CHAPTER 1: OVERVIEW OF NATO BACKGROUND ON SCRAMJET TECHNOLOGY

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1 Introduction

The purpose of the present overview is to summarise the current knowledge of the NATO contributors. All the topics will be addressed in this chapter, with references and some examples. This background enhances the level of knowledge of the NATO scramjet community, which will be used for writing the specific chapters of the Report. Some previous overviews have been published on scramjet technology worldwide. One of the most documented within the available overviews on scramjet technology worldwide is [D1] from Dr. Tom Curran.

NASA, DOD, the U.S. industry and global community have studied scramjet-powered hypersonic vehicles for over 40 years. Within the U.S. alone, NASA, DOD (DARPA, U.S. Navy and USAF), and industry have participated in hypersonic technology development. Over this time NASA Langley Research Center continuously studied hypersonic system design, aerothermodynamics, scramjet propulsion, propulsion-airframe integration, high temperature materials and structural architectures, and associated facilities, instrumentation and test methods. These modestly funded programs were substantially augmented during the National Aero-Space Plane (X-30) Program, which spent more than $3B between 1984 and 1995, and brought the DOD and other NASA Centers, universities and industry back into hypersonics. In addition, significant progress was achieved in all technologies required for hypersonic flight, and much of that technology was transferred into other programs, such as X-33, DC-X, X-37, X-43, etc. In addition, technology transfer impacted numerous other industries, including automotive, medical, sports and aerospace.

The future development of scramjet and hypersonic technology within the USA falls under the NASA Advanced Space Transportation Program and yet to be defined DOD interests. A complete plan will be completed in 2002. This current ASTP program is a comprehensive program designed to complete technology development and demonstration by 2018, leading to a Space Shuttle replacement vehicle IOC by 2025. This program is focused on the NASA third generation goal, which is to reduce cost and increase reliability and safety. Assuming 1000-2000 flights per year, the 3rd generation goals are $100/lb. of payload to LEO, and a 10^{-6} failure rate. These stretch goals can not be achieved using rocket propulsion. This program includes system analysis, focused and generic research and technology development, and ground and flight technology demonstrators. Systems studies are being used to evaluate numerous vehicle architectures: single stage to orbit (SSTO) and two stage to orbit (TSTO), vertical and horizontal takeoff, hydrogen, hydrocarbon and dual-fuel, as well as alternate propulsion systems. Propulsion systems generally fall into two categories: Rocket-based and turbine-based. Both approaches use dual-mode scramjets over much of the flight envelope, from Mach 3 or 4 to Mach 12 to 15. Rocket based combined cycle (RBCC) systems use rockets in the scramjet duct for low speed acceleration and/or orbital insertion. Turbine based combined cycle (TBCC) systems use turbine based engines for low speed acceleration, and some form of rocket for orbital insertion. In TSTO systems, the propulsion options double. The program is also developing the critical technologies identified by the system studies. These range from structures and materials, to tires to operational and integrated vehicle health monitoring (IVHM).

The French national Research and Technology Program for Advanced Hypersonic Propulsion (PREPHA) ended in 1999. It aimed at acquiring a first know-how for the hydrogen-fueled scramjet, which could be combined with other airbreathing modes (particularly ramjet) and rocket mode for powering future reusable space launchers. It gave the opportunity to acquire a first know-how in scramjet and dual-mode ramjet components design (inlet, combustor, injection struts, nozzle) and hypersonic airbreathing vehicle system studies (design and performance evaluation for space launchers, missiles and experimental flight vehicles) [A1]. The French Aeronautics and Space Research Center (ONERA) and EADS Aerospatiale Matra Missiles (now in the new MBD.A Missiles Systems European group and its "MBDA-F" French subsidiary) have been major contributors to the PREPHA Program.

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In France, after the end of the National PREPHA program, MBDA-F and ONERA have taken the initiative in starting further works to preserve the intellectual and material investment and to improve mastery of hypersonic airbreathing propulsion. Since 1997, ONERA and DLR are leading the in house research program JAPHAR (Joint Airbreathing Propulsion for Hypersonic Application Research) [A5]. This program aims at studying a hydrogen fueled dual mode ramjet working in the Mach number range from 4 to 8. It also aims at defining a methodology for ground and flight performance demonstration. That includes the definition of a possible experimental vehicle able to autonomously fly in the given Mach number range [A6]. MBDA-F leads a cooperation with Moscow Aviation Institute (MAI) to develop a dual-mode dual fuel ramjet, operating from Mach 3 to Mach 12 with a variable geometry. MBDA-F and EADS-Launch Vehicles (EADS-LV) are also developing an innovative technology for fuel-cooled composite material structures. Under the aegis of the French MoD, MBDA-F and ONERA are leading the PROMETHEE R&D program to improve knowledge on hydrocarbon fueled dual mode ramjet for missile application. Its aims at developing a propulsion system able to power a missile from Mach 2 to Mach 8. First phase is expected to end in 2002 [A16].

Previously, in Germany, some cooperative work has been performed with Russia on hypersonic flight testing issues and on scramjet flowpath technology at TsAGI (Jukowsky, Russia).

Contribution to the education of students is also one of the important elements of this scramjet technology effort. Students, young scientists or technicians are often enthusiastic to be associated, even for only several months on scramjet technology development efforts [B10], [B18].

2 System studies

2.1.1 Reusable space launchers

The USA focused on SSTO technology during the NASP era. The program failed because of naively optimistic projected costs and schedules. After NASP (1984-1994) NASA initiated several hypersonic technology programs: the LaRC/DFRC Hypersonic X-Plane Program, Hyper-X, in 1996; the GRC Trailblazer in 1997; and the MSFC Advanced Reusable Transportation ART technology program in 1997, Bantam in 1997, Spaceliner-D and finally, just “Spaceliner” in 1999. Of these programs only Hyper-X and ART build on the technology gains of the X-30 program. The Hyper-X Program focused on extending scramjet powered vehicle technology to flight, elevating as much technology as possible, and validating, in flight, the design systems, computational fluid dynamics (CFD), analytical and experimental methods required for this complex multi-disciplinary problem. The smaller ART program focused on rocket based combined cycle (RBCC—i.e. single duct air augmented ramjet/dual mode scramjet)) wind tunnel testing of alternate airframe integrated scramjet flowpath concepts.

Currently, all US space access focused hypersonics propulsion is incorporated under the NASA Marshall Space Flight Center (MSFC) led “third-generation” (Venture-star replacement), “Spaceliner” Program. The third generation goal is to reduce cost and increase reliability and safety. Assuming 1000-2000 flights per year, the 3rd generation goals are \$100/lb. of payload to LEO, and 10^{-6} failure rate. These stretch goals can not be achieved using rocket propulsion. This program includes systems analysis, focused and generic research and technology development, and ground and flight technology demonstrators. Systems study being used to evaluate numerous vehicle architectures including permutations of single stage to orbit (SSTO) or two stage to orbit (TSTO), vertical or horizontal takeoff, hydrogen, hydrocarbon or dual-fuel, as well as various propulsion systems. Propulsion systems studied generally fall into two categories: Rocket-based or turbine-based. Both approaches use dual-mode scramjets over much of the flight envelope, from Mach 3 or 4 to Mach 12 to 15. Rocket based combined cycle (RBCC) systems use rockets in the scramjet duct for low speed acceleration and/or orbital insertion. Turbine based combined cycle (TBCC) systems use turbine based engines for low speed acceleration, and some form of rocket for orbital insertion. In TSTO systems, the propulsion options double. Current high fidelity analysis is limited to NASP derived air breathing launch vehicles. These single stage vehicles close at near one million pounds take off gross weight for 50,000 pounds to low earth orbit. Low speed engine selection and uncertainty in combined/combination engine performance and weight can change vehicle weight by 40 percent, as shown in figure 1. Two stage systems tend to fall within this same band, but have an uncertainty in closed weight.

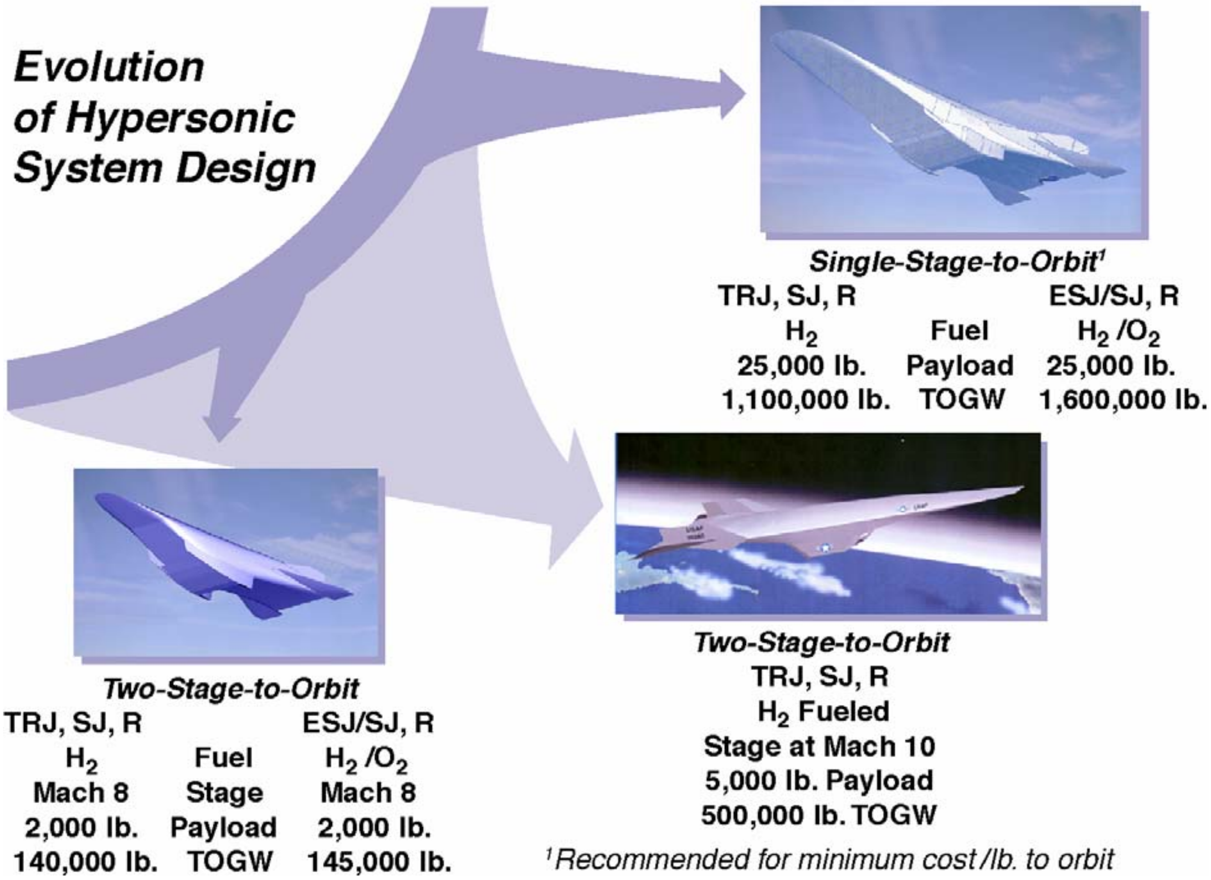

Figure 1: Vision Vehicle (Space Access).

During the 1980s, several national programs were undertaken around the world to study combined propulsion for space launchers and to acquire needed technologies. Some system studies were performed in France to evaluate combined propulsion interest for space launchers. These studies, considering TSTO and SSTO vehicles, concluded that combined propulsion didn't present any advantage if the airbreathing phase is limited to subsonic combustion (maximum flight Mach number 6/6.5). Then it was decided to continue studies by exploring the possibility to use supersonic combustion. In this way, the French national Research and Technology Program for Advanced Hypersonic Propulsion (PREPHA) started in 1992 under the aegis of Ministry of Defense, Ministry of Research and Technology, National Space Agency ([C7]).

Four concepts of combined propulsion systems, using slush hydrogen as fuel, have been considered at the beginning of the PREPHA: two twin-duct concepts which are turborocket-scramjet-rocket and turbojet-dual mode ramjet-rocket; two one-duct concepts which are rocket-dual mode ramjet-rocket and ejector dual mode ramjet-rocket ([C17] and [C18]). After integration of these propulsion systems on the generic vehicle, trajectories simulation allowed selection of the rocket—dual mode ramjet—rocket concept (in separate ducts) and to improve, step by step, airframe and propulsion system design [C19]. Finally, these studies concluded that there was the feasibility of a vehicle able to fulfill the mission with a total take-off mass of 487.3 metric tons without payload or 540 t with a payload of 5 metric tons [C20].

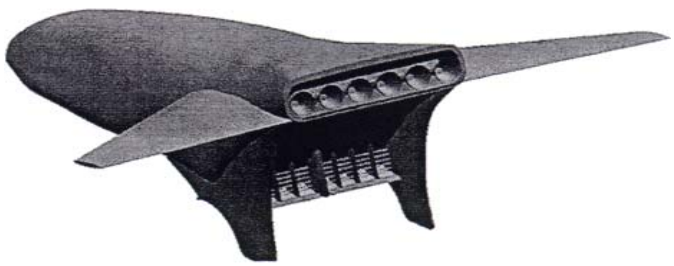

Figure 2: PREPHA vehicle (550 metric tons Take-off weight)

In addition, some ESA (European Space Agencies) programs have been conducted in the 90's utilizing SSTO with scramjet and other cycles [B12]. In two of these programs (FESTIP 1 and 2), Belgium has realized studies on pre-design and trajectory calculations for SSTO and TSTO using scramjets, high speed ramjets, RBCC and other cycles [E1].

In the scope of the WRR co-operation between MBDA-F and MAI, several topics have been addressed, including system studies (for the air-breathing engine point of view) and technological work [B3, B9, B24, B25, B26]. The results obtained during the WRR Prototype development phase in term of propulsive performance and cooling system technology have been used to determine how the fully variable geometry of the two-mode ramjet could really impact on the global performance of a SSTO vehicle [B26]. A WRR-type engine has been defined and integrated to the generic SSTO vehicle designed during the PREPHA Program. The accessible performance of this vehicle have been compared with those obtained with the same vehicle powered by the final version of the fixed geometry two-mode ramjet designed during the PREPHA Program. This study indicated a real interest of variable geometry for the combustor: the increase of performance seems much higher than the increase of weight (actuators…). In parallel with the propulsion-oriented WRR space launcher system studies by MBDA-F and MAI , ONERA is still continuing some system studies to assess the interest of a possible use of a high-speed airbreathing system for space launcher application [A20].

2.1.2 Missiles

It is generally assumed that the first application of high-speed airbreathing propulsion will be missiles or the strategic UAV ([A13] and [A14]). In France, after several in-house studies [D3], a generic missile is studied within the PROMETHEE program, in order to more deeply study the military application and to develop some needed specific technologies ([A15] and [A16]). The technical program is oriented by the conceptual design of a generic air-to-ground missile. A variable geometry engine concept has been selected. Its preliminary design study is under enhancement. This design study allows performances to be determined and to mass budget characteristics to numerical flight simulation codes in order to estimate the achievable global performance of the missile. The generic mission which has been considered is the air-launch air to ground concept, but other may be derived.

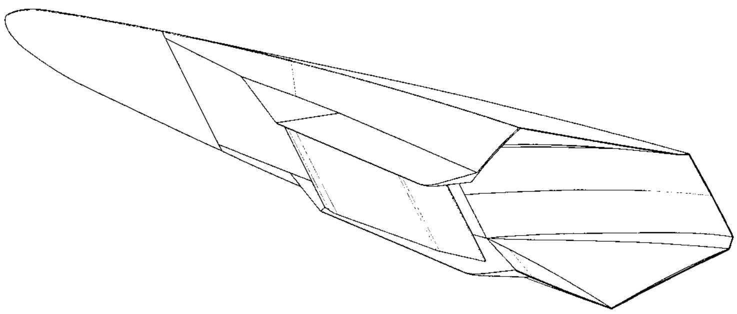

Figure 3: artist view of PROMETHEE missile.

In the US, a lifting body scramjet missile concept was developed under the DARPA ARRMD. Demonstration of the hydrocarbon-fueled engine is currently underway in ground facilities and it will be flown on the X-43C. Within the DOD several military hypersonic programs emerged in the USA after NASP. These programs include the USAF AFRL Hypersonic Technology (HyTech) program [D7], the Defense Advanced Research Projects Agency (DARPA) Affordable Rapid Response Missile Demonstrator (ARRMD) Program, the USN Rapid Response Missile Program and the Army Scramjet Technology Development Program.

Figure 4: ARRMD Missile**.**

2.1.3 Aircraft

In France, only very preliminary studies have been performed with scramjet-powered aircraft (i.e., the Mach 5 civil transport or higher Mach number commercial or military aircraft, mainly from the propulsion point of view.) No application of hypersonic civilian transports is currently being considered in the US. In addition to the missiles programs, the USAF Aeronautical Systems Center, in collaboration with the Air Combat Command, has conducted a Future Strike study, which included hypersonic aircraft. With this renewed interest in hypersonic vehicles, requirements are being discussed which can only be met with hypersonics systems. These include the USAF CONUS-based Expeditionary Aerospace Force concepts, and "control of the complete aerospace continuum" [U15]. Some potential hypersonic aircraft concepts and capabilities are presented in figure 5.

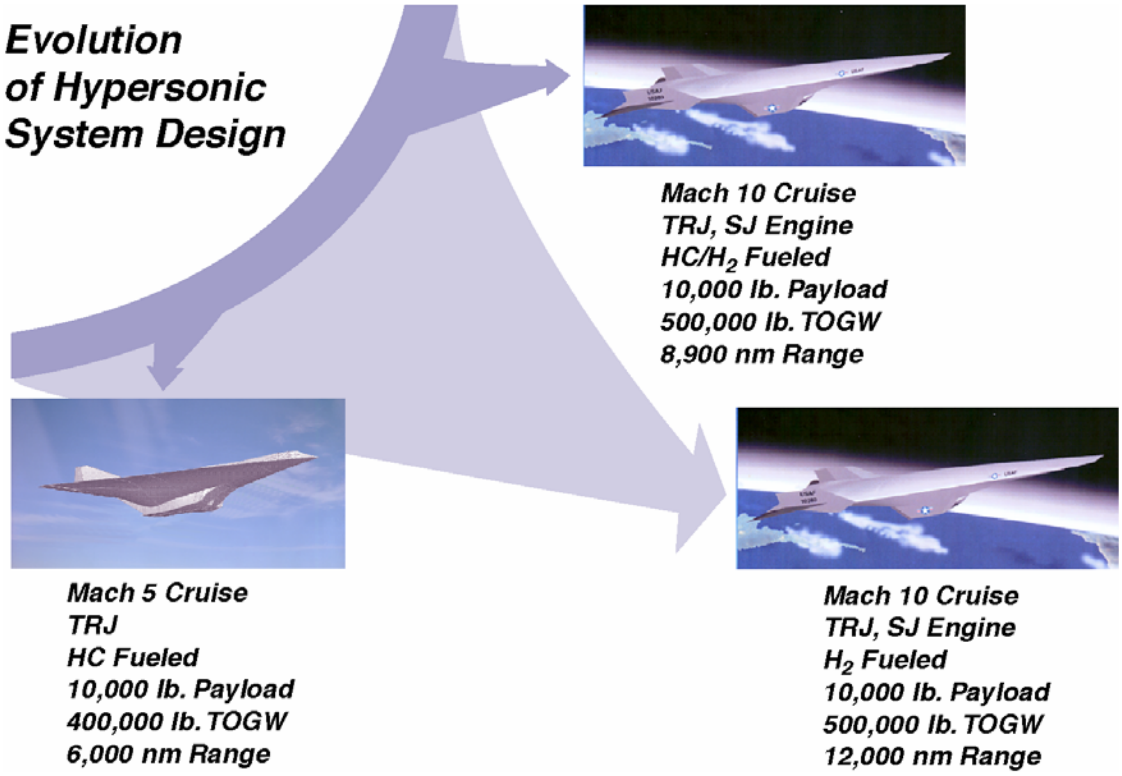

Figure 5: Hypersonic aircraft concepts.

3 Design tools

3.1 Test facilities

It is currently necessary to simulate the flight operation of a scramjet engine in ground based facilities.

The NASA Langley Research Center Scramjet Test Complex is made up of five facilities, the Direct Connect Supersonic Combustion Test Facility, the Combustion Heated Scramjet Test Facility, the Arc-Heated Scramjet Test Facility, the 8-ft. High Temperature Tunnel, and the Hypersonic Pulse Facility. The Langley Direct-Connect Supersonic Combustion Test Facility (DCSCTF) is used to test ramjet and scramjet combustor models in flows with stagnation enthalpies duplicating that of flight at Mach numbers between 4 and 7.5. Results of the tests are typically used to assess the mixing, ignition, flameholding, and combustion characteristics of the combustor models. The facility operates “directly connected” to the combustor model with the entire facility test gas mass flow passing through the model. The combustor model may exhaust freely (into the test cell), or directly (connected) to an air-ejector or to a 70-ft diameter vacuum sphere. Nozzle geometric simulations can also be added at the exit of the combustor models.

The Langley Combustion Heated Scramjet Facility (CHSTF) has historically been used to test complete (inlet, combustor, and partial nozzle) subscale scramjet component integration models in flows with stagnation enthalpies duplicating that of flight at Mach numbers from 3.5 to 6. The CHSTF uses a hydrogen, air, and oxygen heater to obtain the flight stagnation enthalpy required for engine testing. Oxygen is replenished in the heater to obtain a test gas with the oxygen mole fraction of air (0.2095). The facility may be operated with either a Mach 3.5 or 4.7 nozzle. Either gaseous hydrogen or gaseous hydrocarbon (both at ambient temperature) may be used as the primary fuel in the scramjet engines tested in the CHSTF. A 20-percent silane, 80-percent hydrogen mixture (by volume) is available for use in the scramjet model as an igniter/pilot gas to aid in the combustion of the primary fuel.

The Langley Arc-Heated Scramjet Test Facility (AHSTF) is used for tests of component integration models of airframe integrated scramjet engines at conditions experienced at flight Mach numbers of 4.7 to 8. Results are used to assess the performance of the scramjet, to optimize the design of the components, and to optimize fueling schemes. Typical models include the inlet, isolator, combustor, and a significant portion of the nozzle and are hydrogen and silane fueled. The flow at the exit of the facility nozzle simulates the flow entering a scramjet engine module in flight, which has been processed by the forebody shock of the vehicle. The total enthalpy of the flight condition is achieved by electrically heating the air with a Linde arc heater. Run times normally range from 30 sec at flight Mach number of 8 simulated conditions to 60 sec at flight Mach number of 4.7 simulated conditions.

The Langley 8-Foot High Temperature Tunnel (8-Ft HTT) is a combustion-heated hypersonic blowdown-to-vacuum wind tunnel that provides duplication of total flight enthalpy for Mach numbers of 4, 5, and 7 through a range of altitude from 50,000 to 120,000 ft. The open-jet test section is 8 ft in diameter and 12-ft long. The test section will accommodate very large models, air-breathing hypersonic propulsion systems, and structural and thermal protection system (TPS) components. Stable wind tunnel test conditions can be provided up to about 60 seconds. Additional simulation capabilities are provided by a radiant heater system that can be used to simulate ascent or entry heating profiles. The high-energy test medium is the combustion products of air and methane that are burned in a pressurized combustion chamber. Oxygen is added for air-breathing propulsion tests. Hypersonic air-breathing propulsion system tests are performed with the propulsion test article (e.g. NASP concept demonstration engine, Hyper-X flight vehicle) attached to a model support pedestal mounted on an external force measurement balance. Propellant fuel (e.g. gaseous hydrogen, liquid hydrocarbon) and purge gases are supplied to the test article by the facility.

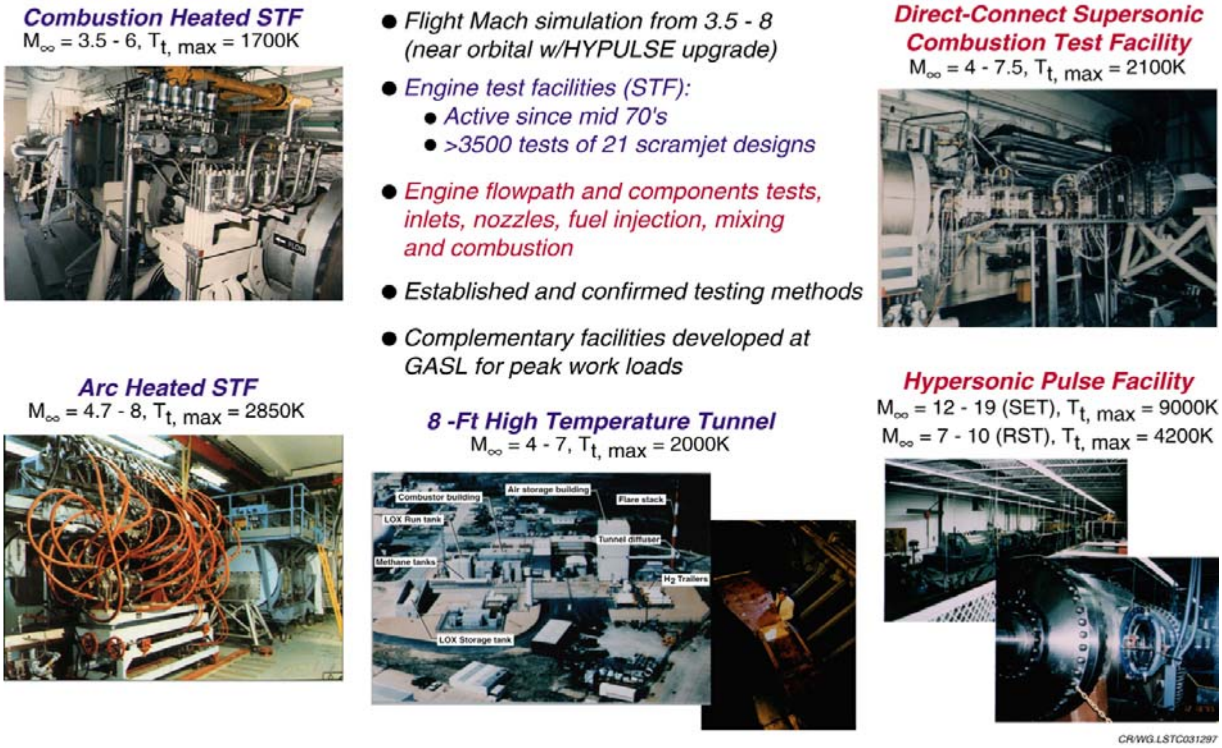

Figure 6: NASA Langley Scramjet Test Complex.

The Hypersonic Pulse Facility (HYPULSE) is a dual-mode shock tunnel facility. It can operate in both a reflected shock tunnel mode and a shock-expansion mode. A 7 ft diameter test section is available for aerothermodynamic and propulsion testing for models up to 15 feet long. HYPULSE is operated as a Ludweig tube to reproduce flight conditions up to Mach 2, as a reflected shock tunnel for flight conditions from Mach 4 to Mach 12, and as a shock expansion tube for flight conditions from Mach 12 to Mach 25. When operated in the tunnel mode, HYPULSE expands the test gas to Mach 6.5 using a 26” diameter axisymmetric nozzle. Optical access is provided for schlieren images and for laser diagnostics, and instrumentation is available for collecting measurements from pressure, heat transfer, and temperature transducers.
The Arnold Engineering and Development Center (AEDC) facilities in Tennessee are used for endurance tests of high-speed engines or currently, engines at moderate Mach number [D5]. The NASA Hypersonic Test Facility at Glenn Research Center is used to do free jet tests up to Mach 7 conditions with no water or CO2 in the incoming air [D6]. A GASL test facility is used to simulate trajectory variations, thanks to a high area-ratio axis-movable nozzle.

Thanks to the PREPHA program, ONERA and MBDA-F extended the simulation capabilities of the French ramjet test facilities up to Mach 7.5 flight conditions and 100 kg/s of air ([A2] and [A3]). Two facilities have been used extensively: the ATD 5 test cell of ONERA Palaiseau and the MBDA-F hypersonic test rig of Bourges-Subdray ramjet test facility. The first facility is limited in size (4 kg/s and 4 MPa with 2400 K of stagnation conditions, 10 seconds of test) but able to reproduce Mach 7.5 conditions. The second facility (Subdray) is currently limited to Mach 6.5 conditions, but with higher mass flow (up to 100 kg/s at 8 MPa, two minutes test duration). These are water-vitiated facilities.

New optical diagnostic methods are under development, which will allow exploration of the flow into the combustion chamber in an industrial facility [A22], [D4]. Up to now the laser-induced optical methods are mostly used in laboratories or more academic supersonic combustion configurations [A16] [A21]. Using available pulsed high enthalpy tunnels to reach higher stagnation test conditions is also under consideration in Europe [D11], [D15].

3.2 System analysis, CFD and modeling

The key to any hypersonic vehicle development or technology program is a credible preliminary system analysis to identify the technical requirements and guide technology development. The complexity of the hypersonic airbreathing system and the small thrust margin dictate that a thorough system analysis be performed before any focused technology development is started. System analysis is executed on four levels. The lowest level**,** designated “0,” does not require a physical geometry. The level zero analysis utilizes ideal engine cycle performance, historical L/D and C_d values for aerodynamic performance, design tables (or weight fractions) for structure and components weight, “rocket equation” for flight trajectory, and estimates for packaging. This analysis does not require a specified vehicle, engine flowpath or systems definition. All higher levels of analysis require a vehicle, engine flowpath shape and operating modes, system definition, etc.

The next level of system analysis, referred to herein as Level 1, utilizes uncertified cycle performance and/or CFD, impact theory, unit or uncertified finite element model (FEM) weights, single equation packaging relations, and energy state vehicle performance. This level of analysis does not capture operability limits, and thus has large uncertainties.

Level 2 analysis utilizes “certified,” methods; i.e., the user has sufficient relevant experience. This level uses the same methods for propulsion, aerodynamics, structure and weights (but certified), trimmed 3-DOF (degree of freedom) vehicle performance analysis and multiple equation, linear or non-linear packaging relations. Certification is only achieved by demonstration that the methods used work on the class of problems simulated (this relates to the method, as well as the operator applying that method). For example, at level 2 analytical models utilize corrections for known errors, such as inlet mass spillage, relevant empirical fuel mixing models [U1], shear and heat flux models [U2], etc. This empirical approach is based on experimental (wind tunnel tests, structural component tests, etc) data. Higher level methods (CFD, FEM) are used to refine the vehicle and propulsion system closure.

The highest design level (level 3) is achieved only by having a significantly large fraction of the actual vehicle manufactured and tested. Wind tunnel and other ground testing provide less verification than flight tests. Although numerous components have been built and ground tested, flight data is required for the highest level of design. This has not yet been done for a hypersonic airbreathing vehicle (see §6 below). Whatever the level of system analysis, closure is achieved by sizing the vehicle so that the propellant fraction required (for the mission) is equal to the propellant fraction available (packaged within the sized vehicle). However, the reported closure weight is only as good as the lowest level of analysis used in the “closure.”

Computational fluid dynamics (CFD) has multiple roles in the design of a hypersonic propulsion system. It primarily serves as an engineering tool for detailed design and analysis [U3]. In addition, results from CFD analyses provide input data for cycle decks and performance codes. Finally, CFD has several applications in engine test programs to develop an engine concept. CFD is first used to guide the test setup and to determine the proper location for placement of instrumentation in the engine. It has also proven to be an effective tool for determining the effects of a facility on testing; for example, the effects of contaminants in a combustion heated facility on an engine combustor test. During and following a test, CFD is useful to predict flowfield measurements as a complement to measured data. Various computational strategies are utilized in inlet, combustor, and nozzle of a scramjet engine.

Computational analyses of inlets typically employ codes that solve the Euler equations, or Euler codes iterated with the boundary layer equations for viscous effects, for initial analyses. More detailed calculations utilize either the parabolized Navier-Stokes equations, or the full Navier-Stokes equations if significant flow separation must be considered. All of the calculations typically solve the steady-state equations so that simulations can be completed in reasonable times. Turbulence is modeled using either algebraic or two-equation turbulence models with empirical compressibility corrections and wall functions. Transition models are not currently being employed. Thermodynamic properties are generally determined by assuming that the inlet flow behaves as a perfect gas or equilibrium air. Calculations are conducted on fixed grids of 100,000 to 3,000,000 points in multizone domains. A limited degree of dynamic grid adaptation is employed when necessary. There is a serious need for the development of advanced transition and turbulence models. This is likely the most limiting area for accurate modeling of inlet flowfields. Promising work is now underway to develop new algebraic Reynolds stress turbulence models with governing equations that can be efficiently solved [U4], [U5]. For non-equilibrium flows, the differential Reynolds stress equations must be solved, however, and further work is necessary for this to be done more efficiently. Advances in large eddy simulation, with the development of subgrid scale models appropriate to high-speed compressible flow, may also allow this technique to be applied to inlet flows in the future [U6]. Finally, work is needed to develop improved transition models for inlet flows, particularly with flows exhibiting adverse pressure gradients. Some models are quite operational to predict the transition beginning zone, but generally not its end.

Computations of combustor flowfields typically employ codes that solve either the parabolized or full Navier-Stokes equations, depending upon the region of the combustor being modeled and the degree of flow separation and adverse pressure gradient being encountered. Steady-state methods are normally used with limited unsteady analyses for mixing studies or the analysis of combustion instabilities. Turbulence is again modeled using algebraic or two-equation models with empirical compressibility corrections and wall functions. There is a limited use of models to account for turbulence-chemistry interactions based on probability density functions. Thermodynamic properties are determined utilizing perfect gas or, in some cases, real gas models. Chemical reaction is modeled with reduced reaction set, finite rate models. For the hydrogen-air reactions occurring in a hydrogen-fueled scramjet, a typical reaction mechanism includes nine chemical species and eighteen chemical reactions, although other mechanisms are employed as the case dictates [U7]. Hydrocarbon-fueled scramjet concepts are modeled with much more complex mechanisms that must be further reduced to allow practical computations. Calculations in each case are typically conducted on fixed structured grids of 200,000 to 8,500,000 points in multizone domains. Typical run times on a Cray C-90 computer range from 10 to over 300 hours. Many of the future technology needs for combustor simulations follow from the needs for inlets described earlier, but several of the additional requirements will be more difficult to achieve. For combustor modeling, a factor of ten improvement in the efficiency of steady-state and temporal Navier-Stokes codes will be needed to carry out the required calculations with the necessary accuracy and design turn-around time. Multigrid methods again offer promise for significantly enhancing convergence rates, but the application of multigrid methods to reacting flows also results in additional challenges for success with the method [U8]. Current research to apply multigrid methods to high-speed reacting flows has resulted in a significant improvement in convergence rates over single grid methods. Dynamic grid adaptation will become even more important for capturing the complex flow structure in combustors, in particular the shock-expansion and vortical structure in the flow. Proper resolution of vortical flow requires very high resolution to conserve angular momentum. Again, there is a serious need for improved turbulence modeling in high speed reacting flows, both to model the turbulence field and to properly couple the effects of turbulence on chemical reaction and reaction on turbulence. Promising work is again taking place in this area using several approaches [D8]. Techniques using velocity-composition probability density functions have been successfully applied to incompressible reacting flows, and this work is now being extended [U9] to model compressible reacting flows. Work is also underway to apply Large Eddy Simulation (LES) techniques to compressible reacting flows. Subgrid scale models for the LES of these flows are currently being developed. Recent work utilizing a filtered mass density function for the LES of turbulent reacting flows appears particularly promising for the future [U6]. Finally, further work is needed to simplify the modeling of chemical reaction in combustor flowfields. Methods for systematically reducing the number of reactions in a full reaction mechanism are required to reduce the computational work [U10].

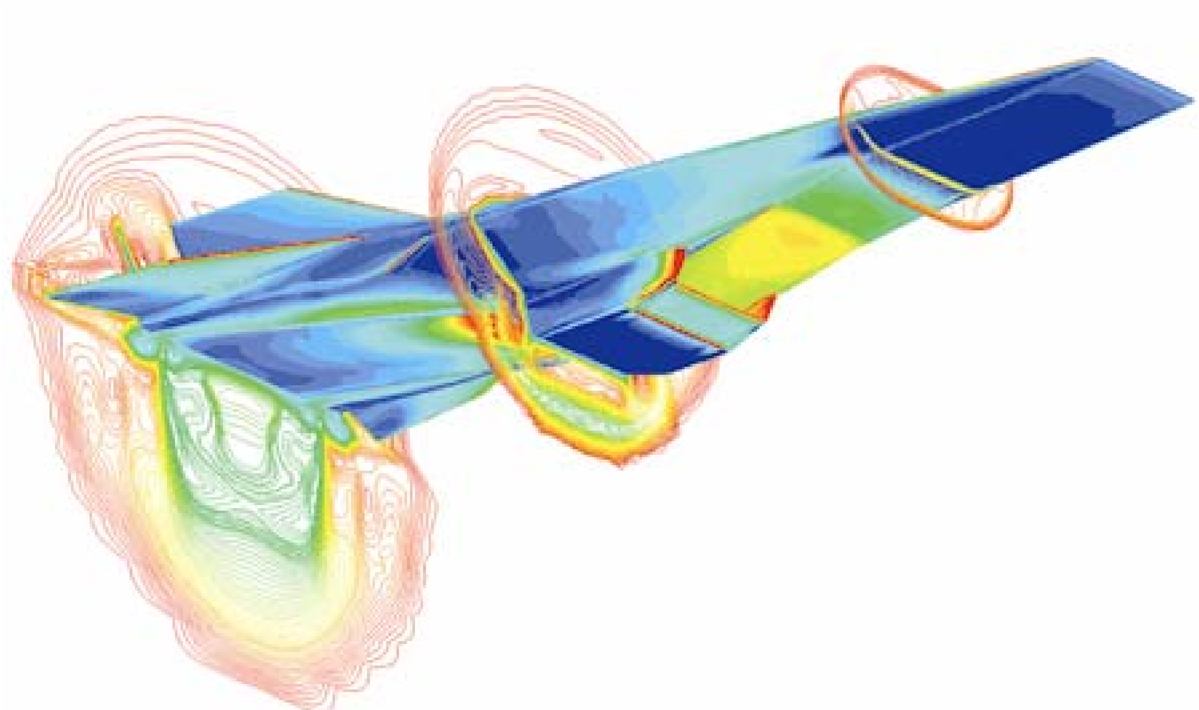

Figure 7: Hyper-X nose-to-tail and generic combustor computational solution.

Computations of nozzle flowfields are usually conducted with Euler codes, or Euler codes iterated with boundary layer calculations for initial engineering design studies; and with either parabolized or full Navier-Stokes codes for more detailed studies. Steady-state methods are normally employed. Turbulence is modeled by algebraic or two-equation models with empirical compressibility corrections and wall functions. Perfect gas or, when necessary, real gas models are used to determine thermodynamic properties. Chemical reaction is modeled with reduced kinetics models as utilized in the upstream combustor flow. Finite rate analyses are required throughout the nozzle to assess the continuing degree of reaction, and to determine the extent of recombination reactions that add to the available thrust. Calculations for complete nozzles are typically carried out on structured grids of 100,000 to 500,000 nodes grouped in multizone domains. Future technology needs for nozzle simulations, even though less demanding, follow very similar lines to the requirements for combustor simulations. Dynamic grid adaptation will be useful for capturing shock structure and resolving possible wall separation due to shock-boundary layer interactions. There is a further need for improved turbulence models, particularly for capturing the nozzle wall boundary layer relaminarization created by the favorable pressure gradient. Algebraic Reynolds stress turbulence models offer significant promise for describing these flowfields [U4], [U5]. The reduced kinetics models currently being applied to nozzle flows appear to be reasonably accurate, although some further work to improve the description of recombination may be warranted. In addition, methods of accurately predicting the combustion process and the recombination process with a small reaction set will expedite solution times.

In Europe CFD is also a big concern for high-speed propulsion development. In this view, a research program is in progress at ONERA and with MBDA-F and several research laboratories to improve the accuracy of physical models thanks to a very detailed basic test. Integration of the improved models into the code and the global validation are led together by ONERA and MBDA-F. This effort is focused on the MSD code, initially developed by ONERA to simulate internal aerodynamic flows, which has been upgraded in cooperation between ONERA and MBDA-F to perform subsonic and supersonic reactive flow simulations. It solves the unsteady, 3D, averaged Navier-Stokes equations by a finite volume algorithm on multi-domain structured curvilinear grids [C2]. It includes multi-species capability and takes into account the variations of gases' thermodynamic properties with the temperature. The MSD code has been used in France by industrial or research labs, for basic configurations up to actual combustors such as CHAMOIS scramjet. It is able to compute the heat release, to predict the ignition process (with hydrogen as fuel), and to roughly represent the liquid injection and associated droplets in turbulent 3D flow [D8], [D2], [C11], [C13].

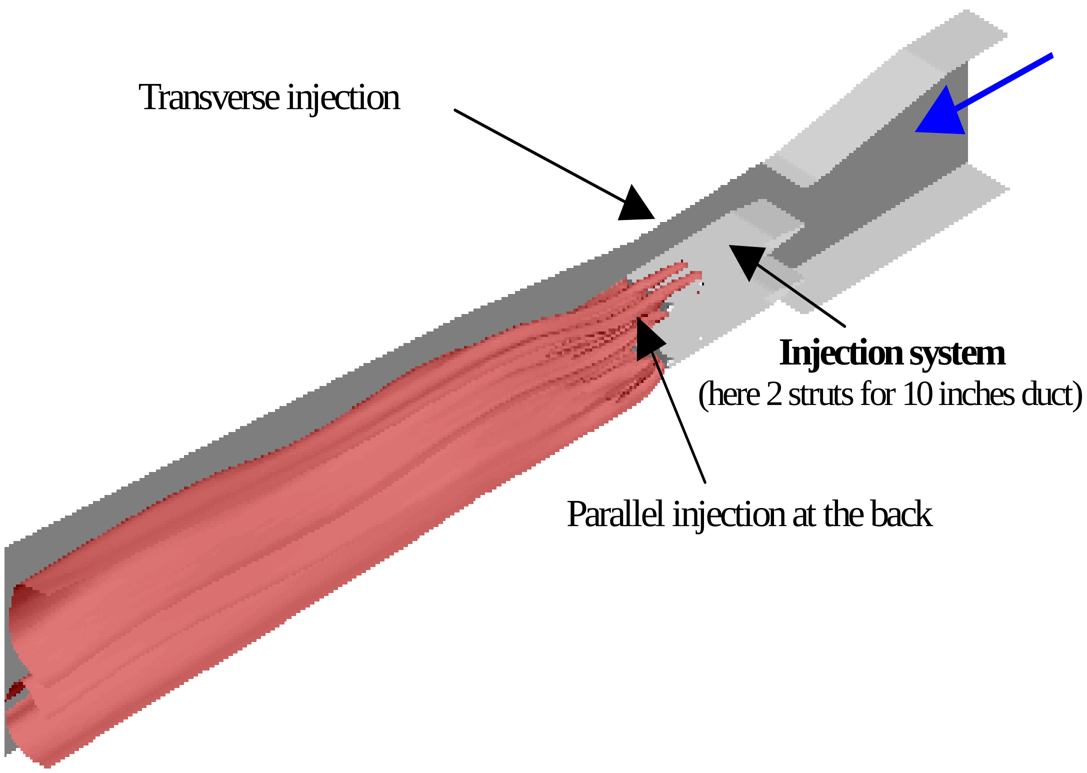

Figure 8: CFD analysis of CHAMOIS scramjet combustor.

In France, since 1993, CFD analysis has been systematically associated with test results. The MSD code and associated models have been demonstrating since 1994 the capacity of reproducing qualitatively the thermal blockage phenomenon due to excessive heat release resulting from insufficient distributed heat release. But they still have difficulties:

- To predict on a sufficiently extended range the hydrocarbon/air kinetic effect (ignition, …)
- To predict the facility effect in case of water or CO_2 vitiation
- To quantitatively predict the thermal blockage ER of a given scramjet combustor
- To quickly perform accurate 3D nose-to-tail flow computations.
- To give quantitative information on hot spots and local heat transfer locations
- To compute a real size, actual, regeneratively cooled scramjet combustor, in case of hydrocarbon fuel (catalytic or thermal decomposition, …) and/or composite materials (non-isotropic and porous).

Two test facilities are used in JAPHAR program for checking the diffusion/mixing/combustion process with all available optical diagnostic systems: the M11 test bench of DLR at Lampoldshausen (test section: 40x50mm², Mach 2 nozzle) and the LAERTE test bench, developed during the PREPHA Program (test section 45x45mm², Mach 2 nozzle). The so elaborated database is used to validate the physical models, issued from PREPHA program and which have been implemented in the MSD code and used for scramjet design and test analysis.
Several LES or DNS codes and basic computations have been performed in France (supersonic mixing or reactive layer, interaction with an incoming shock, turbulence enhancement, …). But, up to now, the results have not been directly used or applied to the validation of CFD modeling.

Because of the time needed to improve the CFD tools and because of the speed of computer capabilities enhancement, it seems that the main effort has to be put on modeling (turbulence, combustion, boundary layer transition and separation, coupling, …).

4 Scramjet flowpath

4.1 Forebody and Inlet

During the PREPHA program, a generic forebody was tested in the ONERA S4MA wind tunnel with wall pressure and thermal flux measurements, utilization of total and static pressure rakes in the plane of the inlet entrance and infrared thermography. Some specific CFD parametric studies have been performed in addition during JAPHAR studies [A8].

Significant inlet issues for hypersonic airbreathing propulsion lie primarily in the areas of configuration and inlet unstart. The design of 2-D, axisymetric, and 3-D inlets for a multitude of applications from auxiliary systems to primary propulsion systems (turbojet, ramjet, scramjet) can be accomplished. Usually these designs accept some pre-idea of the capture shape and overall requirements (mass flow, contraction ratio, pressure rise, total pressure recovery, etc). Computer codes are sometimes used to generate the grids for complex three-dimensional shapes, although 2-D and sidewall compression configurations do not require this step. Shock diagrams are laid out, followed by boundary layer thickness corrections and then applications of variable geometry to meet the Mach number range.

After completing the preliminary design, the inlet is evaluated experimentally. Full Navier-Stokes computer codes are then used, first to be validated by comparison with the experimental data and then to evaluate inlet performance and other parameters that are not easily obtained by the experimental program. Often models are not tested (or cannot be) over the Mach number range and data are extrapolated with the aid of analytical tools. Scale is also extrapolated as small models are expanded to flight vehicles.

Knowledge of inlet starting is important for all inlets, but is especially valuable for fixed geometry designs. Usually the surest way of obtaining inlet-starting information is through an experimental program. Because inlet contraction ratios are usually increased to the limit to obtain maximum performance, the “startability” of an inlet is typically very hard to predict with accuracy. This is understandable considering the possibility of having two or more real experimental flow solutions. Small variations in the tunnel flow, incoming boundary layer characteristics, and even model wall temperature can affect starting. Inlets can also pulse-start in the wind tunnel, and this must be compensated by providing a method of unstarting and then restarting the inlet during test. Small, relatively inexpensive models are usually used to get this information, but extrapolating the data to larger models and to flight scale is not assuredly accurate. Larger inlet models may be found to start easier in a large test facility, but an even larger flight article may not start again if it has to swallow a thick vehicle boundary layer that was not be simulated in the wind tunnel test.

PREPHA considered a fixed combustor geometry to limit the technological difficulties. Then, a propulsive stream tube geometry was adapted thanks to a variable capture area inlet which gave as much as possible an adapted geometrical contraction ratio variation as a function of the Mach number. This kind of inlet has been studied in France for several years [C8]. The basic concept [C9] has been adapted to the needs and constraints of the SSTO generic vehicle. A modular model was tested at Mach 5 and 7 in the ONERA R2Ch wind tunnel with a particular device helping the inlet start. This allowed a geometrical contraction ratio superior to 4 in spite of high deviation angles limiting the length, and then the mass, of an operational inlet. A new model definition has been realised for a second experiment at Mach 2 to 5.5 (ONERA S3MA wind-tunnel) aiming at defining maximum contraction ratio, effects of forebody boundary layer thickness, and maximum deviation angle on the compression ramps.

During JAPHAR program, two types of 2D inlets were studied: a mixed, external/internal, compression inlet studied at DLR with testing in the H2K and TMK wind-tunnels, and an internal compression inlet, designed from the PREPHA program, tested by ONERA in the S3MA wind-tunnel from Mach 3.5 to Mach 5.5 [A8].

4.2 Fuels

Candidate scramjet fuels include hydrogen, hydrocarbons, pyrophorics, and exotic high-energy-density fuels. Hydrogen ignition and combustion will occur in moderately-heated air under very lean conditions, and is rapid enough that scramjet combustion is possible over a reasonable length. Furthermore, because H_2 ignition and combustion can be sustained at strain rates 10 to 30 times higher than in flames using gaseous light-hydrocarbons (HCs) at typical temperatures, hydrogen is the necessary/preferred fuel for airbreathing scramjets based on reactivity alone. Also, liquid H_2 is very effective for active cooling of vehicle structures, which is required at high speed. Unfortunately, liquid (or slush) H_2 is difficult to store and handle on a routine basis, and it has three to four times a lower energy density than typical storable hydrocarbons.

Although HCs heated during active cooling will become more reactive, such increases are limited without significant decomposition. So-called storable endothermic fuels may be catalytically hydroformed, dehydrogenated and/or cracked in-situ, so that additional heat is absorbed and resultant fuel fragments (including H2 and CO) become more reactive. However, such heterogeneous catalytic processes are difficult to accomplish, reproduce, and control without forming significant carbon deposits on catalysts and within fuel passages and injectors. Pyrophorics (e.g. 20 mole percent silane in H2) ignite spontaneously and burn when injected into air, and thus make good ignition and piloting aids. However they are not endothermic, and they usually carry molecular weight penalties, are toxic, and produce troublesome condensed phase products (e.g. silica). Finally, fuel chemists working over the last 40 years have devised a number of so-called exotic high-energy-density fuels (e.g. cubane, various strained-ring compounds, polymeric BxNyHz, and liquid H2 gelled with light HCs), and/or organic additives (e.g. nitrates, nitrites, nitro compounds, ethers and peroxides) with improved reactivity and energy release. Typical problems with these materials are stability, safe storage and handling under field conditions, toxicity, and increased cost; however, these problems are not necessarily insurmountable.

The use of gaseous or liquid hydrogen is currently planned in France, Germany, Italy, and the USA. In the US, the HyTech program has cracked the endothermic barrier by catalytic regenerative cooling. This allows operation to 1300F using JP7 fuel with little if any cooling. In France, for supersonic missiles without active cooling, new very high density fuels have been formulated and flight-tested. Production, ageing, storage, regulation, injection and combustion are demonstrated thanks to a specific advanced development [C4]. Some preliminary work has been done on the use of liquid hydrocarbon for regeneratively-cooled higher speed engines, within the scope of the PROMETHEE program [D10], [D11] or during PTAH-SOCAR cooled structures experimental evaluation [D12]. This work is currently under development in France.

4.3 Combustor

In the past, the design of a combustor flowpath utilized an experimental procedure consisting mostly of trials and errors. With direct-connect tests, a supersonic nozzle is attached to a facility heater with the nozzle exit flow conditions simulating the combustor entrance conditions for a ramjet or scramjet. A combustor duct, containing fuel injectors, is attached to the supersonic nozzle and the area variation of the combustor flow path is altered (experimentally) to achieve desired pressure and reacted fuel distributions. With freejet engine tests (or, semidirect-connect tests), a ramjet or scramjet engine, typically with a truncated forebody and a truncated aftbody/nozzle, is placed within a facility test cabin, and tests are then conducted during which the engine geometry is varied such that the desired performance is achieved. More recently, engines (or test articles) have been constructed with high contraction ratio inlets which compress the freestream flow to higher levels than have been attempted in the past. Pre-test calculations are typically conducted where a CFD solution of the inlet yields the flow properties at the throat. A simple chemical equilibrium quasi-one-dimensional calculation is then conducted to indicate how much fuel injection and combustion could be achieved within the combustor before the flow becomes choked. These relatively simple calculations alert the researcher of any possible performance problems to be expected before construction or testing of the engine occurs.

The design of the combustor flowpath must also include choosing the location and type of fuel injectors. Various fuel injection mixing "recipes" are available to help the engineer with this task [U11, U16]. The "Langley Mixing Recipe" was developed during the early 70's as a way to correlate fuel injection mixing efficiency with downstream distance for scramjets operating in the mid-speed range of Mach 4 to 8. More recently, computational methods have been used to study and optimize fuel injector components and to assess and optimize fuel injectors installed in the engine flowpath [U12]. Combining computational methods with earlier engineering design techniques has been found to offer the best strategy for scramjet combustor design.

Within the scope of PREPHA, an experimental combustor, named CHAMOIS, has been developed by MBDA-F. In spite of its limited dimensions (entrance area of 212 x 212 mm^2), this combustor presents as much as possible the same difficulties as a large operational combustor such as fuel injection by struts, wall/injection strut interaction, strut/strut interaction, upstream flow non uniformity (boundary layer and shock waves). One-, two- and three-dimensional numerical studies have allowed the definition of its combustor geometry. Then, several CHAMOIS test series (1994, 1995, 1996, and 1997) have been successfully performed. The tests have been done in connected pipe mode, with uniform or heterogeneous incoming airflow, in the MBDA-F Bourges-Subdray test facility under Mach 6 conditions ([C12] and [C13]). A liquid-kerosene-fueled CHAMOIS combustor has been tested in the same facility in 1997 and the flow has been computed [D2].

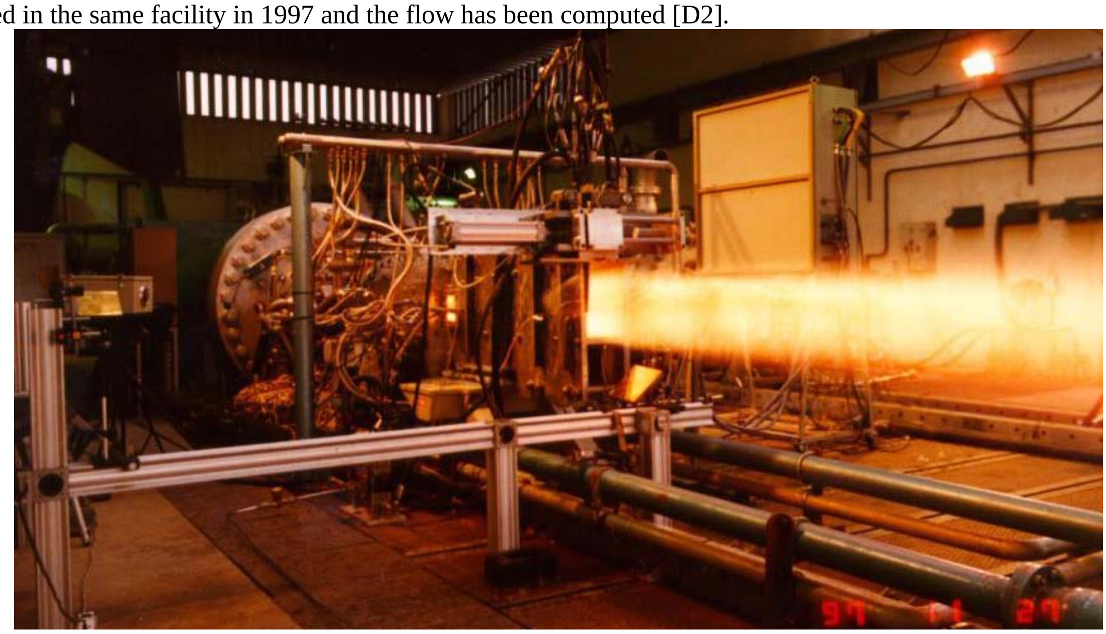

Figure 9: CHAMOIS scramjet testing in Bourges-Subdray.

During PREPHA, in order to obtain some data at Mach 7.5 flight conditions and to observe the water vitiation effects, a new small combustor (100 x 100 mm^2 at the entrance) has been developed for a complementary test in ATD 5 facility at Mach 7.5 conditions with vitiated air and at Mach 6 conditions with more or less vitiated air thanks to the heat exchanger supplying the test facility (1000 K pure air). Moreover, this small combustor, called MONOMAT, has been used to analyse the transonic combustion mode in order to confirm the feasibility of the thermal choked dual mode ramjet [D13].

The JAPHAR dual mode ramjet powering the vehicle exceeds capacity of the ATD5 test facility at ONERA Palaiseau Test Center, which provides Mach 7.5 flight conditions but for a limited air mass flow of 4 kg/s (water vitiated air). Considering that it is not possible to design a subscale model of the engine by following a scientific methodology, it has been decided to develop a smaller model, based on the same concept, but not homothetic, and to validate the whole design methodology on this concept. The vehicle engine and its integration to the airframe are being studied only by numerical simulation. Then, a stainless steel heat-sink model (entrance cross section of 100x100mm²) has been designed and manufactured by ONERA. It is equipped with only one strut at the upstream injection level and two struts at the downstream injection level. Today, ONERA is performing a direct connected pipe test. The combustion chamber has been tested at Mach 4.9, 6.3 and 7.4 conditions. It has been possible to obtain subsonic combustion with a stable thermal throat at the end of the chamber. At higher Mach number, supersonic combustion was sustained.

With partial support of DGA, MBDA-F and the Moscow Aviation Institute (MAI) are developing a large scale prototype of a dual mode dual fuel ramjet with a fully variable geometry combustion chamber (Ref [B25]). This engine, called Wide Range Ramjet (WRR), has the challenging characteristics such as operation from at least Mach 3 up to Mach 12, use of movable panels during operation along the trajectory, modification of the internal geometry by a control-command computer connected with sensors on the engine in order to maximize the performance in real time, use of subsonic and then supersonic combustion, use of kerosene and then hydrogen as fuels, and a large scale engine (entrance area of $0.05m^2$, several meters length). This engine is under final manufacturing and is to be tested at Bourges-Subdray when the corresponding funding will be available. Then the cold structural framework of the WRR Prototype has been manufactured and major components have been developed and tested including kerosene with mixed bubbles of hydrogen, an ignition device, and a 3D-shape injection strut [A10]. More than half of the necessary cooled panels called Heat Protective Elements have been realized. Control code has been written, tested and validated with simulation of prototype operation (waiting for the test), taking into account the transient actual behaviour of each actuator. The PROMETHEE combustor mock-up (212 mm width, scale 1 in height, stainless steel heat sink) has been designed at MBDA-F and will be tested in ONERA ATD test cells in 2002.

4.4 Nozzle

Design of the scramjet internal nozzle and the external nozzle is performed in concert with the combustor design activity using a similar design strategy as described earlier. The nozzle flowfield is characterized by much of the flow physics of the combustor, but there are additional requirements including high velocities and high initial temperatures, significant divergence and skin friction losses, potential relaminarization of the flow, energy-bound chemical radicals that will not relax in a finite nozzle length, and excited vibrational states. The favorable pressure gradient in the nozzle eliminates concerns with shock-boundary layer interaction and separation. Nozzle design still utilizes facility testing, but significant success had been achieved with computational modeling and design using programs ranging from Euler through full Navier-Stokes codes.

PREPHA gave to France several results for nozzle and aft-body design for scramjet-powered vehicles. Basic research has led to a better knowledge of the evolution of the boundary layer along the expansion ramp and the interaction between the jet and the external boundary layer at the exit of a Single Expansion Ramp Nozzle (SERN). In the field of concepts definition, a numerical approach has been used to determine the influence of different parameters including length of the cowl, movable or fixed flap, and the expansion ramp profile [C10]. In order to allow a general evaluation of the FLU3M code in the case of the nozzle and afterbody, a generic model with an internal hydrogen burner has been tested in S4MA wind tunnel in Modane [C4].

4.5 Integration with other modes

4.5.1 Dual-mode scramjet (subsonic then supersonic combustion in the same engine)

Dual-mode ramjet design operational limits are generally set by vehicle architecture (SSTO/TSTO, etc.) and engine cycle selection. For SSTO RBCC vehicles, the dual mode scramjet generally is designed to operate from Mach 3 to 12 - 15. It must include variable geometry for control of contraction ratio and combustor area-length. For an over/under TBCC, the scramjet will be expected to operate from Mach 3.5 – 4.2 to Mach 15. The higher scramjet “takeover” speed is based on high-speed turbine-based engines, which remain more efficient than scramjets to a higher Mach number, as US studies concluded. The USA’s dual-mode scramjet design is essentially that discussed in the next paragraph.

During the PREPHA program, different airbreathing propulsion systems, with a fixed geometry duct, were considered and their comparison led to selection of a dual mode ramjet concept (subsonic combustion up to Mach ~6 flight conditions then supersonic combustion) with a first quasi-constant cross section combustion chamber, used for supersonic combustion (first injection level), placed upstream of a diverging one, which is used for subsonic combustion with thermal throttling (second injection level). The fixed-geometry dual-mode scramjet of JAPHAR is based on the same concept as the double combustion chamber dual mode ramjet, selected during the PREPHA Program and also studied during the Radiance project. The WRR Prototype used a movable geometry (during the test) and a geometrical throat. Its test (planned in 2002) will give extensive time-dependant information on ram to scram transition by controlled contour modification.

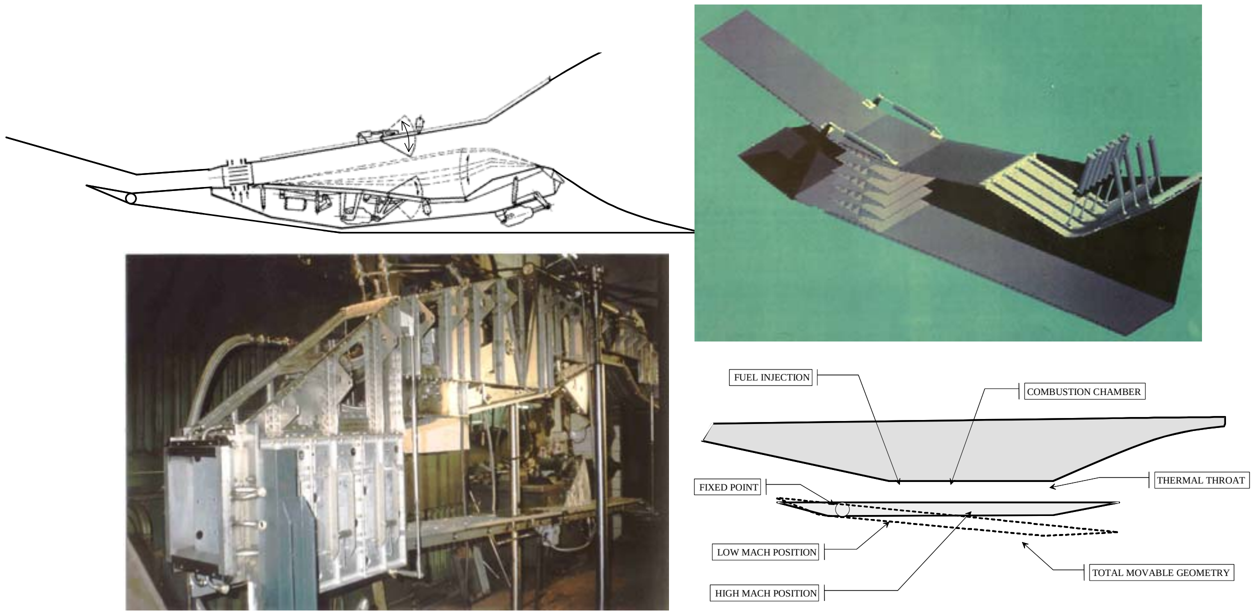

Figure 10: Examples of variable geometry dual-mode ramjets.

For the PROMETHEE missile, the selected engine is a variable geometry dual-mode ramjet operating from Mach 1.8 to Mach 8. The geometry variation is achieved by the cowl wall rotation around an axis placed upstream the minimum cross-section. At low flight Mach number, the combustion is subsonic thanks to thermal choking.

4.5.2 Rocket integrated in the airbreathing duct

For missile applications, due to the 2D cross section of the scramjet combustor, solid propellant boosters (required to accelerate the vehicle from its launch up to airbreathing engine start) are not integrated but are external and jetissonable. For single stage to orbit (SSTO) access to space (ATS) missions, a rocket is required for the final stage of boost, orbital insertion, orbital maneuvering and de-orbit. A candidate for a SSTO vehicle is the rocket based combined cycle (RBCC) engine. RBCC engines utilize an air-augmented rocket for low speed, a ramjet/scramjet for mid-speed, and a rocket for high-speed operation. The NASA Marshall Space Flight Center is leading a program to develop an RBCC propulsion system to demonstrate the technology for future launch systems. A RBCC propulsion system may be tested in a program (X-43B) that is follow-on to the Hyper-X program later in this decade.

Only paper studies were realised in Western Europe on this topic. In the scope of the French PREPHA program, the study of a generic SSTO vehicle led to conclude that the best type of air-breathing engine could be the dual-mode-ramjet (subsonic then supersonic combustion) [A1] [B2] with separate rockets for take-off and final acceleration. An extensive technology work has been investigated [B4] [B5] [B6] [B7] [B8] [B13].

4.5.3 Detonation-based cycles (ODWE and PDE)

Work was begun in the early 1990’s at NASA Langley to study the feasibility of both Oblique Detonation Wave Engines (ODWE) and shock induced combustion engines. The premixed shock induced combustion (PMSIC) concept utilizes a strong shock to initiate combustion of a fuel-air premixture at the entrance to the engine combustor with the idea of significantly shortening the combustor of a scramjet. Computational studies were completed, and model design was begun, but the program did not continue to the stage of testing [D16]. There was also work during this period to computationally study oblique wave detonation engines [U13] and pulse detonation engines.

The WRR concept is –theoretically- designed to be used in ODWE mode instead of conventional scramjet after Mach 10. Several 1D and 2D computations have only been performed. ODWE basic studies have been experimentally demonstrated at ENSMA/CNRS laboratory at Poitiers, under Mach 10 conditions, with hydrogen as fuel.

There is currently an effort still underway in USA and in France to explore pulsed detonation engine (PDE) concepts. Some preliminary integration of PDE or PDR (rocket) concepts into a scramjet have been investigated [U17]. One of the common scientific challenges of the PDE and ODWE is the theory and the mastering of detonation of non-perfectly mixed gases. Recent studies have shown that realistic, attainable performance limits the useful lower and upper Mach number limit of the PDE. Therefore, a careful realistic review of this engine will continue as it is being developed, to determine if it is useful for hypersonic systems.

4.5.4 Association with turbo-ducts

NASA Langley and Air Force studies show that scramjets integrated with a high speed turbojet, in an "over-under" configuration, may provide optimum vehicles in terms of take-off gross weight, reliability, operating cost and safety. Turbojets have high reliability, vis-a-vis rockets. Integration of the two engine flowpaths is currently in the conceptual stage. Previous design, dating to the late 60's, has included wind tunnel tests of propulsion-airframe integration issues for these over-under engine systems.

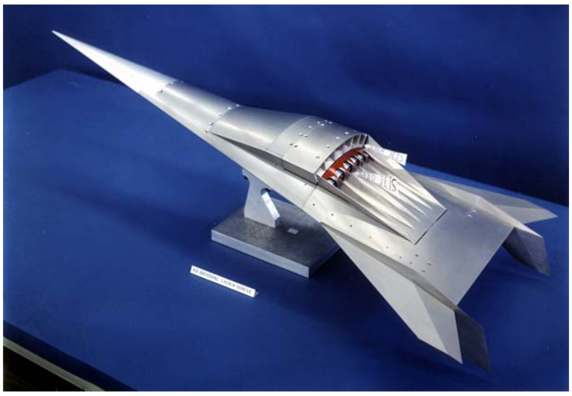

Figure 11: "Over-under" turbo-ramjet**.**

4.5.5 Cooled-air secondary duct

Liquid-air systems have been studied in the USA, Europe and Japan. US studies have shown that liquid air rocket based combined cycle engines are competitive with turbojet-scramjet systems for space access [U16]. Several paper studies have been performed in Europe on this topic, in particular during ESA/CEPS studies (non-liquefied air, see [B12]) and PREPHA (air collection during cruise, see [B2]).

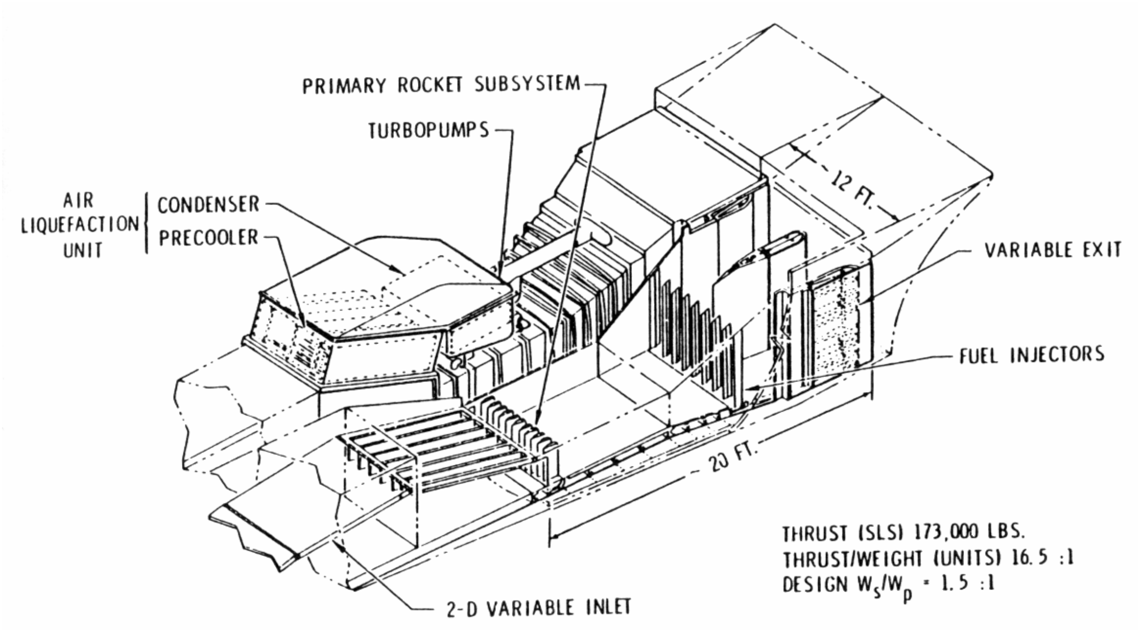

Figure 12: ScramLACE system.

4.6 Airframe integration

Hypersonic airbreathing configurations are characterized by a highly integrated propulsion flowpath and airframe systems [U13]. Propulsion/Airframe Integration (PAI) research for this class of vehicle is focused on understanding various component interactions and their effects on integrated vehicle aero-propulsive performance. Advanced airframe-integrated concepts seek to exploit these interactions to maximize performance and improve stability and control characteristics. Investigations of these phenomena require a range of analytical, computational and experimental methods [U15].

Airframe-propulsion integration has been studied extensively in France and Germany, but only using computations of different levels [A1] [B2] [D11] [B12] [B19] [B25]. Testing demonstration and associated methodology has been prepared, but no specific experimental work has yet been conducted in Western Europe.

Much of the US present capabilities and experience in this area is derived from support of various NASA hypersonic programs, such as the National Aerospace Plane (NASP) and the Hyper-X (X-43A) Program. A survey of work from these programs represents the state of the art in this research area. The development of the X-43A configurations provides an opportunity to evaluate testing and analytical capabilities and highlight some areas of opportunity for improvements in methodology leading to the development of a full-scale scramjet-powered flight vehicle. Among the advancements in PAI, experimental wind tunnel testing of a complete scramjet powered vehicle configuration was accomplished for the first time, and powered aerodynamics was validated at Mach 7.

5 Materials and structures

Structural concepts for hypersonic vehicles and propulsion systems evolve with the design of the overall system. Many of the current concepts use either cold integral or non-integral graphite/epoxy LH2 tanks (developed and successfully tested under the X-30 and used in the DC-X and X-33). A mechanically attached insulated multi-wall insulation (IMI) thermal protection system (TPS) is used on the windward side, and the now obsolete tailored advanced blanket insulation (TABI) is baselined for the lee side. This combination provides a lightweight TPS with durable external skin. Wing and tail structure is titanium metal matrix composite, developed for the X-30. The engine primary structure is graphite-polyimide (being demonstrated on the X-37). Regeneratively cooled copper, aluminum, and high temperature superalloy panels are utilized in the engine and a convectively cooled process developed and verified for the X-30 cooled engine and vehicle sharp leading edges.

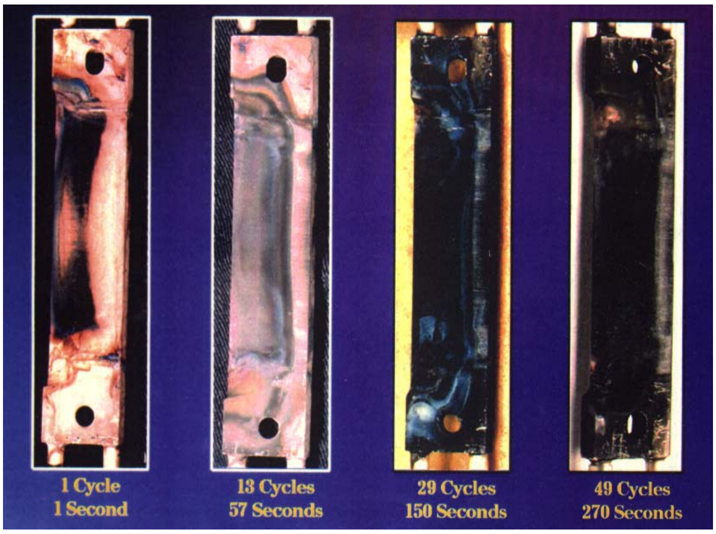

Figure 13: Test of engine heat exchangers and leading edges (Ref. U17).

The WRR project gave and will give the opportunity to acquire a strong know-how for the design and the experimental validation of the active cooling systems usable in a dual mode ramjet for the different components including injection struts, combustor walls, movable panels and hinges [A11]. In the framework of the WRR program 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, called HPEs (Heat Protective Elements), are 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 both 3D configurations, in particular multi-layer architectures, such as the “two stages” HPE (material FeCrAl), and heat exchange enhancement systems in the cooling channels. Each of these HPE concepts has been tested in the MAI facility, in which a hydrogen fueled scramjet combustion chamber is used as a high temperature gas generator. The tested HPE (100x200 mm^2) is placed at the scramjet chamber exit. A wedge is facing the tested HPE in order to create a shock wave whose interaction with the HPE increases the heat flux. For both metallic and composite materials versions, the future developments of cooled panels, or cooled integral structure, will take advantage of the large database collected during the WRR cooperation. Reference [D14] gives a synthesis of this know-how.

Figure 14: Hot test of WRR component (scramjet leading edge).

Recent French studies lead to considering combustion chamber technologies based on the use of thermo-structural composite materials cooled by the fuel. In this field, very limited works have been performed by EADS and SNECMA during the PREPHA program with basic tests performed at ONERA [A1]. Under the aegis of DGA and the USAF, ONERA and SNECMA are working with Pratt & Whitney in the A3CP program for endothermic fuel cooled composite materials structures ([A17]). The developed technology consists of manufacturing channels in a composite material sheet (C/SiC) and brazing a second composite material sheet to constitute a cooled panel. The program is dealing with the different difficulties related to this kind of cooled panel including composite materials brazing technology able to sustain high temperature (> 1000 K), compatibility between material and endothermic fuel and possible catalyst, and fuel leakage through composite material porosity. A first hot test of A3C panel is planned in 2002.

Between 1993 and 1996, MBDA-F and EADS-Launch Vehicles (EADS-LV) led the project St ELME (French acronym for Advanced Injection System Through High Mach Number Flow). This project consisted in designing, manufacturing and testing in the CHAMOIS scramjet a high performance scramjet injection strut [A18].

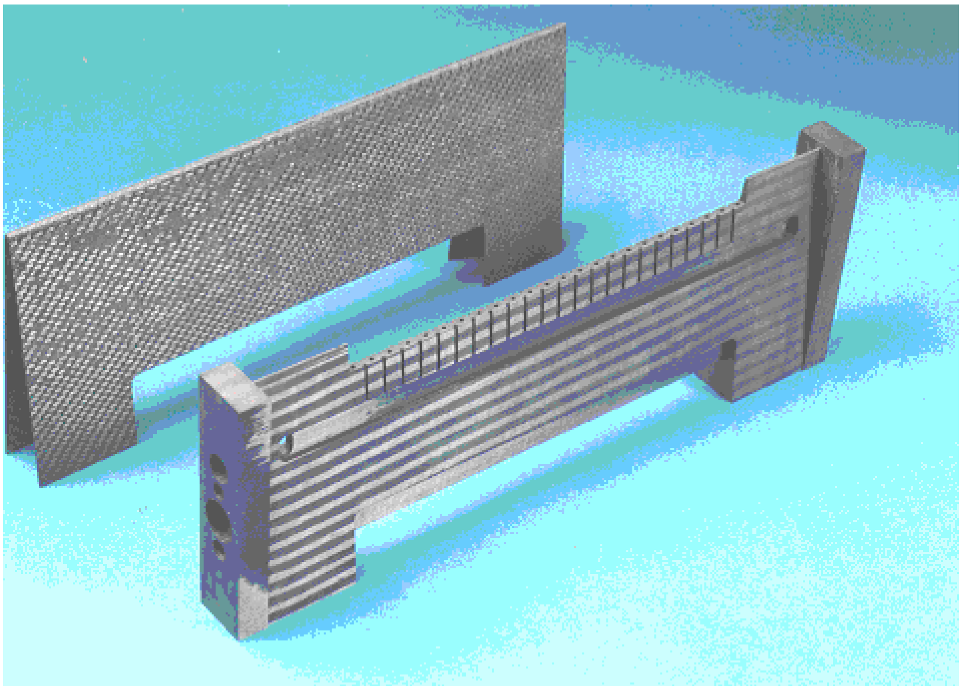

Figure 15: SAINT-ELME injection strut in thermostructural composite.

Today, MBDA-F and EADS-LV are focusing their in-house effort on the development of a low cost, highly reliable and effective technology for the fuel-cooled composite material structure, particularly usable for the walls of a ramjet/scramjet combustion chamber. This technology, called PTAH-SOCAR (French acronym for Weaved Wall Applied to Hypersonic – Simple Operational Composite for Advanced Ramjet), takes advantage of the EADS-LV know-how in the field of pre-form manufacturing and particularly of its mastery for weaving the fibrous structure [D12]. Three different composite panels have been successfully tested since July 2001 with gaseous nitrogen and liquid kerosene as coolant and with a maximum wall temperature of 1850K. In connection with flowpath design and engine integration, long time testing of flight worthy scramjet engines have been deeply investigated in the US (see §6.3, [A17], [B16]).

6 Flight testing

6.1 Hyper-X

The primary goals of the Hyper-X Program are to validate the airframe-integrated, dual-mode, scramjet-powered vehicle in flight and provide databases for validation of design methods and tools [U14]. This will be accomplished using data from the X-43-A vehicle under powered conditions at Mach 7 and 10, and unpowered conditions down to subsonic flight. In preparation for these X-43 flights, refinement of the vehicle design using optimization methods was required to assure that the small, compact X-43 vehicle accelerates. In addition, every detail of the hypersonic system was evaluated, including the high Mach number, high dynamic pressure stage separation. The most extensive hypersonic aerodynamic, propulsion and thermal database ever generated for this class of vehicle is being used to develop autonomous flight controls, size TPS and reduce risk for this first ever scramjet-powered hypersonic flight. [B14]

The X-43A mission of June 2001, the first in a series of three, was lost moments after the X-43A and its launch vehicle were released from the wing of the NASA B-52 carrier aircraft. Following launch vehicle ignition, the combined launch vehicle and X-43A experienced structural failure, deviated from its flight path and was deliberately terminated. The board studying the June 2 loss of the first X-43A mission expects to find more than one factor responsible for the loss. After complete analysis of the failure, a new flight test will be planned.

Figure 16: The Hyper-X vehicle integration.

6.2 CIAM –"Kholod" Scramjet Flight Tests

This axisymetric, dual-mode scramjet had been flight tested by Russia, first with internal funding (1991), then in 1993 and 1995, in cooperation with France [C14], with participation of three specialists taking part in the present RTO subgroup. Then the last tests were performed within the scope of a CIAM-NASA cooperation. This test series provided ground and flight data at similar freestream conditions, showing similar results. This test also provided insight into autonomous flight controls.

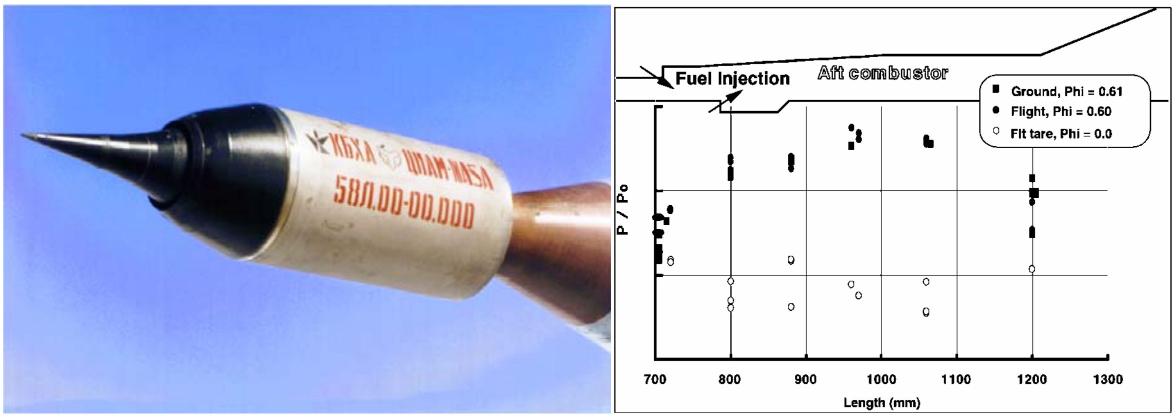

Figure 17: Kholod Russian experimental system.

6.3 ASTP - Hypersonic Development for Space Access

The future development of scramjet and hypersonic technology within the USA falls under the NASA Advanced Space Transportation Program and yet to be defined DOD interests. A complete plan will be completed in 2002.

The ASTP program is a comprehensive program designed to complete technology development and demonstration by 2018, leading to a Space Shuttle replacement vehicle by 2025. Propulsion systems generally fall into two categories: rocket-based and turbine-based combined cycles. Both approaches use dual-mode scramjets over much of the flight envelope, from Mach 3 or 4 to Mach 12 to 15. The program is also developing the critical technologies identified by the system studies. These range from structures and materials, to tires to operational and Integrated Vehicle Health Monitoring (IVHM).

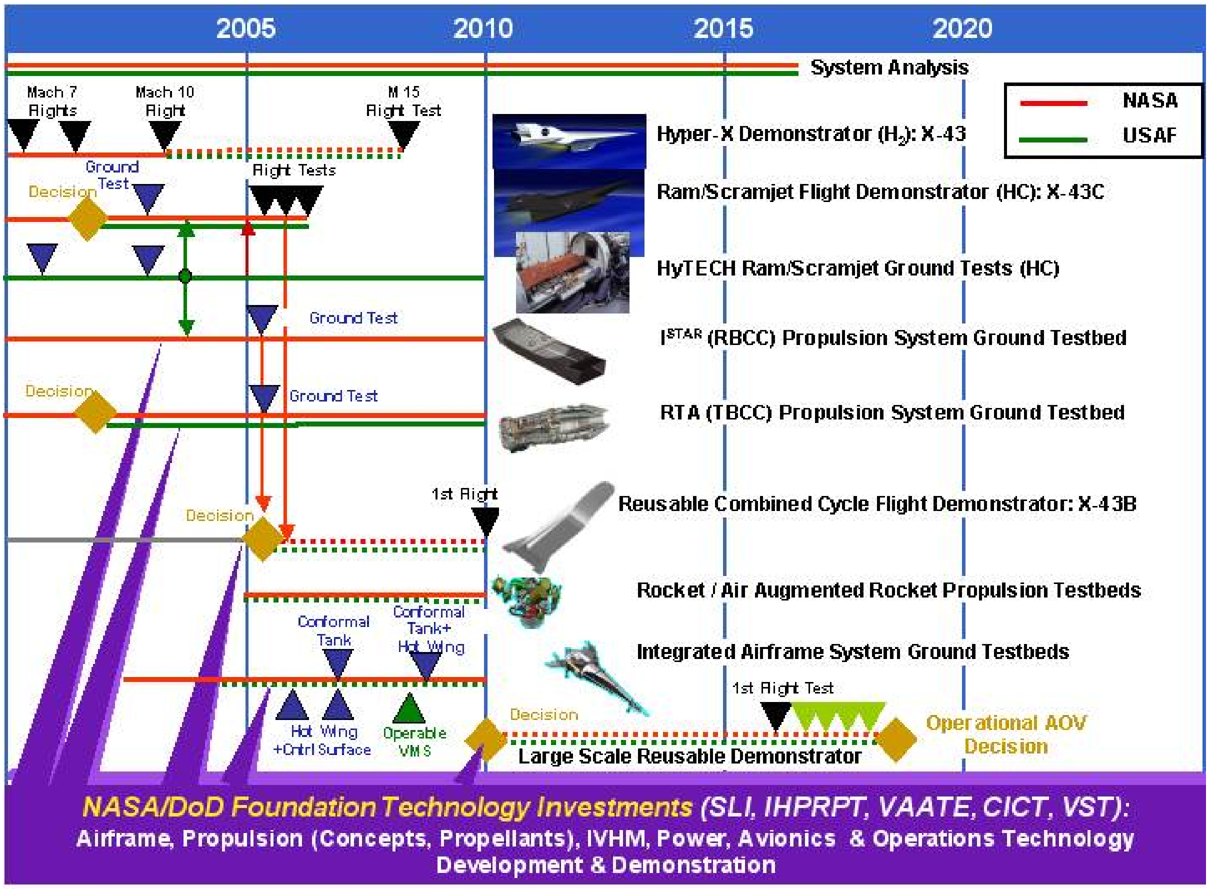

Figure 18: The Advanced Space Transportation Roadmap.

In addition, the program features both ground and flight demonstrations of key technologies, such as scramjet, RBCC and TBCC propulsion systems and major airframe structures. In the current plan, and in the conceptual designs, plans are being developed for scramjet flight demonstrators, ground testing RBCC and TBCC demonstrators, and a flight demonstration of either the RBCC or TBCC configuration. The current program is also calling for a large-scale flight vehicle, which would demonstrate all propulsion and other system technology. This phased, incremental program is designed to focus technology development, take to flight only systems which have heritage in ground/wind tunnel development, delay vehicle architecture selection until flight data is obtained, and meet the NASA and USAF 2025 IOC.

The first flight demonstrator is currently called the X-43C. It is a slightly scaled up version of the X-43 with 3 HyTech hydrocarbon engines, which were developed for the AARMD program. These flightweight, hydrocarbon-fueled and -cooled engines were developed and will be supplied by the USAF. NASA will provide the vehicle, integrated to fit the existing engines, and fly three vehicles using the same approach used by the Hyper-X Program, i.e. rocket-boosted, and not recovered. These vehicles will be boosted to Mach 5, demonstrate acceleration from Mach 5-7, duel-mode ramjet mode transition from ramjet to scramjet mode, and engine performance, operability and durability. Test times will be on the order of 4-6 min. This small step will produce the first regenerative cooled, flightweight scramjet powered vehicle, and represents an affordable, incremental step up from Hyper-X. (The Hyper-X team, with help from the USAF and other NASA centers will execute this program, with first flight scheduled for 2005). This will also be the first flight weight scramjet engine system built in the USA since the NASA Hypersonic Research Engine, which completed wind tunnel testing in 1972.

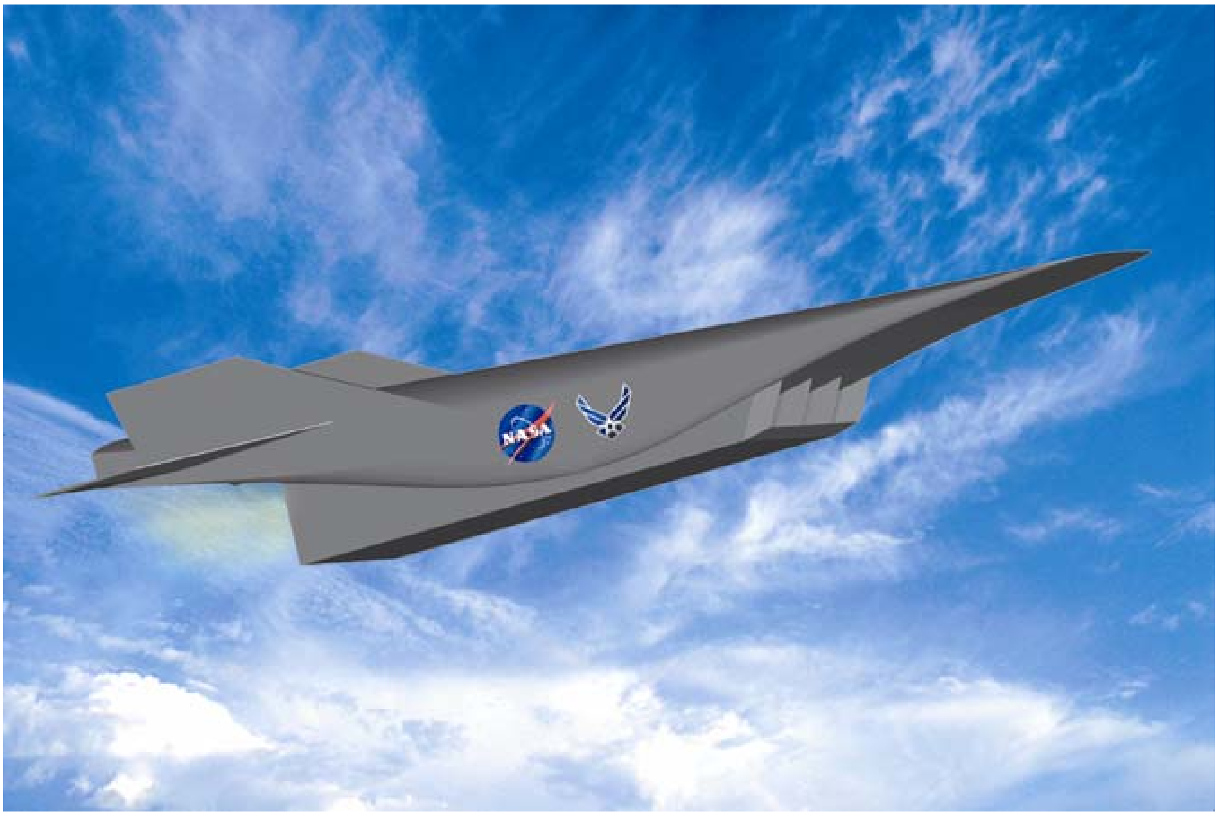

Figure 19: The X-43C Vehicle.

Following ground development and testing of both a flightweight RBCC and TBCC, and completion of a conceptual design for flight testing, one engine system will be selected for flight-testing. As currently envisioned, this engine system will be tested using an air launched version of the X-43 lifting body configuration called the X-43B. The vehicle will be dropped from the NASA B-52, will then accelerate to Mach 7, and glide to a dead stick landing, much like the X-15. Current studies indicate that by using hydrocarbon fuel to reduce vehicle size, this vehicle should be between 13 - 15 meters long, and weigh about 25,000 pounds at drop. Two vehicles are currently envisioned. Each will be required to fly 25 missions without major engine replacements/repairs. This reusable flight vehicle will provide the first real data on operation/costs for this class of vehicles. Depending on budgets, these vehicles can be flying by 2008-12. This vehicle is likely to be a jointly funded DOD-NASA program with the first flight in 2008.

Other ground demonstrators are being considered in the US for large airframe structural elements.

Following completion of the X-43C, the ASTP will evaluate system studies and requirements for "Hypervelocity" scramjet operations. A hypervelocity scramjet demonstrator is being investigated. This would utilise LH2, be rocket boosted, and validate LH2 fueled/cooled scramjet engine operation in the Mach 12-15 speed range. Because of budget considerations, this vehicle is envisioned as Hyper-X scale.

The final demonstrator leading to an operational vehicle will fold together all of the available technology, will be based on the selected vision vehicle, and demonstrate all propulsion modes and other key technologies. Engine cycle(s), vehicle architecture, number of stages, and fuel will be based on a down select from the system studies. Clearly this vehicle demonstrator will be a large undertaking. But, it will have significantly lower risk than the X-30 or X-33 because of the incremental approach that will be utilized. This demonstrator is scheduled for ASTP in 2010 - 2012, and first flight 2015-2017.

6.4 Europe flight testing issues (probably with Russia)

Considering difficulties and cost of test facilities on one hand, the extreme sensitivity of the aero-propulsive balance on the other hand, it is clear that scramjet technology development needs a large phase of flight experimentation. A demonstrator of an operational vehicle being very expensive and the associated technical risk being very high, such a flight experiment should begin with the development of small experimental vehicles.

The limited French participation within the basic flight experiments, undertaken by CIAM from Moscow, of boosted “Kholod” axisymetric hydrogen-fueled engines was a first step [C14]. But, the design of the tested Kholod engines is very close to the HRE or ESOPE combustors design (tested on ground in USA and in France in 1970s) and the limited height of the combustion chamber is not representative of a large operational scramjet. Beyond this first step, an analysis of needs allowed to evaluate the capacity of a large set of typical experimental vehicles to comply with these requirements [C15].

Assuming obtained results, ONERA and MBDA-F sketched some self-powered experimental vehicles [C16].
For JAPHAR experimental vehicles, a height of 100mm has been chosen for the combustion chamber entrance to be representative of a space launcher application. A combustion chamber of this height requires the use of injection struts. Due to the chosen height of the combustion chamber, the generic experimental vehicle is a relatively large one (~10m long). Two 400mm wide propulsion system modules power it. On the base of a preliminary design, some design studies have been performed to optimise the general configuration of the experimental vehicle [A7] and particularly the forebody shape [A8].

All these efforts, prepared with discussion with the scientific community (ONERA, MBDA-F, CNRS, DLR, Russian institutes), would lead to a small scale experimental vehicle, able to demonstrate in flight, whatever the final applications, the capacity of developing a dual mode ramjet to propel a vehicle from Mach 2 to Mach 8.

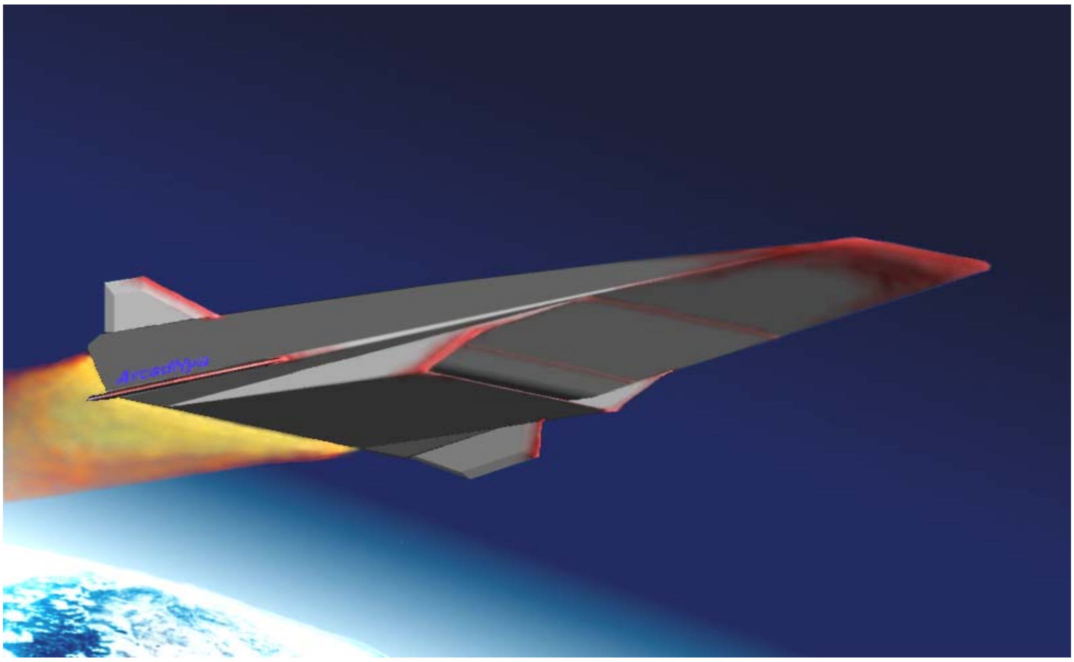

Figure 20: Artist view of airbreathing hypersonic experimental vehicle.

7 Conclusions

Significant advancements have been made in high-speed airbreathing propulsion, particularly within the USA and France. Many of these advancements have been discussed in this introductory chapter. These advancements are finally being exploited in the first years of this new century.

The hypersonic community will benefit from:

- System studies and mission analysis
- CFD enhancement
- Ground testing.

Two major issues will be:

- Ground testing of a flight-worthy regeneratively-cooled scramjet engine (planned in 2003 in the US, in 2008 in France)
- Flight testing of an autonomous vehicle with demonstrated understanding of the aero-propulsive balance (planned in 2002 in the US and 2008 in France).

Progress in high-speed airbreathing propulsion is critical for the NATO Alliance. Access-to-space requirements that must be addressed include reusable vehicles that provide rapid access to low earth orbit at significantly reduced cost and with increased reliability. There is also a need for development of hypersonic, airbreathing missiles to provide rapid response on target and to counter threats from other hypersonic weapons. Finally, hypersonic aircraft concepts must be considered that can rapidly reach any critical area on earth and provide a platform for reconnaissance and defence.

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