Pulse-mode Performance Characteristics of a Small Liquid-monopropellant Rocket Engine

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Abstract
The 70 N-class monopropellant hydrazine thruster may be employed in the space launch vehicle for attitude control, was tested through ground firing test for the design verification. Performance evaluation of thruster was carried out in the pulse-mode operation with various propellant inlet pressure. The primitive variables which are thrust, propellant flow rate, temperature, and pressure for evaluation of thruster were measure in each component. As a result, it is verified through performance test that response and repeatability of thrust and pressure are very superior, and the performance of development model thruster was 40.45 N in thrust, and 153.9 s in specific impulse at atmospheric condition.

1. Introduction

The propulsion system for attitude control of space launch vehicles can be classified into cold gas, mono-propellant, bi-propellant, and solid rocket motor system. For a selecting rocket propulsion system, the trade-off study should be performed, because each propulsion system has advantages and disadvantages and there are a number of considerations e.g., performance, mission capability, economics, safety, reliability, and so on[1]. The attitude control system requirements used in space vehicles are as follows; nominal thrust level, thrust duration in steady-state firing mode, minimum impulse bit, impulse repeatability in pulse-mode firing, response characteristics, as well as total impulse depending on a deliverable propellant quantity.

Table 1 is shown characteristics of chemical propulsion systems. The cold gas system has lower specific impulse and impossible to throttling. And, solid rocket motor has outstanding thrust performance but cannot operate restart, pulse-mode operation, and not easy throttling. In contrast to the above systems, mono-, bi-propellant systems provide many advantages in terms of attitude and velocity control equipped on space launch vehicles, satellite, and exoatmospheric kill vehicles, because these systems have superior thrust performance and restart, pulse-mode operation, and throttling capability are very excellent.

The current representative monopropellant hydrazine (N_2H_4) has strong point such as long/stable storability, high specific impulse, easy throttling, low pollutant by plume, and low flame temperature. So, propulsion system using

Table 1: Characteristics of chemical space propulsion system [2]

Classification	Cold gas	Monopropellant	Bipropellant
Specific impulse (s)	50	225	290
Thrust range (N)	0.05 – 0.1	0.5 – 2,000	> 1,500
Impulse range (N·s)	$< 10^4$	10^4 - 10^6	10^5 - 10^9
Min. impulse bit (N·s)	0.001	0.01	0.1
Complexity	Least	Midrange	Most
Spacecraft contamination	N	N	Y
Restart	Y	Y	Y
Pulsing	Y	Y	Y
Throttling	Y	Y	Y

hydrazine offer many advantage over the alternative approaches to performing the attitude control and station keeping function, though the application of bipropellant systems tends to increase recently along with the maturity of technology development.

This paper summarizes a ground hot-firing test results for the 70 N-class hydrazine thruster under development. And, the thruster performance characteristics are presented in terms of thrust, impulse bit, propellant supply pressure, chamber pressure, and temperature at the pulse operation mode.

2. Design specification of thruster and test procedures

The design specification of 70 N-class hydrazine thruster is summarized in Table 2. The hydrazine thruster generated thrust through supersonic nozzle by chemical reaction of propellant throughout the catalyst in thrust chamber. This development model is designed to produce 67 N (15 lb_f) of nominal steady-state thrust at an inlet pressure 2.41 MPa (350 psia). As depicted in Fig. 1, the thruster is composed of a solenoid-operated propellant valve and thrust chamber assembly (TCA) that consists of thermal-barrier tube, propellant feed tube, injector, multiple thrust chamber, and converging-diverging nozzle. Test and evaluation model (TEM) is configured and fabricated so as to measure and sample the performance parameters such as pressure, temperature, and gas product composition at specific positions of the thruster.

Table 2: Design specification of 70 N-class liquid-monopropellant hydrazine thruster

Parameter	Unit	Specification
F_v	N	67±5
I_{sp}	s	225±8
A_e/A_t	-	50
$P_{chaber,upper}$	MPa	1.55
$P_{chamber,lower}$	MPa	1.38
$\dot{m}_f$	g/s	29.2

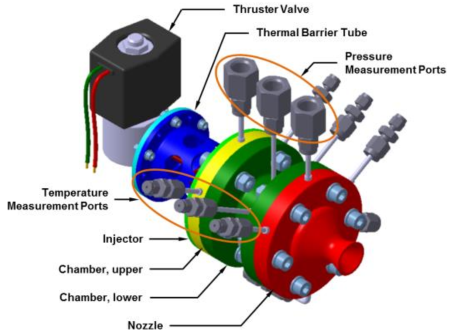


Figure 1: Configuration of 70 N-class hydrazine thruster (TEM)

The supersonic nozzle expansion ratio of flight model is set to 50 for the operation outer space or at high altitude, whereas that of the TEM is modified to 10 in order to prevent the thrust loss owing to flow separation and shock occurring inside the nozzle at the atmospheric condition[3]. Shape of the feed tube is determined to minimize heat soak-back which may cause the self-decomposition of hydrazine at pulse-mode operation. A monopropellant grade hydrazine (99.09% of purity) per MIL-PRF-26536F[4] is fed into double-stage catalyst chamber charged into with granular iridium-coated alumina.

Test measurement rig (TMR) applying single component system was constructed for the measurement of thrust in TEM. Test configuration of the hydrazine thruster installed on TMR is shown Fig. 2. TMR has the on-line calibration capability which enables remotely a compensation for the variation of structural sensitivity caused by thermal effects and hysteresis during firing test.

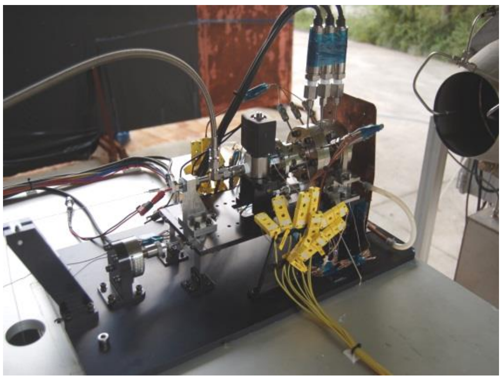


Figure 2: Configuration of TMR and thruster (TEM)

This paper present pulse-mode firing results for TEM with various propellant inlet pressure. Period and electrical pulse width were set to 5.0 s and 0.5 s, respectively. Thruster is, at first, exposed to a burn-in process for the mechanical and chemical stabilization of catalyst bed, and then pulse firing is conducted according to the test procedures.

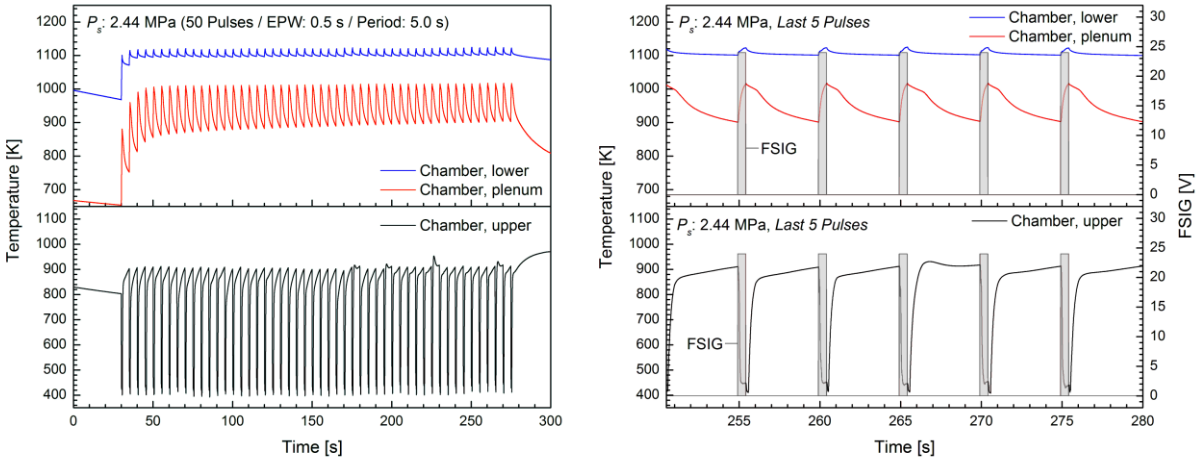
3. Test results and discussion

Liquid hydrazine discharged from injector decomposes across the double-stage thrust chamber producing ammonia (NH_3), nitrogen (N_2), and hydrogen (H_2) according to the following two-step reactions:

$$Exothermic:\ 3N_2H_4 \rightarrow 4NH_3 + N_2 - Q_1 \qquad \textbf{(1)}$$

$$Endothermic:\ 4NH_3 \rightarrow 2N_2 + 6H_2 + Q_2 \qquad \textbf{(2)}$$

While the reaction of Eq. (1) is exothermic, Eq. (2) is endothermic[5]. Length to diameter ratio of reaction chamber and size/shape of catalyst particles are intimately associated with the flow residence time of decomposition gases. The size of reaction chamber is related to ammonia dissociation which is endothermic reaction and it has influence in temperature of product gas. Furthermore, specific heat ratio defined as thrust chamber temperature is affected to theoretical performance of thruster. Therefore, test and evaluation process according to the dimensional variation of catalytic reactor is indispensable for the design optimization of TCA. A number of test cases were drawn for the parametric test, among which test results for a standard model are to be presented in this work. Variation of the temperature measured at radial center of upper, lower, and plenum chamber is shown in Fig. 3.



(a) Full range　　　　(b) Last 5 pulses

Figure 3: Thermal behavior at pulse-mode firing with 2.41 MPa of propellant supply pressure

In the Pulse-mode firing test, saturation temperature of lower and plenum chamber were 1125 K, and 1017 K, respectively. In case of upper chamber, inner temperature behavior has repeated process which is cooling by cold propellant and increase of temperature by decomposing hydrazine and heat transfer from lower chamber. And, the temperature change of upper chamber started after 40 ms from firing signal. In addition, chemical reaction of thruster is finished within 35 ms after closing signal of FCV.

In case thermal conduction so called heat soak-back occurs from thrust chamber to upstream of TCA, it can lead to vaporization of liquid propellant in fuel supply line or FCV. This process can lead to explode of propellant supply line by spontaneous ignition of propellant vapor and combustion instability caused by vapor lock at injector. Therefore, thermal-barrier tube and feed tube should be designed to diminish of thermal conduction from reaction chamber, because they are directly connected to FCV[6]. Additionally, auto ignition temperature and flashing point of hydrazine are 543 K and 324 K. Therefore, temperature of HEA (head-end assembly) and FCV should be kept below 323 K. The temperature behavior of HEA and FCV are shown Fig. 4. Maximum temperature of feed tube and thermal-barrier tube are 650 K and 641 K. But, upstream temperature of components has been less than 320 K and

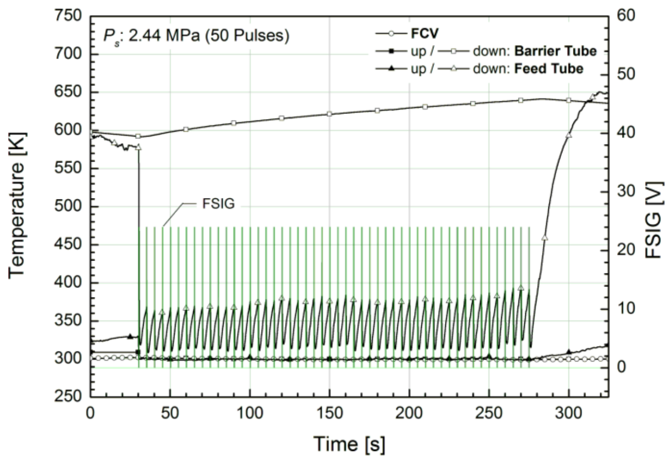


Figure 4: Temperature variation of HEA

FCV temperature has stayed 302 K, also. So, for the reason mentioned above, it is confirmed that structure HEA is suitable designed for evasion of soak-back.

While pulse-mode operation, repeatability of chamber pressure is used as an important factor for evaluation of thruster. Fig. 5 is shows overall variation of pressure at an inlet pressure 2.41 MPa. The average pressures of upper, lower, and plenum are 1.61, 1.43, and 1.16 MPa, respectively. After 4th pulse, it is confirmed that pressure behavior of 5th to 50th pulses have been stabilized. Because, reaction chamber is heated by initial pulse operation and catalyst has stabilized thoroughly. Consequently, the data after 5th pulse is employed for performance evaluation of thruster.

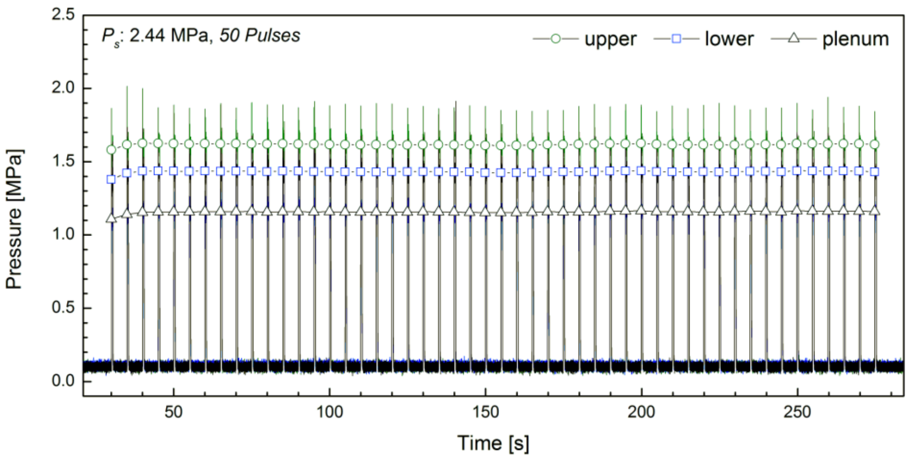
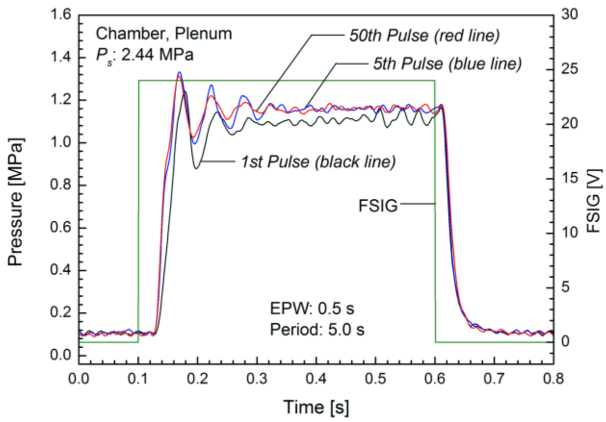


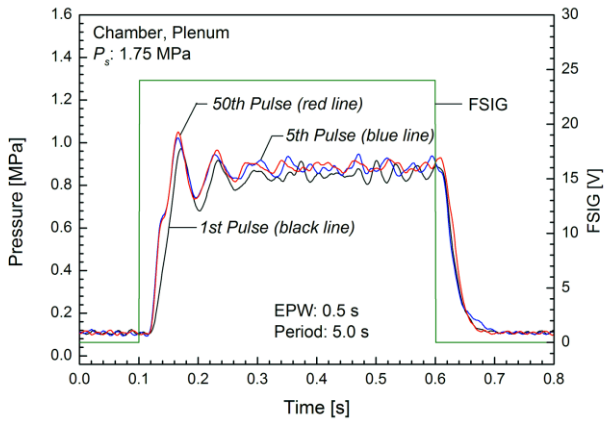
Figure 5: Overall variation of chamber pressure at 2.41 MPa of propellant inlet pressure

Figure 6 shows pressure behavior of pulse-mode operation at plenum chamber. The overlaid traces are shown for various propellant supply pressure and pulses (1st, 5th, and 50th pulse). The pressure traces shows very similar tendency and pressure level at all inlet pressure. The 1st pulse at all propellant supply pressure indicates tardy pressure rise and low pressure level. A cause of this phenomenon is low initial temperature of reaction chamber. In the high inlet pressure, pressure trace seems to be stable behavior. But, in the low pressure, it is observed that pressure behavior has irregular pressure perturbation, because supplied propellant from injector doesn’t have pressure gradient to flow across catalyst bed. One common approach to prevent the perturbations is use of coarse catalyst. The use of coarse catalyst causes decline of pressure drop in thrust chamber. By using the above method can lead to the stabilization of pressure, but response characteristics such as ignition delay may be occurred falling off in performance. In this test, the thrust chamber is charged fine catalyst to secure safety for the improvement of chemical decomposition efficiency of hydrazine.

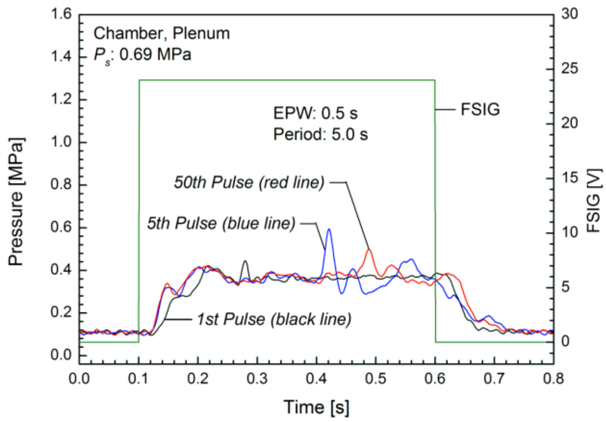
The thruster for attitude control of space vehicle is mainly used to pulse-mode operation. Generally, response characteristics are evaluated by a few terms such as ignition delay, rise time, and decay time. Table 3 shows pulse response time at 2.41 MPa of propellant supply pressure. The pressure behavior is identified similar reproducibility



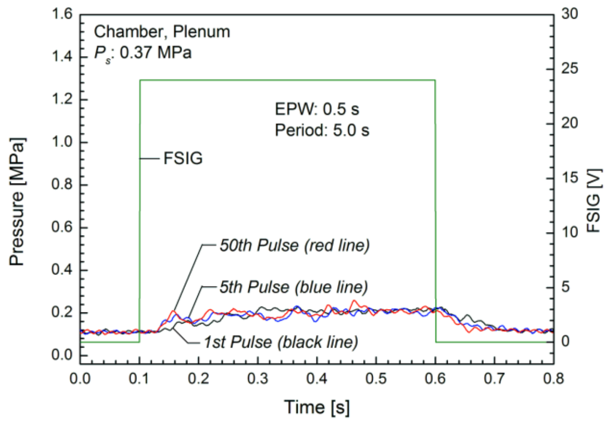
(a) P_s = 2.41 MPa



(b) P_s = 1.72 MPa



(c) P_s = 0.69 MPa



(d) P_s = 0.34 MPa

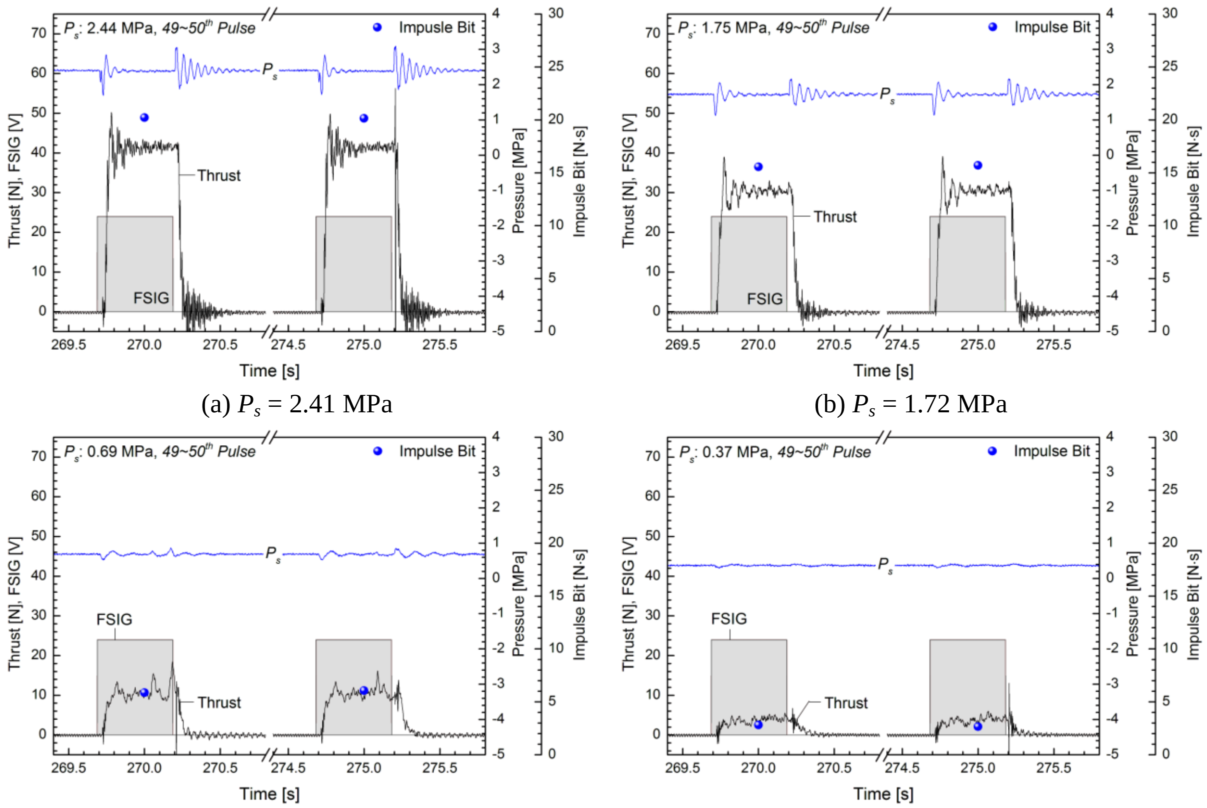
Figure 6: Plenum chamber pressure traces for various propellant inlet pressure and pulses

Table 3: Summary of pulse response characteristics

Classification	Time	SD (σ)
Ignition Delay [ms]	27.3	3.3
Rise time [ms]	56.2	1.8
Decay time [ms]	46.6	2.1

after 5^{th} pulse. Therefore, these time values were derived by arithmetic average values which used from 6^{th} to 50^{th} pulses. Response time is determined on the basis of steady-state chamber pressure. The thruster response characteristics are affected by such factor as mechanical characteristics of propellant valve, feed line hydraulic delay, ignition delay, and so on. The pressure rise time is divided into several events which are time to decompose enough hydrazine to fill the void space in the reaction chamber and heat the whole catalyst bed[2]. In this test, the ignition delay, rise time, and decay time were 27.3 ms, 56.2 ms, and 46.6 ms, respectively. By calculation of standard deviation(σ) of rise time which is 1.8 ms, it is verified that repeatability of pressure is very superior. But, the need for improvement of response characteristics is indispensable. So, ignition delay of thruster could be reduced through increasing of bed loading or use to fine catalyst. But, this approach can be increase of the catalyst loss rate[7]. Also, ignition delay and decay time is enough room for improvement through performance betterment of propellant valve. Furthermore, if initial temperature of catalyst bed increase, the ignition delay time will be more decrease.

Thrust behavior is shown in Fig. 7 along with various propellant supply pressure, impulse bit at the pulse-mode operation. Each performance parameter behavior is represented 49 ~ 50^{th} pulses to enable the identification due to short operating time. The thrust curves have very similar tendency and magnitude after stabilization process of reaction chamber. In addition, according to the decrease of propellant supply pressure, it is confirmed that the thrust



(a) P_s = 2.41 MPa

(b) P_s = 1.72 MPa

(c) P_s = 0.69 MPa

(d) P_s = 0.34 MPa

Figure 7: Variational behavior of thrust and impulse bit with various propellant inlet pressure

level is diminishing, also. As with pressure behavior, each thrust in the high supply pressure shows similar tendency. But, in the low pressure condition, thrust trace is shown irregular behavior. Therefore, additional evaluation test for performance improvement of thruster is required through the parametric test for optimize of reaction chamber and catalyst.

Comparison of thruster performance with various supply pressure is shown in Table 4. The measured thrust and impulse bit were 40.45 N and 20.22 N·s at an inlet pressure 2.44 MPa, respectively. The steady-state thrust of thruster used for this test is 40.71 N at 2.43 MPa of propellant supply pressure. And, it is confirmed that the pulse thrust efficiency compared to steady-state thrust is 99.4%.

Table 4: Comparison of the thruster performance with various supply pressure

Supply pressure	2.41 MPa	1.72 MPa	0.69 MPa	0.34 Mpa
Thrust [N]	40.45	30.87	11.78	5.42
Impulse Bit [N·s]	20.22	15.44	5.89	2.71
Specific Impulse [s]	153.87	150.18	129.37	101.25

4. Conclusion

The ground hot-firing test was carried out for the 70 N-class hydrazine thruster. The purpose of this test was to verify for design qualification of development model hydrazine thruster. So, the overall operating performance of thruster was confirmed and drawback of components that need improvement is reviewed through test results. The test results of Pulse-mode operation is described in this paper and performance evaluation was performed by thrust, impulse bit, and response characteristic.

The development model of thruster is verified to produce 40.45 N of pulse-thrust at 2.43 MPa of propellant supply pressure. And, pressure rise time was observed in 56.2 ms and its standard deviation is 1.8. As results, it is confirmed that the thruster has superior response characteristics and thrust performance. But, relatively tardy ignition delay and high ammonia dissociation are magnified as necessity of performance improvement. It is possible to improve through change of catalyst size and length adjustment of thrust chamber. Therefore, parametric test of reaction chamber is to be performed by additional ground firing test.

Acknowledgment

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