УДК 531.383

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THE MODEL HYBRID ROCKET MOTOR. COMPUTATIONS, DESIGN AND FIRE TESTS

Introduction

Rocket propulsion is classified into two fundamental types of engines: liquid and solid. Every type includes specific features which make it usable for a space vehicle. Although liquid rocket engines are characterized by high performance and the ability of thrust control, they are complex and expensive. Solid rocket motors are quite simple and not as expensive as liquid rocket engines. However, they are not throttlable, and additionary the attainable specific impulse is significantly lower than in the liquid engine.

We investigated the hybrid rocket engine, which has the main advantages of both liquid and solid propulsion systems. The principal of operation of such engine is that fuel and oxidiser are stored in different phases: solid fuel and liquid oxidiser or inversely. The first option is more commonly used, since solid oxidizers do not have a good mechanical and chemical properties as compare to solid fuels. Positive aspects of this idea is the ability of thrust magnitude control and restart of the motor, which influence on safety and application range. Moreover, the structure of a rocket involving a hybrid motor is not as complex as with liquid engine. It makes the hybrid less expensive and offer better performance then solid rockets .

The present development tendency of a rocket propulsion, apart from cost reduction, requires the application of environmentally friendly propellant. The Warsaw University of Technology and the Institute of Aviation conduct their own research on environmentally safe propulsion systems. The experimental hybrid rocket motor has been designed and manufactured. There have been performed several fire tests in order to verify the main properties of the propellant composition applied in the motor.

Propellant combination

The main disadvantages of a hybrid rocket motor are problems with combustion instability, low burning rate of a solid fuel grain and variation of mixture ratio during operation. These aspects causes that hybrids have never been used as a space propulsion systems. Nowadays a possible solution is investigated. The suggestion is that a small amount of solid oxidiser, integrated with a fuel grain, may increase the propellant regression rate and decrease combustion instabilities.

Considering all essential features of a solid propellant fuel grain, including the chemical equilibrium optimization, the most suitable composition was designed. It contains aluminium powder, ammonium nitrate, HTPB and additives: dioctyl adipate, ammonium dichromate and carbon. Aluminium increases combustion temperature. Ammonium nitrate was chosen because of its nontoxic decomposition products, low cost and high value of heat of decomposition. HTPB is the most common binder applied to solid rocket fuel grain since its high performance. Additives are essential to improve mechanical properties and increase the regression rate.

The target liquid oxidiser is 90% hydrogen peroxide or liquid oxygen. However, to simplify and advance the initial research, several fire tests were performed with application of gaseous oxygen.

Thrust chamber structure and geometry

One of the main assumptions was to design and manufacture a small experimental hybrid rocket motor (fig. 1). The research also assumed the adaptation of the existing test stand. All these conditions were defined to minimize the overall research cost and keep safety at highest possible level. The thrust chamber was made of carbon steel. Internal dimensions of the chamber were constrained not to exceed the capability of the test stand. Combustion chamber internal diameter equals 50 mm and the length is 80 mm. These dimensions determine the solid fuel grain geometry.

Wall thicknesses of all elements were over-dimensioned as to prevent possible exceeding the yield point of the material at high temperature. Although the motor case is massive, it has no significant influence on thrust measurement.

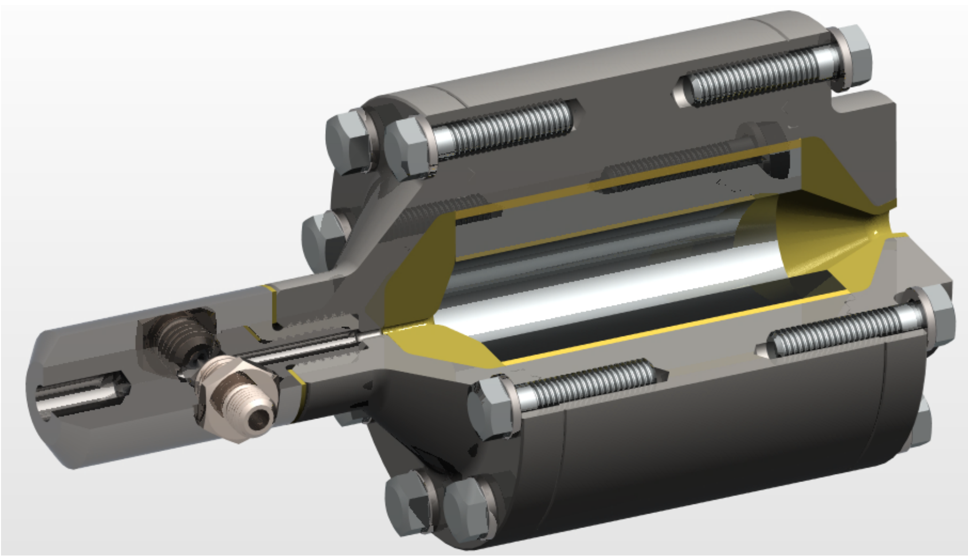

Fig. 1. Hybrid rocket motor structure

Thermodynamics and performance calculation

In order to design the rocket motor nozzle geometry and to estimate thrust magnitude, it is essential to make general assumptions. The assumed chamber pressure is 1 MPa. As a result, the chamber to exit pressure ratio is about 10. The estimation of propellant performance was made using the CEA code (Chemical Equilibrium with Applications). The program makes possible to calculate significant parameters as: specific impulse, temperature, thrust coefficient, nozzle area ratio. It also helps to optimize propellant composition. Calculated specific impulse value is 2030 m/s. The estimation of thrust magnitude requires also the information about propellant mass flow rate. It mainly depends on the solid fuel burning rate and the combustion area. The assumed burning rate value is 0,25 mm/s. Verification of this assumption is expected to be performed after several fire tests. Calculated thrust value is 17,8 N.

Fire Tests

The research stand contains motor test stand and data registration system (fig. 2).

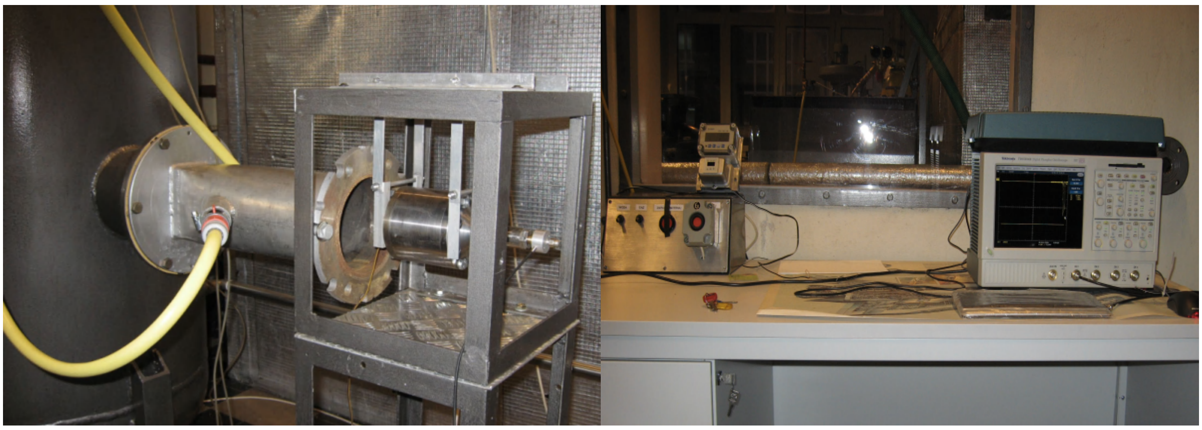

Fig. 2. The motor test stand and data registration system

The data acquisition system enables all necessary measurements to be conducted simultaneously. However, only thrust was measured during the earliest motor tests. Following experiments also included thrust chamber pressure measurement. A few fire tests have been carried out so far using gaseous oxygen as an external oxidiser. The change of the thrust in time was measured and may be seen on (fig. 3).

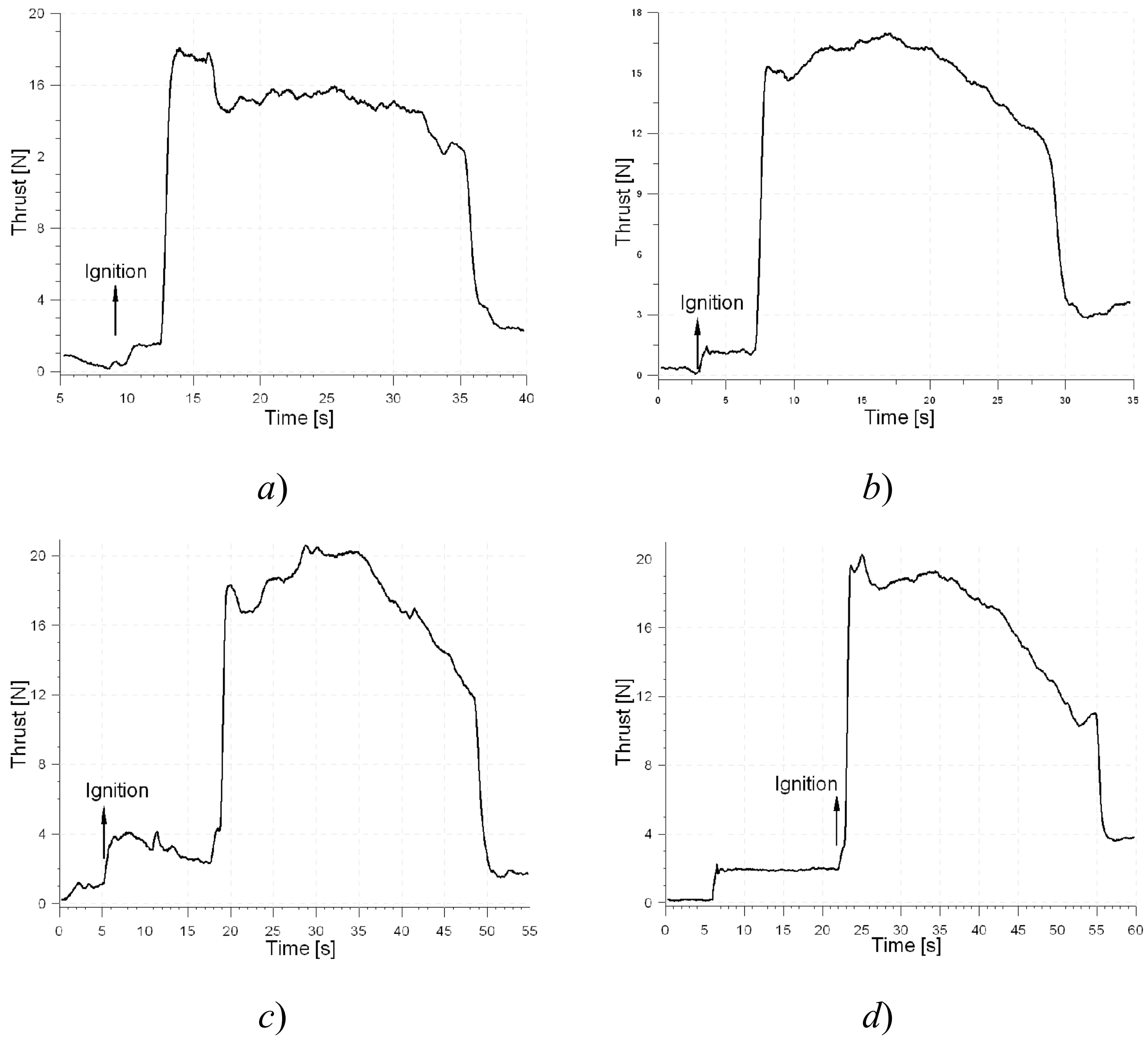

Fig. 3. Thrust measurement

As far as the thrust values are concerned there is a little difference between results of the experiment (fig. 4) and initial computations. The measured thrust was within range of 15 to 20 N (calculated value was 17,8 N). Low value of the thrust just after the ignition was the result from a small amount of oxidiser injected to combustion chamber. After increasing oxygen mass flow rate the thrust of motor was sharply increased and was staying on a high level till time when the feeding oxidiser had been cut off. Furthermore, the lower pressure in combustion chamber from initially assumed and differences in real burning rate of solid propellant in relation to calculated burning rate, may be classified to the reasons of thrust force variations. The chamber pressure was measured with the piezoelectric sensor (water cooled) and pressure values were in the range of 0,4 – 0,7 MPa. Moreover, the nozzle was cooled by water and had passed all tests very well.

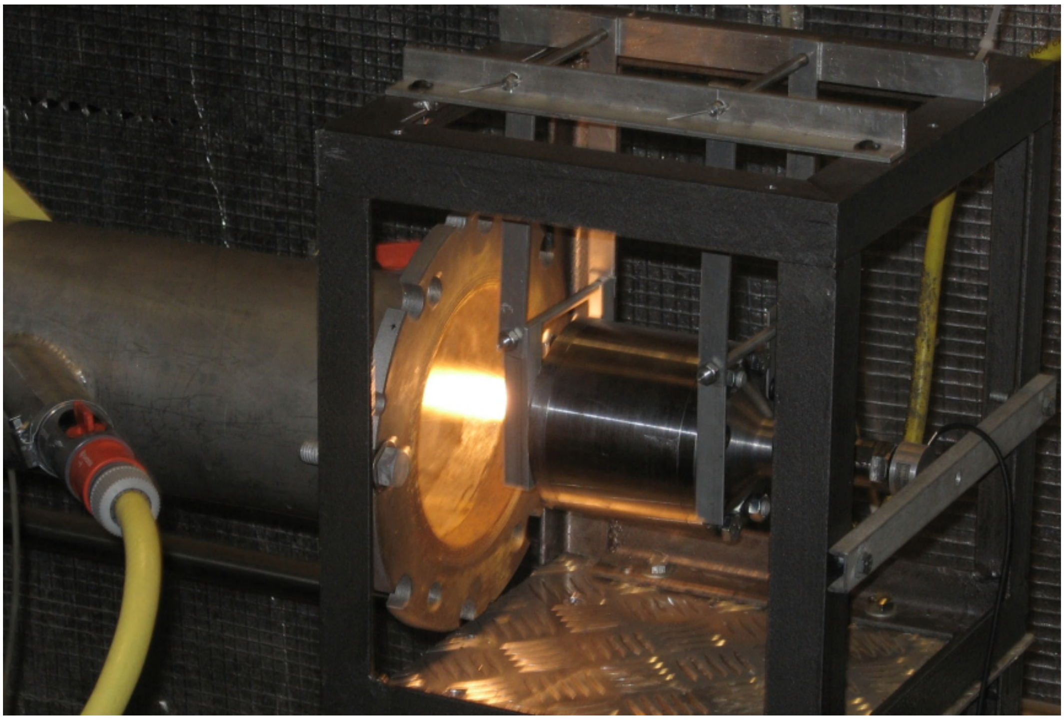

Fig. 4. Motor fire test

Conclusion

The experimental hybrid rocket motor was designed and tested. The motor utilize aluminium with HTPB as a fuel. Also addition of small amount of ammonium nitrate to the fuel was tested. As the oxidizer gaseous oxygen was used. The initial fire tests have been performed and the correctness of research conception has been confirmed.

However, further research and development of a hybrid motor require better understanding of the main combustion phenomena. It is necessary to control internal ballistics processes, such as a solid fuel regression rate and mixture formation. Solving these problems is only the first step to develop of a hybrid propulsion system.

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