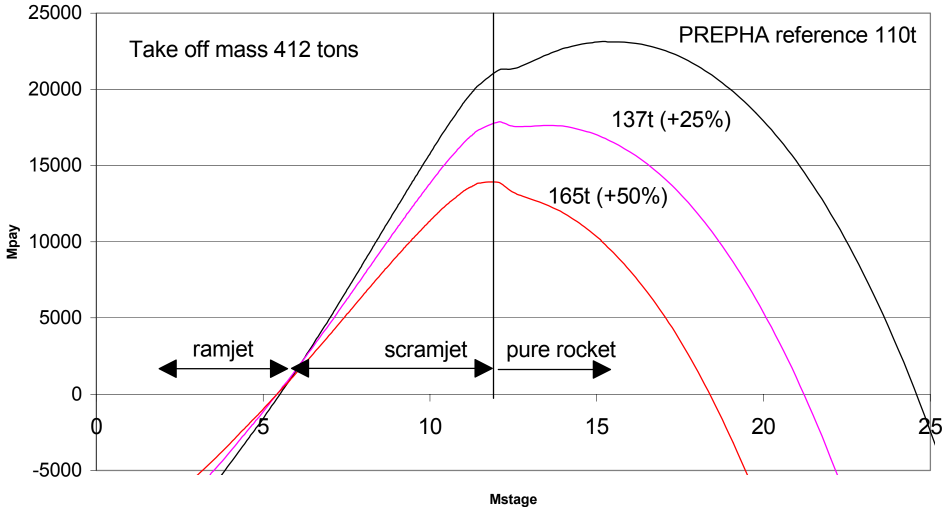

Fig. 3 –Effect of staging Mach number on payload for a TSTO using a low performance second rocket stage (Isp = 340 s and dry mass = 12% of total mass)

Even if the NEO is not a real SSTO, it would be design as a SSTO and would take into account all the requirements related to the flight outside the atmosphere (attitude control for example) and to the atmospheric re-entry. Then, considering that the number of reusable space launchers will be limited and that these vehicles will remain some kinds of prototypes, it could be possible to integrate step by step some performance to finally reach the real SSTO mission.

Remark : It has to be noticed that the design of the vehicle could take into account the possibility to really place the vehicle without payload in orbit in order to make possible the capture of a payload in orbit before returning on ground.

Such concept appears very attracting and is currently studied within MBDA France. A few specific information elements are given on that study in the last part of the present paper.

Military application

Nevertheless, considering following points, it appears clearly that military application and more specifically missile application could be developed first:

- The space application draws maximum benefit from air breathing propulsion when using it up to Mach 10-12, in order to optimize the staging of the different propulsion modes. On the contrary, the military interest of high speeds can be reached significantly below this domain. Mach 8 should not be very far from the upper limit for missile applications, and the Mach domain to be addressed through the related studies is then reduced.
- In its whole flight envelope, the space launcher has to provide a very large acceleration, which is one of the key parameters to provide sufficient payload performances into orbit. A cruising military system has naturally less needs in terms of acceleration capability at high speed.
- Test facilities, developed in the frame of PREPHA, were designed to test components of the propulsion system of a launcher at much reduced scale and in a limited Flight Mach number conditions range (up to Mach 7.5 when operational engine would have to operate up to Mach 10/12) but they nearly enable to test a missile engine at full scale. This situation contributes to reduce the uncertainties remaining after ground tests to get to flight tests. Scale effects will necessary have to be addressed for space application first by numerical simulation and then, may be, by combining step-by-step demonstration vehicles.

- Finally, it is clear that if a flight demonstration was made using a vehicle whose size would have been chosen minimal for together preserving the demonstration interest of the operation, and limiting the cost, this minimal size would probably be not very far from the size of a missile (4 to 6m). Consequently, the success of the flight demonstration would validate the methodology used to develop the experimental vehicle, so that this methodology would also be applicable for any kind of vehicle of similar size and level of integration. With this basis, the development of an operational vehicle could be started with sufficient design margins and limited technical risk. On the contrary, even if such a demonstration would add a lot of credit for general use of dual mode ramjet, space launcher application would still require further developments before an operational vehicle program can be started.

Then, different possible military applications can be proposed (Ref [14]):

- tactical missile when penetration is the key factor or when pure speed is necessary against critical time targets,
- high speed reconnaissance drone with improved mission safety and response time capability (Ref [15], [16]),
- global range rapid intervention system based on previously mentioned missiles and drones,
- global range military aircraft or UCAV,
- short response time space launching system.

From an European point of view, it is clear that a global range military aircraft is out of possibilities (at least largely beyond the first quarter of the XXIst Century). In the same way, the development of a specific military space launcher can not be foreseen in the first half of the Century. Nevertheless, one can dream that before this time a fully reusable space launcher, mainly developed for civilian missions, will be able to provide rapid access to space for military operations (unpredictable flight over hostile zone, refurbishment or repairing of damaged satellites…).

On the contrary, it is probable that missiles and drones could appear within the two next decades.

In any case, it is clear that military application of high-speed air breathing propulsion corresponds with high value armament systems which can not be supported by only one European Nation. In the same time, related possible mission correspond with a large and direct involvement of political people. By this fact, the use of high-speed air breathing propulsion for military purposes in Europe does not just need a large, but achievable, technology step. It needs also an important, but maybe feasible, progress in European integration for Foreign Affairs and Defence issues.

RECENT FRENCH BACKGROUND

During last fifteen years, a large effort has been undertaken in France, to improve knowledge on hypersonic air breathing propulsion, acquire a first know-how for components design and develop needed technologies.

MBDA France and ONERA brought major contributions to this effort by participating in different Research and Technology programs: PREPHA, WRR, JAPHAR, PROMETHEE...

Within the scope of the French National Research and Technology Program for Advanced Hypersonic Propulsion - PREPHA, they acquired a first know-how in scramjet and dual-mode ramjet components design (inlet, combustor, injection struts, nozzle) and hypersonic air breathing vehicle system studies (definition and performance evaluation for space launchers (Fig. 4), missiles and experimental flight vehicles) (ref [17], [2]). They also improved their test facilities and numerical means (Ref [18]) and started some activities on flight experiments.

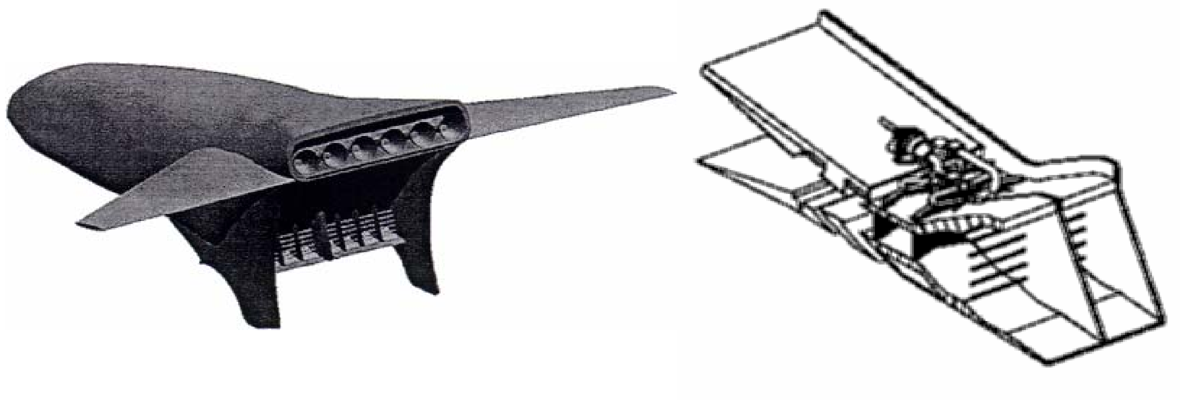

Fig. 4 – PREPHA - Generic SSTO vehicle and its air breathing propulsion system

From 1995 to 2000, and with a partial support of French Administration, MBDA France has been leading a cooperation with the Moscow Aviation Institute (MAI) to develop and test at ground a wide range ramjet dual mode ramjet (Mach 2-12), with fully variable geometry, and using kerosene and hydrogen (ref [19] to [22]). MBDA France has also been leading a cooperation with the Institute of Theoretical and Applied Mechanics (ITAM) from Novosibirsk to develop variable geometry air inlet able to operate in a very large Mach number range (2 to 8 or more) (Ref [23] to [25]- Fig.5).

Fig.5 – Mach 2/Mach 8 variable geometry air inlet tested in ITAM wind tunnel

MBDA France is also working with Astrium Space Transportation to develop fuel-cooled composite structures. First achievement of this cooperation was the development on a C/C hydrogen-cooled injection strut (ref [26]). Partners are now developing an innovative technology for C/SiC endothermic fuel-cooled or hydrogen-cooled structures of a complete combustion chamber (ref [27] to [30]).

In parallel, from 1997 to 2001, ONERA led an in house program in cooperation with its German counterpart DLR. This program, named JAPHAR, aimed at pursuing the basic studies engaged during PREPHA by studying a hydrogen fuelled dual mode ramjet, able to operate in the range Mach 4-8, and by developing a methodology to demonstrate the in flight aero-propulsive balance of an experimental vehicle which would use this engine (ref [31] to [37] – Fig.6).

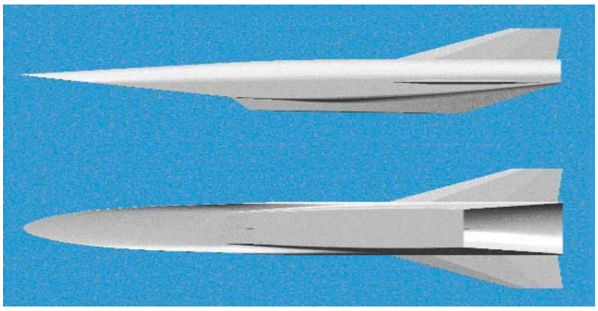

Fig.6 – JAPHAR - generic experimental vehicle

By another way, ONERA was cooperating with SNECMA (ref [38]) to develop a composite materials technology for endothermic fuel-cooled structures in the frame of the A3CP program.

In 1999, French MoD started the PROMETHEE Program. This program, led by MBDA France and ONERA, aims at studying the main difficulties associated to hydrocarbon fuel dual mode ramjet in order to acquire a first know-how in the design and operation of a propulsion system capable of powering a hypersonic cruise missile, and taking directly some operational constraints into account (ref [39] to [41]). To federate these studies, the application to a long range air-to-ground generic missile is considered (Fig. 7).

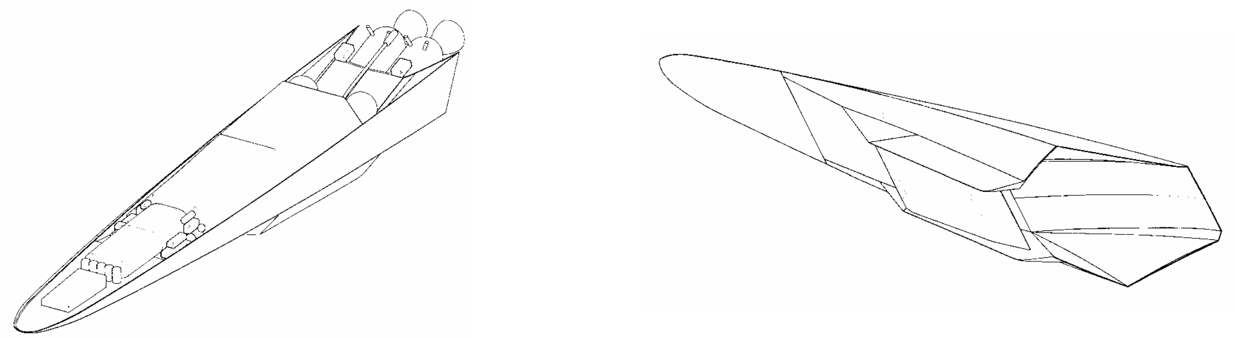

Fig. 7 – Generic Air-to-Ground missile

KEY TECHNOLOGY ISSUES

The feasibility of previously described application mainly depends of two key technology issues:

- capability to predict with a reasonable accuracy and to optimize the aero-propulsive balance (or generalized thrust-minus-drag),
- development of needed technologies for the propulsion system as a low weight, high robustness fuel-cooled structure for the combustor.

Aero-propulsive balance sensitivity

For an air breathing propulsion system, the net thrust (i.e. the thrust which can effectively be used for compensating the drag and accelerating the considered vehicle including the propulsion system) is the difference between the thrust provided by the exit nozzle (momentum of accelerated hot gas coming from the combustion chamber) and the drag due to air capture by the inlet. As a matter of fact, atmospheric air has initially no speed. During capturing process, some energy has first to be paid to accelerate the incoming air in the upstream direction up to 40 to 75 % of the vehicle speed. On the contrary, hot exhaust gas must be ejected through the nozzle in the rear direction at a speed exceeding flight speed (in vehicle reference).

This fact can be illustrated as follows:

- at flight Mach number 2, a net thrust of 1 is obtained by producing a thrust of 2 by the nozzle which compensates an air capture drag of 1,
- at Mach 8, a net thrust of 1 is obtained by a nozzle thrust of 7 while air capture drag is 6,
- at Mach 12, a net thrust of 1 is obtained by a nozzle thrust of 12 while air capture drag is 11.

Then, the higher the flight Mach number, the more sensitive the net thrust. At Mach 8, for example, an error of 5 % on nozzle performance leads to a reduction of 35 % in net thrust. At Mach 12, the same error will result in 60% net thrust reduction.

Then, it is more and more mandatory to optimize the integration of the propulsion system into the vehicle airframe, and vehicle and propulsion system components are operating in a much coupled way which would require testing the overall system to determine the global performance.

But, the higher the flight Mach number, the more difficult to simulate right flight conditions with on-ground test facilities. Generally, in such test facilities, air is heated up to total temperature before being accelerated through a nozzle to enter the test section at the right Mach number. What ever the heating process may be, that generally leads to the creation of radicals, and very often some pollution into the incoming air, which can globally change the combustion process.

This problem is largely increased when heating process is based on pre-combustion (hydrogen, gaseous or liquid hydrocarbon fuel) and oxygen completion. In this case, chemical nature and thermodynamic characteristics of the incoming air are modified, that creates change of ignition delay and modification of thermodynamics into the propulsion flow path.

By another way, for large and very large vehicles, some scaling effects are difficult (or impossible ?) to solve. Then a specific development methodology has to be defined by combining large scale partial tests (possibly corresponding with very large, then expensive test facilities) and numerical simulation in order to be able to ensure design margins for the development of an actual full scale system; the only one validation of this methodology accessible prior to the full scale development being acquired by numerical simulation.

Propulsion system concept

As already mentioned, past studies performed in France demonstrated that combined propulsion could have an interest for space launcher only if the air breathing mode can provide a significant part of the total speed increment.

For a TSTO, a limited part of the total speed increment given by the air breathing mode will not make the launcher unfeasible but will not improve the performance (payload mass/total take-off mass) by comparison with a full rocket system: reduction in needed fuel mass being compensated by the complementary dry mass of the air breathing engine.

For a SSTO, it is clear that the complementary dry mass corresponding with the air breathing mode and its integration into the vehicle will directly reduce the possible mass of payload. Then, the benefit provided by the air breathing mode in term of specific impulse improvement must be sufficient to compensate all these negative terms:

- large Mach number range of operation,
- high installed specific impulse allowing good acceleration level in the whole air breathing mode,
- low dry mass.

Different types of air breathing combined cycles were considered for the system studies performed within the framework of the PREPHA program (Ref [1] and [11]). These studies showed that the use of turbo machinery is not pertinent by comparison with systems based on ramjet. As a matter of fact, one can only take advantage of a turbine based combined cycle in term of specific impulse on a limited Mach number range (maximum up to Mach 6) while it corresponds with a very important increase of dry mass:

- the engine by itself is heavy,
- its combination with a ramjet/scramjet system is very difficult and leads to complex and heavy air inlet consecutively ensuring the supply of a large air mass flow to two different air ducts.

At the contrary, a ramjet/scramjet (dual-mode ramjet DMR) system can be used on a large Mach number range (up to Mach 12) and corresponds with more simple system using a single air duct, avoiding complex transition phase within the air breathing mode and more limited induced dry mass addition.

Moreover, such a ramjet/scramjet system is more capable to integrate some improvements like in-flight oxygen collection or extension to higher Mach number by adding an Oblique Detonation Wave mode or major evolution like MHD by-pass or other heat release principle which could be developed and validated during the development or the in-service life of the vehicle.

A large effort has been led, mainly in USA, on the RBCC (Rocket Based Combined Cycle) concept. Some system studies have been performed in France on that concept (particularly in the framework of the PREPHA program). It has never been confirmed that such integration of the rocket mode into the air breathing duct can improve the global performance. As a matter of fact, in order to obtain a ramjet effect at low Mach number (between 0 and 1.5/2) and then improve the specific impulse, one must reduce rapidly the thrust of the rocket mode (very rich propellants mixture). But, this action dramatically reduces the global thrust and then the vehicle specific impulse (acceleration capability). Then, it appears preferable to use the rocket mode at full power (eventually without any ramjet effect) up to the minimum Mach number for which the air breathing mode is able to provide alone a sufficient thrust to obtain an improved vehicle specific impulse. By another way, if one tries to integrate into the air breathing duct the rocket engines ensuring the final acceleration, that leads to strong integration constraints limiting the achievable performances for the air breathing mode, particularly at high Mach number (supersonic combustion), and generates new difficulties related to the thermal sizing of the air breathing combustion chamber. Finally, such RBCC system make more complex the attitude control during the flight outside the atmosphere, rocket engines thrust being not easily directed to the vehicle centre of gravity.

Need of a variable geometry concept

As a very large flight Mach number range must be considered for the dual-mode ramjet (i.e. 1.5 to 12), a variable geometry is mandatory to provide the best acceleration capability of the air breathing mode.

A fixed geometry combustion chamber associated to a variable capture area air inlet was considered (Fig.8). But, due to the fixed minimum section of the air inlet (equivalent to the fixed section of the combustion chamber entrance), the thrust was limited at low Mach number because of the blockage of incoming air. Moreover, if one tries maintain the air inlet capture area within the bow shock up to the maximum Mach number of the air breathing mode, the air inlet size is limited and consecutively the available thrust reducing the vehicle specific impulse.

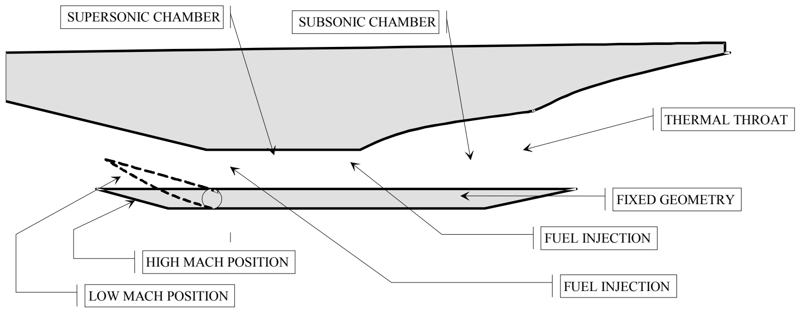

Fig.8 – fixed geometry concept of PREPHA engine

A different concept has been developed with the Moscow Aviation Institute. This concept, called WRR has a fully variable geometry –air inlet + combustion chamber (Fig.9). Then, performances can be increased by comparison with the previous concept as it is shown in Ref [42]. Nevertheless, this concept has the same limitation as the PREPHA concept (i.e. fixed minimum section of the inlet). Then, one cannot take all the benefit of the complexity related to a fully variable geometry system.

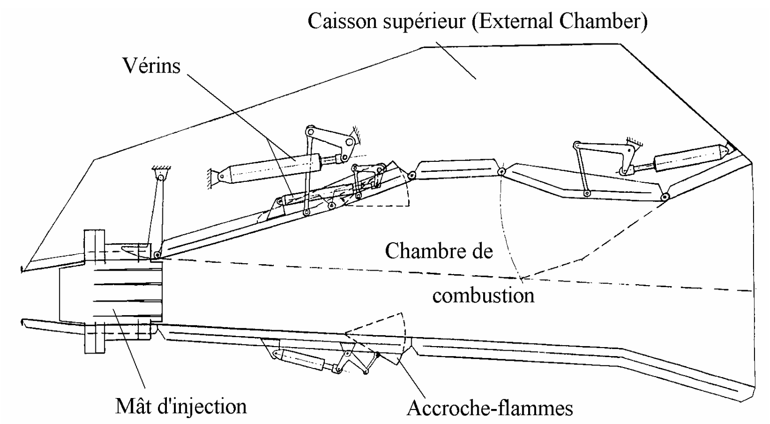

Fig.9 – variable geometry WRR engine

Other concepts have been studied, which consist in modifying in the same time the minimum section of the air inlet and the geometry of the combustion chamber by using a simple movement of the engine cowl. PROMETHEE program is focused of a rotating cowl concept (Fig.10), while PIAF studies, performed with MAI, are focused of a translating cowl (Fig.11).

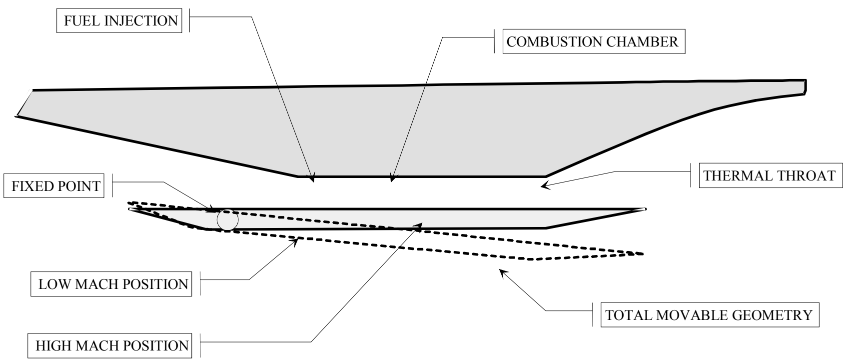

Fig.10 – variable geometry PROMETHEE engine

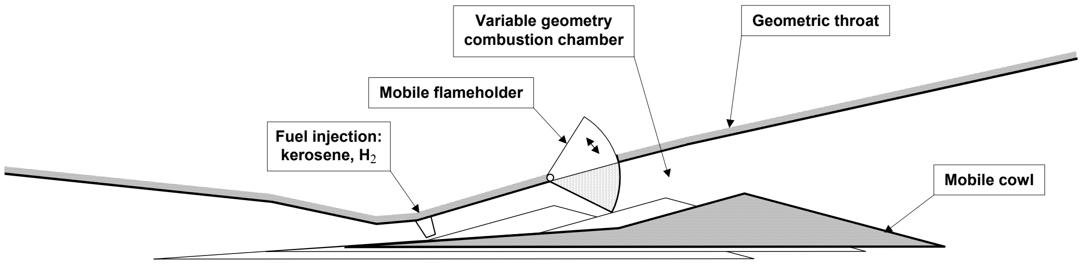

Fig.11 – variable geometry PIAF engine

For such concepts, having at disposal a variable minimum section for the air inlet avoid the need of a large variation of the air inlet capture area (i.e. increase when the Mach number increases). Then the limitation of engine size due to the bow shock is reduced and air breathing engine can be larger at low Mach number providing high thrust level and then better vehicle specific impulse. In these conditions, it is possible to switch to air breathing mode earlier increasing subsequently the overall performance.

Air breathing engine integration

Dual-mode ramjet has obviously the drawbacks of its advantages: a low specific thrust associated with a high specific impulse. The size of the required engine is then quite big, and its weight is about 1000 kg per square meter of air inlet capture area (a benefit of 30 % can be expected by using ceramic composite materials (Ref [43])).

Most of the current launchers projects have quite conventional shapes and the need to integrate a large air breathing propulsion system leads to very low structural efficiency for the flat airframe which is mainly a pressurized fuel tank.

However, other concepts could be studied to try to ensure better trade-off between air breathing propulsion system needs and airframe structural efficiency (Ref [13]).

The first example of these possible vehicles consists in twin axi-symmetric fuel tanks, which are linked by a large 2 D air breathing propulsion system while the rocket engines are placed on the base of cylindrical fuel tanks (Figure 12).

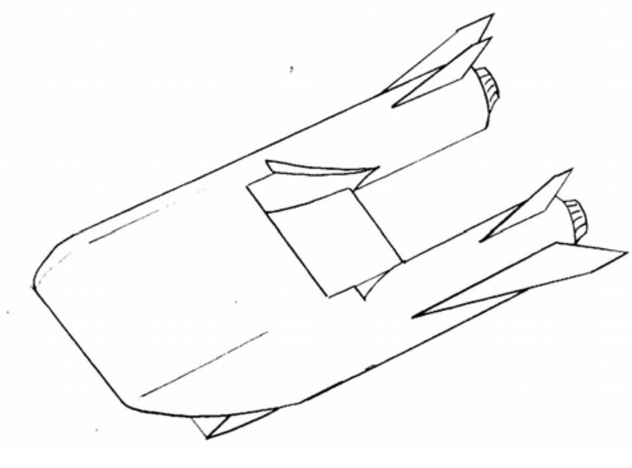

Figure 12 : Twin fuel tank concept

This configuration lends itself a very large, fully variable geometry air breathing system, which has no limitation at low Mach number and then can provide a very high level of thrust. The upstream position of the air breathing mode make possible an inversed SERN nozzle that contributes to the lift at low Mach number rather than increasing the apparent weight. This said, the problem of large base drag created by the two rocket engines still remains and the re-entry phase (air inlet closed of-course) is questionable.

Another concept can be proposed as shown on Figure 13. It is based on a double cone fuselage, which corresponds with a very good structural efficiency. The air breathing engine is semi-annular and takes advantage of a very large air capture section provided by the cone. It can be considered as a series of relatively small modules, which could be more easily tested on ground. The rocket mode can be integrated in the external part of the SERN nozzle. The wing is designed to provide protection of the propulsion system during the re-entry phase (180° vehicle turn before re-entry). Telescopic aerodynamics could provide performing pre-compression while allowing a large nose radius during the re-entry phase.

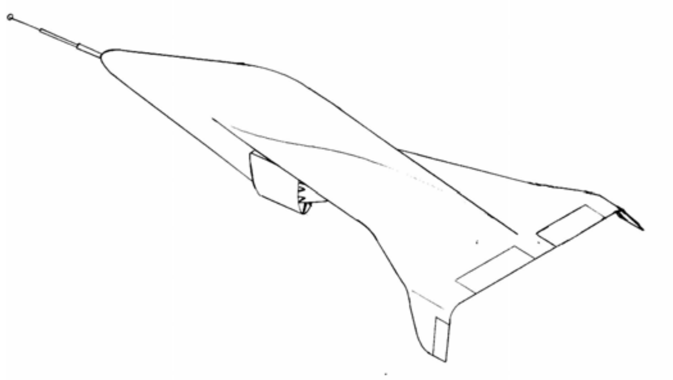

Fig. 13 – double cone airframe concept

A completely axi-symmetric concept can be also proposed as shown on Figure 14. This concept will be a little bit further discussed hereafter.

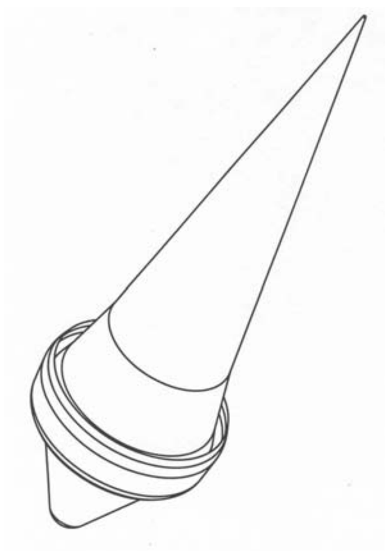

Fig.14 – Fully axi-symmetric concept

Rocket engines are placed in the downstream cone, which constitutes the external part of the SERN nozzle. It is then very easy to control the vehicle including during the flight outside the atmosphere while taking advantage of maximum expansion. The movable cowl of the air breathing engine can be moved upstream to the maximum diameter of the fuselage in order to create a circular wing on the back of the vehicle, allowing to land horizontally after re-entry and deceleration phases (already designed for very large thermal loading). Such a concept leads to a very large engine (2 to 2.5 times larger than that of PREPHA for the same vehicle size). Then the air breathing phase is very efficient and can be performed with a high slope angle, which dramatically reduces the duration of the atmospheric flight (Mach 0 to 12 in 250 seconds instead 1100 seconds for the PREPHA generic vehicle) and then improves the overall efficiency and maybe relaxes the constraints for the sizing of the thermal protection system.

Development methodology

The here above described extreme sensitivity of the aero-propulsive balance on one hand, and the corresponding limited capability of ground test facilities to represent right flight conditions on the other hand make mandatory the definition of a specific on-ground development methodology coupling very closely experimental and numerical approaches.

In such a methodology, the in-flight performance can be predicted only by a nose-to-tail numerical simulation. Then on-ground test facilities will be used to performed partial test of vehicle and propulsion system components separated or coupled one to the other.

These tests have different goals:

- to allow components design tuning and verify a minimum performance,
- to verify, step by step, the ability of numerical simulation to predict accurately performance in conditions as close as possible to the actual flight,
- to acquire a minimum knowledge related to coupled operation of components.

Obviously, such methodology is very challenging. So, before starting any operational development, it must be demonstrated that applying this approach will give an accurate value of the performance, allowing to guarantee design margins and to identify properly right directions for optimizing system design. That is why; a flight experimental program is a mandatory step towards future operational developments.

First approaches for a flight test program

In 1993/1995, a first flight test program was performed with system Kholod developed by the Central Institute of Aviation Motors from Moscow (CIAM). But, this program was based on a hydrogen fuelled axi-symmetrical engine placed on top of a Russian SA6 missile during the whole flight (Fig.15 - Ref[43]).

Fig.15 – Kholod system developed by CIAM

A successful test was performed at Mach 5.7. A new test aiming at flying at Mach 6.3 had a failure. Nevertheless, engine configuration was considered having low interest for future application. Moreover, due to the engine/booster integration it was not possible to establish a clear thrust-minus-drag. Then it was decided to stop this cooperation which has been pursued by CIAM with NASA for a complementary flight.

Beyond this first experiment and in the framework of the PREPHA program, an analysis of flight test needs allowed evaluating the capacity of a large set of typical experimental vehicles to comply with these requirements (Ref [44]). This study resulted in the demonstration of the mandatory need of testing an autonomous vehicle to assess the aero-propulsive balance. It also demonstrated that a small and simplified vehicle would be sufficient for this essential demonstration even if it would not give right answer to some others flight experimental needs.

Assuming obtained results, ONERA and MBDA France sketched a few self-powered experimental vehicles (Ref [45]).

In 1997, ONERA and DLR started the already mentioned JAPHAR program. This program aimed at defining a development methodology and at designing a flight experimental vehicle allowing validating this methodology by flying between Mach 4 and Mach 8. The propulsion system size – height of engine entrance section equal 100 mm – led to a relatively large vehicle (~11 meters long) which corresponds with a cost largely exceeding possibly available budget in France (and Europe up to now).

Finally, MBDA France had a limited participation in the HyShot experiment led by the University of Queensland – Australia and QinetiQ (Ref [46]).

The LEA flight test program

Beyond all current technology development works mentioned here above, and on the base of previous acquired results, MBDA France and ONERA started a flight test program, called LEA, in January 2003 with the support of French Administration (Ref [47] and [48]). In order to limit the cost, this flight test program would be performed with a 4 meters long experimental vehicle having no technology demonstration purpose (use of existing technologies as often as possible) (Fig. 16). In the same view this vehicle would be non- recoverable, then non-reusable. Specifically addressing the aero-propulsive balance, this flight test program is supposed to be performed in cooperation with Russian organizations.

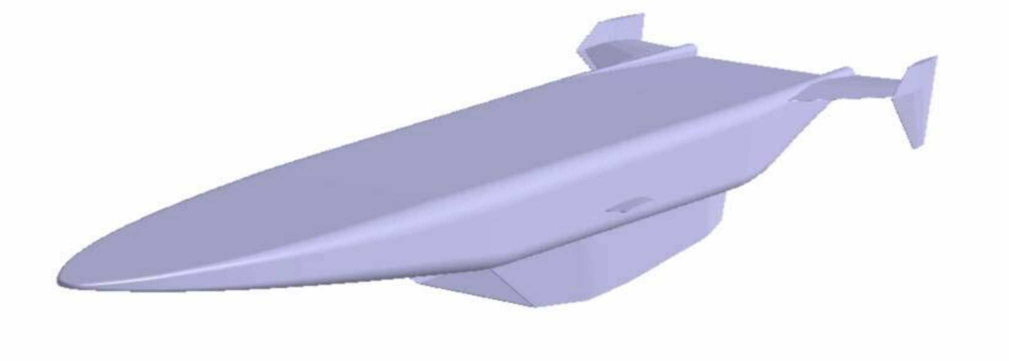

Fig. 16 – CAD view of LEA vehicle

Today, different possible configurations are still considered for the launching and accelerating system and corresponding Russian partners. Nevertheless, an air-launched experimental system would be preferred because it gives some flexibility and reduces range clearance problems.

The test principle consists in accelerating the flight experimental vehicle specimen thanks to an air-launched booster up to the given test Mach number, chosen in the range 4 to 8. Then, after booster separation and stabilization, the experimental vehicle will fly autonomously during 20 to 30 seconds. During this flight, the air breathing propulsion system will be ignited during about 5 seconds with a fuel-to-air equivalence ratio variation (Fig. 17).

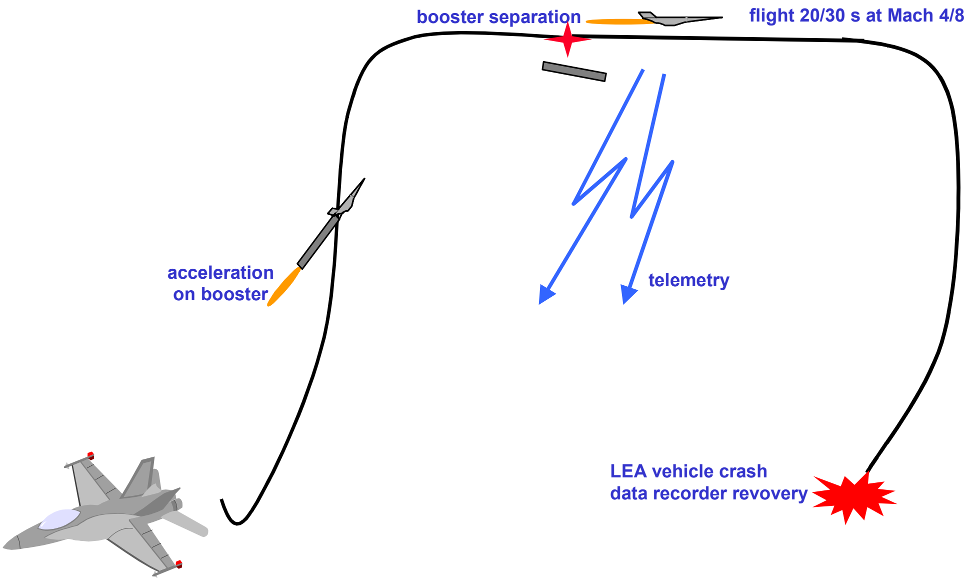

Fig.17 – LEA flight testing sequence

The vehicle would be specifically instrumented to give a precise evaluation of the aero-propulsive balance with and without combustion and to determine the contribution of each propulsion system component to this balance. All measured parameters will be transmitted to ground by telemetry and recorded with an on-board data recorder which will be recovered after the crash of the vehicle.

The program should result in 6 flight tests, planned between 2010 and 2012 for exploring Mach 4 to Mach 8 Mach number range. As explained previously, and beyond a detailed understanding of the aero-propulsive balance, such a flight test program will give the opportunity to define, implement and validate a development methodology applicable to any future operational development.

For this experimental vehicle, the airbreathing propulsion system concept has been chosen by taking into account all results acquired during engines developments performed these last years. The finally selected concept is a variable geometry one using a simple translation movement of the engine cowl, like PIAF engine (Fig. 11) but with a thermal throttling. Nevertheless, as each flight test will be performed at a quite constant Mach number, a fixed geometry engine will be used on board of each LEA test vehicle, this engine configuration being representative of the selected variable geometry concept at the tested flight Mach number.

A parametric study has been performed on the possible technology for the combustion chamber.The finally selected solution is based on metallic heat sink solution with a high temperature low thermal conductivity coating.

The fuel has also been chosen. The most part of French experience in supersonic combustion is related to Hydrogen. But, considering the very low density of Hydrogen, it is preferable to avoid this fuel in order to limit the size of the tank, then the size of the vehicle and consecutive difficulties to find a possible acceleration system complying with the needs (integration constraints, needed total energy release…).

On the other hand, liquid hydrocarbon fuel could be considered. But, our experience is very limited with such a fuel and it would be difficult to ensure a robust ignition and a good combustion efficiency without previous reforming in a regenerative cooling system (simplest technology used on board of the experimental vehicle).

Finally, a mixture of gaseous Methane and gaseous Hydrogen has been selected. By using this mixture, it is possible to increase the fuel density then limit the fuel tank size. It will be also possible to vary the H2/CH4 ratio during the flight to ensure a robust ignition and control the heat release along the combustor.

Some specific works have been performed to adapt used computation codes to this particular fuel. These codes have been validated thanks to basic experiments led in updated ONERA LAERTE test facility. Moreover, ONERA ATD 5 test facility has been updated to allow future CH4/H2 tests for the LEA engine. By waiting, a first test series has been performed with already existing JAPHAR combustion chamber to acquire a first experience with such a fuel.

The forebody has been specifically studied. Some parametric studies have been carried out in order to determine a set of design parameters allowing a satisfactory pre-compression while complying with technology constraints.

On the base of an air inlet design and corresponding performances, a first design of the combustion chamber has been realized thanks to 1D, then 2 and 3D computation. On this basis, a full scale mock-up is under manufacturing for future test in ATD 5 ONERA test facility.

Due to the particular configuration of the afterbody/nozzle, a specific effort is still under progress to well understand the interaction between the propulsive jet and the external flow to accurately determine the effect of propulsion on external aerodynamic (Fig.18).

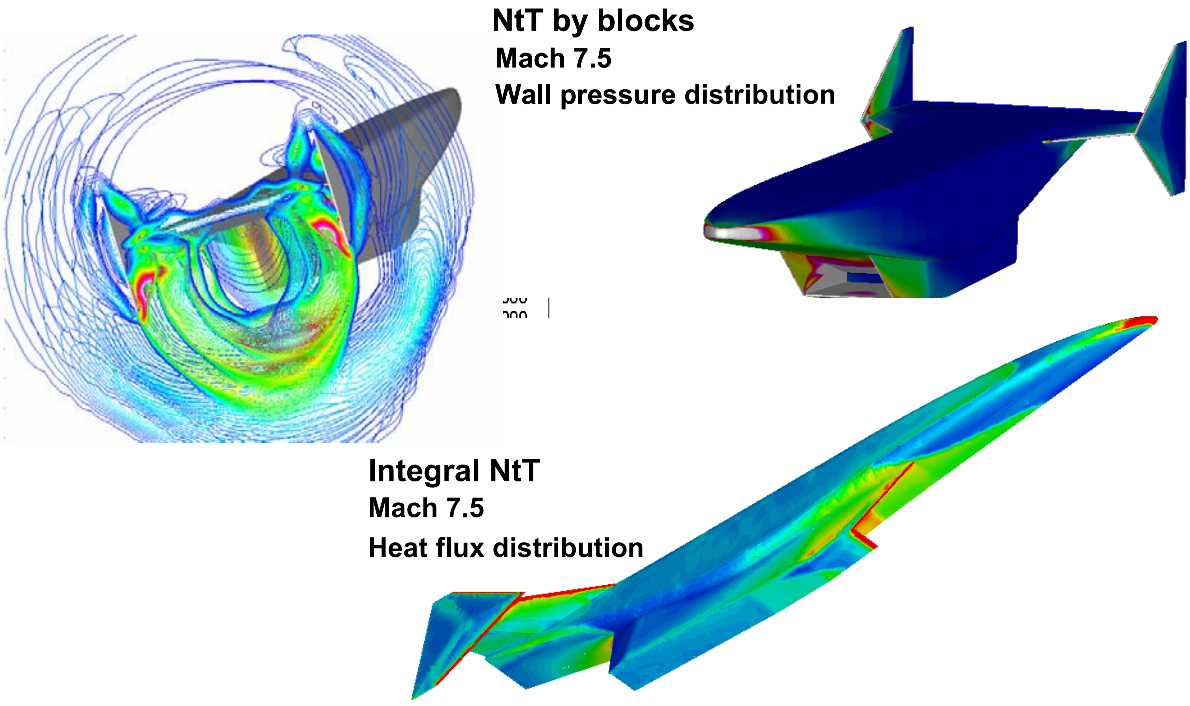

Fig.18 – Some example of Nose-to-Tail computation results

Aerodynamic behaviours of the LEA vehicle and of the Flight Experimental Composite constituted by LEA and its booster have been evaluated by computation for preparing future aerodynamic tests.

Finally, a large effort has been dedicated to the development of Nose-to-Tail computation tools. Thanks to this, two approaches – NtT computation by blocks or integral NtT computation – are available and daily used to evaluate and optimize the aero-propulsive balance of the vehicle (Fig.18).

All the previous elements have been used in a detailed flight simulation in order to obtain a first evaluation of reachable maximum LEA/booster separation conditions. This flight simulation allows simulating a complete flight test sequence including LEA/booster dropping from air carrier, acceleration on booster, separation, descent trajectory of booster, LEA autonomous flight up to final crash.

Other activities have also been carried out to chose the basic technologies used for the LEA vehicle and its propulsion system and a preliminary design has been performed and validated by a Preliminary Design Review (Fig.19).

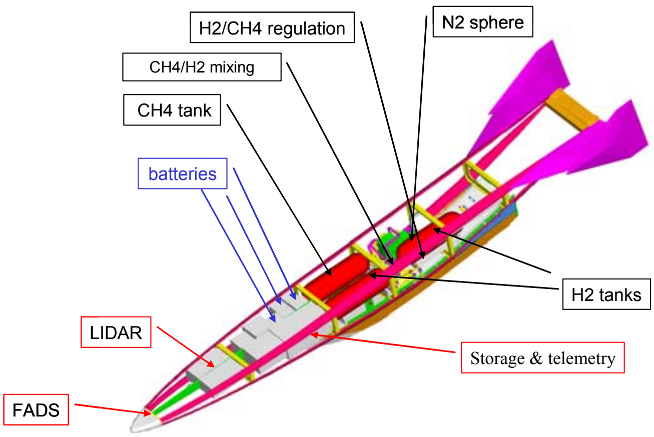

Fig.19 – LEA vehicle Internal layout

By another way, a general approach for on-ground testing has been defined but it still remain to be refined and confirmed. Indeed, as Fig.20 shows and on the base of previous studies [49], a large part of the on-ground testing program should be realized in the S4Ma wind tunnel located in ONERA Modane test Center in the French Alps.

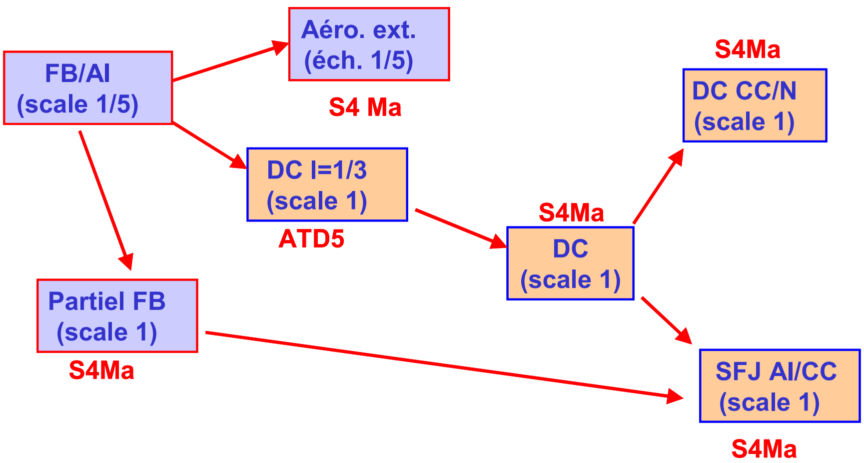

Fig. 20 – general approach for on-ground testing

It is intended to upgrade this test facility in order to take advantage of the existing alumina pebble bed heater which allows to perform test with air non vitiated by water vapour up to Mach 6.5 conditions (1800 K). Thanks to a complementary pre-burner or to an updating of the pebble bed heater, tests corresponding to Mach 7.5/8 flight conditions should be also easily feasible. Detailed design studies, as for example free jet test configuration (Fig.21), have been performed to verify the feasibility of such an upgrading and evaluate precisely the corresponding cost.

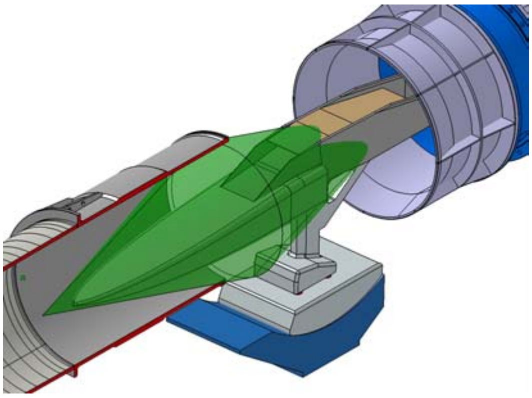

Fig. 21 – Study of LEA free-jet test installation in S4Ma ONERA test facility

Combustion chamber technology

The combustion chamber technology covers different aspects which contribute to ensure the thermal and mechanical strength:

- variable geometry needed to optimize the performance on the overall flight Mach number range,
- fuel used as coolant for combustion chamber structure,
- fuel-cooled structure itself.

Variable geometry

Some developments have already been performed in the scope of the WRR program led by MBDA France in cooperation with MAI. Corresponding technologies (cooled hinges, high temperature sealing system between movable and fixed walls…Fig.22) have been tested separately, then reused in the PIAF combustion chamber which have been successfully tested at MAI in the Mach number range from 2 to 7.5.

Fig.22 – WRR Prototype with removed lateral wall

Endothermic fuel

The cooling capacity of the hydrocarbon fuels is less than hydrogen one, and cannot easily ensure the cooling of the combustion chamber of the dual-mode ramjet, which is absolutely mandatory at high Mach numbers first to ensure thermal resistance of the combustion chamber and second also to improve mixing and combustion process (gaseous fuel with radicals promoting ignition – see Fig.23) and maximize the net thrust (by re-injecting heat lost along combustor walls into the propulsive thermodynamic cycle).

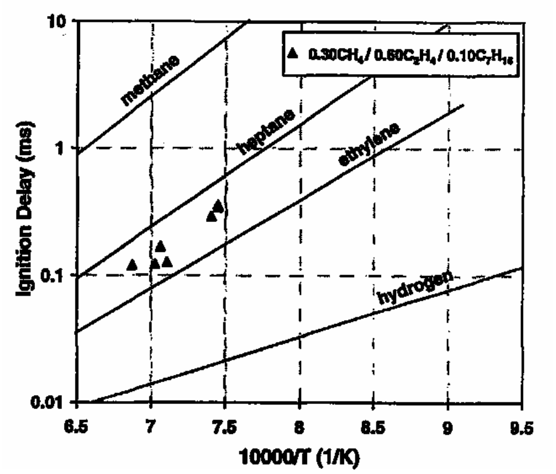

Fig.23 – example of ignition delay for a hydrocarbon mixture representing fuel reforming products

In order to obtain sufficient cooling capability, one way is to use the endothermic effect which is produced by thermal reforming of the liquid fuel into lighter components. Figure 24 shows how the decomposition of the fuel can significantly increase the thermal absorption capacity.

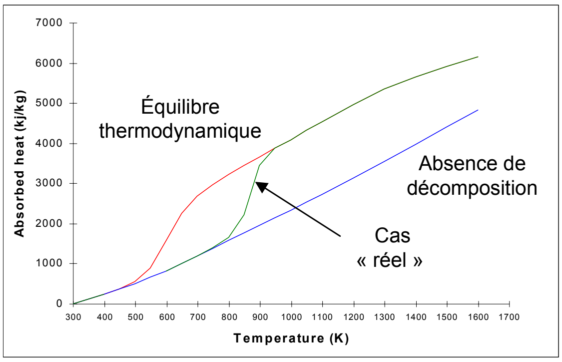

Fig.24 - Endothermic effect

In order to analyze and get to a better understanding of the different thermo chemical mechanisms associated to fuel reforming process inside the cooling system, a pilot system has been developed (Fig.25). The pilot system is organized around a tube which is placed inside a thick copper bloc. The block is heated by electrical resistances progressively, and it then operates as a heat capacity when the fuel flow is started into the tube. This system aims at studying, for different mass flow rates and heating conditions, the evolution of the composition of the products at the exhaust of the tube.

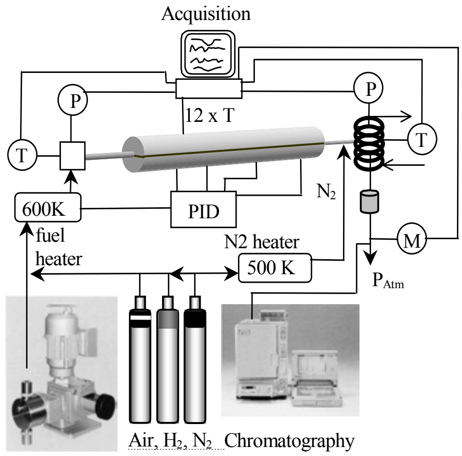

Fig. 25 - Pilot system in operation at ONERA

Complex reactions are located in the cooling circuit elements. The design of a combustion chamber cooled by this method requires perfect mastering of the following points:

- thermal loads evaluation, along the whole flight trajectory
- structural design of an integrated cooled chamber
- fuel thermal reforming process
- fuel injection system capable of managing the different situations encountered during the flight (variable mass flow rate, thermodynamic state of the products, different location of the injection points...)

Some works are led in cooperation with CNRS to be able to simulate the complete operation of the cooling circuit (Ref [50] and [51] - Fig.26) taking into account in the same computation the reforming process (Fig.27)

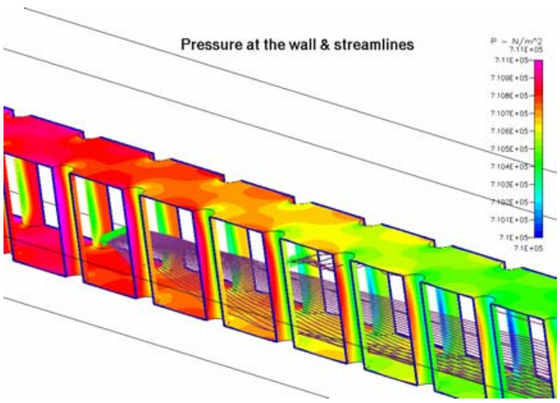

Fig.26 – Example of computation in Ptah-Socar cooling circuit

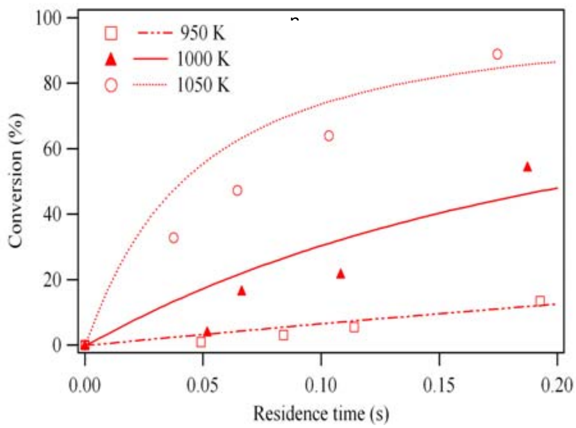

Fig.27 – Endothermic fuel conversion as function of residence time

At the beginning of the flight, the wall of the combustion chamber is not necessarily hot enough to reach the conditions where decomposition of the fuel is obtained. The tuning of the engine has to be made, especially for a certain part of the flight trajectory, using at least partially liquid fuel. On the contrary, when the missile is flying at very high speed, the fuel can reach a very high level of decomposition. Its chemical composition becomes complex, and the combustion parameters of this hot multi-component fuel is no more like the combustion of the original cold liquid fuel.

However, it is not easy to tune the engine using a technological chamber (integrating cooling system), and it also very difficult to produce appropriate mass flow of appropriate composition with external devices. Then the problem becomes to ensure sufficient similitude for the combustion using more convenient and available fuels.

Combustion chamber fuel-cooled structure

From the point of view of materials, the major technological difficulty for the development of hypersonic air breathing vehicles, powered by dual-mode ramjet or scramjet, is to design and realize the structure of the combustion chamber. As a matter of fact, the combustion chamber and the fuel injection and/or flame stabilization systems, possibly placed in the flow, must be able to sustain a rarely so severe thermo-mechanical environment.

Moreover, in the case of variable geometry combustion chamber, it is necessary to ensure a sufficiently controlled geometry to ensure tightness between movable and fixed surfaces of the chamber.

These elements lead to consider combustion chamber technologies based on the use of thermo-structural composite materials cooled by the fuel. By comparison with metallic solutions, such a technology should give large design margin and should correspond with relatively low cost systems while ensuring good reliability and safety characteristics.

In this field, very limited works have been performed by EADS Space Transportation (now Astrium ST) and SNECMA during the PREPHA program and led to basic test, performed at ONERA (Ref [2]). In parallel, MBDA France and Astrium ST led in house development of hydrogen or hydrocarbon fuel-cooled injection strut (St ELME) (Ref [52] – Fig.28).

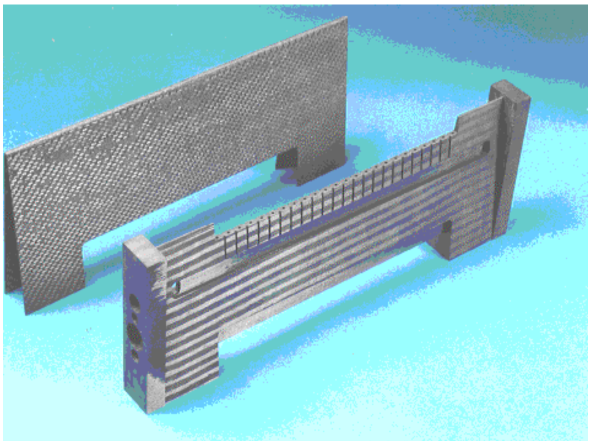

Fig.28 – H2 cooled C/C injection strut

By another way, the cooperation led by MBDA France and MAI for developing a fully variable geometry dual-mode ramjet was also an opportunity to acquire a large know-how in the field of fuel-cooled structures. More than 30 concepts of cooled panels have been developed for protecting the fixed and movable combustion chamber walls.

Most of these studied cooled panels were based on metallic structures. Then, in order to maintain the temperature of the hot wall under the relatively limited capacity of the steel alloys it is necessary to use:

- 3D configurations, in particular multi-layer architectures,
- heat exchange enhancement systems in the cooling channels.

Each of cooled panel has been tested in MAI test facility, in which hydrogen fueled scramjet combustion chamber is used as high temperature gas generator. The tested cooled-panel (100x200 mm²) is placed at the scramjet chamber exit (Fig.29). A wedge is facing the tested panel in order to create a shock wave whose interaction with the tested panel increases the heat flux, which can reach 3 MW/m².

Fig. 29 – Cooled panel test facility at MAI

Today, MBDA France and Astrium ST are focusing their in house effort on the development of a low cost, highly reliable and effectiveness technology for the fuel-cooled composite material structure, particularly usable for the walls of ramjet/scramjet combustion chamber. This technology, called PTAH-SOCAR, takes advantage of the Astrium ST know-how in the field of pre-form manufacturing and particularly of its mastery for weaving the fibrous structure (Ref [53] to [56]).

By comparison with more classical solution, this know-how enables us to create a C/SiC structure, which has the following advantages:

- It avoids any bonding system (brazing, gluing…), which constitutes a weakness for classical systems for sustaining the high pressure of the internal cooling flow and implies limitation of maximum temperature in the bonding region.
- It enables us to obtain a complete combustion chamber structure in one part. That led to :
  - limit the connecting problems to one inlet and one outlet connection,
  - avoid any problem generally encountered with classical plane cooled panels for realization of the corners of a 2D combustor,
  - increase the reliability by limiting the possible leakage problems.
- It considerably reduces the part of the cooled-structure wall, exposed to the hot heat flux, which is not directly in contact with the coolant fuel.
- It avoids to machine internal channels and, by this way, makes easier the integration of specific systems like injectors, injection struts or flame-holders.

Moreover, this technology can be easily adapted to other applications as injection struts or cooled nozzle expansion ramp.

A first test series has been performed with PTAH-SOCAR cooled-panels densified by CVI process (Fig.30). Tests were performed with nitrogen and kerosene as coolant. By reducing step by step the coolant mass flow it has been possible to obtain the maximum temperature on the hot wall which would be reached in an operational application (Fig.31).

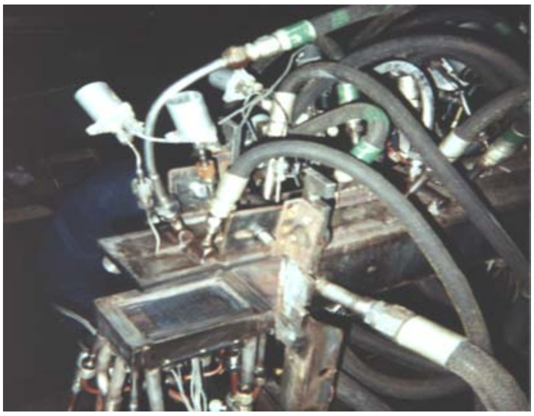

Fig.30 – PTAH-SOCAR panel in test at MAI

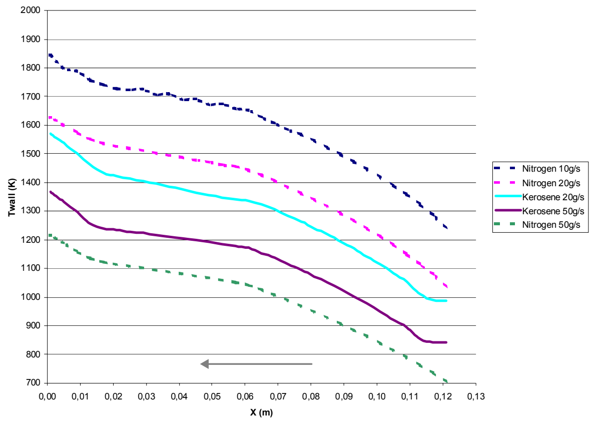

Fig. 31 – Hot wall temperature evolution

A key point for using composite materials for fuel-cooled panels is their natural porosity which could generate safety hazard and reduce the performance (with natural porosity, the total injected fuel would completely leak through combustion chamber walls and no fuel would reach the injection system). It is then very important to master the permeability of the material. However, a preliminary study indicates that a small leakage to the combustion chamber side could be interesting from the point of view of performance. As a matter of fact (Fig. 32):

- If the porosity is too large, an important part of the fuel mass-flow will leak out through the combustion chamber wall. Then, specific impulse will decrease (no combustion of the corresponding mass of fuel) or it will create thermal overloading on the wall (in case of partial or total combustion of the fuel damaging the wall).
- If the porosity is limited, the fuel leaking out through the wall will reduce skin friction losses while the main part of fuel will burn normally resulting in an improved specific impulse by comparison with a perfectly fuel-tight material.

In order to solve this problem, but also to reduce the production cost, the PTAH-SOCAR has been adapted to liquid infiltration by using the LSI process developed by Astrium ST Germany). This new version has already been tested at component level (mechanic and thermal characterisation and cooled panel). After several iterations, this technology present today the permeability needed to ensure an optimum fuel leakage in the case of a propulsion system for a missile.

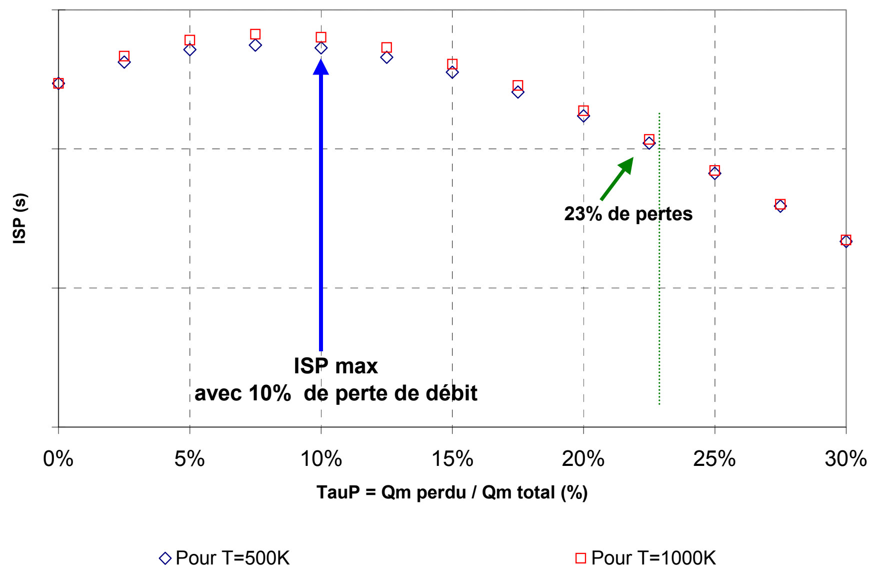

Fig.32 – Fuel leakage effect on specific impulse

On the basis of these technology development works, a demonstration part constituting a portion of cooled combustion chamber, called PSD, has been designed, manufactured and successfully tested in 2006 at ONERA test bench ATD 5 (Fig. 33 - Ref [57]).

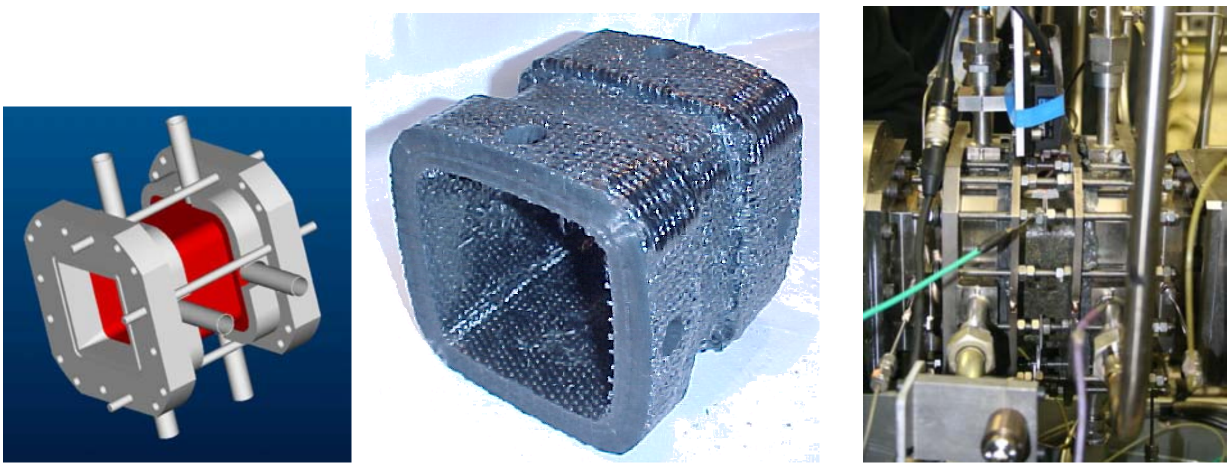

Fig.33 – PTAH-SOCAR PSD demo tested at ONERA ATD 5 test bench

Further works

Beyond the works already in progress, the test facility, developed by MBDA France and ROXEL in their Bourges Subdray test centre in the framework of PREPHA program (Ref [49] – Fig.34), is under upgrading. The new test facility, called METHYLE, will allow to perform long endurance test in representative conditions to pursue and reinforce technology development by using a modular water-cooled dual mode ramjet combustion chamber able to integrate different kind of testing parts as for : element of variable geometry, sealing system, fuel-cooled structure, measurement techniques, engine control system…) (Fig.35).

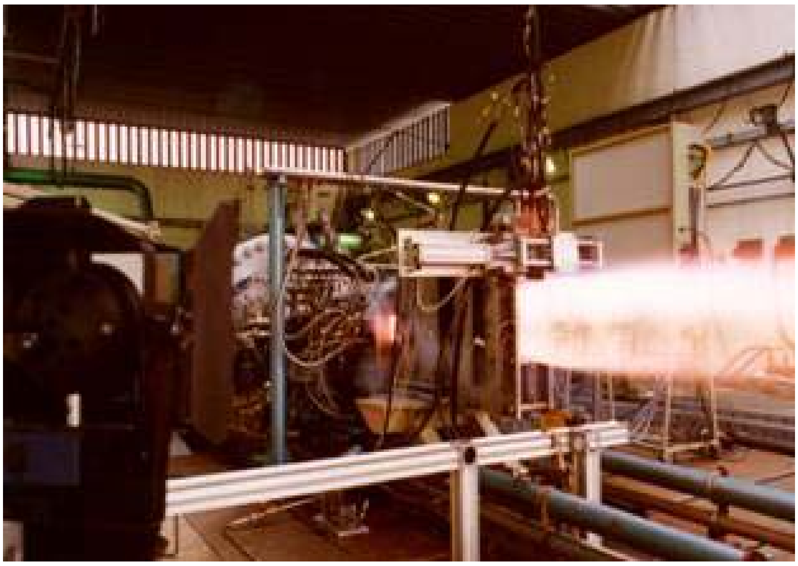

Fig.34 – Hypersonic test bench in Bourges Subdray test centre

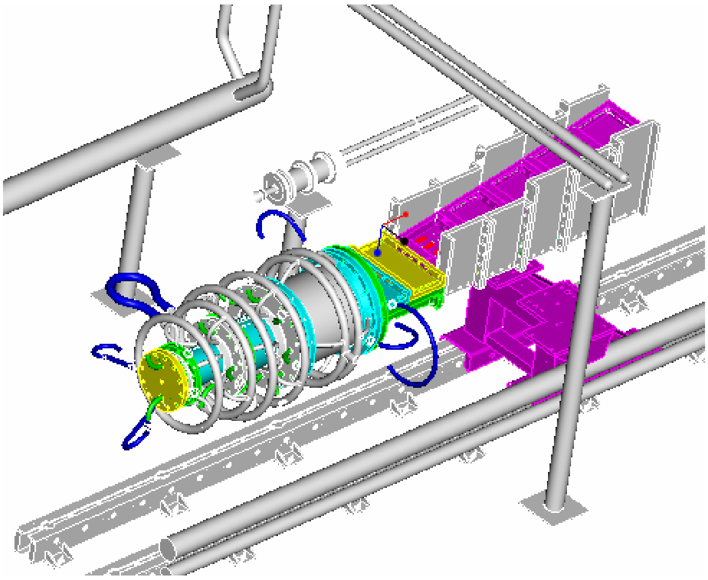

Fig.35 – METHYLE test facility (under development)

FURTHER POSSIBLE DEVELOPMENT

The LEA flight test program constitutes a very important first step in the definition and the validation of a development methodology for hypersonic air breathing vehicles. Nevertheless, if we consider the possible application of high-speed air breathing propulsion to future reusable space launcher, it is clear that the air breathing phase will have to be extended up to Mach 10/12.

In that view, a minimum R&T program has been proposed (Ref [58]). It includes an extension of the flight domain of the LEA vehicle (LEA +) thanks to the upgrading of the present acceleration system or by selecting another one with higher capabilities. At least, taking into account the corresponding background and associated working partnership, it should be possible to define the most efficient flight test program (in term of scientific and technological return to financial investment).

It has to be noticed that such extended flight experimental program could take advantage of other already existing experimental systems and programs as, for example, the HyShot program which could be used to perform partial technology flight validation for LEA vehicle and propulsion system or for instrumentation.

But, the budget which could be potentially available in Europe within the next years for such a flight test program will be limited. By another way, the on-going LEA flight test program between Mach 4 and Mach 8 has to be first performed. That is why, considering these two points, a proposal has been submitted to ESA regarding a preliminary and less ambitious flight test program, called EAST for European Advanced Scramjet Test, which could be performed by 2010.

The EAST program would consider a subscale (~1/4) twin engines configuration derived from LEA vehicle (Fig.36). EAST would not be a simple supersonic combustion experiment within an academic combustor but would consist in testing the system forebody / air inlet / combustion chamber / partial nozzle during a captive flight on top of a booster derived from a sounding rocket system (Fig.37). The EAST experiment would be fixed on the booster thanks to a strut equipped with a thrust measuring system.

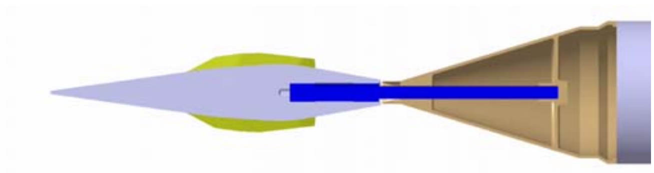

Fig. 36 – EAST configuration

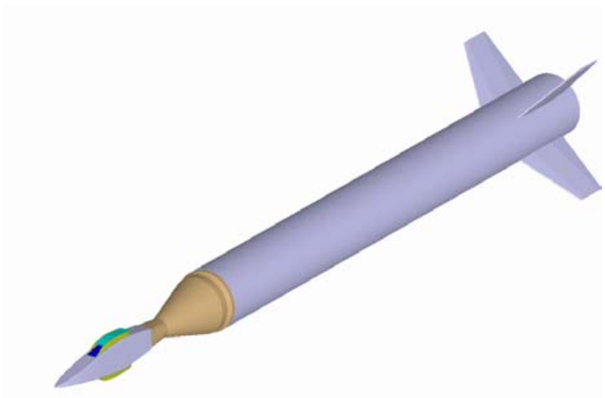

Fig. 37 – EAST on top of the sounding rocket

Such a program, dealing with an integrated propulsion system, would allow extending the already defined development methodology by taking into account new ground test possibilities as, for example, high enthalpy short time wind tunnels F4 at ONERA Fauga or HEG at DLR Göttingen and to acquire a first flight validation. By another way it would be possible to take advantage of the quite complete propulsion system configuration to flight test the needed improvements of LEA technology to sustain higher flight Mach number conditions.

Beyond these technology development efforts, the need was also clearly identified to restart system studies taking advantage of recent progress made regarding knowledge, tools and technology and focusing on more innovative airframe/propulsion system concepts enabling better trade-off between structural efficiency and propulsion system performance.

In that field, a fully axi-symmetric configuration has been considered for a reusable micro-space launcher (10 kg payload). The vehicle, based on a double cone fuselage providing good structural efficiency, is constituted by a main stage powered by air breathing propulsion, combined or not with liquid rocket mode. A “kick stage”, powered by a low performance solid rocket engine provides the final acceleration (NEO concept) (Fig.38).

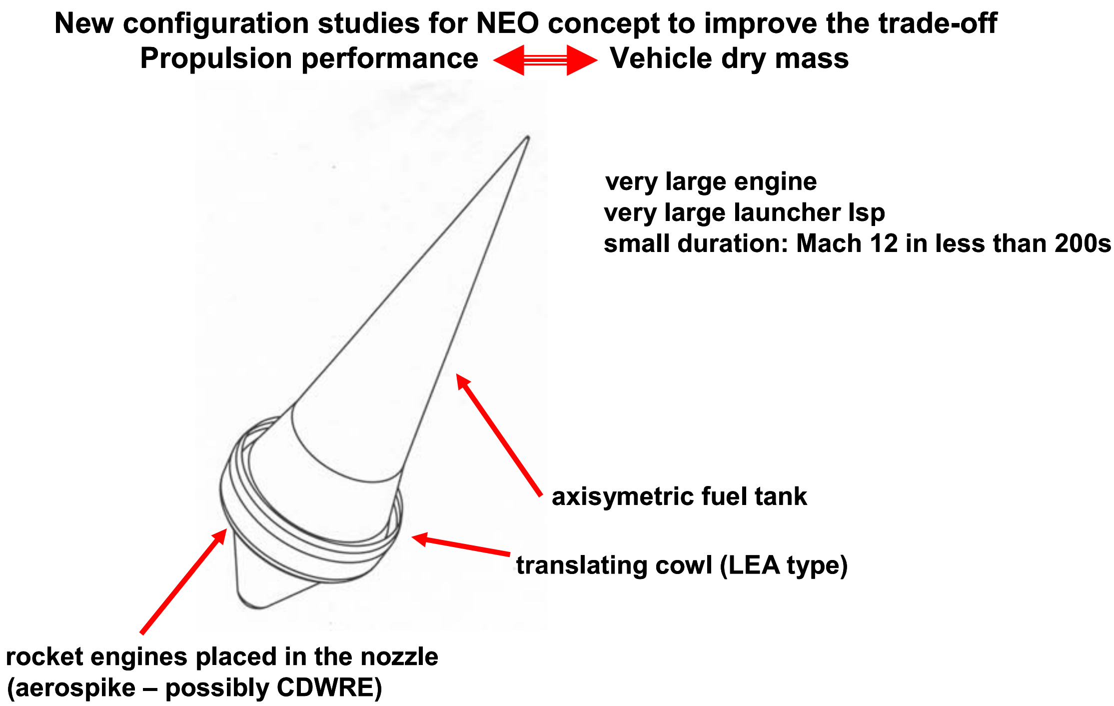

Fig.38 – Axi-symmetric NEO concept

The air breathing engine is annular and can also be constituted by a series of modules, each easily testable on ground (in any case corresponding to a limited scaling effect by comparison with possible ground testing). This engine has fully variable geometry one using a concept derived from the LEA engine. Taking advantage of the translating movement of the cowl, the rear part of the engine can be closed to ensure a safe re-entry on the back using the aerospike nozzle as blunt nose providing the needed thermal protection.

A preliminary design has been performed for different variants : one using a separated booster and a purely air breathing main stage, a second one using a booster and a main stage combining air breathing and rocket mode, a third one without separated booster, the main stage ensuring the initial acceleration in liquid rocket mode and a complementary acceleration phase in rocket mode beyond the air breathing propulsion system operation (Fig.39).

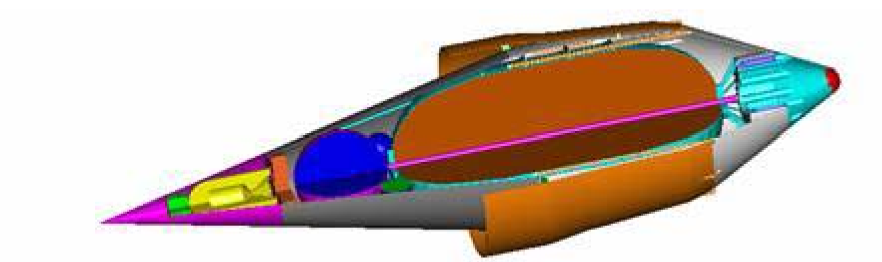

Fig. 39 – Axi-symmetric NEO micro-space launcher – internal layout

On this basis, performance assessment has been carried-out thanks to trajectory simulation. The main conclusions are the following:

- The very large air breathing engine delivers a very large thrust level.
- This thrust level provides a very high acceleration capability and then a very good launcher specific impulse.
- It also allows starting the air breathing mode very early (at about Mach 1.5) reducing the needed on-board oxidizer mass then the corresponding tank mass.
- The acceleration capability allows to reach Mach 12 in less than 200s (to be compared with 1000 to 1200 s needed with an airplane-like vehicle (PREPHA vehicle for example).
- Then the air breathing mode, and consecutive trajectory in the atmosphere, doesn't lead to thermal protection oversizing.
- The large air breathing engine corresponds with a limited dry mass increase as it is mainly constituted by a short annular cowl.
- All the previous elements lead to the feasibility of a reusable main stage with a staging at Mach 14.5 corresponding to a 8.5 m long, 3.0 m diameter vehicle with a gross take-off weight of about 4.5 metric tons, able to place in a 250km circular orbit a payload of 10 kg + 30 kg of avionics.
- Finally, there will be a clear advantage in replacing the liquid rocket engine of this third variant by a continuous detonation wave rocket engine for which integration will largely easier and more efficient (including a very simple way to control the thrust vector and to close the engine for the re-entry phase).

CONCLUSION

The ramjet/scramjet concept constitutes the main air breathing propulsion system which can be used in a very large flight Mach number range up to Mach 10/12 and then could allow developing future fully reusable space launcher and military systems.

Beside international activities, mainly in USA and Japan, a permanent Research and Technology effort has been pursued in Europe since twenty years. Today, the effort led in France aims at addressing the two key technology issues which are the accurate prediction of the aero-propulsive balance of an air breathing vehicle flying at high Mach number and the development of high-temperature structures for the combustion chamber able to withstand the very severe environment generated by the heat release process while ensuring reliability and limited mass and should allow to conclude on the feasibility and interest of the two possible application within the next five to six years (2012/2013).

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