Acceleron Aerospace Journal || AAJ.11.2106-2453
Volume 3, Issue 4, pp (525-533)
Research Paper
E-ISSN- 2583-9942 || DOI: 10.61359/11.2106-2453

Specimen Design of Hydrogen Peroxide Systems for Attitude Control

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Abstract: Hydrogen peroxide (HP) is one of the most desirable alternatives to hydrazine and its derivatives, the most common type of fuel utilized in today's rocket fuel industry. Research interest in HP as a rocket propellant has recently increased due to concerns over the carcinogenic nature of hydrazine-based fuels and the dinitrogen tetroxide rocket propellant combination. Hydrazine is a dangerously unstable, highly toxic, and carcinogenic substance, whereas HP is considered a green fuel. HP has a higher specific gravity and lower cost compared to its counterparts. HP cells are made with catalysts such as nickel, platinum, and palladium, and their efficiency is significantly higher than that of other cells. HP can be used as a monopropellant, bipropellant, or hybrid propellant in various compositions. The exothermic decomposition of HP yields water and oxygen. The rate of decomposition depends on factors such as temperature, concentration, pH, and the catalyst used. After treating the water with suitable salts, it can be consumed by astronauts, giving HP an advantage over other propellants.

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1. Introduction

The hypergolic propellant combination of monomethyl hydrazine and dinitrogen tetroxide is highly effective and has a long history of successful use in various launch vehicles. However, even in minute concentrations, these chemicals are hazardous and carcinogenic, both classified as "acute poisons." In recent years, there has been increasing interest in researching potential alternatives to current propellants. Green propellants encompass a family of propellants (solid, liquid, hybrid, mono- or bipropellant) that meet criteria for low toxicity, negligible pollution impact, good storability, broad applications, and stellar performance and efficiency. These propellants have the potential to reduce overall costs while minimizing negative environmental impacts. One of the popular liquid rocket propellants belonging to the green propellant family is concentrated hydrogen peroxide. The exothermic decomposition of hydrogen peroxide produces steam and oxygen. Approximately 47% of the reaction products from anhydrous hydrogen peroxide is oxygen, making its use as an oxidizer (at high concentrations) very efficient. Concentrations of 70% hydrogen peroxide are used in propulsion applications, while concentrations above 85% are frequently utilized in rocket propulsion. Hydrogen peroxide decomposes to form water and oxygen, with a $\Delta H°$ of –2884.5 kJ/kg and a ΔS of 70.5 J/(mol·K):

$$2H_2O_2(conc) \rightarrow 2H_2O_{(vapour)} + O2_{(gas)}$$

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****** Received: 09-September-2024 || Revised: 28-September-2024 || Accepted: 30-September-2024 || Published Online: 30-September-2024.

Another benefit of hydrogen peroxide is its high density, which enhances the volumetric performance of propulsion and power systems that utilize it. Engineers can handle and use hydrogen peroxide safely due to its low toxicity, low irritability, low corrosivity, and minimal volatility, provided they receive the requisite training and adhere to established safety protocols and procedures.

2. Methodology

After thorough investigation, an HP laboratory-scale engine was designed. Some of the specifications considered for the engine include: 1) the propellant used is hydrogen peroxide (H_2O_2) at 90% concentration, and 2) the nozzle entry stagnation pressure is 2 MPa. The theoretical performance parameters of the engine, obtained using the NASA CEC71 program, were utilized for the engine design.

The propellant and pressurizing tanks are designed as spherical vessels. Titanium alloy Ti-6Al-4V is selected for the tanks due to its high strength and corrosion resistance. The pressurizing gases considered are helium or nitrogen. The maximum reservoir pressure of a standard industrial gas bottle is around 20 MPa. For pressure-fed systems, the propellant tanks operate at an average pressure ranging from approximately 1 to 4 MPa.

Two imperative criteria considered for catalyst bed design are: 1) bed loading factor and 2) residence time. A silver screen catalyst bed is selected due to its market availability and its ability to enhance the decomposition of the propellant at high temperatures, which in turn improves engine performance. The loading factor typically averages around 20, while loading factors of 5 to 10 are often used in small diameter chambers for low-temperature starts. Residence times range from 0.001 to 0.04 seconds for various types of thrust chambers and propellants. The material used for the catalyst bed is Stainless Steel 316, due to its excellent corrosion resistance and resistance to oxidation at high temperatures.

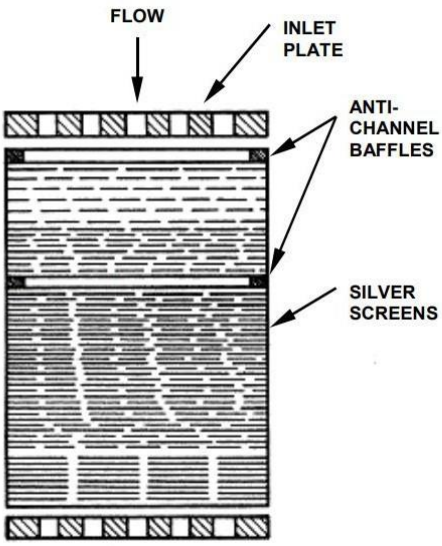

Figure-1 Silver Screen Catalyst Bed [16]

In the combustion chamber of a monopropellant engine, high-pressure conditions are required to accelerate the hot gas mixture. A feed system is necessary to pressurize and transport the propellant from the tank to the combustion chamber, with a pressure drop of 0.7 MPa provided for propellant injection. The orifice diameter for a typical injector ranges between 0.5 mm and 2.5 mm.

A typical conical nozzle is considered for the engine. The half-angle of the nozzle's convergent cone section, denoted as θ_{half}, ranges from 20° to 45°, with a typical value of 30°. The half-angle of the divergent cone, denoted as α_{half}, ranges from 12° to 18°, with a standard value of 15°.

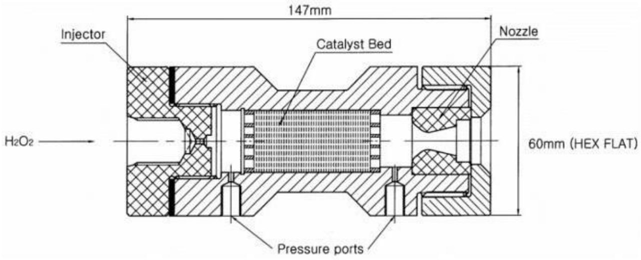

Figure-2 HP Rocket Engine capable of producing 100 N thrust

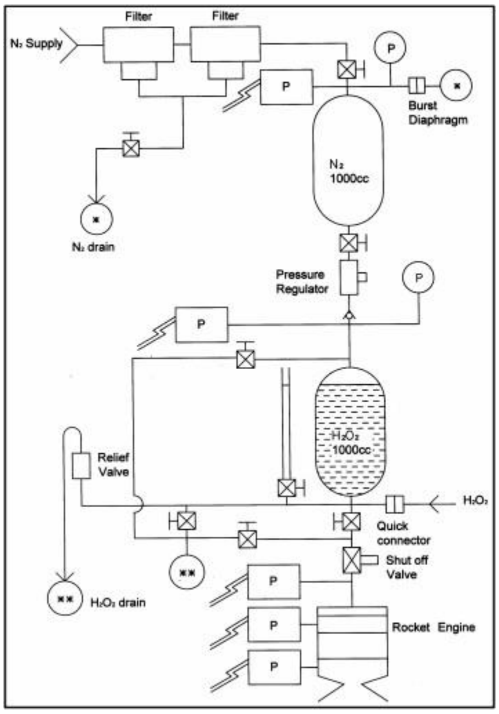

Figure-3. Block Diagram of HP Rocket Engine

3. Propellant Tank Design

The dry mass of the rocket set for vertical firing is 4000 kg, and the theoretical Isp of HTP is found to be 153 seconds. Initially, the propellant mass is calculated using the Tsiolkovsky rocket equation. The required Δu for the vehicle to return from any Near-Earth object is typically 60 m/s.

$$\Delta\mathbf{u} = \mathrm{I_{sp}g_0ln}\left(\frac{m_0}{m_f}\right) \rightarrow \mathrm{m_{prop}} = 163.1968\ \mathrm{kg}$$

Propellant mass fraction, $\zeta = \frac{\mathrm{m_{prop}}}{m_0} = 0.0392$

Volume of propellant, $V_{\mathrm{prop}} = \frac{\mathrm{m_{prop}}}{\rho_{\mathrm{HTP}}} = 0.1125m^3$

Density of HTP at 20^0C=1450 kg$/m^3$

The pressurizing gases are helium and nitrogen. The maximum charging pressure for either gas reaches 20 MPa. In pressure-fed systems, the propellant tanks operate at an average pressure ranging from 1 to 4 MPa, with the propellant tank pressure set at 3 MPa.

Current deep space communication methods rely on Radio Technology and Laser Communication. Due to varying weather and atmospheric conditions, integrating these methods into a single satellite or spacecraft model is recommended. This integration enhances the overall communication standard. Radio Technology is unaffected by weather conditions and can cover large areas with reliable links but has a lower data transmission rate compared to Laser Technology. Conversely, Laser Technology, while offering high data transmission rates, is affected by Clear-Weather Laser Optical Systems (CFLOS) and other atmospheric conditions. When CFLOS is poor, the likelihood of losing the communication link increases. However, these two methods can complement each other, mitigating each other's disadvantages. The main challenge of this symbiotic model is its cost, which may be addressed through global cooperation among countries with advanced communication technologies.

Isentropic condition (He)

$$\mathrm{P_0 V_0}^{\gamma} = \mathrm{P_{prop} V_{prop}}^{\gamma} \rightarrow V_0 = 0.05226\ m^3$$

By ideal gas law,

$$\mathrm{P_0 V_0 = m R_0 T_0 \rightarrow m_{He} = 1.7173\ kg}$$

Similarly for isothermal condition,

$$\mathrm{P_0 V_0 = P_{prop} V_{prop} \rightarrow V_0} = 0.0195\ m^3$$

$$m_{He} = 0.\ 6407\ \mathrm{kg}$$

Similarly, these calculations were performed for $\mathbb{N}_2$ as pressurising gas.

Table. 1. Parameters for pressurising gases

S. No	Pressurising gas	Isentropic		Isothermal	
		Volume (m^3)	**Mass (kg)**	**Volume (m^3)**	**Mass (kg)**
1	He	0.05226	1.7173	0.0195	0.6407
2	N2	0.03841	8.8326	0.0195	4.4844

The propellant and pressuring tanks are taken as spherical vessels. Ti-6Al-4V has high strength and corrosion resistant.

$$V_{prop} = 0.\ 1125\ m^3 \rightarrow D_{prop} = 599.\ 03\ \mathrm{mm}$$

$$\sigma_{\mathrm{yield}} = 880\ \mathrm{MPa},\ \sigma_{\mathrm{perm}} = 440\ \mathrm{Mpa}$$

Thickness of the propellant tank, t_{prop} = 2.402 mm

These calculations were conducted for He and N2 tanks.

Table. 2.Dimensions of pressurizing tanks

	D_{tank} (mm)	t_{tank} (mm)
He	463.86	10.5423
N2	418.5912	9.5134

4. Catalyst Bed Design

A laboratory-scale 100 N HTP (90% concentration) monopropellant rocket engine facility, designed by Ahn Sang-Hee et al., is considered [3]. SS-316, known for its corrosion and oxidation resistance at high temperatures, is used. Taking into account a corrosion allowance of 1.5 mm, the total thickness of the material is set to 2 mm.

Loading factors of 5 to 10 are often used in small diameter chambers for low temperature starts. [2]

$$\mathrm{I_{sp}} = \frac{\mathrm{F}}{\mathrm{g_0 \dot{m}_p}} \rightarrow \mathrm{\dot{m}_p} = 0.06665\mathrm{kg/s}$$

Pack frontal area (m^2) = $\frac{\mathrm{m_p\ (kg/s)}}{\mathrm{Loading\ factor}}$[2]

Taking loading factor as 20,

$$\text{Packing frontal area} = 0.003333\ \mathrm{m}^2$$

∴Diameter of the pack=65.138 mm

$$(\mathrm{T_{cc}})_{theo} = 1029.54\ \mathrm{K}\ [3]$$

$$\eta_c^* = 0.90[3]$$

CC temperature,

$$\mathrm{T_{cc}} = (\mathrm{T_{cc}})_{theo} \times \eta_c^{*2}[3]$$

$$T_{cc} = 1029.54 \times 0.9^2\ \mathrm{T_{cc}} = 833.9274\ \mathrm{K}$$

$$\Delta t = 1 - 40\ ms\ [1]$$

Assuming residence time (Δt) of 15 ms, Catalyst bed length, $L_{cb} = \frac{R_u}{M}\frac{m_p T_c \Delta t}{\frac{\pi}{4}d^2 P_0}$ [3]

$$L_{cb} = 47.0476\ \mathrm{mm}$$

Considering a margin of 20% in order to maximise the combustion process,

$$\mathrm{L_{cb}} = 56.4571\mathrm{mm}$$

$$\sigma_{\mathrm{yield}} = 290\ \mathrm{MPa},\ \sigma_{\mathrm{perm}} = 145\ \mathrm{MPa}$$

Thickness of the catalyst bed, $t_{cb} = 0.4492$ mm

Corrosion allowance of 1.5 mm was provided.[6]

∴Total thickness=0.4492+1.5=1.9492 mm

5. Injector Design

Catalyst bed packing frontal area, A_{cb} = 0.003333 m_2

Assuming there are 10 injectors per sq. inch, $\mathrm{n_{total}} = 52$

Assuming the mass flow rate for all injectors to be equal,

Mass flow rate through each injector = $\frac{\dot{\mathrm{m}}_\mathrm{p}}{\mathrm{n_{tatal}}}$ = 0.001282 kg/s

Assuming $\mathrm{C_d}$ for the orifice as 0.8,

$$A_{\mathrm{orifice}} = \frac{\dot{\mathrm{m}}_\mathrm{p}}{\mathrm{C_d}} \times \sqrt{\frac{1}{2 \times \Delta \mathrm{p} \times \rho_{\mathrm{HTP}}}}\ [1]$$

$$A_{orifice} = 3.546 \times 10^{-8}\ m^2$$

$$\mathrm{D_{orifice}} = 0.2125\ \mathrm{mm}$$

If the orifice diameter is less than 2.5 mm, the propellant disintegrates into larger drops. If the orifice diameter is less than 0.5 mm, clogging occurs.

Suppose the orifice diameter is 0.6 mm,

$$m_0 = \frac{\dot{m}_p}{n} = C_d A\sqrt{2 \times \Delta p \times p_{HTP}} \rightarrow n = 7$$

∴ Mass flow rate through each injector=9.5211 g/s

$$A_{orifice} = 2.642 \times 10^{-7}\ m^2 \rightarrow D_{orifice} = 0.5792\ \text{mm}$$

6. Nozzle Design

Nozzle AR, $$\epsilon = \frac{A}{A^*} = \frac{\left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}}\left(\frac{P_c}{P_e}\right)^{\frac{1}{\gamma}}}{\sqrt{\left(\frac{2\gamma}{\gamma-1}\right)\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}} \text{ [7]}$$

AR in terms of Mach number M is given by,

$$\frac{A}{A^*}.\frac{P_e}{P_c} = \frac{1}{M}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\left(1 + \frac{\gamma-1}{2}M^2\right)^{-\frac{1}{2}} \text{[7]}$$

Mass flow rate through choked nozzle,

$$\dot{m} = \frac{A_t P_c}{\sqrt{T_c}} \times \sqrt{\frac{\gamma}{R}} \times \left(\frac{\gamma+1}{2(\gamma-1)}\right)\left(\frac{\gamma+1}{2}\right)^{-\left(\frac{\gamma+1}{2(\gamma-1)}\right)} \text{ [7]}$$

Throat diameter, $\mathcal{D}_t = 3.2$ mm

Throat length, $L_t \leq 0.25\ D_t$ [12]

$$\mathcal{L}_{convergent} = \frac{\mathcal{D}_c - \mathcal{D}_t}{2\tan\theta_{half}} \text{[21]}$$

$$L_{divergent} = \frac{D_e - D_t}{2\tan\alpha_{half}} \text{[21]}$$

$$\theta_{half} = 30^0, \alpha_{half} = 15^0$$

$$\lambda = \frac{1}{2}(1 + \cos\alpha_{half}) \text{[21]}$$

$\lambda = 0.983$ where λ is the Nozzle Divergence Correction Factor

Table. 3. Nozzle parameters for $L_{convergent}$ of 53.64 mm and an optimum L_t of 0.8 mm

S.No.	Altitude (km)	A/A*	Pc/Pe	M	$L_{divergent}$ (mm)	L_{total} (mm)
1	0	2.877	0.5066	0.6319	4.1569	58.5968
2	25	32.4453	784.5801	5.3615	28.0416	82.4815
3	30	71.5445	1670.8438	5.2615	44.5362	98.9761
4	40	193.8929	6968.6411	6.624	77.176	131.6159
5	50	364.877	25069.4926	9.279	108.0907	162.5306

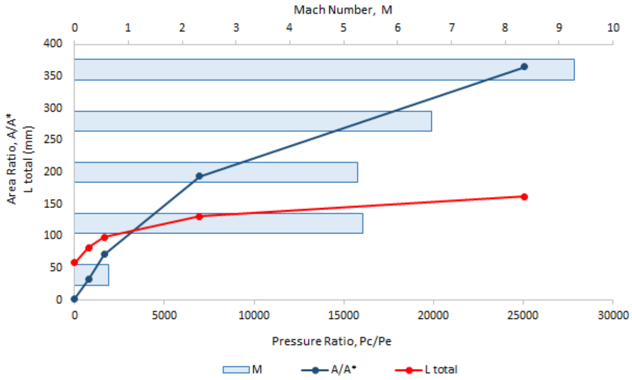

Figure-4. Nozzle Parameters

7. Case Studies

I. Boeing RocketDyne AR2-3

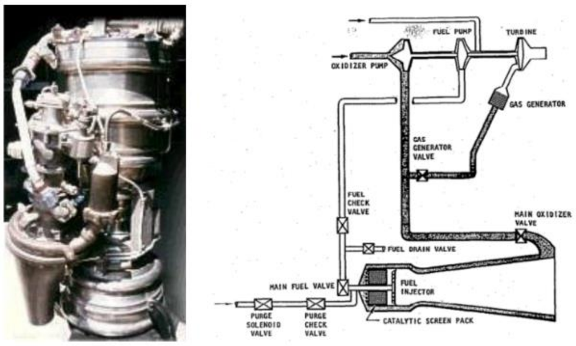

Figure-5. AR2-3 and its operating schematic [8]

Under the Future-X Demonstrator Engine project, the AR2-3 engine was evaluated for potential use in the Boeing X-37 (Orbital Test Vehicle). It was previously used in the Lockheed NF-104A. The engine has a specific impulse of 246 seconds and produces a thrust of 29.34 kN (6600 lbf). JP-4 or JP-5 served as the fuel, while hydrogen peroxide was employed as the oxidizer. In the Lockheed NF-104A, JP-4 was specifically used. The engine could operate solely on hydrogen peroxide (90%) as a monopropellant in emergency conditions. After combustion, the fuel in the AR2-3 engine is expelled through a nozzle with a 12:1 area ratio **[8]**.

II. Upper Stage Flight Experiment (USFE).

Figure-6. USFE Engine Schematic [10]

NASA's Marshall Space Flight Center, in collaboration with the Air Force Research Lab and Orbital Sciences Corporation, developed a low-cost liquid upper stage that utilizes HTP (90%) and JP-8. JP-8, the military version of "Jet A-1" used in commercial aviation, includes anti-icing and corrosion inhibitors. The USFE (Upper Stage Flight Experiment) is a pressure-fed, all-composite stage, significantly reducing the weight compared to an all-metal design **[9]**. The USFE engine includes an ablative chamber, nozzle assembly, fuel injector, catalyst bed for converting HTP (90%) into oxygen and superheated steam, oxidizer dome with a gimbal mount, and propellant feed lines. It produces a thrust of approximately 44.482 kN (10,000 lbf) and has a specific impulse between 275 and 298 seconds in a vacuum **[11, 13]**.

III. Soyuz Thruster

Figure-7. Soyuz HP Thruster [22, 23]

The Attitude Control System consists of HP thrusters (24 for the Soyuz-TMA and 8 for the Soyuz-MS) in the Re-entry/Descent Module of the spacecraft. These thrusters are used to perform maneuvers while in orbit and during re-entry **[14, 15]**. Each thruster is capable of generating a nominal thrust of 220 N, with a vacuum specific impulse I_{sp} of 160 seconds **[18]**.

8. Conclusion

A laboratory-scale 100 N HP (90% concentration) monopropellant rocket engine facility designed by Sang-Hee et al. was taken as a reference **[3]**. The propellant tanks were considered spherical and made of Grade 5 Titanium alloy, with the pressurizing gases being helium and nitrogen. The silver screen catalyst bed was designed based on bed loading factor and residence time. Seven injectors were required for propellant injection, determined by the pressure drop for injection. Due to growing interest in low-toxicity, high-performance propellants, the use of HP as a monopropellant or oxidizer in bipropellant liquid rocket engines is under consideration. Current projects include POLON (a Polish microsatellite propulsion module) **[19]** and ISRO's Gaganyaan, which aims to send humans to low Earth orbit and return them safely to Earth **[20]**.

9. References

[1] Sutton G.P., Biblarz O., “Rocket Propulsion Elements”, Seventh Edition, Wiley India Pvt. Ltd., 2001.
[2] Bollinger L., Goldsmith M., Lemmon Jr. A., “Liquid Propellants and Propellants”, Progress in Astronautics and Aeronautics, Martin Summerfield, Series Editor-in-Chief, Volume 2, 1960.
[3] Ahn Sang-Hee, Choi Tae-Hoon, S. Krishnan, Lee Choong-Won, “A Laboratory Scale Hydrogen-Peroxide Rocket-Engine Facility”, 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 Jul 2003.
[4] Palmer K., Catalan E.V., Lekholm V., Thornell G., “Investigation of exhausts from fabricated silicon micronozzles with rectangular and close to rotationally symmetric cross- sections”, J. Micromech. Microeng. 23 (2013) 105001.
[5] Rahner C., Al-Qureshi H.A., Stainer D., Hotza D., Fredel M.C., “Numerical Evaluation of a Light-Gas Gun Facility for Impact Test”, Modelling and Simulation iEngineering Volume 2014, Article ID 501434, 6 pages.
[6] Anderson W., “Rocket Engines”, Encyclopedia of Energy, Volume 5, 2004 Elsevier Inc.
[7] P. Balachandran, “Fundamentals of Compressible Fluid Dynamics”, PHI Learning Private Limited, Delhi-110092, 2014.
[8] Butler, K., “AR2-3 engine refurbishment and gas generator testing”, 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 June 1999, Los Angeles, CA.
[9] Csgnetwork.com. (2022). Aviation Fuel - Jet Fuel Information. http://www.csgnetwork.com/jetfuel.html
[10] Guerrero J., Hamilton B., Burton R., Crockett D., Taylor Z., “Upper stage flight experiment (USFE) integral structure development effort”, Composite Structures 66 (2004) 327–337.

[11] Anderson W.E., Boxwell R., Crockett D.V., Ross R., Lewis T., "Upper-Stage Flight Experiment", Orbital Sciences Corporation, Launch Services Group; Chandler, AZ; C. McNeal, NASA Marshall Space Flight Center, AL; and K. Verdarame Air Force Research Lab/VS, Albuquerque, NM (1999).
[12] Sapkota J., Xu Y., Sun H. (2019) Numerical Study on Response Characteristics of Solid Rocket Pintle Motor. J Aerosp Techol Manag, 11.
[13] Ross R., Morgan D., Crockett D., Martinez L., "Upper Stage Flight Experiment 10K Engine Design and Test Results", 36th AIAA/ASME/ SAE/ASEE Joint Propulsion Conference, Huntsville, AL, July 17-19, 2000.
[14] Spaceflight101.com. (2022). Soyuz MS – Spacecraft & Satellites. Accessed from https://spaceflight101.com/spacecraft/soyuz-ms/ on 30 August 2024.
[15] Spaceflight101.com. (2022). Soyuz TMA-M – Spacecraft & Satellites. Accessed from https://spaceflight101.com/spacecraft/soyuz-tma-m/ on 30 August 2024.
[16] Ponzo J., "Small Envelope, High Flux 90% Hydrogen Peroxide Catalyst Bed", 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit 20-23 July 2003.
[17] Valentini D., Pasini A., Pace G., Raiji H., "Green Propellant Thruster Design for LEO Platforms Active Debris Removal", Space Propulsion 2018.
[18] Nammo.com. (2022). 220 N Monopropellant Thruster. https://www.nammo.com/product/220n-h2o2-thruster/
[19] Gramatyka J., Paszkiewicz P., Grabowski D., Parzybut A., Bodych D., Wróblewski K., Surmacz P., Pietrzak K., "Development of POLON—A Green Microsatellite Propulsion Module Utilizing 98% Hydrogen Peroxide", Aerospace 2022, 9,297.
[20] Economicstimes.indiatimes.com. (2021, July 14). ISRO holds hot test of liquid propellant Vikas engine for Gaganyaan. https://economictimes.indiatimes.com/news/science/isro- holds-hot-test-of-liquid-propellant-vikas-engine-for- gaganyaan/articleshow/84413976.cms
[21] Ackley M. (2019, June 12). How to calculate Mission Specified Propulsion System Parameters – Part II. Valispace. https://www.valispace.com/how-to-calculate-mission- specified-propulsion-system-parameters-part-ii-2/
[22] Icollector.com. (2022). Soyuz image thruster. Accessed from https://dygtyjqp7pi0m.cloudfront.net/i/12628/12710524_1.j pg?v=8CEFC13340464E0 on 30 August 2024.
[23] Wade M. (2019). Soyuz 7K-LOK. Astronautix. Accessed from http://www.astronautix.com/s/soyuz7k-lok.html on 30 August 2024.

10. Conflict of Interest

The author declares no competing conflict of interest.

11. Funding

No funding was received to support this study.