TRANSACTIONS OF THE INSTITUTE OF AVIATION ISSN 0509-6669
No. 3(240), pp. 7-17, Warsaw 2015 eISSN 2300-5408
DOI: 10.5604/05096669.1194936

DEVELOPMENT AND DESIGN OF A 10 N MONOPROPELLANT THRUSTER

SAMANTHA BOOTH, JORDAN KENTON, PATRICK NIENHAUS, ERICH ZAHN

Institute of Aviation, al. Krakowska 110/114, 02-256 Warsaw, Poland
The Ohio State University, Columbus, Ohio 43210, U.S.
booth.202@osu.edu, jordanbkenton@msn.com, nienhaus.3@osu.edu, zahn.12@osu.edu

Abstract

Monopropellant thrusters are commonly used on spacecraft for a variety of purposes, most frequently for maintaining the spacecraft's orbit. A majority of monopropellant thrusters available and in use today are hydrazine engines. Hydrazine is toxic, unsafe, and not environmentally friendly. An alternative propellant is hydrogen peroxide; it is environmentally friendly, non-toxic, cheaper, and easier to store than hydrazine. This paper covers the design process of a 10 N monopropellant thruster that uses hydrogen peroxide. A thruster capable of low thrust is preferred for orbit keeping; not much power is needed for maintaining orbit, and low thrust engines are smaller and lighter.

Keywords: monopropellant thruster, hydrazine, hydrogen peroxide.

1. INTRODUCTION

Most modern spacecraft in use today, from LEO satellites to planetary explorers, need some way to generate thrust. Thrust is needed for station keeping, soft landings, de-orbiting, and other maneuvers. Monopropellant thrusters are convenient for these types of applications. The theory and mathematics behind developing rocket engines is well-established and most of the design can be created by using any rocket propulsion textbook; however, despite the simplicity of design, one aspect has remained constant over the years. The propellant used for these types of engines has mainly been hydrazine. Hydrazine and hydrazine derivatives have been used as rocket fuel for decades, and consequently these fuels have developed a strong foothold in the industry. However, hydrazine is a danger to the environment and most importantly, a danger to the people who handle it. It is toxic and a known carcinogen, making it dangerous and expensive to handle.

This 10 Newton monopropellant thruster was developed to use hydrogen peroxide. 98% hydrogen peroxide, or high-test peroxide (HTP), is becoming more popular for use as a fuel. Called Type 98, 98%+ HTP is the highest grade able to be manufactured [13]. It is much safer than hydrazine, cheaper, and non-toxic. HTP was the best candidate to make a "green" monopropellant engine. Once the propellant was decided, the rest of the design revolved around those properties. NASA's Chemical Equilibrium with Applications program, or CEA, was used to analyze the decomposition reaction of the HTP. The program provided the information necessary to develop the rest of the design.

2. FUEL

Hydrogen peroxide has many advantages for use as a monopropellant. The chemical decomposes into steam and oxygen gas and is able to decompose spontaneously at high temperatures or with a catalyst [4]. Additionally, the reaction produces high exhaust temperatures of gases with low molecular weights. Hydrogen peroxide has a high density, low vapor pressure, is non-toxic, non-corrosive, and non-reactive [4][12]. Additionally, it remains in a liquid state through a wide range of temperatures and at ambient temperature, making the chemical easier to handle [4][8]. HTP has desirable chemical properties and its exhaust gases are one of the safest of the commonly used monopropellants. With respect to safety, while non-toxic, HTP can cause irritation. Small droplets are quoted by researchers to be the biggest problem due to their unnoticeable yet painful nature [8]. At the same time, these droplets are easily prevented from making contact with skin. Based on this particular characteristic, when compared to other chemicals like hydrazine, accidental release protocols for hydrogen peroxide are significantly less restrictive. The release response for HTP is to rinse the affected areas with water [15]. The environmental effect of hydrogen peroxide exhaust gases, when compared to gases like chlorine, is totally benign [9]. Additionally, hydrogen peroxide has a very low storage hazard class, especially when compared to hydrazine. Michael Carden of SpaceX states that "some view hydrogen peroxide as an unstable and dangerous propellant, but when handled properly, it is safer than most other options" [3]. Carden also goes on to claim that hydrogen peroxide is able to be stored in plastic containers and if stored in more long term containers, can last a "few decades" with stabilizers.

Most of the disadvantages of working with HTP deal with the lower specific impulse. HTP is quoted to have a specific impulse of approximately 160-195 seconds [4]. Despite the fact that hydrogen peroxide has a 20% lower propulsive performance than hydrazine in particular, its significantly higher density effectively counteracts its lower specific impulse. Based on its higher volume specific impulse, hydrogen peroxide is generally recommended for use in systems with great aerodynamic drag losses or volume constraints.

3. MATERIALS

The materials studied for use in the design of this engine were compared using several criterions: compatibility with HTP, material properties, ease of fabrication, and material cost. Each component was then assessed for what would be its key design parameter. The reaction chamber would need to have favorable high-temperature characteristics as well as reasonable strength properties. The nozzle would need to have good strength properties as well as good high-temperature characteristics, though not as high as the reaction chamber. Using compatibility tables for HTP, a large list of materials was compiled to be looked at in more detail. The materials chosen to be used were steel 316 and Inconel 625.

Both of these materials are compatible with HTP and have very strong mechanical properties [5][7]. However, Inconel has superior heat characteristics over steel. Inconel and steel both melt at near the same temperature but Inconel maintains its strength for much higher temperatures, which is why it was chosen for use in the reaction chamber and nozzle sections. Though Inconel is more difficult to work with because of its high heat strength, both materials are easily manufactured and are quite cheap [6][11][16].

3.1. Thermal Analysis

A thermal analysis was needed to understand the temperature distribution and heat flow throughout the body. This is important for components surrounding the engine as well as for the engine itself. Components near the engine that are temperature sensitive must be considered and kept below their given critical temperature level. The engine can be adversely affected by severe temperature gradients as well. Metals, when subjected to large temperature gradients, will expand on the heated face more than the cooler side which can lead to severe internal stress, potentially causing failure. An analysis of thermal effects is necessary to understand how a body will cope with its loading and to avoid problems resulting from extreme temperatures.

In order to start the thermal analysis of the engine, the test chamber was simplified to a cylinder. Bartz equation was then considered to approximate boundary conditions and thermal loads.

$$h_g = \left[\frac{0.026}{(D^*)^{0.2}} \left(\frac{\mu^{0.2} C_p}{Pr^{0.6}} \right)_0 \left(\frac{P_0}{C^*} \right)^{0.8} \left(\frac{D^*}{r_c} \right)^{0.1} \right] \left(\frac{A^*}{A} \right)^{0.9} \sigma \tag{1}$$

$$\sigma = \frac{1}{\left[\frac{1}{2} \frac{T_{wh}}{T_{0g}} \left(1 + \frac{\gamma - 1}{2} M^2 \right) + \frac{1}{2} \right]^{0.8-0.2\omega} \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{0.2\omega}} \tag{2}$$

h_g = convection coefficient; Pr = Prandtl number; C^* = Characteristic velocity; D^* = Throat diameter; r_c = Throat radius curvature; P_o = Back pressure; A^*/A = Throat to exit area ratio; μ = Viscosity; σ = Correction factor; γ = Heat capacity ratio; M = Mach number; T_{wh} = Temperature of the wall being heated; T_{0g} = Temperature of the hot gas; ω = Value from the viscosity-temperature equation.

Since the contents of the chamber were a mixture of gases from the decomposition process, values for viscosity and the heat capacity ratio had to be computed based on individual material properties and their percentage composition. After these values were found, h_g could then be estimated and used to analyze the setup.

Siemens NX 9 with the Nastran solver was used to set up a thermal analysis by creating a cylinder, approximating the reaction chamber, and setting its material properties to that of Inconel 625. Then, heat loading and convective dissipation parameters were set as well as temperature conditions.

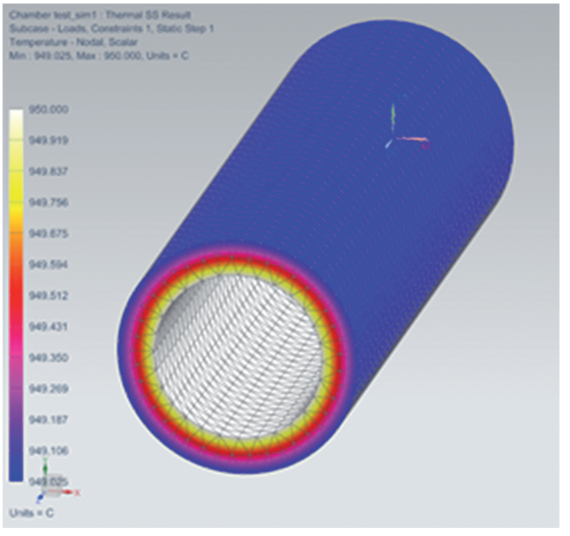

Figure 1. Results of Thermal Analysis

Several different analyses were performed with varying levels of accuracy for the shape of the test section as well as differing values for temperatures and loadings. This gave an overall picture of the thermal processes taking place throughout the thruster. With this understanding, decisions could be made about wall thickness, distance from valve, and other temperature sensitive variables.

4. DESIGN

The design of the engine consists of 4 major parts: the venturi nozzle, the shower plate, the decomposition chamber, and the nozzle. The first component after the valve is the venturi nozzle, which adjusts the mass flow through the system. The shower plate is the entrance to the decomposition chamber, and it spreads out the HTP for a uniform flow. The decomposition chamber holds the catalyst fixed bed. This is where the HTP decomposes into steam and oxygen gas through the use of manganese oxide as a catalyst, and is also the point of maximum temperature. The nozzle condenses and then expands the decomposed HTP, creating the thrust. Each component's design is described in greater detail in the following sections. The assembly is shown below.

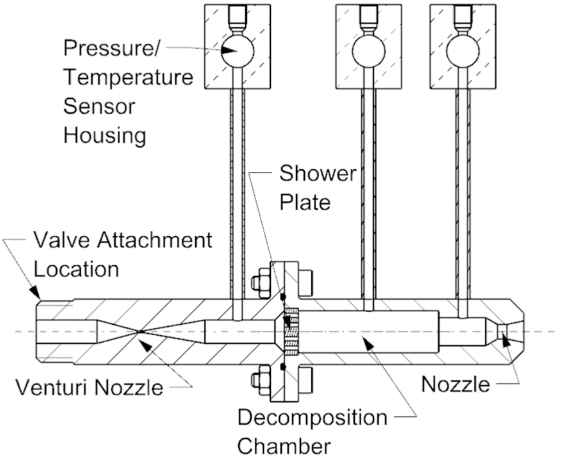

Figure 2. Assembly of Thruster

4.1. Venturi Nozzle

As seen in figure 2, the inlet section of the nozzle is screwed into the end of the thruster valve. The Venturi nozzle was threaded to fit the valve. The rest of the length of the inlet section was used to prevent excessive heat from the decomposition reaction reaching the valve.

A 30-degree inlet angle and a 20-degree outlet angle were calculated, with the connection to the valve as the inlet and the connection to the decomposition chamber as the outlet. Based on the parameters of the valve, the inlet diameter was 3.9688 mm, and the outlet diameter was 9 mm, based on the geometry of the decomposition chamber. The throat diameter was 0.4 mm, and the length of the throat was 1.4 mm. The required inlet and outlet angles and diameters were held constant throughout the design of the Venturi nozzle, making it possible to use trigonometry to determine the necessary length. Based on this calculation in addition to the length of the inlet section and the connection to the decomposition chamber, a length of 53 mm was determined for the venturi nozzle.

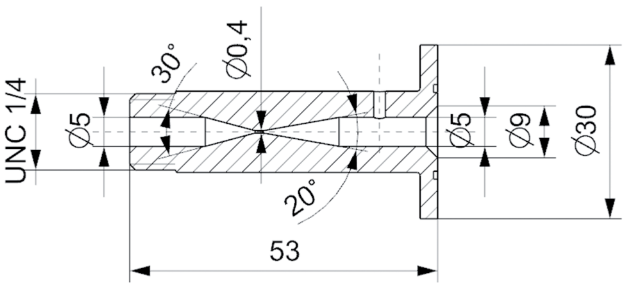

Figure 3. Drawing of Venturi Nozzle

The Venturi nozzle will be connected to the decomposition chamber using 6 screws, seen in figure 2. In order to avoid leakage through the plate a 14 x 1 mm O-ring will be placed in the divot.

4.2. Shower Plate

The design of the injector is a crucial part of any rocket engine. A poor design can lead to an inefficient engine reducing the total impulse and thrust, thus it is important that the design fit what the specific engine needs to function properly. Since this engine does not need to atomize the propellant before it reaches the catalyst, a shower head design was implemented. The shower plate, or faceplate, is 10 mm in diameter. The diameter was determined by how the thruster was to be assembled as a whole. The faceplate needed to sit in between the venturi nozzle and the catalyst bed. To make the assembly of the thruster easier the faceplate will be housed just before the catalyst bed in a smooth hole just large enough to fit the part. This hole will be 3 mm in length, the same length as the faceplate. Sitting in this extra hole the faceplate will remain stationary and will keep the catalysts inside the catalyst bed. The faceplate can be easily removed to inspect and replace the catalyst after testing. The sizes of the holes in the faceplate were determined based on cost to manufacture and the size of the pellets used to catalyze the propellant. Each pellet was about 3 mm in diameter, and the cost of machining holes increases the smaller the hole is. Keeping this in mind the holes were chosen to be 0.9 mm in diameter. This size also allows for several holes to be placed over the surface of the faceplate to allow for a more even distribution of the propellant over the area of the catalyst bed. The design is shown in figure 4. Several displacement and stress analyses were run on older shower head designs with more and smaller holes to determine if the faceplate could handle the pressure loads it will experience during operation.

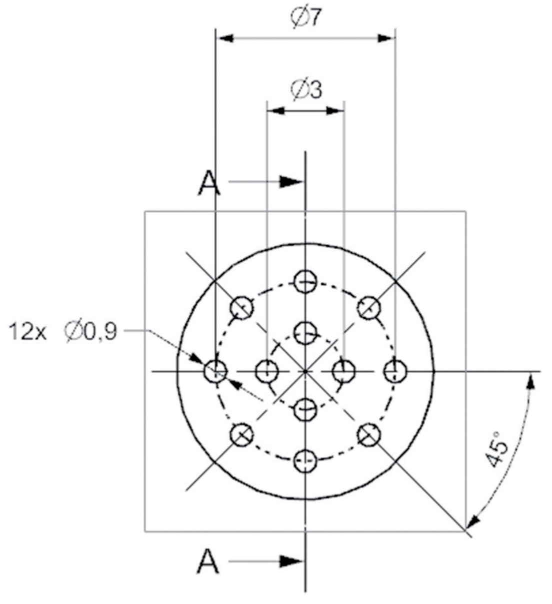

Figure 4. Drawing of Shower Plate

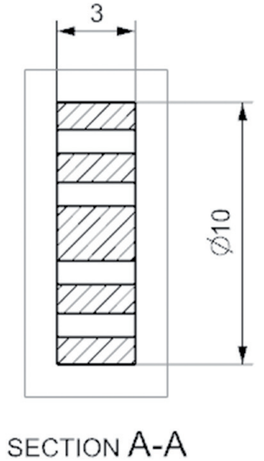

Figure 5. Drawing of Shower Plate Cut-Away View

4.3. Decomposition Chamber

The diameter of the catalyst bed and decomposition chamber was determined based on the following equations:

$$A_{cp} = \frac{\dot{m}}{\phi}$$

$$D_{cp} = \sqrt{\frac{4}{\pi} A_{cp}}$$

A_{cp} is the area of the catalyst bed, $\dot{m}$ is the mass flow, determined by NASA CEA to be 0.0078 kg/s; ϕ is the flux, determined by the Institute of Aviation research team to be 150 kg/s*m^2; and D_{cp} is the diameter of the catalyst bed. Based on the known information, a desired diameter of approximately 9 mm was determined.

In order to determine the L/D (length to diameter) ratio needed to accomplish the optimal 95% decomposition of HTP, a study of comparable tests was conducted. These tests were conducted using the pellet and/or gauze catalyst bed design, as is intended for the thruster the group is designing. The flow rate and corresponding L/D ratio of these studies were graphed and an L/D ratio of approximately 3 was optimal for the desired mass flow of 0.0078 kg/s. Based on this L/D ratio and the known diameter of the catalyst bed, a bed length of 30 mm was chosen.

4.4. Nozzle

The design of the nozzle starts with the two following equations [10]:

$$c^* = \frac{P_c A_t}{\dot{m}} = \frac{\sqrt{\gamma R T_c}}{\gamma \sqrt{\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}}$$

$$C_F = \frac{F}{P_c A_t} = \sqrt{\frac{2\gamma^2}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \frac{P_e - P_\infty}{P_c}\frac{A_e}{A_t}$$

The two equations above help define the two most important nozzle parameters: throat area and area ratio. C^* is a function of the propellant properties and C_F is a function of the geometry of the nozzle. Knowing properties of the propellant from the NASA CEA calculations, the ideal throat area can be calculated. Anticipating a chamber pressure of 10 bar, a c^* of 1016 m/s (given from CEA simulations) and a mass flow rate of 0.0078 kg/s (calculated from an expected I_{SP} of 130 s and desired 10 N thrust), the throat radius can be found to be about 1.55 mm.

Determining the area ratio is dependent on the application. For a prototype to be tested at ground level, a large area ratio is unnecessary. For testing purposes and to verify the design, it is much simpler to use an area ratio that will produce ideally expanded flow at the nozzle exit. In order to find this area ratio, the following equation is used [10]:

$$\frac{A_t}{A_e} = \left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}}\left(\frac{P_e}{P_c}\right)^{\frac{1}{\gamma}}\sqrt{\frac{\gamma+1}{\gamma-1}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

The above equation depends on the pressure ratio. Again, chamber pressure is assumed to be around 10 bar. For ideally expanded flow, the exit pressure is equal to the ambient pressure.

The test chamber has an ambient pressure of about 1 bar; using this ratio, an area ratio (exit over throat) of 2.11 is calculated. This makes the exit radius about 2.30 mm.

The next step is to design the contours connecting the decomposition chamber to the throat and the throat to the exit. The region from the chamber to the throat, the converging section, is all subsonic flow. Therefore, any losses are quite small, and the half angle could range anywhere from 20-45 degrees [1]. For the region from the throat to the exit, the diverging section, 2 possible methods can be used. The first method is similar to the converging section; a simple, conical shape with a half angle from 12-18 degrees [1]. The advantage to this method is that it is simple to design and manufacture. The disadvantage is that on large thrusters, this method is inefficient and can lead to significant losses. Also, the flow can exit the nozzle non-uniformly. The other method is the optimum bell shape design. This design was first introduced by G.V.R. Rao, and utilizes the method of characteristics to optimize the length of the nozzle, while still having uniform exit flow [10]. For this thruster, which is quite small, an optimum bell shape is unnecessary.

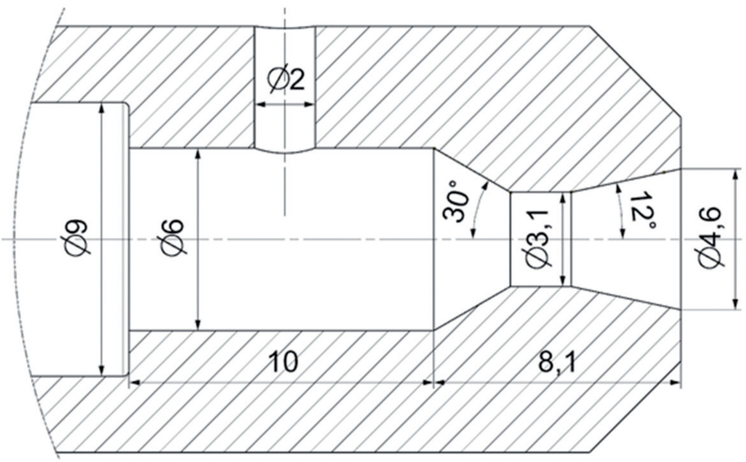

Figure 6. Drawing of Nozzle

The figure above shows the final design of the nozzle. On the left end, the change in diameter from 9 mm to 6 mm indicates where the decomposition chamber ends and the nozzle begins. The diameter change creates a 1.5 mm lip. This lip is there to give a space for a 9 mm diameter steel mesh to sit. The main purpose of the mesh is to keep the pellets inside the decomposition chamber, and from exiting through the nozzle. After the decomposition chamber, there is a 10 mm cylindrical section. The purpose of this is to get pressure and temperature readings from the flow before it enters the converging section. The 2 mm diameter vertical hole is where the pressure and temperature sensor housing will weld. The housing is large in comparison to the thruster, so there had to be enough room for all three housings to fit. The converging section starts with a 30° half-angle cone, shrinking down to the 1.55 mm throat, and finally expanding again through a 12° half-angle cone to the exit. The length of the nozzle is just over 8 mm.

5. EXPECTED RESULTS

The difficulty of this design is in the manufacturing. For key parameters such as the throat and exit diameters, hundredths of a millimeter can influence the thrust level. However, manufacturing hundredths of a millimeter accurately is not reliable and can be expensive.

Additionally, during testing, the heat will expand the nozzle, affecting the performance. Another area of concern is with the mesh right before the nozzle. If this mesh fails, the catalyst will shoot out the back of the nozzle, which is a significant problem. Earlier versions of the design accounted for this issue by welding the mesh into place and welding the nozzle onto the chamber; however, this design was scrapped to simplify the welding and manufacture process. Another area of concern is the o-ring sealing the connection from the venture nozzle to the decomposition chamber. For longer tests, the o-ring could melt, which could lead to pressure losses and performance losses.

There are also areas that will have losses that are hard to account for without test data. For example, the lip from the end of the decomposition chamber to the nozzle. A sudden change in shape like that will have some energy loss. Also, the 10 mm cylindrical section that the flow travels through before entering the converging section is effectively a second "chamber", in which the flow will lose some heat.

Ideally, the thruster will reach values close to what is expected: 10 N of thrust and a specific impulse of 130 s. Any problems can be addressed in another iteration of design, with the help of test data. If a thruster this small using an environmentally friendly propellant is successful, it would be beneficial for satellites. Especially small satellite missions, which are becoming more and more popular [2].

6. CONCLUSION

Detailed above is the in depth characteristics of the 10 N Monopropellant Thruster designed for the Institute of Aviation in Warsaw through their joint internship program with members of The Ohio State University.

The different aspects of the thruster can be summarized to acquire a more general idea of the design. This thruster was designed to use 98% Hydrogen Peroxide. This propellant is relatively safe to handle and work with. The Institute also prepares the propellant in house making it cheaper to use than if they had to purchase it from an outside dealer. 98% HTP is also one of the cheapest monopropellants available today; this coupled with the high chemical density is why 98% HTP was the best choice for this engine. The materials were then chosen based on their compatibility with HTP. It was determined that Inconel 625 was the best match for this thruster because it should handle the heat of the decomposition reaction and holds its strength at high temperatures. Inconel 625 is also nonreactive with HTP.

The overall design of the thruster follows a similar design to most low powered thrusters. From back to front the valve controls the flow of the propellant to the venturi nozzle which corrects the mass flow through the shower plate. The shower plate then distributes the propellant over the decomposition chamber. The propellant then decomposes and is accelerated out the converging diverging nozzle to create the 10 N of thrust.

If this thruster is to be used on an actual satellite, then a few things will change. First, the specific impulse will increase in a vacuum. This will affect the mass flow required for 10 N of thrust, and in turn, affects the throat area of the venturi nozzle and the end nozzle. More importantly, flow exiting a nozzle into a vacuum can never be ideally expanded; therefore, the larger the area ratio, the larger the range of possible thrust values. The actual area ratio on a practical design can be dependent on the application. There are practical limits to this however. A large exit area means a larger nozzle, and more weight. Also, if a nozzle is too large, condensation can occur near the exit. This will lead to losses in performance [10].

REFERENCES

[1] Braeunig, R. A. (2012). Basics of Space Flight: Rocket Propulsion. *Basics of Space Flight: Rocket Propulsion*. Web: http://www.braeunig.us/space/propuls.htm.

[2] Buchen, E., and DePasquale, D. (2014). Nano/Microsatellite Market Assessment. SpaceWorks Engineering. Web: http://www.sei.aero/eng/papers/uploads/archive/SpaceWorks_Nano_Microsatellite_Market_Assessment_January_2014.pdf.

[3] Carden, M. Is Hydrogen Peroxide the Best Rocket Fuel? *DIY Space Exploration*. Web: http://www.diyspaceexploration.com/hydrogen-peroxide-best-rocket-fuel/.

[4] Cervone, A., Torre, L., D'Agostino, L., Musker, A. J., Roberts, G. T. , Bramati, C., and Saccoccia, G. (2006). Development of Hydrogen Peroxide Monopropellant Rockets. AIAA. Web: http://www.esa.int/gsp/ACT/doc/PRO/ACT-RPR-PRO-JPC2006-HP%20Rockets%202006-5239.pdf. *42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit.* AIAA, 9-12 July 2006. Sacramento, California. Web: http://arc.aiaa.org/doi/abs/10.2514/6.2006-5239.

[5] Cole-Parmer – Your Technical Experts in Fluid Handling, Electrochemistry, and Laboratory Equipment. *Cole-Parmer US*. Web: http://www.coleparmer.com/.

[6] MEPS (INTERNATIONAL) LTD., Independent Steel Industry Analysts, Steel Prices, Steel Publications, Steel News, Steel Data, Steel Consultancy, Steel. *MEPS (INTERNATIONAL) LTD., Independent Steel Industry Analysts, Steel Prices, Steel Publications, Steel News, Steel Data, Steel Consultancy, Steel*. Web: http://www.meps.co.uk/.

[7] Online Materials Information Resource – MatWeb. *Online Materials Information Resource – MatWeb*. Web: http://www.matweb.com/.

[8] Rarata, G. 98% Hydrogen Peroxide for Propulsion, 98% HTP (High Test Peroxide). The Institute of Aviation, Warsaw. Web.

[9] Rusek, J. J. (2004). Hydrogen Peroxide for Propulsion and Power Applications: A Swift Perspective. *Proceedings of the 2nd International Conference on Green Propellants for Space Propulsion (ESA SP-557)*. *SAO/NASA Astrophysics Data System (ADS)*. Web: http://adsabs.harvard.edu/full/2004ESASP.557E...1R. Proceedings of the 2nd International Conference on Green Propellants for Space Propulsion (ESA SP-557). 7-8 June 2004, Chia Laguna (Cagliari), Sardinia, Italy.

[10] Sutton, George P., and Biblarz, O. (2001). *Rocket Propulsion Elements*. 7thed. New York: John Wiley & Sons. Print.

[11] Vacuum Grade Super Alloy Scrap Page. N.p., n.d. 12 June 2015. Web: http://www.metalprices.com/metal/super-alloys/super-alloy-inconel-625.

[12] Ventura, M., and Mullens, P. (1999). The Use of Hydrogen Peroxide for Propulsion and Power. *General Kinetics, LLC.* 19 June 1999. Web: http://www.researchgate.net/publication/268456496_The_use_of_hydrogen_peroxide_for_propulsion_and_power. *35th Joint Propulsion Conference and Exhibit*. Los Angeles, CA, U.S.A, 20-24.06.1999. Web: http://arc.aiaa.org/doi/abs/10.2514/6.1999-2880.

[13] Ventura, M., and Yuan, S. (2000). Commercial Production and Use of Hydrogen Peroxide. *General Kinetics, LLC.* Web: http://www.researchgate.net/publication/269238041_Commercial_production_and_use_of_hydrogen_peroxide. *36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*. 16-19.07.2000, Huntsville, Alabama. Web: http://arc.aiaa.org/doi/abs/10.2514/6.2000-3556.

[14] Wade, M. H2O2. *Encyclopedia Astronautica*. Web: http://www.astronautix.com/props/h2oosene.htm.

[15] Wernimont, E. J. (2006). System Trade Parameter Comparison of Monopropellants: Hydrogen Peroxide vs Hydrazine and Others. *42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit.* AIAA, 9-12 July 2006, Sacramento, California. Web: http://www.hydrogen-peroxide.us/chemical-handling-toxicity/AIAA-2006-5236_hydrogen_peroxide_versus_hydrazine.pdf, http://arc.aiaa.org/doi/abs/10.2514/6.2006-5236.

[16] World Steel Association – Home. *World Steel Association – Home*. Web: https://www.worldsteel.org/.

PROJEKTOWANIE I PRZYGOTOWANIE RAKIETOWEGO SILNIKA KOREKCYJNEGO O CIAGU 10 N NAPĘDZANEGO PALIWEM JEDNOSKŁADNIKOWYM

Streszczenie

Rakietowe silniki korekcyjne napędzane paliwem jednoskładnikowym są powszechnie używane w statkach kosmicznych do różnych celów, głównie do utrzymywania statku na orbicie. Większość dostępnych I używanych obecnie rakietowych silników korekcyjnych to silniki hydrazynowe. Hydrazyna jest toksyczna, niebezpieczna I szkodliwa dla środowiska. Alternatywnym paliwem jest nadtlenek wodoru. Jest on przyjazny dla środowiska, nietoksyczny, tańszy i łatwiejszy do przechowywania niż hydrazyna. Praca ta opisuje proces projektowania rakietowego silnika korekcyjnego o ciągu 10 N, który używa nadtlenku wodoru. Silnik korekcyjny, który wytwarza niską siłę ciągu jest preferowany do utrzymywania statku na orbicie ponieważ nie potrzeba dużej mocy do utrzymania statku na orbicie, a silniki o małym ciągu są mniejsze i lżejsze.

Słowa kluczowe: rakietowy silnik korekcyjny, hydrazyna, nadtlenek wodoru.