PROPULSION SYSTEMS FOR HYPERSONIC FLIGHT

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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 airbreathing 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 airbreathing propulsion : reusable space launcher and military systems.

By combining the high-speed airbreathing 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.

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 airbreathing propulsion depends of two key points : development of needed technologies for the fuel-cooled structure of the propulsion system, capability to predict with a reasonnable accuracy and to optimise 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 (thermics, mechanics …).

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

R&T efforts led today in France should allow to better estimate advantages provided by high-speed airbreathing proulsion and to build a coherent development capability including methodology, facilities and adapted technologies.

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Introduction

The ramjet/scramjet concept constitutes the main airbreathing 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 airbreathing 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 airbreathing 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 airbreathing 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 turbomachinery, 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 ddition to an already hot airstream 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.

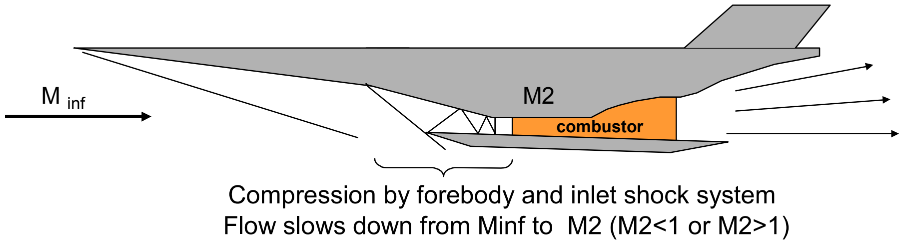

Figure1 : scheme of ramjet/scramjet (DMR) system

To overcome these 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 unburnt to the nozzle. If so, we obtain a supersonic combustion regime, and the engine is called a supersonic combustion ramjet, or scramjet.

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.

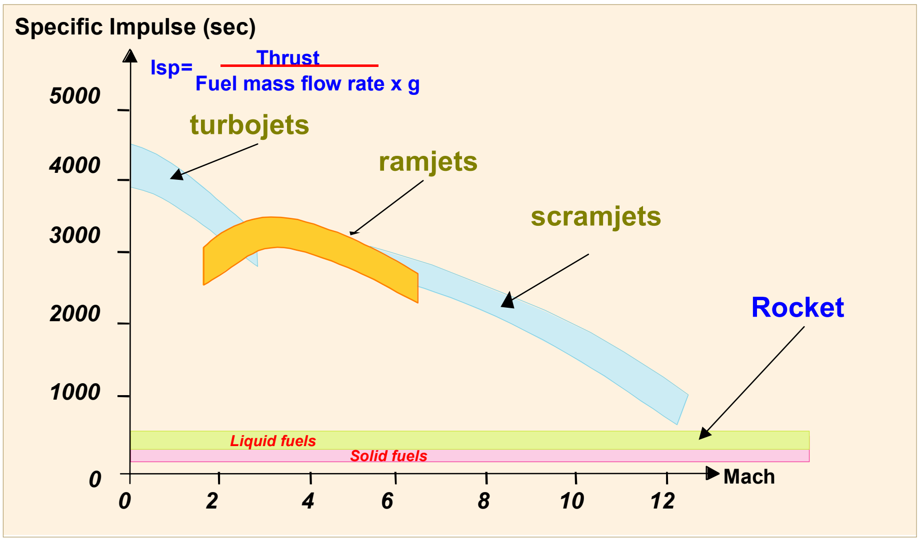

Figure 2 : achievable performance with ramjet/scramjet

Possible application

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

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

Space launcher application

By combining the high-speed airbreathing 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 airbreathing 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 worlwide effort is still under progress for developing the high-speed airbreathing 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 airbreathing mode is 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 airbreathing mode and 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 complexe propulsion system would correspond with an unacceptable risk level.

TSTO

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

Remark : it would be also possible to place the airbreathing 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 in orbit, atmospheric reentry of an airbreathing 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 Fance were considering different kind of combined propulsion systems, for the first stage, with an airbreathing 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 airbreathing 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 ease the development of the second stage. But, it correspond with a very complex first stage vehicle.

For all the previously studies, the staging was very close to the end of the airbreathing 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 this end of the airbreathing 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 airbreathing 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. 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.

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 reentry. 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.

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 airbreathing 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 forseen 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 repearing of dammaged 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 airbreathing 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 responsibles. By this fact, the use of high-speed airbreathing 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 Defense issues.

Recent French background

During last fifteen years, a large effort has been undertaken in France, to improve knowledge on hypersonic airbreathing 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 airbreathing vehicle system studies (definition and performance evaluation for space launchers (Fig. 3), 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.

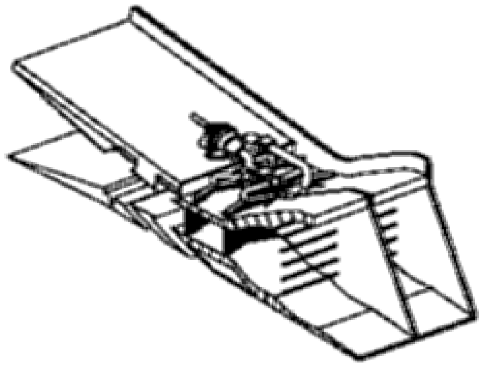

Fig. 3 – PREPHA - Generic SSTO vehicle and its airbreathing propulsion system

Since 1995, and with a partial support of French Administration, MBDA France is 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 is also 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.4).

Fig.4 – Mach 2/Mach 8 variable geometry air inlettested in ITAM windtunnel

MBDA France is also working with EADS Space Transportation to develop fuel-cooled composite structures. First achievment 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 fueled dual mode ramjet, able to operate in the range Mach 4-8, and by developing a methodology to demonstrate the in flight aeropropulsive balance of an experimental vehicle which would use this engine (ref [31] to [37] – Fig.5).

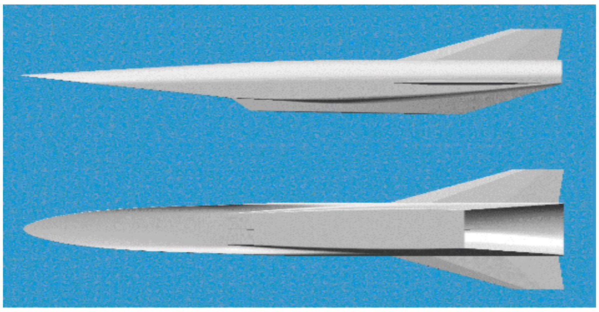

Fig.5 – JAPHAR - generic experimental vehicle

By another way, ONERA is 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. 6).

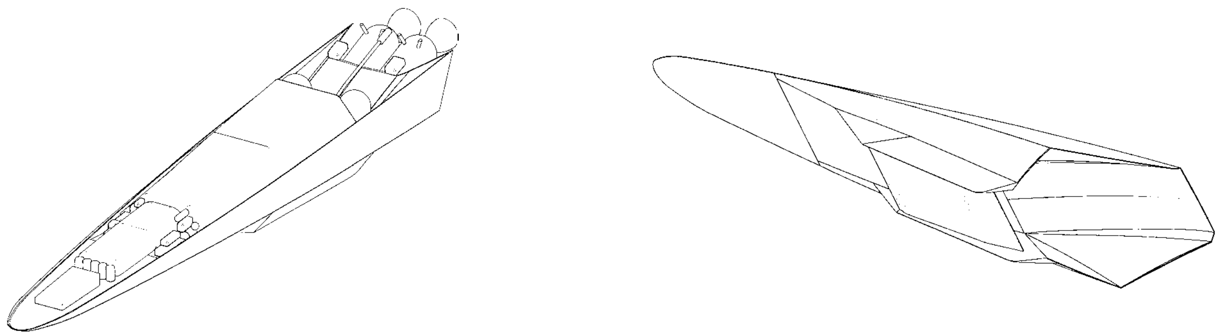

Fig. 6 – Generic Air-to-Ground missile

Key technology issues

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

- capability to predict with a reasonnable accuracy and to optimise 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 airbreathing 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% thrust reduction.

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

But, the higher is the flight Mach number, the more it becomes 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 air which can change 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 flowpath.

By another way, some scaling effect are difficult (or impossible) to solve with similarity rules. Then, the overall system should be tested at full scale that implies very large, and extremely expensive if feasible, test facilities.

Propuslion system concept

As already mentioned, past studies performed in France demonstrated that combined propulsion could have an interest for space launcher only if the airbreathing 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 airbreathing 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 airbreathing engine.

For a SSTO, it is clear that the complementary dry mass corresponding with the airbreathing mode and its integration into the vehicle will directly reduce the possible mass of payload. Then, the benefit provided by the airbreathing 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 airbreathing mode,
- low dry mass.

Different types of airbreathing 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 turbomachinery 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 airbreathing 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 Oblic

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 airbreathing 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 airbreathing 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 airbreathing duct the rocket engines ensuring the final acceleration, that leads to strong integration constraints limiting the achievable performances for the airbreathing mode, particularly at high Mach number (supersonic combustion), and generates new difficulties related to the thermal sizing of the airbreathing 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 center 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 airbreathing mode.

A fixed geometry combustion chamber associated to a variable capture area air inlet was considered (Fig.7). 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 airbreathing mode, the air inlet size is limited and consecutively the available thrust reducing the vehicle specific impulse.

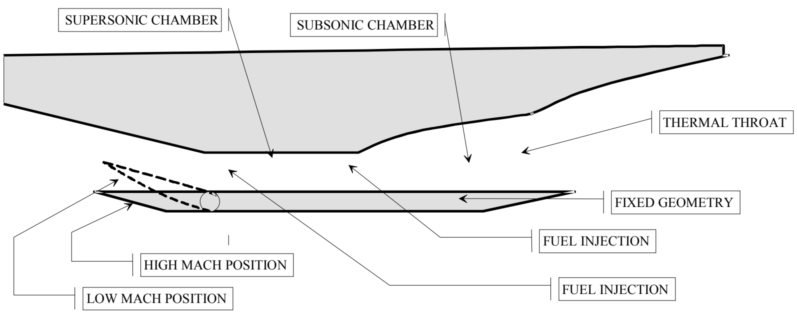

Fig.7 – fixed geometry concept of PREPHA engine

An other concept has been developed with the Moscow Aviation Institute. This concept, called WRR has a fully variable geometry –air inlet + combustion chamber (Fig.8). 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.

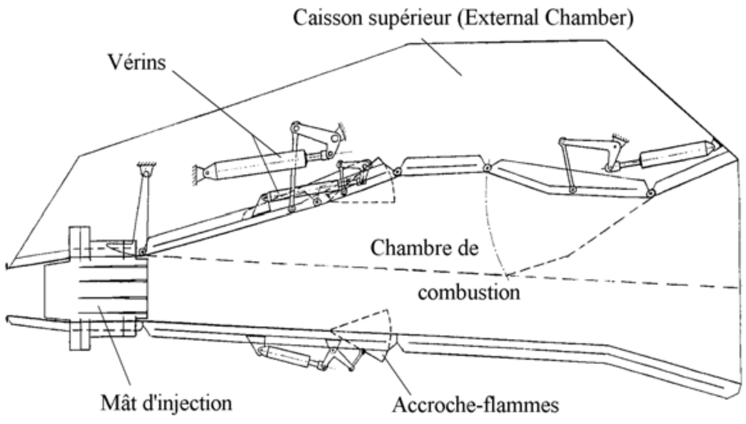

Fig.8 – variable geometry WRR engine

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.9), while PIAF studies, performed with MAI, are focused of a translating cowl (Fig.10).

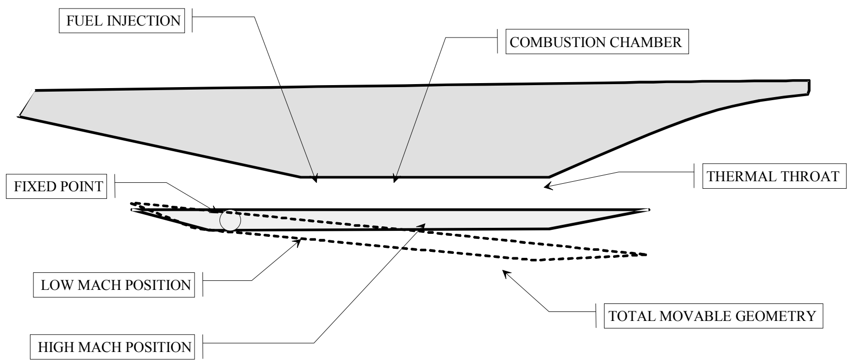

Fig.9 – variable geometry PROMETHEE engine

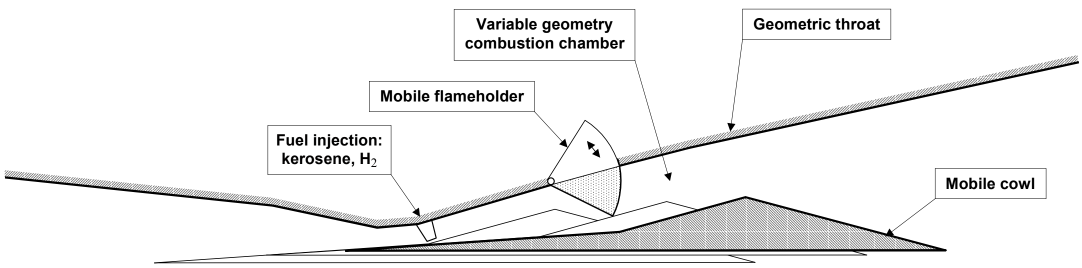

Fig.10 – variable geometry PIAF engine

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 airbreathing engine can be larger at low Mach number providing high thrust level and then better vehicle specific impulse. Then, it is possible to switch to airbreathing mode earlier increasing subsequently the overall performances.

Airbreathing 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 airbreathing 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 airbreathing propulsion system needs and airframe structural efficiency (Ref [13]).

The first example of these possible vehicle consists in twin axisymmetric fuel tanks, which are linked by a large 2 D airbreathing propulsion system while the rocket engines are placed on the base of cylindrical fuel tanks (Figure 1).

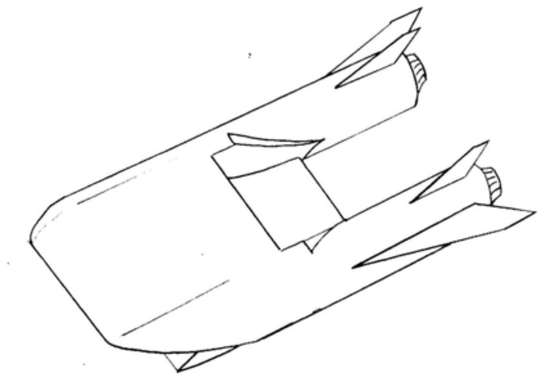

Figure 11 : Twin fuel tank concept

This configuration lends itself a very large, fully variable geometry airbreathing system, which has no limitation at low Mach number and then can provide a very high level of thrust. The upstream position of the airbreathing 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 reentry phase (air inlet closed of-course) is questionable.

An other concept can be proposed as shown on Figure 12. It is based on a double cone fuselage, which corresponds with a very good structural efficiency. The airbreathing 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 reentry phase (180° vehicle turn before reentry). Telescopic aerodynamics could provide performing pre-compression while allowing a large nose radius during the reentry phase.

A completely axisymmetric concept can be also proposed as shown on Figure 13. It is also based on a double cone fuselage providing good structural efficiency. The airbreathing 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 PIAF concept (Fig.10). 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 airbreathing 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 reentry 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 airbreathing 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.

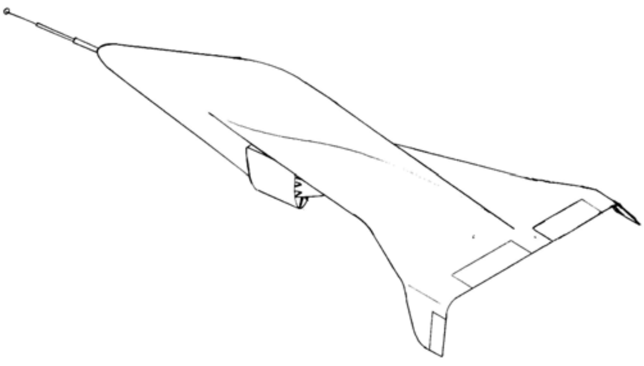

Fig. 12 – double cone airframe concept

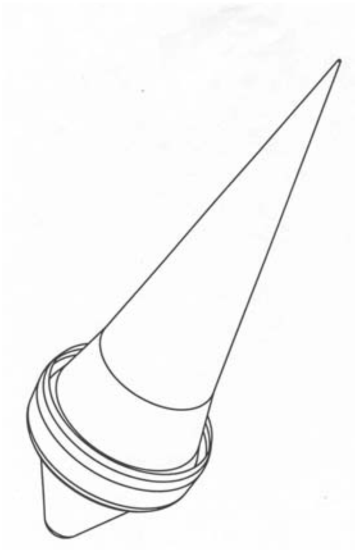

Fig.13 – Fully axisymmetric concept

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 two by two.

These tests have two 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.

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 the Central Institute of Aviation Motors from Moscow (CIAM). But, this program was based on a hydrogen fueled axisymetrical engine placed on top of a Russian SA6 missile during the whole flight (Ref[43]).

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 to evaluate 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 mentionned JAPHAR program. This program aimed at defining a development methodology and at designing a flight experimental vehicle allowing to validate 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 hereabove, and on the base of previous acquired results, MBDA France and ONERA started a flight test program, called LEA, in Januar 2003 with the support of French Administration (Ref [47] and [48]). In order to limit the cost, this flight test program would be realized with a minimum experimental vehicle (5 meters long) without any technology demonstration purpose (use of existing technologies as often as possible) (Fig. 14). 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 organisations.

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 airbreathing propulsion system will be ignited during approximatively 5 seconds with a fuel-to-air equivalence ratio variation (Fig. 15).

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 transmited 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.

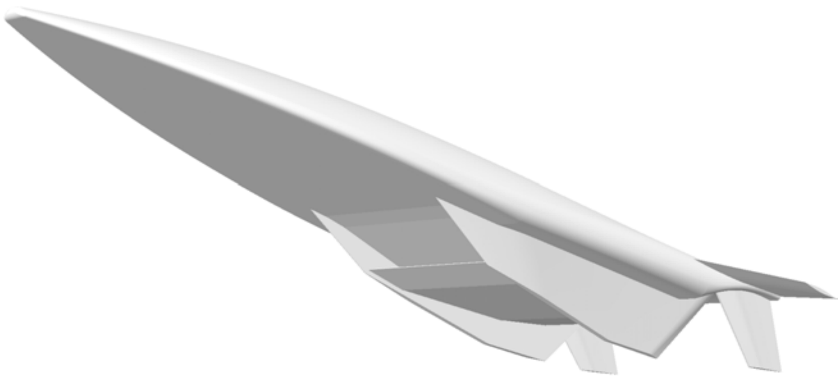

Fig.14 – LEA flightexperimental vehicle

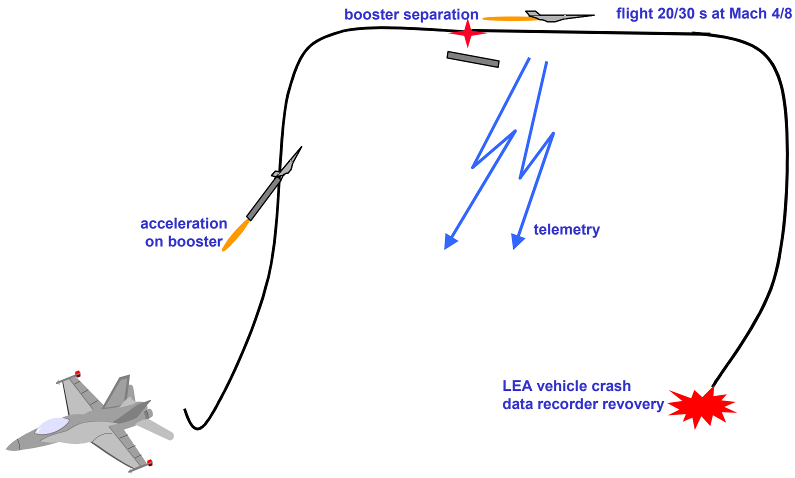

Fig.15 – LEA flight test sequence

A general approach for on-ground testing has been defined on the base of previous studies (Ref [49] – Fig.16). A large part of the on-ground testing program should be realized in the S4Ma windtunnel located in ONERA Modane test Center in the French Alps. It is intended to upgrade this test facility in order to take advantage of the existing alumina peeble bed heater which allows to perform test with air non vitiated by water vapor up to Mach 6.5 conditions (1800 K). Thanks to a complementary pre-burner, tests in conditions corresponding to flight at Mach 7.5/8 should be also easily feasible.

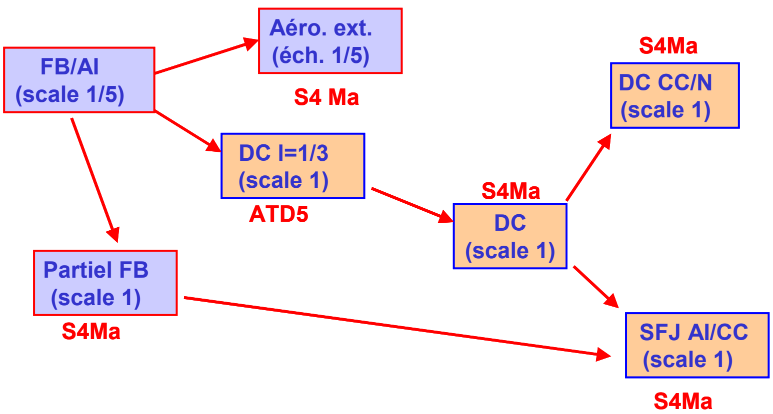

Fig. 16 – general approach for on-ground testing

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…) 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.

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.17) and maximize the net thrust (by reinjecting heat lost along combustor walls into the propulsive thermodynamic cycle).

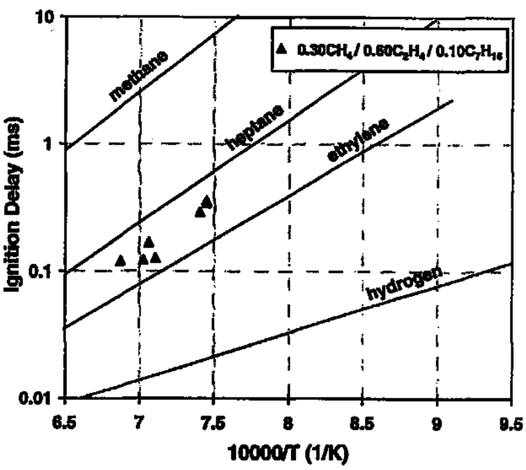

Fig.17 – example of ignition delay for a hydrocarbon mixture representing fuel reforming products

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 18 shows how the decomposition of the fuel can significantly increase the thermal absorption capacity.

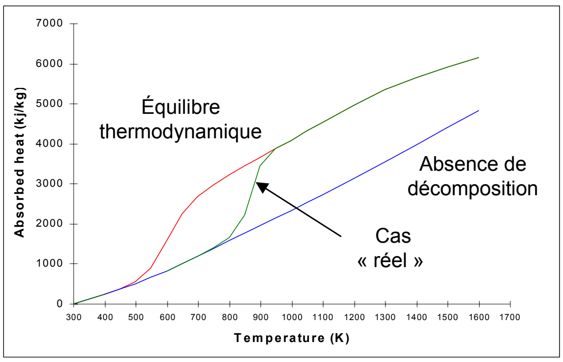

Fig.18 - Endothermic effect

In order to analyze and get to a better understanding of the different thermochemical mechanisms associated to fuel reforming process inside the cooling system, a pilot system has been developed (Fig.19). 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.

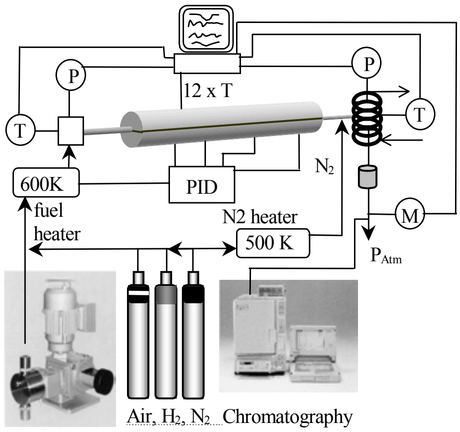

Fig. 19 - Pilot system in operation at ONERA

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.20) taking into account in the same computation the reforming process (Fig.21)

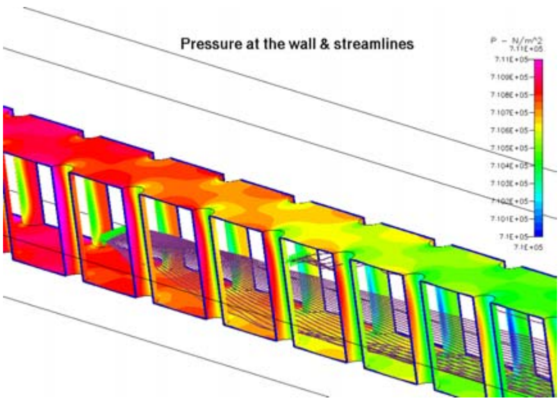

Fig.20 – Example of computation in Ptah-Socar cooling circuit

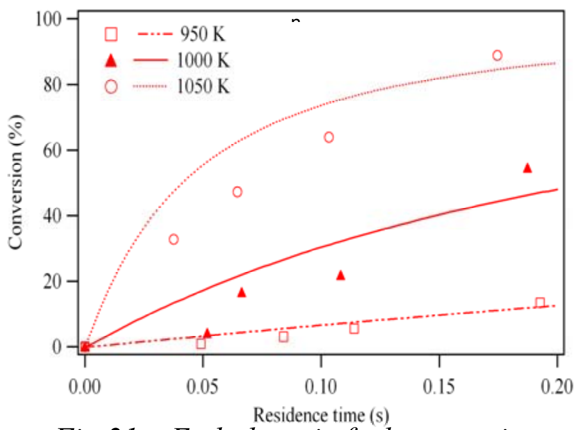

Fig.21 – Endothermic fuel conversion as function of residence time

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 airbreathing 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 (EADS ST) and SNECMA during the PREPHA program and led to basic test, performed at ONERA (Ref[2]). In parallel, MBDA France and EADS ST led in house development of hydrogen or hydrocarbon fuel-cooled injection strut (St ELME) (Ref [52] – Fig.22).

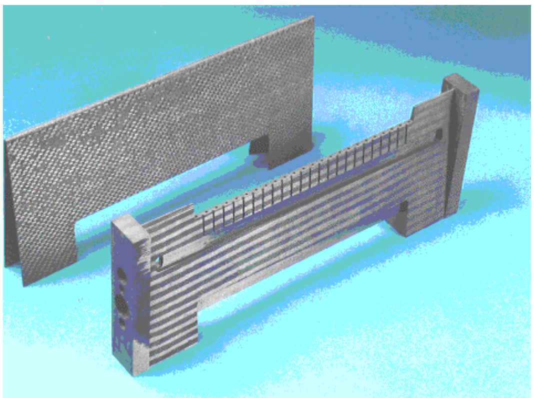

Fig.22 – H2 cooled C/C injection strut

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 pannel has been tested in MAI test facility, in which a hydrogen fueled scramjet combustion chamber is used as high temperature gas generator. The tested cooled-pannel (100x200 mm²) is placed at the scramjet chamber exit (Fig.23). A wedge is facing the tested pannel in order to create a shock wave whose interaction with the tested pannel increases the heat flux, which can reach 3 MW/m².

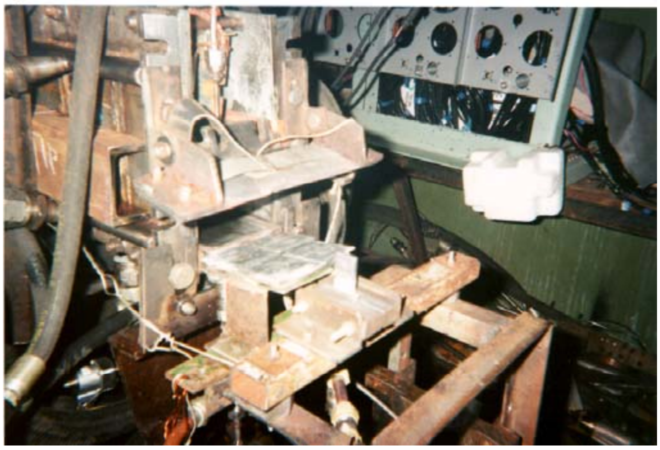

Fig. 23 – Cooled pannel test facility at MAI

Today, MBDA France and EADS 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 EADS 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-pannels densified by CVI process (Fig.24). 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.25).

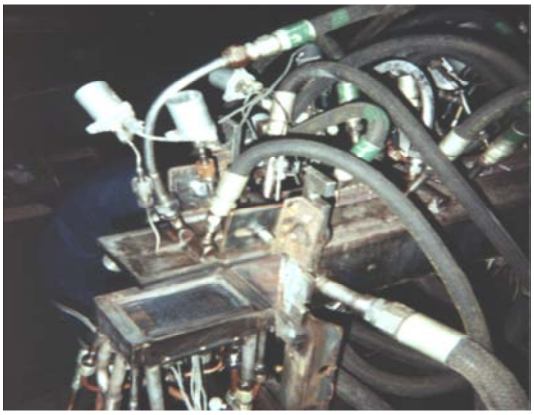

Fig.24 – PTAH-SOCAR pannel in test at MAI

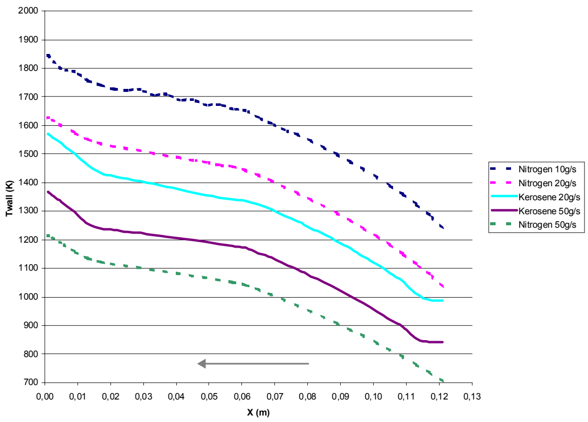

Fig. 25 – Hot wall temperature evolution

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. 26) :

- 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 EADS ST Germany). This new version has already been tested at component level (mechanic and thermic 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.

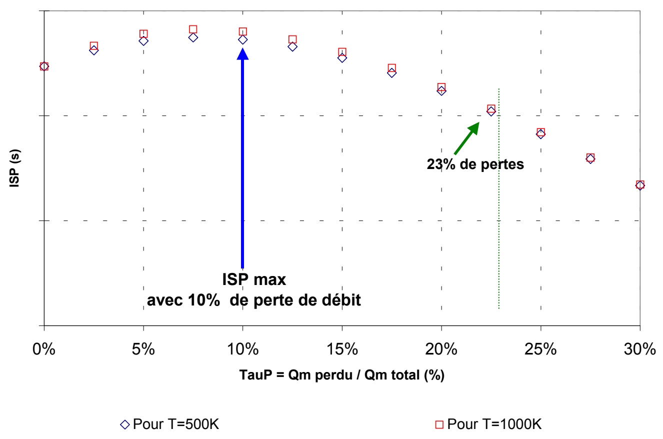

Fig.26 – Fuel leakage effect on specific impulse

A portion of cooled combustion chamber realized with this technology will be tested at ONERA in a few months.

Further works

Beyond the works already in progress, the test facility, developed by MBDA France and ROXEL in their Bourges Subdray test center in the framework of PREPHA program (Ref [49]), 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…)

Conclusion

The ramjet/scramjet concept constitutes the main airbreathing propulsion system which can be use in a very large flight Mach number range up to Mach 10/12 and then could allow to develop 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, this effort is mainly led by France and aims at addressing the two key technology issues which are the accurate prediction of the aero-propulsive balance of an airbreathing 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 seven or height years (2011/2012).

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