STUDENT HYBRID THRUSTER TESTING AND RESEARCH

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ABSTRACT

After the first student-hybrid-thruster campaign at DLR-Lampoldshausen in 2011 the Institute of Space Propulsion consequently developed the student-test program. A stronger and improved test rig was fabricated, advanced measurement techniques were applied, more detailed test objectives were defined and testing was accompanied by advanced CFD and thermochemical calculation. In many ways the hybrid thruster appears as an ideal research object for rocket propulsion students. The thruster provides aspects of liquid propulsion, phase transition, solid propulsion, ignition, combustion and rocket nozzle flow. System pressure and combustion temperature are in the order of the state of the art of rocket engines, but for all this the students can fully handle the thruster, rig and testing by their own due to small mass flow and short test duration. Nevertheless the students are introduced to working principles and standards for rocket engine test facilities. The latest test campaign focused on the investigation of the exhaust jet. The experimental studies and advanced measurements were compared to a numerical Navier-Stokes calculation of the flow field.

1. INTRODUCTION

Rocket propulsion science contains an overwhelming variety of research, development and testing subjects. It is a challenge for every scientist and engineer in this field to understand not only his special subject deeply but as well its overall context in space science. Especially for students of propulsion science the German Aerospace Center (DLR) as an establishment of research and as the national space agency has started a program to support young academics to step into this thrilling technology at the frontier of science.
School pupils are encouraged in school labs to enter space studies and university students are trained in summer schools. A campus and a student test field are in progress [3] and the first students have already practised on a student test rig, laid down their master thesis and stepped into recent research programs. On their way the students learn to apply modern computer codes for computational fluid dynamics (CFD), to use advanced measurement and diagnostic systems and they are trained to prepare and execute hot testing of propulsion systems according to principles and standards for rocket test facilities. In the direct vicinity of the large rocket engine test facilities the students are carefully coached by experienced senior test engineers to conduct their own developments.

2. DEVELOPMENT OF THE TEST PROGRAM

The student testing was developed from a simple thrust suspension point to a comfortable, mobile student test rig with thrust measurement. Also the test specimen, a micro hybrid thruster was modified in order to introduce measurement sensors for combustion pressure and temperature. The next generation of thruster

will allow higher thrust and longer test duration. In the first test campaigns [1,2] the validation of the test rig and measurement system was in the focus as well as the reproducibility of test results. The second test campaign focussed on the expansion process of the oxidiser in the injection element between run tank and chamber. Another subject of the second campaign was the investigation of the regression rate of the fuel package [9]. The results of the third campaign are content of this publication, it has the focus on the characteristic of the exhaust jet.
The students are introduced to test management procedures such as risk management, test readiness validation, failure management, and introduced to test documentation such as test plan, test request, measurement requests and test execution procedures. They are trained to prepare and use the test rig as well as the control and command system of the rig and specimen and the measurement acquisition, recording and processing system.
For each campaign the students establish the test ensemble consisting of mobile components again, but on the other hand they take profit of the progress and results of previous campaigns.

3. OVERVIEW OF STUDENT HYBRID THRUSTER TOPICS

The hybrid thruster provides three main subjects for student research, the liquid oxidiser supply, the combustion of the solid fuel package and the flow downstream the combustion chamber (nozzle flow and exhaust jet).
The oxidiser in this test program is nitrous oxide (N_2O) in liquid state at its saturation point. The supply to the combustion chamber is either liquid with phase transition at the chamber inlet (through the injector) or gaseous N_2O taken from the gas fraction of the storage bottle. The injection of the liquid N_2O is very much alike the injection in large cryogenic rocket engines. Due to the theoretical and experimental work on this subject the students reach a deep and solid understanding of the injection process of modern rocket engines.
The combustion process of the solid fuel offers another package of research subject. The prediction of the pyrolysis, the investigation of the regression rate and the understanding of the recirculation influence to the overall characteristic and performance of the chamber are subjects which contribute to recent work and theories in hybrid and solid propulsion.
The transonic and supersonic flow of the nozzle and exhaust offers another interesting field of research. From one-dimensional flow obtained from NASA programs [7] the student go forward to two-dimensional CFD calculation [12] and furthermore investigate the effect of post combustion and real gas effects, which is a vast package of subjects again when we consider that the flow is composed of several gas species.

4. CALCULATION OF THE PROPULSION PROCESS

4.1 THERMOCHEMICAL CALCULATION

The thermochemical calculation of the combustion process provides the properties of the gas in the nozzle flow and in the exhaust. All the features and properties of the used propellants have to be well-known to have a comprehensive overview of the combustion process.
The nitrous oxide (N_2O) is supplied in liquid state in a small bottle at saturation pressure and in the amount of 8 grams. This oxidiser is very common in the space propulsion and all its features are easily available in literature, especially the thermodynamic ones as function of the temperature (i.e. enthalpy of formation, entropy…). Most of the software used for the study of the thermochemical process present an internal library with these requested data.

Different and more complex is the discussion about the fuel, Polyethylene Terephthalate (commonly called PET). Its chemical formula is $C_{10}H_8O_4$ which identifies the single monomer, a recurring unit of a more complex polymer chain produced by the reaction between Ethylene Glycol, well-known in literature, and Terephthalic Acid, more complex and based on the aromatic ring in it.
All its properties must be known for the thermochemical calculations and in particular the heat of formation should be defined at certain temperature.
For the definition of it the *additivity of group properties of Van Krevelen-Chermin method* [13] is recalled and, according to this method, the molecular structure of the polymer is sufficient for the evaluation of the heat of formation that can be easily found by ([5], [8]):

$$\Delta H_f^0 = \sum \text{contribution of component groups} + \sum \text{structural correction} = A + B^*T \qquad (4.1)$$

The group contributions are considered as linear functions of the temperature and these, together with structural corrections, are based on experimental data [14]. The heats of formation of the PET monomer at T = 25 °C = 298 K is ([8], [14] and [13])

$$\Delta H_f^0 = -268008 \, {}^{J}\!/_{mol} = -268.008 \, {}^{kJ}\!/_{mol}$$

but it is possible to calculate it in any condition. A more detailed description of this calculation is presented in [4].
At this point, the whole features to establish and define the combustion process are known and easily derivable.

4.2 COMBUSTION

An overall understanding of the combustion process is essential requirement for a complete investigation of the performance of the thruster and its exhaust. This study provides the chemical composition of the burned mixture flow with the support of Nasa CEA, software developed by Gordon & Mc Bride [7]. Specifying the kind of problem of interest (i.e. rocket, detonation, assigned enthalpy and pressure problem…) and setting the input data requested, the software is able to return the thermodynamic, thermochemical and transport properties of complex mixtures. This NASA software is based on the well-known equations and relations for chemistry and thermodynamics, assuming all gases as ideal and the interaction among phases negligible and then the perfect gas law can be used [7].
The case of the micro hybrid thruster presented in this paper concerns a *rocket problem* and considers a *mixture ratio o/f* of 7.1. This value of *o/f* comes from an experimental evaluation of the fuel package mass and its consumption in several tests considering only the amount of propellant burned during the combustion process [9]. The amount of N_2O burned is only 8 g and the PET average one is 1.1303 g. The o/f ratio, together with the combustion chamber pressure measured by sensor, is used as input data for NASA CEA simulations.
Due to the fact that the PET is not included in the NASA CEA library, a new definition of this reactant is needed in form of a user-provided name and properties. It means that the name, amount, chemical formula and heat of formation must be entered in the software by the user.
The combustion is considered *frozen* in agreement with what has been done in Tau Code (see Chap.4.3).
The input data for the simulation of the combustion process with NASA CEA and its results are shown respectively in table 4.1 and 4.2.

Oxider	N_2O
Fuel	$C_{10}H_8O_4$
Mixture ratio o/f	7.0778
Heat of formation PET	-268.008 kJ/mol
Chamber pressure	16.6 bar
Supersonic Area ratio A_e/A_t	4.2539

Table 4.1 Combustion process simulation - NASA CEA input values

Temperature (CC)	2528.58 K
Density (CC)	2.4203 kg/m^3
Isentropic factor Υ (CC)	1.2397
Mach (exit nozzle)	2.742
Specific impulse (exit nozzle)	1829.3 m/s

Table 4.2 Combustion process simulation - Main resulting values

4.3 CFD CALCULATION OF THE EXHAUST

The numerical study of the micro hybrid thruster´s exhaust gases under examination is made by using DLR internal software Tau Code, whose main modules are the preprocessing, flow solver and adaptation. It is a powerful software to study and predict the viscous and inviscid flow behaviour in complex geometries from subsonic to hypersonic flow regime, employing hybrid unstructured grids. Tau-Code, designed for massively parallel computations, is not able to generate a grid but comprises tools for grid modification, adaptation and deformation.

In order to completely describe the flow field, a wide range of turbulence models are available, ranging from simple algebraic approaches to full Reynolds-Stress Transport models [12]. However, Tau Code does not assume any combustion model for the mixture flow. This is a software limitation which implies the study of a frozen flow along the nozzle without considering any combustion products.

In the case under analysis, to compute the flow behavior a RANS turbulence model is chosen, which represents also the natural mode of operation for Tau Code. In particular a one-equation model is used, which refers to the Standard Spalart-Allmaras model [11] in order to solve the transport equation. For the discretization of the convective fluxes of the RANS and turbulence equations, the upwind scheme is used. The implicit Backward-Euler method is the scheme for time-stepping solution. Different levels of adaptation (maximum 30% of new points for each one) are applied taking into account the presence of high gradients due to the shocks.

A complex structure of the simulation for the flow field solving is implemented, which consists in the changing of the *Courant-Friedrichs-Lewy number* and several grid adaptations to guarantee the convergence of the method and make the *y*+ value (a non-dimensional wall distance for a wall-bounded flow) as much as possible close to 1. The y+ is a good indicator for the goodness of the mesh of the model: a value close to 1 is the desirable value for a good grid resolution and a convergent solution in a near-wall modelling [6]. This is shown fig.4.1 while fig.4.2 shows the good convergence of the simulation.

At the end the simulation presents 200'000 iterations providing a good compromise between result and computational time.

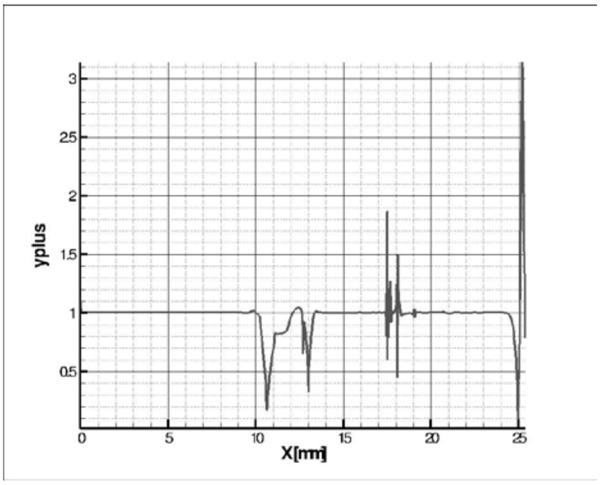

Fig.4.1 y+ in the nozzle

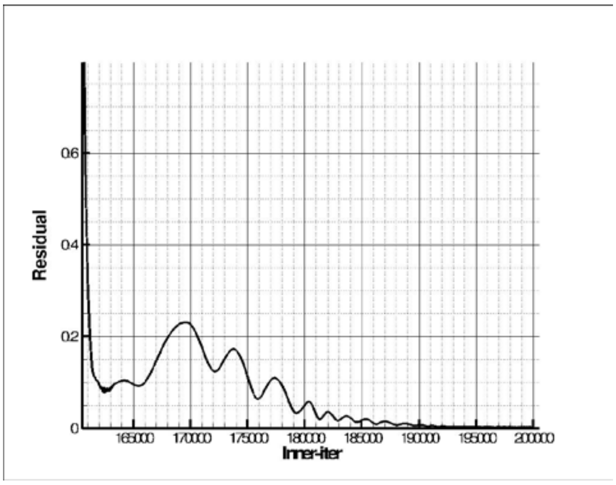

Fig.4.2 Residual vs Iterations

Using the results given by NASA CEA (Table 4.2), a precise and close to the real phenomena CFD simulation can be performed with Tau Code, providing a set of results reliable and realistic. From these results with post processing analysis, it is possible to reproduce the exhaust flow field and investigate its features, such as the Mach distribution (Fig.4.3) or the velocity distribution (Fig.4.4).

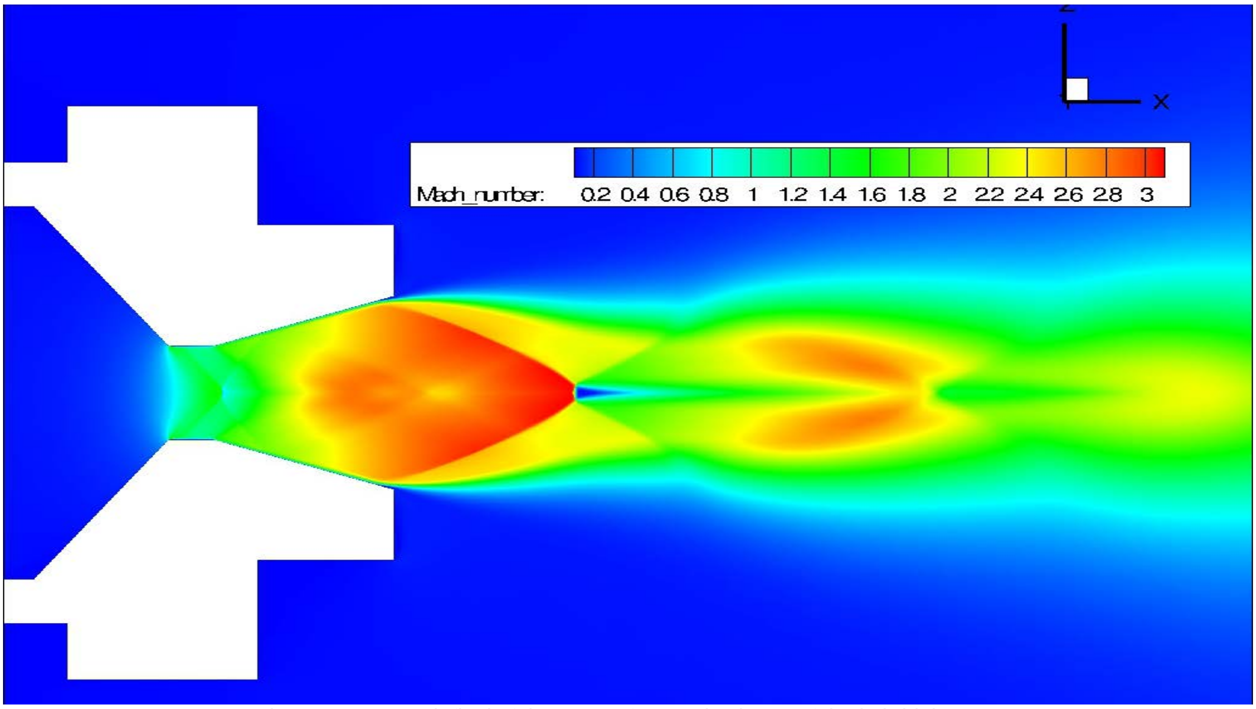

Fig.4.3 CFD Mach field of the nozzle flow and exhaust of the hybrid thruster

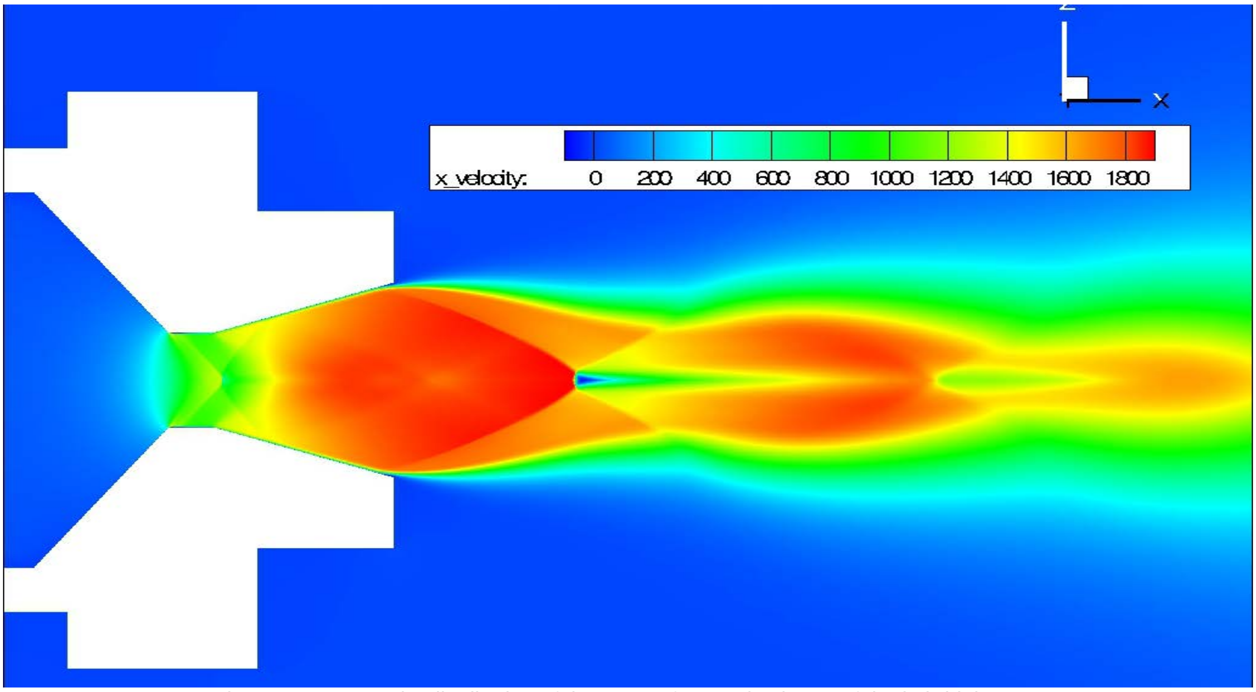

Fig.4.4 CFD velocity distribution of the nozzle flow and exhaust of the hybrid thruster

Further analysis gives the possibility to understand better the behavior of the system. In particular it is possible to focus the attention on the distribution of the Mach number along the symmetry axis of the nozzle, only for short part of the field behind the nozzle (approximately a distance of 55mm from the exit) because the whole field is not of interest and the Mach number goes to zero when the flow is extinguished.
The length of the nozzle is 25.4 mm, marked with red line in the Fig.4.5 and inside the throat the Mach number passes from a supersonic condition to a subsonic one and then returns to values higher than 1. This behavior leads to a loss in the nozzle performance and then it turns into limitation in the thrust. This is due to the shape of the nozzle: the walls of the throat are parallel and this is not a perfect geometry to reach the optimal performance for the nozzle.

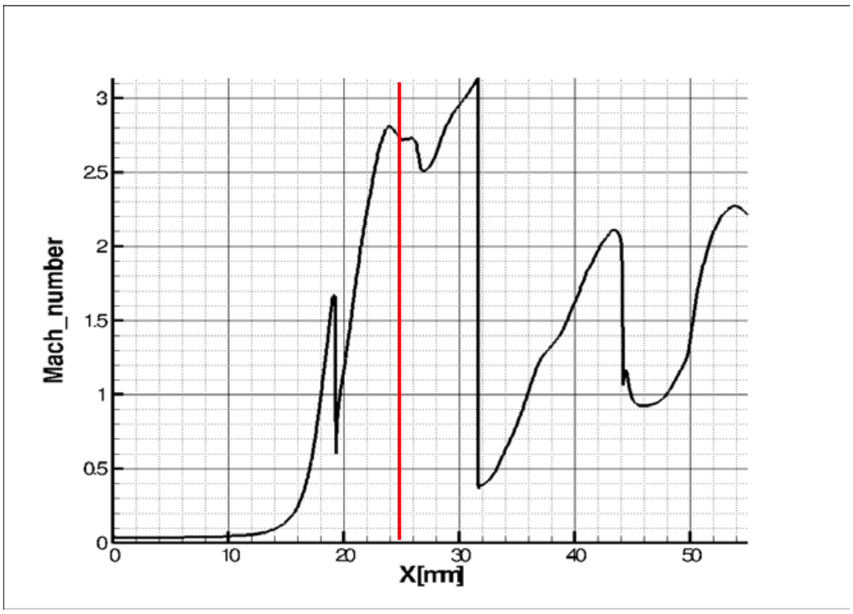

Fig.4.5 CFD velocity distribution of the nozzle flow and exhaust of the hybrid thruster

4.4 COMPARISON: CEA VS CFD

Having at disposal the combustion process simulation by NASA CEA and the CFD calculation of the flow field, a first check and comparison of the results shows how close these two models are in terms of thrust, mass flow rate and specific impulse. The relative results are presented in the table 4.3 and the deviation between these two models is anyway small, around 15%, thus giving a positive feedback about the goodness of the models.

	Nasa CEA	Tau Code	Error
Thrust	17.9 N	15.15 N	15.3%
Mass flow rate	10.589 g/s	9.198 g/s	13.1%
Mach	2.742	2.716	1%
Specific impulse	186.54 s	185.97	< 0.5%

Table 4.3 Comparison of numerical results - NASA CEA vs Tau Code

The deviation between NASA CEA and Tau Code results has to be also investigated deeper but in first analysis it is possible to say that this is due to a very important approximation made in the Gordon-McBride software: all the transformation during the simulation is considered to be isentropic, this hypothesis is not present in Tau Code. It has to be considered also that NASA CEA involves a one-dimensional flow while Tau Code computes two-dimensional flows.

5. HYBRID TESTING

For experimental investigation the hybrid thruster HT-01 is tested on the test rig PVII at the DLR test center in Lampoldshausen. Optical diagnostics are applied to investigate the flow field of the exhaust. The tests are performed according applicable standard (DIN, EN, EC-rules, ECSS) and the test team applies standard testing procedures (test preparation, execution, documentation, risk and failure management e.g.)

5.1 HYBRID THRUSTER HT 01

The hybrid thruster HT 01.e used in this campaign is shown in the picture below, it consists in a case, made of aluminium. The case contains all the elements necessary:

- The nozzle (convergent-divergent)
- The fuel package
- The injector body
- Piercer
- Diaphragm
- N2O bottle
- Arming device

Four O-rings are needed to avoid gas losses, and two snaps rings are positioned at the beginning and at the end of the case to keep the parts in position.
In the current version (HT 01.e, e for experimental) of the thruster it is possible to measure the temperature and the pressure inside the combustion chamber, the case has two holes which permit to connect two sensors, and a metal ring is positioned between the nozzle and the fuel package to allow the measurement in this configuration. A second flange with another two holes gives access for the ignition system. The thruster HT 01.e has the same internal dimensions as the flight version HT 01.f except for the space of the separation rings.

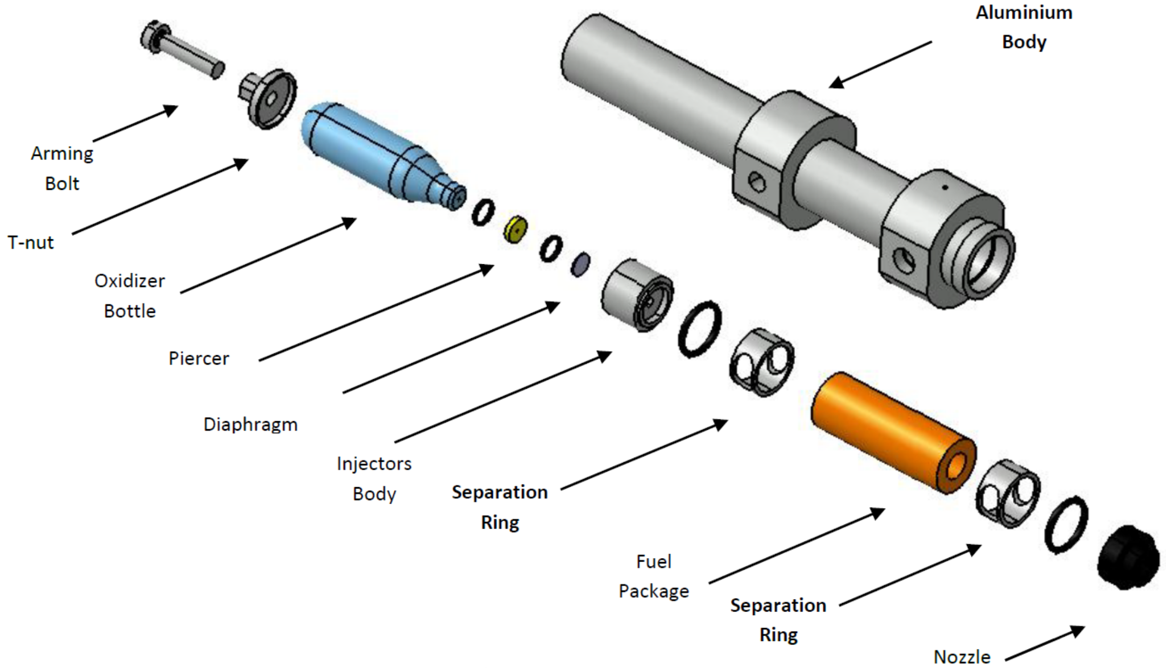

Fig. 5.1 Thruster assembly

5.2 TEST RIG PVII

The test rig PVII consists of a solid steel table with a sledge on it, which is in contact to a force (thrust) sensor. On the sledge a tube takes in and bears the thruster. A horizontal and vertical firing direction is possible. A plexiglas protection roof covers the test position.

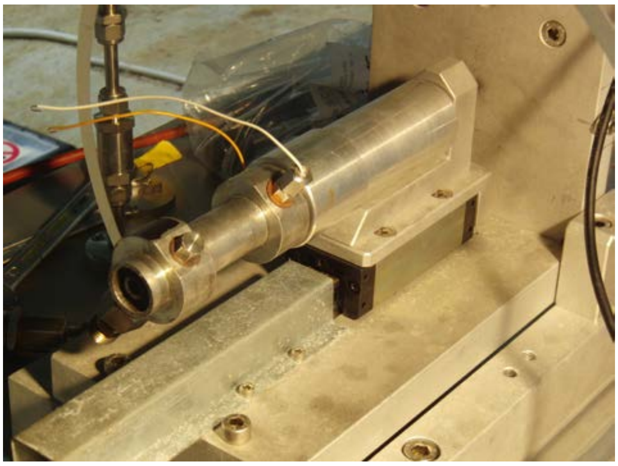

Fig.5.2 Thruster HT 01.e on the test rig PVII

5.3 TEST OBJECTIVES

The first objective was to select a reference point in order to define conditions for the numerical simulation and calculation, hence the objectives were to confirm the calculation with experimental result. For this test campaign the following test objectives were defined:

Primary objectives

- Numerical modelling of the exhaust flow
- Experimental set up
- Better understandings of the exhaust gas field
- Investigation & visualization of the phenomena by IR and schlieren imaging
- Verify the accordance between numerical and experimental data

Secondary objectives

- Optimization of test process
- Improve of ignition & start-up

5.4 MEASUREMENT OF BASIC PARAMETERS

During the several tests performed, the basic parameters are monitored. In particular the signals of the temperature and pressure in the chamber and the thrust generated by this little rocket are recorded during all the experiments.
As temperature sensor, a thermocouple type S is used due to the high temperature reached in the combustion chamber.
Many tests are performed on the hybrid thruster. Among them, two of these are used as reference tests and for the calibration of the sensors. These are the test 6 and 7 (T006 & T007) which present the following graphs for pressure, thrust and temperature:

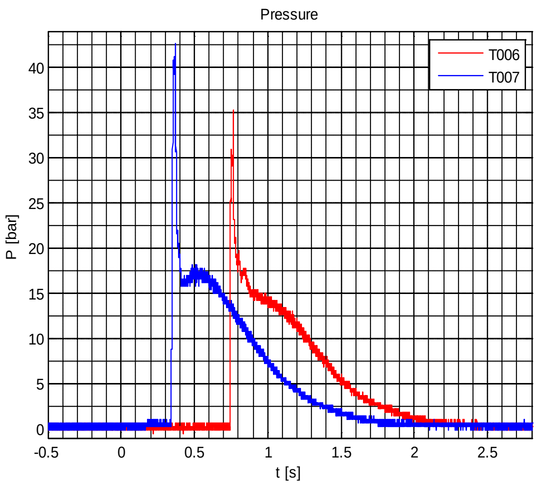

Fig.5.3 Pressure trend

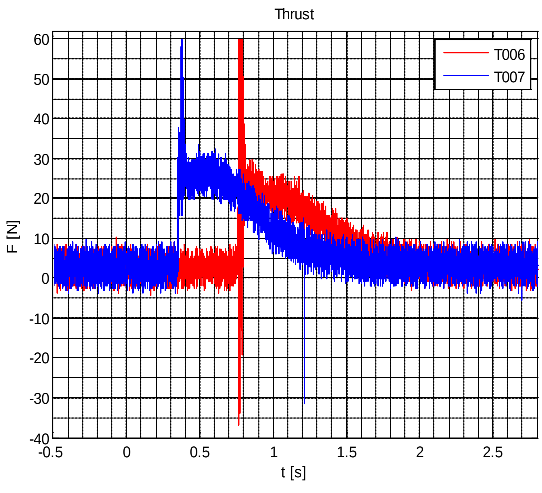

Fig.5.4 Thrust trend

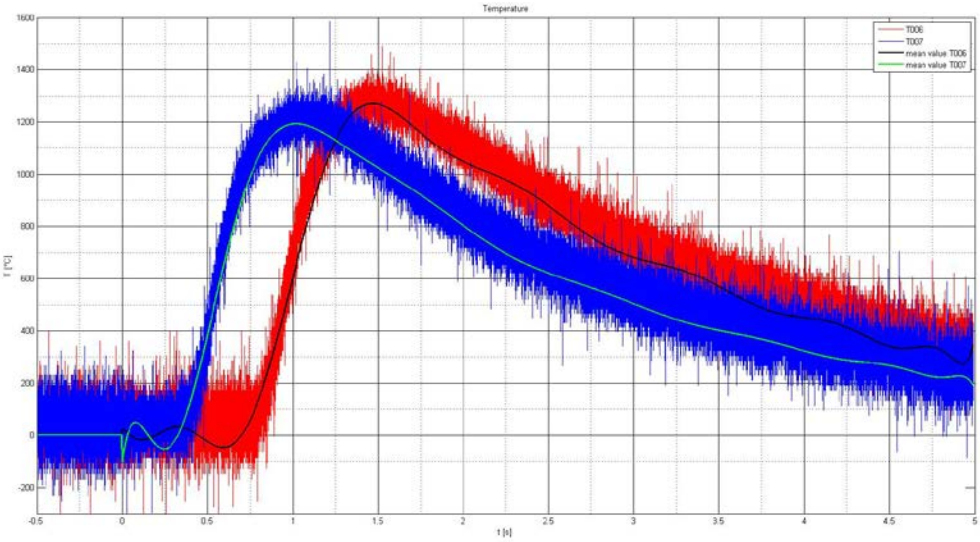

Fig.5.5 Temperature trend

It is possible to notice a delay of about 0.5 s in the startup of the test T006 due to the fact that this test showed an effective delay between the moment the igniter makes sparks and the start of combustion.
It is necessary to evaluate and find on these curves one point, defined as reference point, chosen from the constant phase (*plateau)* of the thrust curve. The reference point, when it fits among all the tests providing the quite similar values of pressure temperature and thrust, can be considered as evidence for the reproducibility of the event studied. The temperature peaks in the table 5.1 are referred to the peak of the mean value temperature curves and not to the absolute peaks.
This point is used also as comparison point for the numerical simulation with NASA CEA and Tau-Code.

Test	Time [s]	Temperature Peak [°C]	Pressure [bar]	Thrust [N]
T006	0.86	1248.7	16.56	23.73
T007	0.45	1193.2	16.57	23.74
Ref. Point	-	1220.95	16.6	23.73

Table 5.1 Reference point

5.5 INFRARED VISUALIZATION

The study of temperature field which characterizes the whole system under analysis is carried out using an infrared camera. In this way it is possible to record and take pictures of the experiment and in particular of the exhaust gases and of all the area surrounding the nozzle. With an infrared camera it is easy to detect the radiations emitted and transduce them into temperature variations producing qualitative impression of the field.
In particular the aim of this investigation is to evaluate the position of the Mach disks from the exit of the nozzle looking both the same in the CFD results (Fig.5.6) and in the experimental infrared pictures (Fig.5.7). This analysis confirms that the temperature fields present the same distribution; and the numerical model and the experimental phenomena fit very well together.

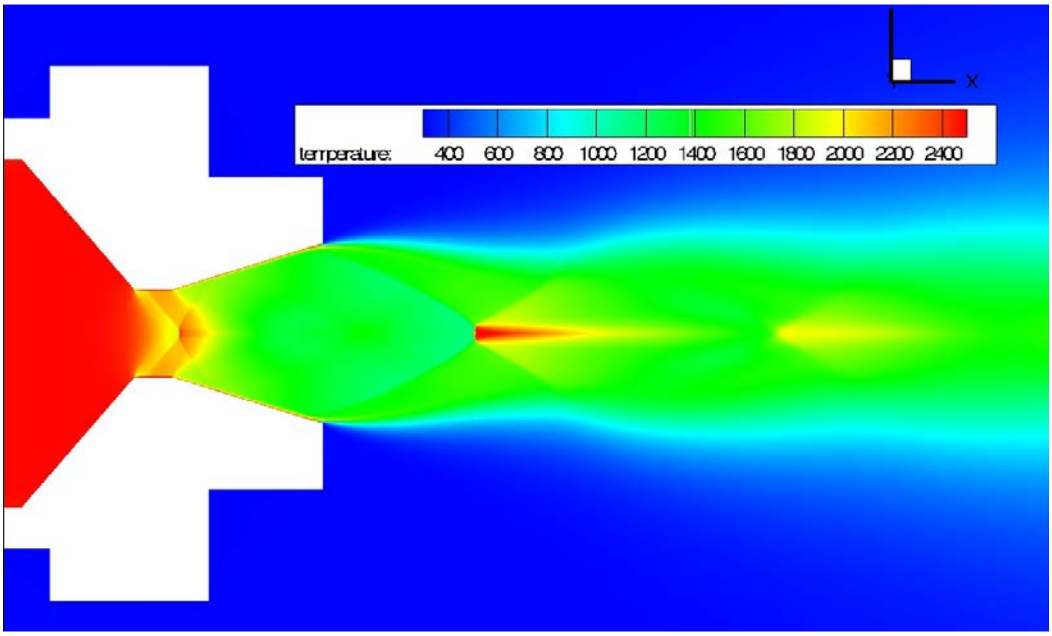

Fig.5.6 CFD temperature exhaust and nozzle field

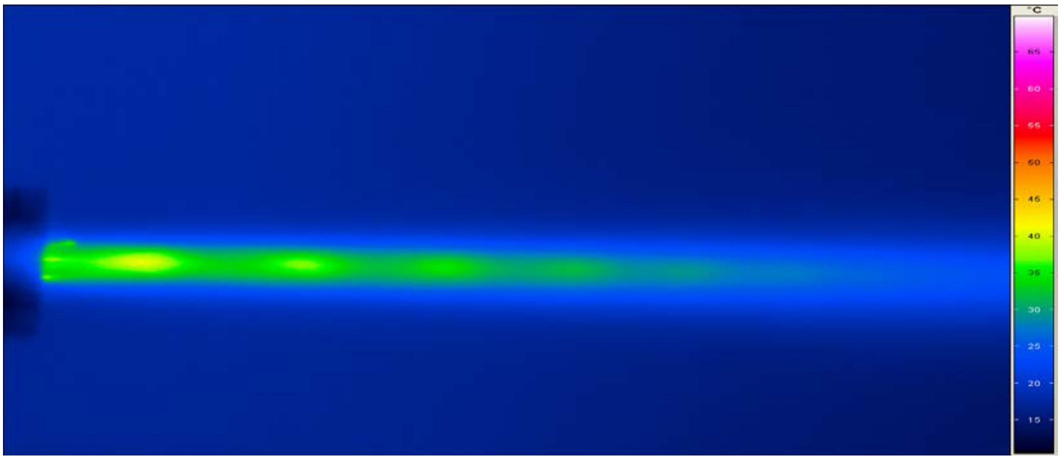

Fig.5.7 Infrared Mach pattern details in the exhaust

The attention is focused on the initial moments of the start-up of the combustion and in particular after 0.5 s from the ignition when the flow is supposed to be fully developed and a plateau is registered.
Making a comparison considering only the first two Mach disk, the Mach patterns are very close to be at the same position in these two cases with small differences of 2-3 millimetres and the following results emerge:

	IR picture	Tau simulation
1° Disk	33.15 mm	32.39 mm
2° Disk	47.8 mm	45.5 mm

Table 5.2 Mach disk comparison

There is a good accordance in the temperature distribution between the reality and the numerical simulation.

5.6 SCHLIEREN VISUALIZATION

The schlieren method is a powerful technique to study the behaviour of a flow in terms of density gradient. Showing the density distribution in the observed area, the basic principle of the schlieren technique is the combination of the optical projection of an object with an indication of its light deflection, which is proportional to the variation of the density. Effectively the schlieren resulting image is an optical image formed by a lens which leads to a conjugate optical relationship to the schlieren object [10].
A knife-edge, able to cut off part of the reflected bended light is an essential element of the schlieren visualization.
The schlieren imagery is used for the study of the micro hybrid thruster to visualize the exhaust gases and observe the flow field and its structures, especially the Mach patterns.
But the main idea to analyze the Mach distribution in the flow field is not possible: any Mach disks or other discontinuities inside the flame are not visualized due to the hot plume surrounding it (Fig.5.8). A reasonable explanation for this is that the density distribution is not uniform especially near the exhaust flow with high density gradient in that area: the plume has higher density in comparison with the inner flow and that means it covers all the phenomena related to the lower density of the real exhaust flow.
A confirmation of this problem is given also by the CFD results and the Fig.5.9, in which we see the a density distribution which is higher in the border region of the jet than inside of it.

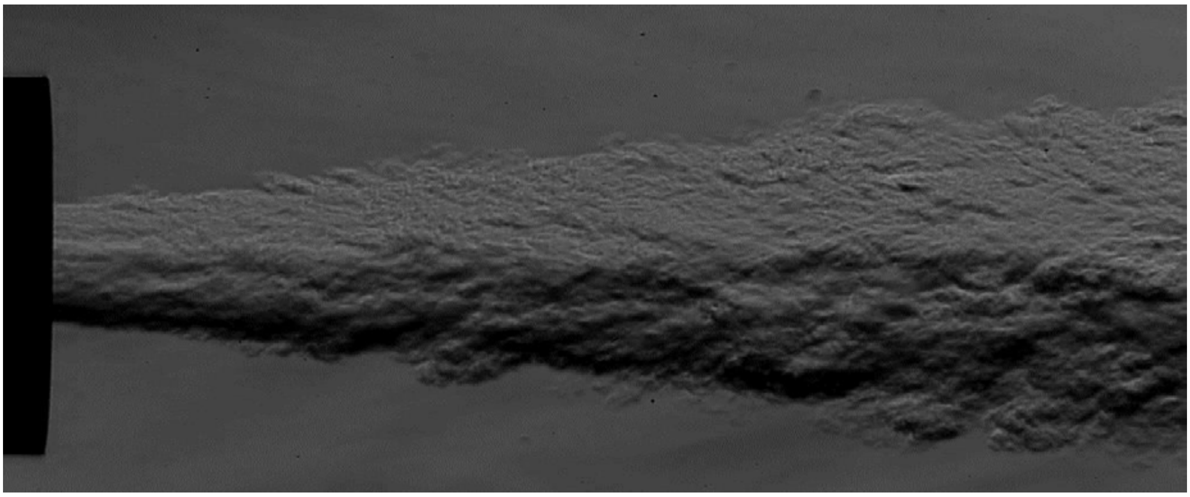

Fig.5.8 Schlieren image of the exhaust

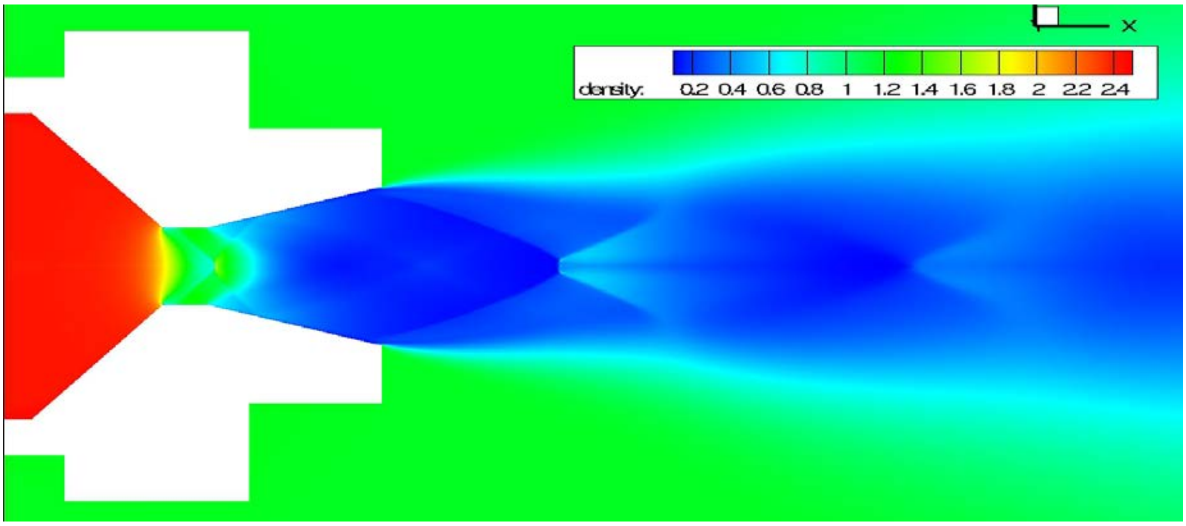

Fig.5.9 CFD simulation: density distribution

6. COMPARISON: TEST VS CALCULATION

The final comparison between the experimental results and CFD calculation is carried out at the end of the test campaign. The sensors used for the experiment produce values useful as input for the CFD simulation: it is a kind of continuous swap between the numerical and test results, using the latter ones as input for the first in a circle on-going update.
Among the basic parameters measured during the tests, the comparison is led in terms of the thrust generated by this engine.
Considering the Tau Code calculated thrust (Chap.4.4) of 15.15 N and the one provided by the sensor, equal to 23.7 N, an error of about 35% is experienced.
The experimental thrust, defined by the reference point chosen from the experimental test, is not so in accordance with the theoretical one: reason for this discrepancy could be the hypothesis of frozen combustion in CFD simulation, while the combustion in the real phenomena is in equilibrium generating different values of thrust.
Anyway it has to be kept in mind that the environmental conditions (reference temperature and pressure) during the experimental tests are variable and different from numerical hypothesis considered stable.
Considering these, it is probably also necessary to find a better reference point on the thrust curve (Fig.5.4), chosen in correspondence of a temperature stable along the process.

7. CONCLUSION

The students have the opportunity to manage with all the issues related to the test management procedures such as risk management, test readiness validation, failure management, with particular attention to test documentation such as test plan, test request, measurement requests and test execution procedures.
A good knowledge of how to use the test rig as well as the control and command system of the rig and specimen and the measurement acquisition, recording and processing system and principles and standards for rocket engine test facilities are acquired.
At the end of this study several considerations about the results are possible which stress the good results of this research.
The slight difference of the numerical results (Chap. 4.4) bares a good accordance between NASA CEA and CFD simulations in terms of basic thruster performances. A further and better confirmation of the goodness of the numerical model is provided by the infrared imagery which confirms the presence of the same Mach field in the exhaust flow during the experiment and what has been investigated numerically.
From the other hand, the results in terms of thrust are not so close together and the schlieren visualization does not provide useful information about.

8. OUTLOOK

Continuous improvements are always possible in the engineering and science and no work could be considered perfect.
There are a lot of possibilities to improve and get better the system here studied: from a better calibration of the temperature sensors to reduce the noise and accelerate its response to the choice of a new reference point in the curve registered.
Currently the most critical aspect of the micro hybrid thruster is the ignition system [4]: this is the main reason of the failures during the test campaign which boast about 85% of success, and a more reliable and satisfactory ignition system could improve this percentage.

The CFD model at the moment provides good and acceptable results in accordance with the rocket theory but new solution to accelerate the convergence and produce better results is always possible.
A bigger and new generation of thruster is at the moment under study with the aim to generate higher thrust (up to 100N) and this development could be a link to further studies of the exhaust having more information and data to compare with the numerical results. These could be the monitoring of the exhaust flow velocity with PIV technique and a new schlieren visualization with different configurations of components to obtain information of the exhaust.
The test campaign with the next thruster (HT 02.e) will be executed in the recently finished test field M11.5 in the DLR test center of Lampoldshausen. For these campaigns the mobile test bench PVII is under modification in order to enable tests up to the life time limit of the specimen. Due to longer test duration constant and more representative flow conditions are expected. Furthermore the testing rule which calls for an evacuated test position is applied. All tests will be remotely monitored and controlled from a control room outside the safety radius.

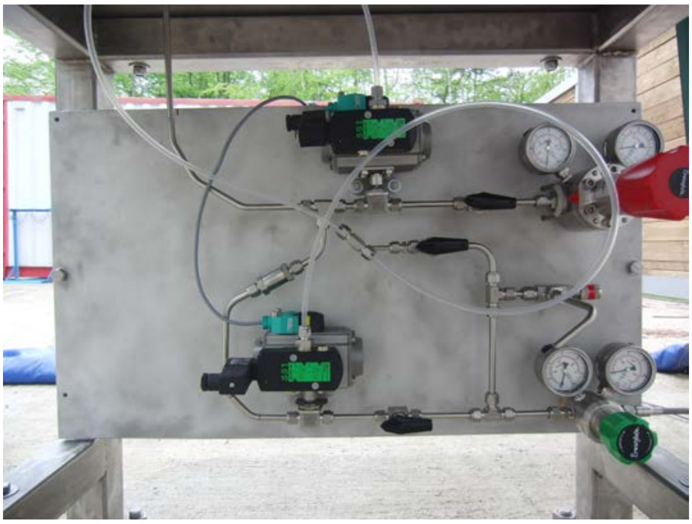

Fig 8.2 Propellant Control Panel for Thruster HT 02.e

Fig. 8.1 Thruster HT 02.e on Test Bench PVII

9. ACKNOWLEDGMENT

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