German Test Facilities for High Speed Air Breathing Propulsion

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Abstract

In combination with flight testing and computational fluid dynamics (CFD) schemes, ground based testing is a mandatory tool to be used in the design and development process of future air breathing propulsion systems. In fact it is the optimised combination of all three tools, which is required to advance the knowledge in this field and to develop the database necessary for the development of operational highly integrated vehicles. In the present paper, the operational German ground based test capabilities for high speed air breathing propulsion are summarised. This overview is restricted to test facilities, which allow testing including combustion, so that cold hypersonic tunnels, which are used for intake, nozzle and fuel off combustor testing, are not included. In the considered facilities, a broad variety of investigations ranging from fuel / oxidizer mixing, ignition, combustion and combustion instabilities up to the testing of integrated systems consisting of intake, combustor and nozzle are performed.

1. Introduction

The objectives of the space transport market to reduce the specific transport costs and to increase the reliability and flexibility of new transportation systems require that future space research programs develop partly reusable or fully reusable vehicles with improved propulsion systems. Air breathing propulsion concepts have the advantage over rocket engines that they use the atmospheric air as oxidizer, permitting an important decrease of on-board propellant mass. The dual mode scramjet is one option for a new and advanced type of engine to be operated at flight Mach numbers up to $M = 8$ and above. The efficiency gains of such a high speed air breathing propulsion system potentially allow improvements in size of payload, cost / kg and reliability of future hypersonic flight vehicles. A possible application could be a space launcher with combined air breathing and rocket propulsion system to improve the average installed specific impulse along the ascent trajectory or a hypersonic transport aircraft utilising e.g. a turbine or rocket based combined cycle. In Europe, research is conducted during more than 20 years to develop high speed air breathing propulsion technology (see e.g. [1], [2], [3], [4]). Although significant advances have been made in developing dual mode (sc)ramjet engines, major scientific and technological challenges remain. In order to develop a methodology for reliable predictions of scramjet powered vehicle performance, a strong link between ground based and flight testing and computational fluid dynamics (CFD) needs to be established.

A key technology in the framework of the development of such new transportation systems is the sustained flight at hypersonic speeds (in the order of eight times the speed of sound and more) through the atmosphere. Any future air breathing propulsion powered hypersonic vehicle design will depend on two key technologies. These are the performance of the propulsion system and the aero-propulsive balance (i.e. thrust-minus-drag balance) of the complete vehicle. Dedicated experimental investigations of the different components of air breathing propulsion units are necessary for the understanding of the physical and chemical processes in both on and off-design conditions. In the framework of the overview of ground based facilities given here, emphasis is put on facilities which allow testing including combustion. Cold hypersonic wind tunnels which are used for intake, nozzle and fuel off combustor testing are not included. Fundamental experimental investigations of fuel / oxidizer mixing and combustion processes in the combustion chamber of dual mode (sc)ramjets are performed in continuously running so called connected tube test facilities. To demonstrate a positive aero-propulsive balance, one significant issue of future flight tests is the measurement of thrust and the correlation of this data with measurements made in ground based facilities and CFD predictions. The positive net thrust or aero-propulsive balance must be achieved in the whole Mach number range of engine operation. The increase of the viscous drag in a combustor with increasing Mach number results in a decreasing ratio of net thrust to drag. Therefore, the control of maintaining a positive aero-propulsive balance becomes more sensitive. This requires an optimised propulsion airframe integration resulting in an extremely coupled development procedure of the system components, namely the intake, combustion chamber, thrust nozzle and airframe. For experimental investigations this means that the complete system must be tested in ground based, so called free jet facilities. Due to the fact that if high scaling factors would be used then not all relevant effects can be accounted for by similarity laws, the system should preferably be tested at full scale. Consequently, if feasible, very large and expensive test facilities need to be used.

The present papers gives an overview of the operational German connected tube and free jet facilities both of which are important tools for the development of future high speed air breathing propulsion systems. Both types of facilities have inherent advantages and disadvantages. The available free jet facility is a short duration test facility with test times in the milliseconds range. It is therefore necessary to combine basic combustor performance investigations in free jet facilities with testing in connected tube facilities with sufficiently long test times in order to e.g. study the possible establishment of unsteady combustion that could cause dramatic vibration problems which must be avoided. However, in order to reach total temperature operating conditions which are related to flight conditions up to M=8, these continuously running facilities require hydrogen combustion pre-heating and the influence of the water vapour content on the combustion process must be quantified. Hence, the cross utilisation of different types of test facilities will allow to investigate different aspects such as details of the combustion process or the integration of intake, combustor and nozzle and will at the same time help to improve the performance predictions of air breathing propulsion systems.

2. Overview of connected tube facilites

In connected tube facilities a test section flow is generated which duplicates the inflow conditions into the combustor of dual mode (sc)ramjets. That means that the flow on the intake and thus the coupling of the intake and combustor flow is not considered. Connected tube facilities require pre-heating of the test flow in order to duplicate high Mach number flight conditions. Up to a flight Mach number of approximately M = 6, electric heater can be utilised. However, in order to generate combustor inflow conditions for higher flight Mach numbers, combustion pre-heating is applied.

2.1 Supersonic combustion test facility of ITLR at the Universität Stuttgart

The supersonic combustion test facility of the Institute of Aerospace Thermodynamics (ITLR) at the Universität Stuttgart (see Fig. 1) is equipped with a continuously running screw compressor delivering a maximum air mass flow of 1.5 kg/s at 10.5 bar. The benefits of the screw compressor are that it enables continuous experiments and it allows to change the total pressure easily, even during an experiment. The air is heated up to a maximum test section inlet temperature of 1500K using a two-staged electric heater of 1MW electric power. The air is accelerated via a Laval nozzle to approximately M = 2 before it enters the scramjet combustor. Hence, the test facility simulates a flight Mach number of approximately M = 6. Due to the utilisation of the electric heater, the air supplied to the combustor is free of vitiation. The experimental investigations focus on various fuel injection and fuel-air mixing devices as well as on ignition and combustion stabilization processes in combustion chambers and the development of advanced optical measurement techniques using a supersonic free jet flame (see e.g. [5] - [7]).

In Fig. 2, a schematic of the test section and combustion chamber configuration as well as the injector geometry used in [6] are shown. The test section consisted of a two dimensional 2.1 Mach number Laval nozzle, a rectangular 226 mm long isolator, and a diverging combustor. The fixed diverging (FD) angle of the combustor was 1° on each side until x = 598 mm. Thereafter, the diverging angle of the variable part, denoted as VD, could be changed within the range 0° to 3° on each side. The test section was made of copper because of its excellent thermal conductivity. Inside cooling channels with pressurized water to protect the combustor wall from the harsh environments during the several hours test run were used. The objective of this investigation was to study mode transition from weak combustion to strong combustion or vice versa. In a scramjet engine this is a critical phenomenon, because the thrust of each mode varies considerably. The mode transition is supposed to interact strongly with a so-called pseudo-shock wave or shock train. In order to control vehicles with scramjet engines, it is, therefore, essential to understand mode transition. Hydrogen was injected into the hot air using a parallel lobed injector (see Fig. 2). The injector was made out of copper and coated with gold to prevent oxidation and hydrogen penetration on the surface. The length and width were 86 mm and 40 mm, respectively. It could generate 4 streamwise vortices using a lobed structure. The width of the lobe was 8 mm and the nearest lobe was only 2 mm apart from the side wall (see Fig. 2). At a certain equivalence ratio, the flame jumped from far downstream to the end of the injector with bright illumination and high wall pressure, i.e., mode transition occurred. In this case of strong combustion, boundary layer separation was suppressed, thereby influencing the pseudo-shock structure. The effects of total pressure and geometry on mode transition were also investigated.

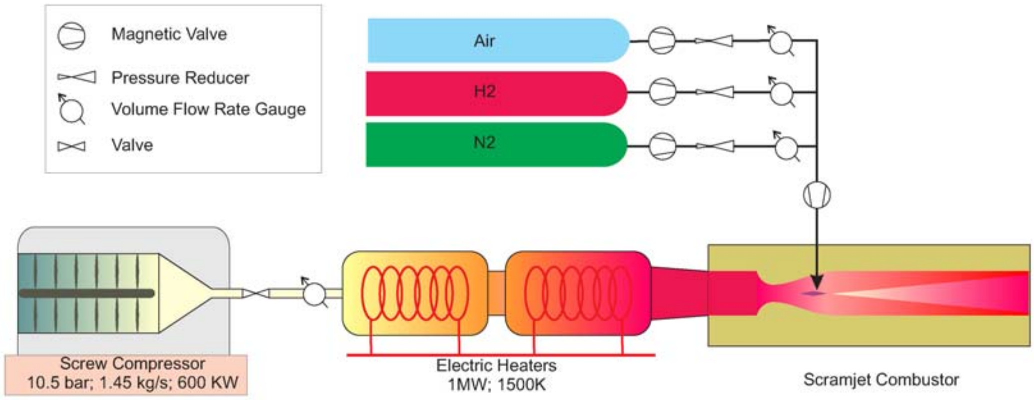


Figure 1: Schematic of the supersonic combustion test facility of ITLR at Universität Stuttgart.

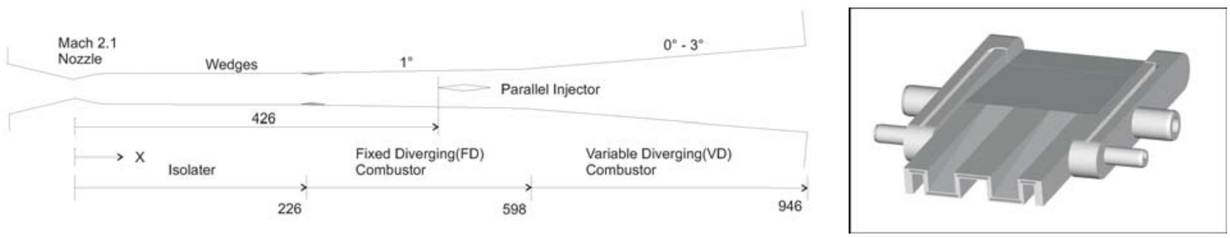


Figure 2: Schematic of the test section and combustor configuration (left) and of the lobed injector (right) used in [6].

In addition to wall static pressures measurements on the top wall of the test section using 40 pressure transducers, Schlieren for flow visualization, 1D spontaneous Raman scattering for the investigation of hydrogen/air mixing in a supersonic flow, pulsed 1D Raman scattering to investigate supersonic hydrogen/air combustion, laser-induced fluorescence (LIF) for the detection of OH distribution in supersonic combustion and techniques such as coherent anti-stoke Raman scattering (CARS), Rayleigh scattering and emission spectroscopy were applied at the ITLR facility [5].

Further, the ITLR facility was extensively used for fundamental investigations during the German Collaborative Research Centre 259 “High-Temperature Problems of Reusable Space Transportation Systems” at the Universität Stuttgart. Various fuel injection techniques of different complexity were investigated at supersonic flow conditions. The experimental results on mixing and combustion processes were compared with numerical simulations. In addition to simple wall and slot injections, ramp injection as well as injection by plane and 3D struts generating turbulent flow vortex structures for enhanced mixing were tested. The various injection devices were evaluated with respect to mixing homogeneity and generated total pressure losses [8]. Current research is directed towards supersonic combustion concept development for dual mode operation and further development of laser based diagnostic measurement techniques within the Graduate School GRK-1095/1 “Aerothermodynamic Design of a Scramjet Propulsion System for Future Spacecraft” funded by the German Research Foundation (DFG) [9].

2.2 Supersonic combustion test facility of LFA at Technische Universität München

In the connected tube supersonic combustion test facility of the Institute for Flight Propulsion (LFA) at the Technische Universität München (see Fig. 3) air is supplied at a pressure of 1MPa and at a maximum flow rate of 0.5 kg/s. An interchangeable Laval nozzle accelerates the flow to supersonic speeds (M = 1.9 - 2.2). The airflow is continuously pre-heated by an electric heat exchanger (700 K) and then vitiated through catalytic pre-combustion of hydrogen on a platinum-palladium solid catalyst up to a maximum of 1200 K. Hydrogen is supplied both to the test section (293 K, different mass flow rates for the pilot and main flames) and to the catalyst, while methane (570 K, max. mass flow rate 75 g/s) is only supplied to the test section for the main supersonic combustion. Oxygen is added to the supplied air upstream of the electric heater and counterbalances the amount reacting on the catalyst.

To monitor the static pressure distribution in the combustor thirteen pressure taps are implemented along the bottom wall. A total pressure probe and a total temperature sensor measure the conditions at the combustor exit. A Schlieren optic setup is used to determine the exit Mach number. The Schlieren images have been exploited also to calibrate the Particle Image Velocimetry (PIV) arrangement and to validate the pertaining results. PIV measurements have been carried out to study the strut wake and the interactions with the surrounding flow. An infrared pyrometer calibrated for a range of 630 K to 2000 K provides information on the surface temperature of the strut and a pulsed Dye laser and an excimer laser have been set up and adjusted to perform planar laser induced fluorescence (PLIF) measurements to detect the concentration of the OH radicals and to investigate the planar distribution of reaction zones. Further, to obtain information on the combustion process and on the aerodynamics of the flow exiting the combustor, a gas sampling probe has been implemented. The probe is made of Alloy 800H and it is water cooled (open cycle). The geometry of the probe tip allows sample collection under supersonic conditions to prevent friction choking and heating up of the gas. This feature and the heat exchange along the probe body support freezing of the reactions. The composition of the samples is then determined by means of gas chromatographic analysis.

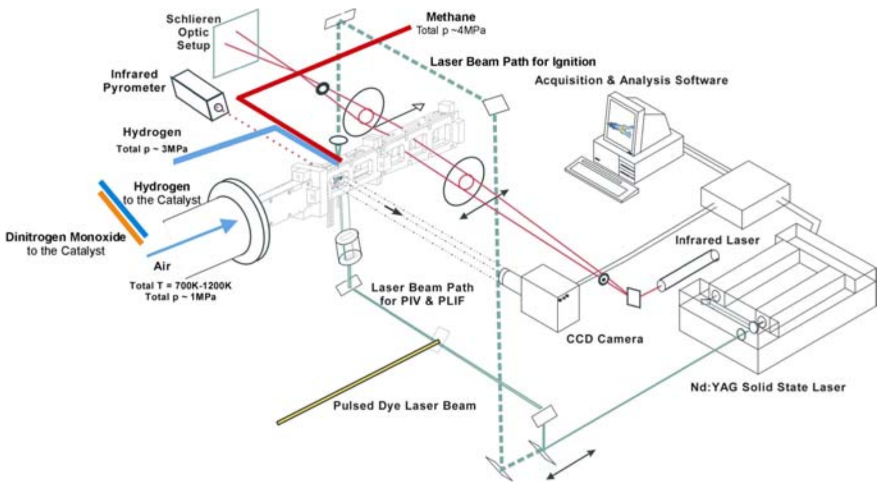


Figure 3: Schematic of the connected tube test facility of the Institute of Flight Propulsion [see e.g. 10].

The research performed in the LFA facility includes investigations on flame stabilization in supersonic combustion, supersonic combustion with strut injection, transition between ramjet and scramjet modes in a dual-mode combustor, methane combustion in a supersonic air flow and staged injection in a dual-mode combustor (see e.g. [11] – [15]).

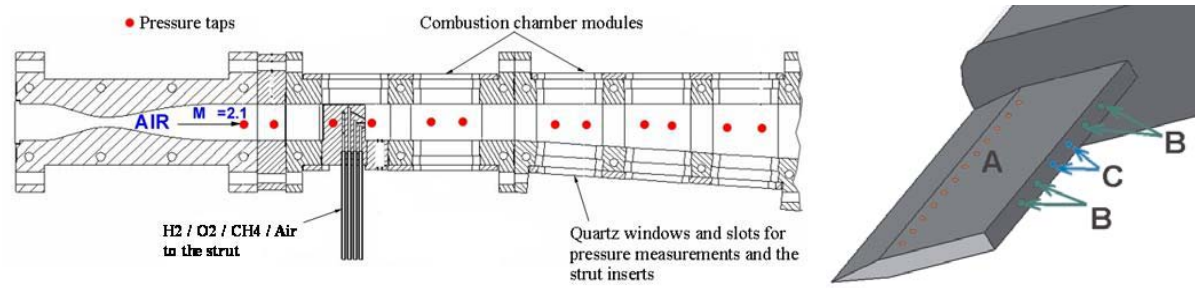


Figure 4: Side view of the modular combustion chamber with strut injection (left) and strut injector (right) [15].

The combustion chamber (see Fig. 4) has a length of 360 mm and an entry cross section of 25×27 mm^2 and consists of two modules. The strut injector is positioned in the first module, which is 160 mm long and has a constant cross section. For staged injection investigations a cavity injector is placed in the second module that diverges by 4° over 200 mm to counteract the pressure increase due to combustion. The walls are made of stainless steel and have quartz windows to enable optical access for non-intrusive measurement techniques. The combustion chamber is not cooled.

Different strut geometries have been tested in the last years: thickness, apex half-angle, trailing edge geometry, length and position in the combustion chamber and in free stream are among the parameters that have been varied. The present struts are made of Alloy 800H and feature an apex half-angle of 7.5°, 23 mm length, 27 mm width and 2 mm thickness. Several injection holes both axial and normal to the main supersonic stream guarantee for a homogeneous fuel distribution. The goal is to exploit the subsonic wake of the strut and the vortices created at its trailing edge to generate a recirculation zone with higher temperature in the middle of the supersonic airflow. Hydrogen and either air or oxygen at sonic speed and at a temperature of about 290 K are injected axially into this region to vary the mixture ratio. A laser beam is focused into the mixing zone to ignite the mixture. The dwell time of the radicals in the recirculation zone is long enough to support propagation of the reaction after few laser pulses so that a pilot flame can be stabilized. Methane or hydrogen is injected at approximately 570K and at sonic speed on both sides of the strut normally to the surface (i.e. to the supersonic stream). The cavity injector has a 4 mm depth and a 20 mm length measured from the front wall to the middle of the slanted rear wall ($\alpha = 20°$). The injection holes are fitted along the direction of the stream lines. Because of its length-to-depth ratio the cavity is classified as open, i.e. the shear layer detaching at the front cavity lip will reattach at the cavity rear wall and not on its floor.

2.3 M11 test complex of DLR Lampoldshausen

Since more than a decade the M11 test complex of the DLR Space Propulsion Institute at Lampoldshausen has been used for experimental research on supersonic injection and combustion ([16] – [20]). At several test positions air heaters allow in connected tube test set-ups to simulate the hot air flow from air intakes under ramjet and partly even up to scramjet relevant conditions. A typical example of an experimental setup is presented in Fig. 5. At the M11-4 test facility with an air heater, a M = 2 Laval nozzle and the scramjet combustion chamber with the hydrogen injector are mounted. High pressure air is supplied to the air heater, where total temperatures between 600 K and 1500 K can be generated by the use of hydrogen-oxygen-burners. The stagnation pressures ranges up to 3.5 MPa and air mass flow rates up to 5 kg/s can be achieved. The oxygen concentration of the vitiated air is kept constant at a mole fraction of 21% by the use of additional oxygen.

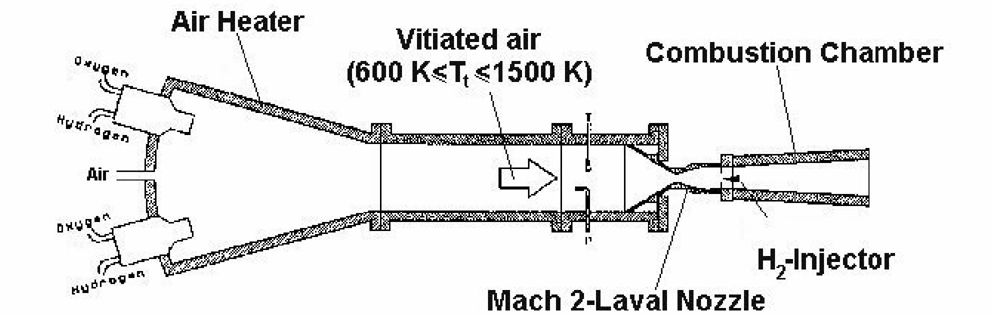


Figure 5: Schematic of the connected tube facility M11-4 at DLR Lampoldshausen (not to scale) [19].

The rectangular entrance cross section of the combustor, which can be seen in the sketch of Fig. 6 in more detail, measures 50 mm in height and 40 mm in width. The upper and the lower wall have a diverging angle of α=3°. A wedge shaped injector is placed in the center of the chamber immediately behind the Laval nozzle. Windows are located on both side walls. They give access to the flow field and combustion process over the whole length of the model combustor for the application of optical diagnostic tools such as observation with cameras, Schlieren or laser-based techniques (PIV, LIF, CARS, etc.).

Scramjet research activities at the M11 facility have been focussed on detailed studies on supersonic flame characteristics and injector performance using a wide set of applied measurement techniques such as Particle Image Velocimetry (PIV), Laser- Induced Florescence (PLIF), spontaneous OH-emission, sampling and pneumatic probes, coherent anti-Stokes scattering (CARS) and high speed Schlieren.

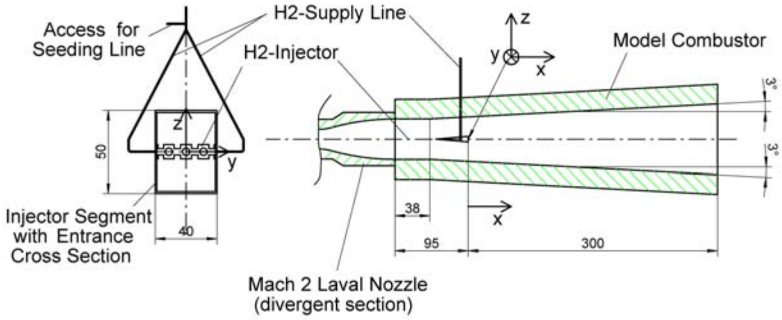


Figure 6: Combustion chamber dimensions and injector installation in the M11-4 facility [19].

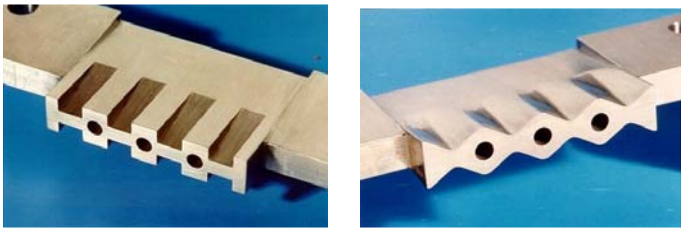


Figure 7: Fuel injectors USCER (left) and WAVE (right) used in the M11-4 facility [19, 20].

The images of Fig. 7 show two examples of injectors, which are installed horizontally, spanning the combustion chamber from the left to the right side wall as can be seen on the rear view on the injector section of the combustion chamber in Fig. 6 (left sketch). The influence of the external shape of two different injectors on the combustion chamber flow with respect to mixing enhancement and pressure increase by combustion heat release was studied within the European LAPCAT program [4, 21]. Both injectors (see Fig. 7) are wedge injectors, which were designed to produce gradients of static pressure, density and flow velocity over their surface, normal to the main flow direction in order to enhance the fuel / oxidizer mixing. The results obtained show that the outer shape of the investigated hydrogen injectors has a distinct influence on the mixing and combustion process inside the model scramjet combustor. Laser diagnostic measurements indicate that for the WAVE injector a more favourable 3-dimensional structure of the hydrogen-air-interface and mixing layer was created, resulting in a noticeably higher heat release in reacting flow cases.

3. Overview of free jet facilities

The utilisation of free jet facilities focuses on the investigation of integrated high speed air breathing propulsion systems consisting of intake, combustor and nozzle. For small scale flight configurations these investigations can be performed at 1:1 scale in the existing German facility.

3.1. High enthalpy shock tunnel Göttingen, HEG

The High Enthalpy Shock Tunnel Göttingen, HEG of the German Aerospace Center (DLR) was designed and built over the period 1989 - 1991 and was commissioned for use in 1991. The free piston driven shock tunnel is operated in reflected mode. It is designed to provide a pulse of gas to a nozzle at stagnation pressures of up to 200 MPa, and stagnation enthalpies of up to 24 MJ/kg. Since its commissioning, HEG has been utilized in numerous space programs. The research activities which have always been strongly linked with CFD investigations range from the calibration process of the facility and the study of basic aerodynamic configurations, which are well suited to investigate fundamental aspects of high enthalpy flows, to the investigation of complex re-entry configurations [22]. A schematic of the facility is shown in Figure 8. The overall length of HEG is 60 m and it consists of three main sections: the driver, consisting of an air buffer and a compression (driver) tube, the shock or driven tube and the subsequent nozzle/test section. The compression tube is separated from the adjoining shock tube by a stainless steel main diaphragm. During a run, HEG uses the high pressure air buffer to drive a free piston down the 33.0 m long compression tube compressing quasi adiabatically a driver gas mixture of helium and argon in a transient process, allowing generation of high driver temperatures and pressures. The driver pressure is controlled by the bursting of the scored main diaphragm. After diaphragm burst, a strong shock wave propagates down the 17.0 m long shock tube and is reflected from the end wall, creating a region of high pressure, shock-heated test gas. When this nozzle reservoir region is formed, the secondary diaphragm, a thin mylar sheet, ruptures and the test gas expands through the convergent-divergent hypersonic nozzle to provide the free stream flow. More detailed descriptions are given in [23] and [24].

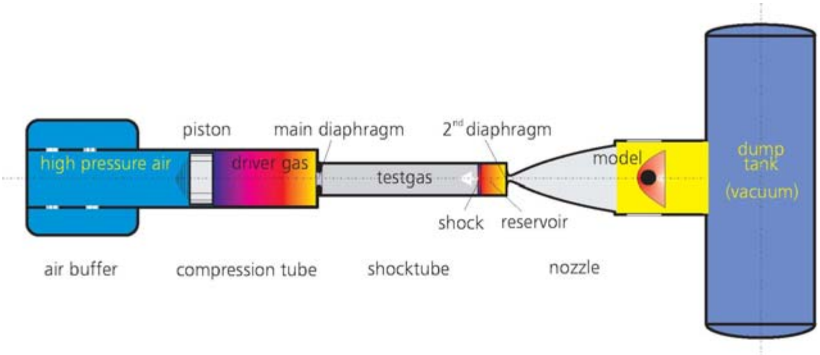
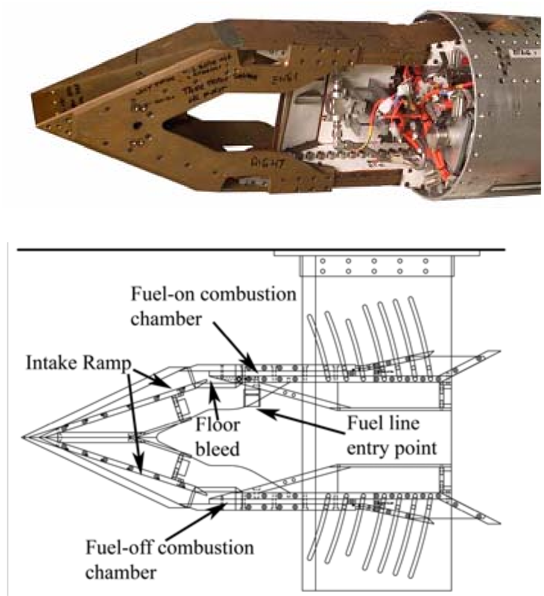


Figure 8: Schematic of the free piston driven shock tunnel, HEG (not to scale).

The extension of the HEG operating range to allow the ground based testing of complete scramjet engines consisting of intake, combustion chamber and nozzle included the design and construction of several contoured nozzles as well as the installation of a gaseous hydrogen fuel injection system for the delivery of hydrogen fuel to the wind tunnel model. The fuel injection system consists of a 12 mm diameter Ludwieg tube and a fast acting solenoid valve. The Ludwieg tube can deliver a pulse of fuel with constant pressure for 13-20 ms with mass flow rates in the range of 1 - 190 g/s. Test conditions at total enthalpies of about 3 MJ/kg are available providing free stream conditions which duplicate atmospheric flight conditions in an altitude range from 20 to 50 km at Mach numbers of 6 and 8. The exit diameters of the contoured nozzles are 400 and 600 mm, respectively. The available test time at these HEG operating conditions is approximately 4 ms.



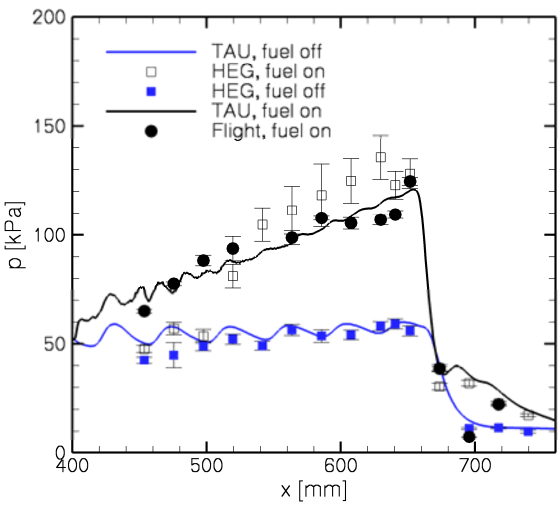


Figure 9: Photograph of the HyShot flight configuration (courtesy The University of Queensland, Australia) and schematic of the wind tunnel model (left); Measured and computed static wall pressure distributions in the HyShot combustor at a flight altitude of 33 km (right).

So far the main emphasis on the scramjet research in HEG was on the analysis of the HyShot [25] supersonic combustion flight experiment configuration using a 1:1 scale model (see left part of Fig. 9). The intake is a simple wedge of 18° half angle and is 100mm wide. The cross-sectional dimensions of the combustion chambers are 9.8 x 75 mm. Fuel was injected into only one chamber in order to allow fuel-on measurements to be compared with fuel-off measurements. The fuel injector was comprised of a series of four holes in the wall of the combustor located 58.0

mm from the leading edge of the combustion chamber. In HEG, the free stream flight conditions were duplicated for an altitude range between 27 and 33 km. On the intake and in the combustor, surface pressure and heat flux measurements were performed. The results of the test campaigns in HEG show good agreement with flight and numerical data and confirm the establishment of supersonic combustion in the HyShot configuration [26]. Important aspects of the flow path development such as the laminar-turbulent transition of the boundary layer flow, the influence of angle of attack, equivalence ratio and dynamic pressure variations and their interdependencies could be investigated in HEG as part of the HyShot post flight analysis. Within the accompanying numerical investigations, performed in the framework of the European LAPCAT project [4], the DLR flow solver TAU was used to determine the free stream conditions in the HEG test section, to simulate the HyShot intake flow field and for the simulation of the turbulent reacting flow in the combustor [27]. In the right part of Fig. 9, the measured (flight experiment and HEG wind tunnel data) and the computed static pressure distributions in the HyShot combustor downstream of the hydrogen injector are shown. Good agreement between flight data, HEG measurements and numerical analysis is observed.

Currently, the HEG operating range for scramjet ground based free jet testing is extended to duplicate M10 conditions in approximately 30 km flight altitude. The exit diameter of the M10 nozzle amounts to 880 mm.

Acknowledgement

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