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DESIGN AND DEVELOPMENT OF A HYDROGEN-PEROXIDE ROCKET-ENGINE FACILITY

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ABSTRACT

The ongoing developmental studies on the application of hydrogen peroxide for propulsion are briefly reviewed. A detailed design-study of a laboratory scale hydrogen peroxide mono-propellant engine of 100 N thrust is presented. For the preparation of concentrated hydrogen peroxide, a distillation facility has been realised. Results of water analogy tests are presented. Initial firings using the concentrated hydrogen peroxide were not successful. Low environmental temperature, low contact area of the catalyst pack, and contamination in the hydrogen peroxide were considered to be the reasons. Addressing the first two points resulted in successful firing of the rocket engine.

Keywords: *Hydrogen peroxide, monopropellant rocket, green propellant, silver catalyst*

1.0 INTRODUCTION

In recent years, there has been a renewed interest in the use of hydrogen peroxide (H_2O_2) as an oxidizer in bipropellant liquid rocket engines as well as in hybrid rocket engines [1-4]. This renewed interest is because of the growing importance in using propellants of low toxicity and enhanced versatility. The use of H_2O_2 in rocket propulsion offers the versatility of operating the engine on a dual mode, namely, a bipropellant mode (either as a bipropellant liquid engine or as a hybrid rocket engine) for a large thrust requirement and a monopropellant mode for a small thrust application. A propulsion unit without a requirement for a separate ignition unit offers a higher system-reliability. H_2O_2 decomposes into a mixture of superheated steam and oxygen to a temperature of around 1000K. This leads to the automatic ignition either with a liquid fuel in a bipropellant engine or with a solid fuel in a hybrid-rocket engine. Thus, the versatility with the additional advantage of automatic ignition makes the "green" H_2O_2 an attractive oxidizer.

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2.0 ONGOING DEVELOPMENTAL PROJECTS

Many developmental studies are in progress around the world in adopting H_2O_2 in rocket propulsion. These studies are towards developing H_2O_2 oxidized bipropellant liquid engines (mostly having kerosene as the fuel) and hybrid rocket engines.

The H_2O_2-oxidized hybrid rocket engines are actively being considered for application in upper stage propulsion. In 1999, NASA awarded a contract to Lockheed Martin Astronautics, along with subcontractors Boeing Rocketdyne and Thiokol, to begin development of a H_2O_2-oxidized hybrid motor for upper stage application in reusable launch vehicles and emerging defense applications [5-7]. The hybrid upper stage propulsion system uses a hockey-puck-shaped, single end burning fuel grain that is slightly oxidized to enhance regression rate and system operability. High concentration (>90%) H_2O_2 is passed through a catalyst pack and aft-mounted injector, which directs the oxidizer toward the face of the fuel grain in a swirling pattern. In 2001, at NASA Stennis the static firings of these 280 and 610 mm diameter motors demonstrated auto-ignition, stable and efficient combustion, extinguishment, and restart of the propulsion system. A follow-on effort at an increased scale is reported to be under consideration [7].

For decades, launch vehicles have accommodated small "piggyback" spacecraft, namely secondary payloads. But, most of these secondary payloads do not have any means of changing orbits once deployed from their host launch-vehicle. Therefore there is a widespread need for small and inexpensive propulsion and guidance modules that can boost small secondary payloads from their drop-off orbits to more desirable orbits. SpaceDev has been awarded in August 1999 a contract to develop the propulsion and guidance modules using the H_2O_2 oxidized hybrid-rocket concept. The micro-kick hybrid motor under this concept is storable, re-startable, throttleable, modular, and scalable. It is about 130mm diameter and 305mm length with a total thrusting time of about 45s. Using the knowledge gained by several test firings of this motor, SpaceDev has begun development of larger, reusable motors in the 45 - 67kN-thrust class [7, 8]. ONERA in France is working on the development of H_2O_2/polyethylene or HTPB hybrid-propulsion system for 100-kg micro-satellites and small tactical missiles [8, 9]. Work on the Hybrid Rocket Technology Demonstrator continues at Purdue University. A flight version of 4kN thrust, four-port, hydrogen peroxide/ polyethylene hybrid rocket motor has been successfully hot fire tested for many times [10, 11].

Aerojet has successfully developed a trifluid propellant injector for H_2O_2–kerosene reusable bipropellant engines [12]. Boeing Rocketdyne is developing H_2O_2 catalyst packs and H_2O_2/kerosene torch igniters for possible applications in orbital maneuvering systems, crew escape systems, and all upper stage and on-orbit applications requiring lower life-cycle costs and improved safety [12].

For the development of all these monopropellant, hybrid, or liquid-bipropellant propulsive-systems using H_2O_2, the catalyst properties and the catalytic system design are fundamental aspects. Catalytic systems traditionally contain packed screen beds made up of the screens of pure-silver or silver electroplated stainless-steel/nickel. However, these catalytic systems have disadvantages such as large pressure drops, high weight, de-activation due to the stabilizers in H_2O_2, and inability to support the decomposition of high-concentration H_2O_2. Hence, there has been an interest in the development of new catalytic systems [13-15].

In consideration of the above review on H_2O_2 propulsion, it was decided to build, as a first step, a laboratory scale 100-N H_2O_2 monopropellant rocket engine facility in the School of Mechanical Engineering, Kyungpook National University. This basic facility is to be used for research in the different areas of H_2O_2 propulsion systems.

3.0 ENGINE DESIGN

The engine uses H_2O_2 of concentration $\geq 90\%$. The thrusting time is to be in excess of 10 seconds. The nozzle entry stagnation pressure = 2 MPa and the nozzle pressure ratio = 15. Using NASA CEC71 program [16], the engine theoretical-performance was calculated and the results are given in Table 1.

There are two important parameters for the design of a screen bed: 1) the average mass flux through the bed (the so called bed-loading) and 2) the average residence time. Among the screen bed systems, pure silver screen is found to be most effective one. Adopted values of mass-flux in proven beds of silver screen vary from 117 to 280kg/m^{2}s [17-20]. Average residence time in the catalyst bed varied from 0.7ms to 1.5ms [18-21].

Table 1: Theoretical rocket performance characteristics of the hydrogen peroxide engine assuming frozen flow and 90% hydrogen peroxide concentration

	CHAMBER	**THROAT**	**EXIT**
p_0/p_e	1.0000	1.8188	15.000
p (*MPa*)	2.0	1.01	0.133
T (*K*)	1029.54	906.39	559.65
$\overline{m}$ (*kg/kg-mol*)	22.105	22.105	22.105
γ	1.2648	1.2764	1.3158
A_e/A_t		1.0000	2.6713
c^* (m/s)		940	940
C_F^0		0.702	1.338

Generally the quality factor for c^* (or c^* efficiency), η_{c^*} is taken as 0.95 for bi-propellant liquid engines and solid propellant motors. Since the engine under consideration is a monopropellant one and the quality of decomposition is very much dependent on the catalyst, a conservative value of 0.90 is assumed for the quality factor. Therefore, estimated experimental c^*_{expt},

$$c^*_{expt} = \eta_{c^*} c^*_{theo} = 0.9 \times 940 = 846 \frac{m}{s} \tag{1}$$

$$\left(C_{F_{sea\ level}}\right)_{theo} = C_F^0 + \frac{A_e}{A_t}\left(\frac{p_e}{p_{0n}} - \frac{p_a}{p_{0n}}\right)$$

$$= 1.338 + 2.6713 \times \left(\frac{1}{15} - \frac{1.01325}{20}\right) = 1.3808 \tag{2}$$

Assuming a quality factor for the thrust coefficient $\eta_{C_F} = 0.95$,

$$\left(C_{F_{sea\ level}}\right)_{expt} = \eta_{C_F}\left(C_{F_{sea\ level}}\right)_{expt} = 0.95 \times 1.3808 = 1.3117 \tag{3}$$

$$\left(I_{sp_{sea\ level}}\right)_{expt} = c^*_{expt} \times \left(C_{F_{sealevel}}\right)_{expt} = 846 \times 1.3117 = 1109.7\ \mathrm{N - s/kg} \tag{4}$$

Propellant flow rate,

$$\dot{m}_p = \frac{F}{\left(I_{sp_{sealevel}}\right)_{expt}} = \frac{100}{1109.7} = 0.09013 \frac{kg}{s} \tag{5}$$

An average mass-flux of 200kg/m^2-s is assumed for the engine [17-20]. Therefore, the diameter of the catalyst bed = 0.02395 m (say, 25 mm).Combustion chamber temperature,

$$T_0 = T_{ad}\eta^2_{c^*} = 1029.4\mathrm{x}0.81 = 834\mathrm{K} \tag{6}$$

For the assumed residence time of 1.5ms, the catalyst-bed length,

$$L_c = \frac{R_u \dot{m}_p T_0 \Delta t}{\overline{m}\left(\pi D_c^2/4\right)p_{0n}} = \frac{8314.3 \times 0.09013 \times 834 \times 0.0015}{22.105 \times \left(\pi \times 0.025^2/4\right) \times 2 \times 10^6} = 0.043\mathrm{m} \tag{7}$$

In order to avoid tunneling effect of H_2O_2 through the catalyst pack perforated stainless steel plates three in number are to be introduced at the beginning, middle, and the end of the catalyst pack. Therefore the total length of the catalyst pack is selected as 55mm.

3.1 Injector Orifice

To effectively de-link the feed system from the engine, generally about 0.6MPa or 10 percent of the chamber pressure, whichever is higher, is provided at the propellant injector. Therefore, a pressure drop of 0.7MPa is provided for the propellant injection. For the mass flow-rate of 0.090 kg/s, assuming the coefficient of discharge for the orifice as 0.8, the orifice diameter is calculated as 1.8mm. As the variation of propellant-injection characteristics are to be considered for the study of engine performance, different orifice diameters from 1.4 mm to 2 mm in steps of 0.2 mm are selected.

3.2 Nozzle Dimensions

The mass flow-rate through the choked nozzle is given by,

$$\dot{m} = \frac{p_{0n} A_t}{c^*_{expt}} \tag{8}$$

$$A_t = \frac{0.09013\text{x}846}{2\text{x}10^6} = 3.8125\text{x}10^{-5}\,\text{m}^2 \Rightarrow D_t = 6.97\text{mm} \qquad \text{say 7mm} \tag{9}$$

$$D_e = D_t\sqrt{A_e/A_t} = 7 \times \sqrt{2.6713} = 11.44\text{mm} \Rightarrow \qquad \text{say 12 mm} \tag{10}$$

A half-cone angle of 13^o is selected for the nozzle.

3.3 Propellant Tank Pressure

For the mass flux of 200 kg/m^2-s, the pressure drop across the catalyst bed is expected to be about 0.85 MPa [17]. Therefore the pressure upstream of catalyst bed = 2.85MPa. With the pressure drop of 0.7MPa across the injector orifice and 0.2MPa across the solenoid valve, the propellant tank pressure = 3.75MPa. A minimum pressure drop of 1.0MPa is to exist at the pressure regulator. Therefore, the minimum pressure upstream of the pressure regulator = 4.75MPa.

3.4 Propellant Tank Volume

Thrusting time is to be in excess of 10s. Assuming an ullage volume of 5 percent of propellant volume and 5 percent of propellant volume for tube-passages and protuberances, with a standard one liter tank available in the market, the propellant volume that can be stored in the tank,

$$V_p = \frac{10^{-3}}{1.1} = 9.091\text{x}10^{-4}\,\text{m}^3 \tag{11}$$

Volume flow-rate of propellant for the engine of 100N thrust,

$$\dot{V}_p = \frac{\dot{m}_p}{\rho_p} = \frac{0.09013}{1400} = 6.4379\text{x}10^{-5}\,\text{m}^3/\text{s} \tag{12}$$

Therefore the maximum-possible thrusting time,

$$t_{max} = \frac{9.091x10^{-4}}{6.4379x10^{-5}} = 14.12s \quad (13)$$

As it is not being envisaged to fix any anti-vortex unit at the outlet within the propellant tank, arbitrarily a time of 12s is fixed as the maximum rated thrusting time. Therefore with the initial propellant volume of 0.9 liter, maximum thrusting time is around 12 s. The assembly drawing of the engine that has been fabricated is shown in Figure 1. The specifications of the engine are given in Table 2.

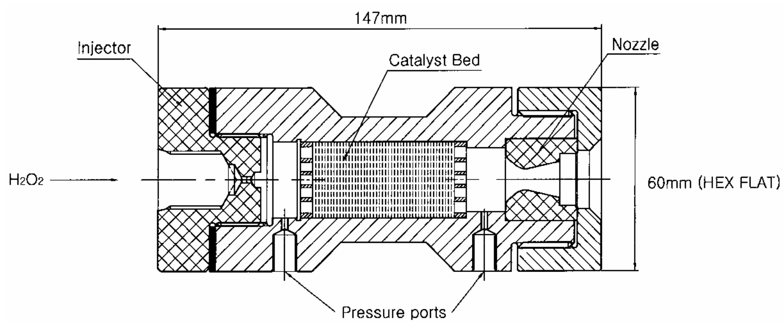

Figure 1: Hydrogen peroxide engine of 100N thrust

Table 2: Specifications of the H_2O_2 engine and its facility

Engine thrust	= 100 N
Estimated specific impulse	= 1110 N-s/kg
Regulated H_2O_2 tank pressure	= 3.75 MPa
Injector pressure drop	= 0.70 MPa
Injector orifice diameter	= 1.8 mm
Nozzle entry stagnation pressure	= 2.0 MPa
Propellant flow rate	= 0.090 kg/s
Catalyst bed-length	= 55mm
Approximate thrusting time	= 12 s
Nozzle throat diameter	= 7 mm
Nozzle exit diameter	= 12 mm

3.5 Hydrogen Peroxide Distillation Unit

Possibly the main impediment in starting the H_2O_2 based rocket research in a university is the difficulty in getting the rocket grade H_2O_2, say 90 percent or more of concentration. To solve this problem, a distillation unit has been realized and this is shown in Figure 2.

In the 20 liter flask, Figure 2, low concentration H_2O_2 solution is stored. The distillation unit is evacuated to a pressure of about 100mm of mercury. The 20 liter flask is heated to a temperature around 70^{o}C. The H_2O_2 solution in the 20L flask starts boiling and the water contained in it evaporates to get condensed in the

10L flask. Thus the concentration of the sample in the 20L flask keeps increasing with time. Cold water is circulated in the condenser for the easy condensation of the water vapor. At any time, the concentration of the H_2O_2 in the 20L flask can be found from the known initial concentration of H_2O_2 solution and its initial volume, and the volume of the water condensed in the 10L flask. Once the required concentration is reached in the 20 liter flask, the heating is stopped. After the unit gets cooled to ambient temperature, the vacuum is released. The concentrated H_2O_2-solution from the 20L flask is collected. The concentration of H_2O_2 in the solution is evaluated accurately by weighing the known volume of the concentrated H_2O_2. If the concentration is found at the desired level, the concentrated H_2O_2 is stored for the use in the rocket. The industrial grade H_2O_2 of 50% concentration and the laboratory reagent grade, a variety purer than the former, of 30% concentration are freely available. For the present studies, the laboratory reagent grade is concentrated to 90% level.

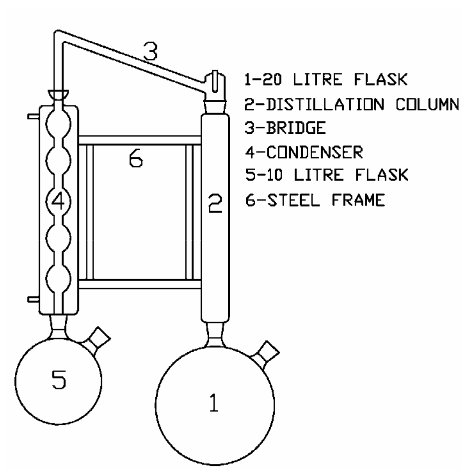

Figure 2: Hydrogen peroxide distillation unit

4.0 TEST FACILITY

The sketch of the realized facility of the H_2O_2 engine is shown in Figure 3. Sufficient safety features have been incorporated by introducing burst diaphragm and relief valve in the test facility. All the control valves are remotely operated by pressurised nitrogen. As the pressure regulator of low flow capacity required for the 100N engine was prohibitively expensive, a pressure regulator of high flow capacity ($c_v = 0.06$) had to be selected and this was made suitable for the 100N engine by adding a bypass orifice [22]. Pressure transducers are fitted at five stations: pressurisation tank, propellant tank, upstream of the injector, chamber pressure upstream of the catalyst bed, and downstream of the catalyst bed.

Propellant is filled into the 1000cc tank through quick connectors. Pressure regulator is set to the required propellant tank pressure. Recording and display of the pressure transducer-readings are initiated. Nitrogen supply is opened and it enters the gas pressurisation tank of 1000cc volume after passing through 40 and 7 micron filters. Once the propellant tank pressure is stabilized, shut-off valve is opened to initiate the engine operation. The engine is fired until the propellant is consumed (~12s for 900cc of propellant). Once the propellant is consumed nitrogen-purging automatically follows to cool the engine.

In order to gain experience in the operation of the facility and also to prove the system, the facility has been tested extensively under simulated condition using water or nitrogen. While using nitrogen, the injector orifice and nozzle throat diameters were altered to simulate the engine operation. A typical recording of the simulated test using nitrogen is given in Figure 4.

5.0 HOT TEST

20-mesh pure-silver screens were used for the catalyst bed. The silver screens were initially pickled with 50% nitric acid and subsequently activated with 2% solution of samarium nitrate. The total catalyst-bed length of 55mm was stacked with 20 mesh silver screens interposed with three perforated separator discs of stainless steel (each of 4mm thick). The total catalyst bed was compacted at 15 MPa.

The initial attempts to fire the engine was not successful. A failed-test result is shown in Figure 5. The test consisted of injecting the concentrated H_2O_2 for two intervals with a gap of about two seconds: first for a short duration of about 1 s, (from ~1.8th s to ~2.8th s, Figure 5) and the second for a long duration of more than 6 s (from ~4.8th s onwards, Figure 5). Only pulsed decompositions (at ~3.4 s and ~4.9s) could be obtained.

The possible reasons for the H_2O_2 not getting decomposed at the catalyst bed could be three. The first could be the low environmental temperature. At the time of the test the atmospheric temperature was around 5^{o}C. Willis [18] reported the most pronounced effect of engine case temperature on starting-time delays and most of his tests were conducted at the case temperature of 200^{o}C. Love and Stillwell [21] maintained the propellant tank at a temperature around 30^{o}C. The second possibility is the insufficient surface contact of the catalyst material with the H_2O_2. In the initial tests 20 mesh silver screens were used. Runckel et al. [20] found 40 mesh silver screens to be better than 20 mesh silver screens. The third reason could be the contaminations in the concentrated H_2O_2. Whitehead [23] explains the importance of reducing the contaminations in preparing a propellant grade concentrated H_2O_2

As the next developmental activity, the propellant tank was jacketed with heater elements and maintained at 35^{o}C. The engine case was also jacketed with heater elements and maintained at a temperature of 60^{o}C. In order to increase the surface area of the catalyst screens, the catalyst pack was compacted at 35MPa. The hot tests with these modifications were successful and a typical test result is shown in Figure 6.

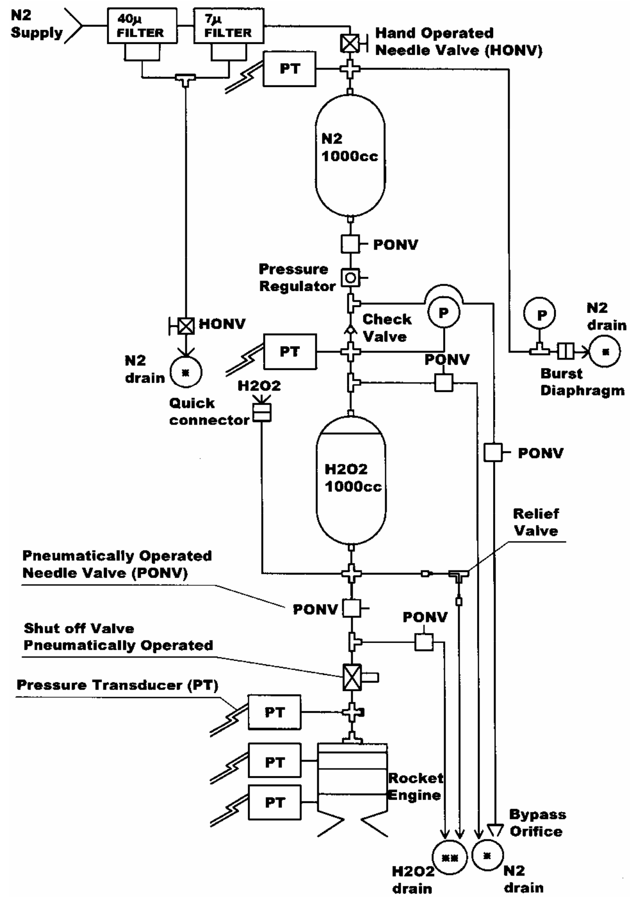

Figure 3: Hydrogen peroxide rocket engine facility

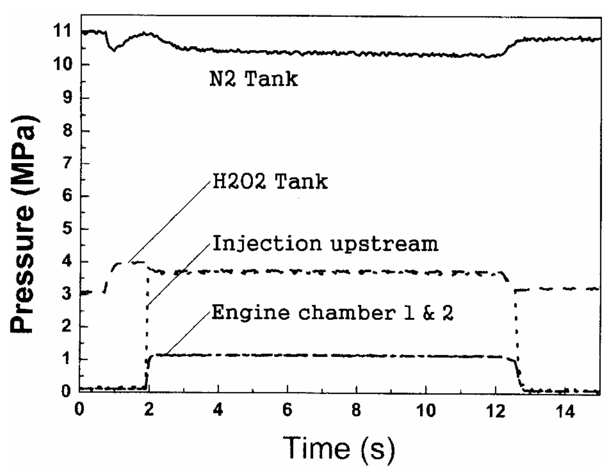

Figure 4: Engine pressure-recordings of a simulated test using nitrogen

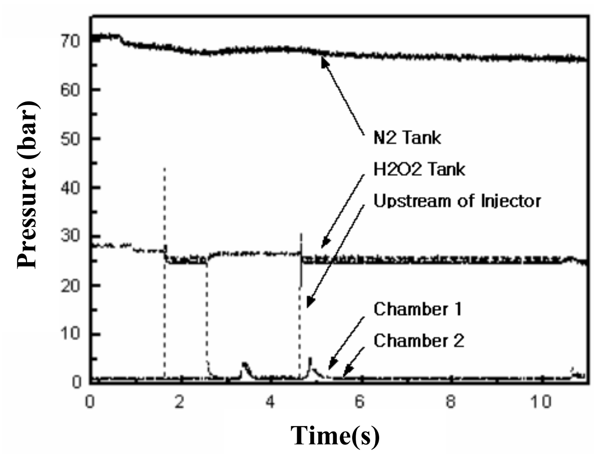

Figure 5: Pressure–time traces of a hot test that failed

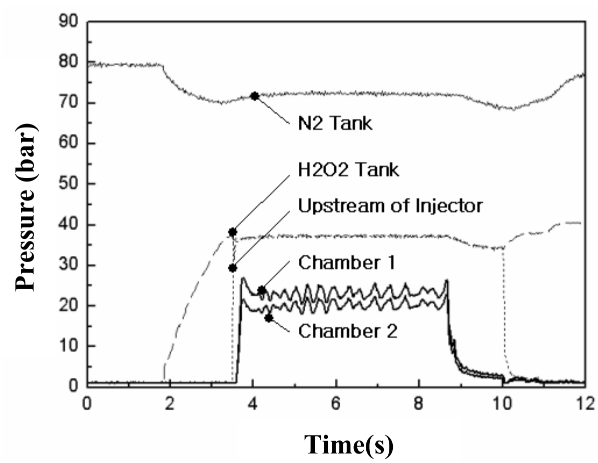

Figure 6: Typical result of a successful hot test

6.0 CONCLUSIONS

Because of the growing interest in using propellants of low toxicity and enhanced versatility, there has been a renewed interest in the use of hydrogen peroxide (H_2O_2) as an oxidizer in bipropellant liquid rocket engines as well as in hybrid rocket engines. A brief review of the ongoing developmental programs reveals that the application of H_2O_2 in rocket propulsion is quite varied: reusable launch vehicles, upper stage propulsion, emerging defense applications, tactical missiles, micro-satellite propulsion, orbital maneuvering systems, crew escape systems, and all upper stage and on-orbit applications requiring lower life-cycle costs and improved safety.

The detailed design of a laboratory scale facility of the H_2O_2 mono-propellant engine (100N thrust) has been presented.

Initial hot tests revealed the needs to have a controlled high temperature environment for engine and propellant. A modification incorporating enhanced temperature for the propellant and engine case and increased catalyst contact area by compacting the catalyst pack at a higher pressure yielded successful firing of the engine.

NOMENCLATURE

A	area (m^2)
c^*	characteristic velocity (m/s)
C_F	thrust coefficient
D	diameter (m)
F	thrust (N)
I_{sp}	specific impulse (N-s/kg)
L_c	length of catalyst bed (m)
$\overline{m}$	molar mass (kg/kg-mol)
$\dot{m}_p$	propellant flow rate (kg/s)
p	pressure (Pa)
R_u	universal gas constant (J/(kg-mol K))
t	time (s)
T	temperature (K)
V_p	propellant tank volume (m^3)

Greek symbols

γ	ratio of specific heats
Δt	residence time in the catalyst bed
η_{c^*}	c^* efficiency
η_{C_F}	nozzle flow quality factor
ρ_p	propellant density (*kg/m³*)

Subscripts

0	stagnation condition
a	atmospheric condition
ad	adiabatic flame temperature condition
c	catalyst bed
e	nozzle exit
exp t	estimated experimental-condition
n	nozzle entry condition
t	throat condition
theo	theoretical value

Superscript

0 adapted condition

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