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One-Newton class thruster using arc discharge assisted combustion of nitrous oxide/ethanol bipropellant

Akira Kakami [a,*], Takuya Ishibashi [b], Keisuke Ideta [b], Takeshi Tachibana [b]

[a] Department of Mechanical Design Systems Engineering, University of Miyazaki, 1-1 Gakuenkibanadai-sakura, Miyazaki, Miyazaki 889-2192, Japan
[b] Department of Mechanical Engineering, Kyushu Institute of Technology, 1-1 Sensui-cho, Tobata, Kitakyushu, Fukuoka 804-8550, Japan

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ABSTRACT

This paper discusses a 1-N class thruster using nitrous oxide (N_2O)/ethanol bipropellant thruster using arc discharge assisted combustion. Bipropellant thruster, which is mainly used in spacecraft for orbit transfer and station keeping, utilizes toxic propellants: hydrazine and nitrogen tetroxide despite relatively high specific impulse. Then a new bipropellant thruster with arc plasma assisted combustion was proposed to develop a non-toxic eco-friendly bipropellant thruster. Both N_2O and ethanol, which can be stored in a liquid form, have neither toxicity nor reactivity to materials such as metal and polymer. Arcjet was utilized to initiate and sustain N_2O/ethanol chemical reaction at 0.4 MPa thrust chamber pressure. A prototyped thruster shows that combustion was successfully started and sustained with the assistance of a 1-kW arc discharge. Thrust measurements at atmospheric ambient pressure yielded a specific impulse of 55 s with a characteristic exhaust velocity efficiency of 60%.

1. Introduction

For attitude control, station keeping and orbit transfer of spacecraft, bipropellant thrusters have been utilized because of their relatively high specific impulse [1]. Conventional bipropellant thrusters for spacecraft use hydrazine and dinitrogen tetroxide (NTO) as liquid propellant. On the other hand, comparatively high freezing points, 274 K for hydrazine and 262 K for NTO, demand temperature management system. The liquid propellant is toxic and reactive to metals and rubbers. Accordingly, exhaust gas treatment systems are necessary for ground testing, and careful selection of materials for tanks and tubes. Hence, green propellant propulsion devices have been developed by companies and universities [2–7,9].

We have proposed to apply nitrous oxide (N_2O)/ethanol bipropellant to space propulsion because the propellant is storable in a liquid form in the space, and compatible to many materials [8]. In the designed thruster, arc discharge was utilized to initiate and stabilize combustion at thrust chamber pressure below 1.0 MPa, which is lower than that for conventional bipropellant rocket engines, and is comparative to that of 1-N class thrusters. Because the lower thrust chamber pressure can yield unstable combustion, the arcjet would keep fired after ignition to assist combustion. We prototyped and tested an N_2O/ethanol bipropellant thrusters and evaluated their performance: thrust, specific impulse and c^* efficiency.

2. Proposed thruster

2.1. Design

Fig. 1 shows a schematic of the prototyped thruster. An injector supplies atomized ethanol droplets with gasified N_2O into a thrust chamber. Simultaneously, an arcjet feeds N_2O arc plasma toward the liquid ethanol. The arcjet plume, which is a high enthalpy flow containing various reactive species such as oxygen atom and nitrous monoxide, impinges ethanol droplets and surrounding cold N_2O. Then, the plume assists the droplet evaporation and sustained chemical reactions between ethanol droplets and N_2O. Arc discharge assisted combustion would allow the proposed thruster to sustain stable combustion. Whereas the thruster had an arcjet, energy used by the thruster is mainly provided by chemical reaction. Hence, the thruster would be categorized into chemical thrusters. The N_2O/ethanol bipropellant thruster using arc-discharge-assisted combustion will provide many advantageous characteristics. N_2O, which is neither poisonous nor reactive below a dissociation temperature of 500 °C, is storable in the space owing to a vapor pressure of 5 MPa at the room temperature and a

* Corresponding author. Tel./fax: +81 985 58 7300.
E-mail address: kakami@cc.miyazaki-u.ac.jp (A. Kakami).

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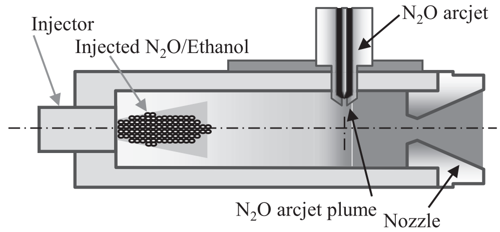

Fig. 1. Designed thruster.

freezing point of −90.9 °C. Relatively high vapor pressure allows N_2O to be used as the pressurant of ethanol. Hence, the proposed thruster would have almost the same structure as the mono-propellant thruster, which requires a pressurant for propellant supply.

Ethanol, which has neither reactivity nor toxicity, is storable in a liquid form in the space without sophisticated temperature management device due to a freezing point of −114 °C. Regarding the injector, both impingement- and coaxial-types are applicable to the designed thruster. Because N_2O is readily evaporated and liquefied by managing its temperature or pressure, a coaxial type injector can be used in the thruster. In coaxial-type injectors, atomization is completed in the vicinity of the injector exit, and droplet diameter would be smaller than that of the impingement-type injector. The smaller droplet diameter enhances characteristic exhaust velocity efficiency (c^* efficiency) and allows reduction in thrust chamber length.

2.2. Performance

Fig. 2 shows the dependence of theoretical adiabatic flame temperature T_f and specific impulse I_{sp} on oxidizer/fuel ratio (O/F) at a target thrust chamber pressure of 0.4 MPa and nozzle area ratio of 10. As illustrated in Fig. 2, specific impulse has a peak value of 288 s at an O/F of 4.0 whereas the theoretical adiabatic flame temperature has the maximum value at 5.7. This is because ethanol has five hydrogen atoms and accordingly, the average molecular weight of the exhaust was lowered at lower O/F. The theoretical calculation yielded specific impulses ranging from 220 to 288 s at O/F ranging from 1 to 7.

The adiabatic flame temperature reaches as high as 3000 K, and decreases below O/F of 4. The augmented flame temperature would damage the thruster, and consequently the thruster should operate

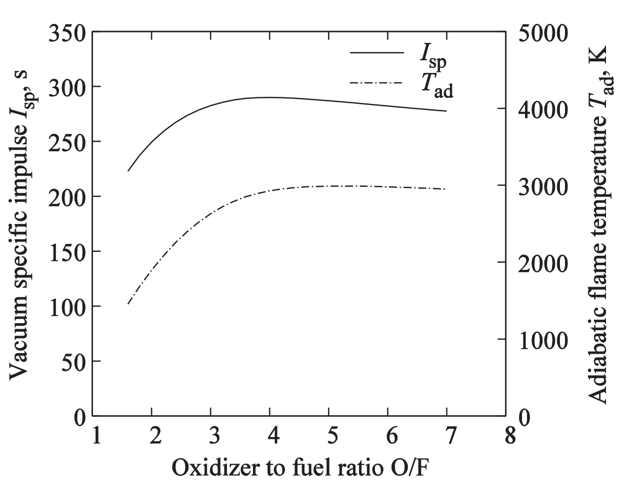

Fig. 2. Theoretical specific impulse and adiabatic flame temperature.

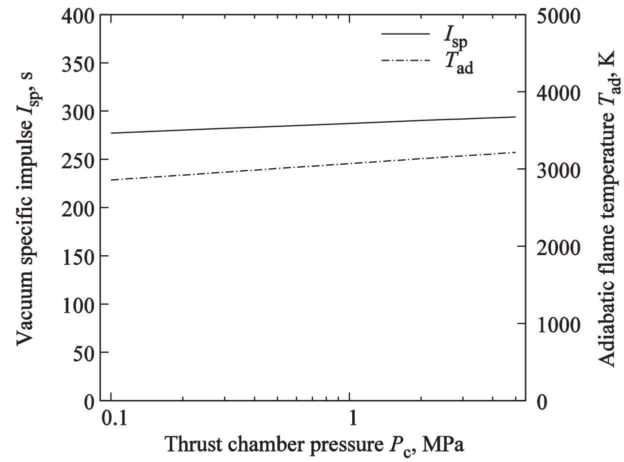

Fig. 3. Theoretical specific impulse and adiabatic flame temperature on thrust chamber pressure.

at O/F below 4 in order to reduce thermal damage on the thruster with the sacrifice of specific impulse. From the point of view, thruster was tested in O/F range of 1–7.

Fig. 3 shows the correlation between theoretical specific impulse and thrust chamber pressure. In general, increase in thrust chamber pressure provides higher specific impulse. In contrast, the arcjet stably produces plasma below 0.4-MPa plenum pressure, and accordingly requires thrust chamber pressure to be below 0.4 MPa in the case with arc-discharge assisted combustion. As shown in Fig. 3, theoretical specific impulse is slightly affected by thrust chamber pressure. Hence, the designed thruster can extract the potential of N_2O/ethanol bipropellant at 0.4 MPa class thrust chamber pressure.

3. Experimental apparatus

3.1. Prototyped thruster

Fig. 1 and Table 1 exhibit a schematic diagram and configuration of the 1-N N_2O/ethanol prototyped thruster. A target thrust and thrust chamber pressure are 1.0 N at atmospheric ambient pressure; a target thrust chamber pressure of 0.4 MPa was selected for stable sustainability of arc discharge. A throat area was determined from the target thrust and thrust chamber pressure using theoretical sonic speed and gas density at the throat. Because the thruster was fired at the atmospheric ambient pressure, the nozzle area ratio is 1.38, which yields optimum expansion in the nozzle for the target thrust chamber pressure. High enthalpy N_2O flow produced with the arcjet was injected into the thrust chamber in perpendicular to the direction of the ethanol droplets and gaseous N_2O supplied with a coaxial type injector. Ethanol was provided with a coaxial-type injector, which atomizes liquid propellant by the sharing force induced by the discrepancy in flow velocities between liquid and gas because the injector produces the smaller droplets than the impingement type injector.

Table 1
Designed thruster configuration.

Target thrust chamber pressure, MPa	0.4
Throat diameter, mm	1.7
Nozzle area ratio	1.38
Thrust chamber length, mm	60, 120
Discharge current, A	15–20
Theoretical Isp (Vacuum), s	204
Theoretical Isp (Atmospheric), s	149