Thermal behavior and energy characteristics of solid propellants

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This paper aims at the energy characteristics for four propellants: HTPB/AP、AP/NC、LS/NC、LS/SQ-2 through the thermodynamic calculations. Thermal degradation due to different proportion of the LS / NC formula and different proportion of the LS / SQ-2 propellants is studied by DSC and TG. Ignition tests due to different proportion of the propellants aim to screen out the best propellant whose energy and heat sensitivity meet the requirements for micro-thrusters. The resulting thrusts and impulses by experiments for the quality of different charge micro were compared with numerical simulations. This study on the thermal behavior therefore provides promising applications of solid propellants for rocket propulsion systems.

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1. Introduction

MEMS-based solid propellant micro thrusters have wide applications for small satellites operating at high attitude and orbit control technology because of its small size, high precision micro impulse and good integration [1-2]. A three-layer sandwich micro propulsion was fabricated by Lewis and Jason, using Lead Styphnate as the propellant. It produced 10^{-4} N-s of impulse and about 100 Watts of power [3]. Modeling and simulation of the micro thrusters using MEMS technologies had been performed by Zhang and Chou [4]. It was found that at sea level, the predicted thrust ranges from 0.76 to 4.38 mN and the estimated total impulse ranges from 1.16×10^{-4} to 4.37×10^{-4}N·s using HTPB / AP / AL as the propellant. In space, the predicted thrust range can be increased to 9.11 mN to 26.92 mN and the estimated total impulse ranges from 1.25×10^{-3}N·s to 1.70×10^{-3}N·s. Rossi et al. [5] successfully fabricated an array of micro-rockets using MEMS technology. Experimental tests of ignition and micro combustion employing a double-base (DB) solid propellant mixed with black-powder (BP) were reported and discussed. The thrust was reported to range from 0.1 to 1 mN for addition ratio of BP (x = 10%, 20% and 30%). Double-base propellants and black powder were mixed with different configurations to produce MEMS-SPMT propellants by Rossi etc [6] in 2007. When the ignition power is 600 mW, the double base propellant with no black powder addition cannot be ignited, however double base propellants containing 10%, 20% and 30% black powder can be ignited easily, the ignition delay is about 150 ms. A three layer micro-thruster array was designed and loading methods of the propellant and test methods of micro thrust properties were investigated by Xiao [7] from Nanjing University of Science and Technology. Numerical simulations of the structural and mechanics properties were performed for the micro thruster. A mixture of Lead Styphnate and nickel hydrazine nitrate was used as the propellant, and its properties of ignition and thrust were investigated. A MEMS-based solid propellant propulsion micro thruster array was developed by Zhang from Tsinghua University and Hu from Northwestern Polytechnical University, and others [8]. Ignition and thrust tests were performed on a prototype with a 6 × 6 array of micro thrusters. It was found that the instantaneous ignition power was less than 1W, the average ignition voltage was lower than 40V, theoretical thrust was about 1.7mN, compared with the electric propulsion which had the same order of magnitude thrust, the working voltage can be greatly reduced. He [9] from Nanjing University of Science and Technology tested the thrust of the micro-thruster with AP/NHN propellant and AP/THPC propellant in chambers with different diameter. The temperature distributions after the ignition of the bridge membrane and the chamber are simulated using ANSYS thermal analysis software. It was found that the bridge membrane temperature went up to above 1000°C in between 10ms to 20ms.

With the miniaturization of thrusters in their sizes, the combustion characteristics of solid propellants for micro-thrusters are very different from propellants of the conventional large-size rocket motors. The combustion performance of propellants directly determines the quality of the thruster working process. Therefore, further study on micro-propellant combustion characteristics was needed. This paper aims to obtain the energy characteristics for four propellants: HTPB/AP, AP/NC, LS/NC, LS/SQ-2 through the thermodynamic calculations; Thermal degradation due to different proportion of the LS / NC formula and different proportion of the LS / SQ-2 propellants is studied by DSC and TG. Ignition tests due to different proportion of the propellants aim to screen out the best propellant whose energy and heat sensitivity meet the requirements for micro-thrusters. The resulting thrust and impulse by impulse tests for the quality of different charge micro thrusters were compared with numerical simulation results. Study on energy and combustion properties of solid propellants for micro-thruster to provide support for research on rocket propulsion systems.

2. Energy characteristics of propellants

Thermodynamic calculations of the combustion chamber and nozzle were performed using the Minimum Gibbs free energy method [10-11] to get some parameters that represent the energy characteristics of the propellants. Results will help to select propellant formulations which meet the requirements by comparing energy parameters among the propellants in different oxidant, different binder and different ratio, then the thermal stability of the selected propellant will be researched. Energy characteristics was calculated by the Chemical Equilibrium with Application (CEA) software developed by NASA Glenn research center [15]. Thermodynamic calculation conditions are as follows; Initial temperature was set to room temperature at 300 K, the initial pressure of the combustion chamber was set to the atmospheric pressure 0.1MP, the combustion chamber area and nozzle throat area $A_c/A_t = \pi \times 0.25^2 / 0.106^2 = 17.47$, nozzle expansion ratio $A_e/A_t = 0.32^2/0.106^2 = 32.04$. Table1 below is the energy characteristic parameters of LS/NC(5:5). Table 2 below gives the energy characteristic parameters of the four types of propellants in different proportion.

Table 1. The energy characteristic parameters of LS/NC(5:5).

	Combustion	Throat	Outlet
Pinf/P	1.0000	1.7617	371.12
T(K)	2676.51	2483.29	1101.30
H(Kj / Kg)	-1334.73	-1710.89	-3914.14
Cp($Kj / Kg \cdot K$)	2.4765	1.9780	2.5038
GAMMAs	1.1602	1.1795	1.1574
MACH	0.035	1.000	4.188
CF	0.0243	0.6700	1.7535
C^*	1295.4	1295.4	1295.4
I_{sp} （m/s）	31.4	867.9	2271.5

Table 2. The energy characteristic parameters of the propellant.

Propellant	Code	Ratio	Energy Characteristics				
			T_f/K	n_g /mol·kg^{-1}	γ	I_{sp}/ N·s·kg^{-1}	C*/m·s^{-1}
LS/NC	A1	80/20	2738.5	24.2	1.1	2087.2	1190.1
	A2	70/30	2715.7	26.1	1.1	2150.3	1226.3
	A3	60/40	2695.3	28.0	1.1	2211.7	1261.4
	A4	50/50	2676.8	29.9	1.1	2271.6	1295.4
LS/SQ-2	B1	80/20	2669..4	23.4	1.1	2028.0	1155.3
	B2	70/30	2616.8	24.9	1.1	2063.6	1175.2
	B3	60/40	2568.5	26.4	1.1	2098.8	1194.6
	B4	50/50	2523.9	28.0	1.1	2133.9	1213.5
AP/HTPB	C1	80/20	2260.6	45.8	1.2	2474.3	1423.4
AP/NC	D1	80/20	2297.9	33.5	1.3	2225.7	1259.1

From Table 2 we can conclude that

(1) Comparing A1 with B1, they both contain the same number and variety of oxidant LS, the Characteristic velocity of A1 is C*=1190.1 m/s which is 34.8m/s higher than the value for B1. The specific impulse I_{sp} is 59.2N·s/kg higher. It can be seen that energy of NC propellant is higher than SQ-2 propellant. The same conclusion can be draw by comparing respectively A2

with B2, A3 with B3, A4 with B4.

(2) C1 and D1 both contain the same number and variety of oxidant AP, the characteristic velocity of C1 is 164.3m/s higher than D1, specific impulse is 248.6N·s/kg higher. It can be concluded that energy of HTPB propellant is higher than NC propellant; Comparing A1 and D1, they both contain the same variety of binder NC, it can be seen that C* of A1 is 69m/s lower than D1, specific impulse is 138.5N·s/kg lower. It can be concluded that energy of LS propellant is lower than AP propellant.

(3) Comparing A1,A2,A3 and A4, represents an increase of the content of NC in the propellant from 20% to 50%, the characteristic velocity and the specific impulse are both increasing. This shows that increasing the content of NC can improve propellant energy. This is also true when the content of SQ-2 is increased, as for the cases B1, B2, B3 and B4. This shows that increasing the content of SQ-2 can improve propellant energy.

The above analysis shows that in the same binder, energy of AP propellant system is higher than the corresponding LS propellant system; In the same oxidant, energy of HTPB propellant is higher than the NC propellant, energy of NC propellant is higher than the SQ-2 propellant; Increasing the content of SQ-2 or NC can improve propellant energy.

AP / HTPB has the highest energy and obviously impulse advantages among the four types of propellants, but as a propellant for micro-thrusters, energy is not the most important factor. Since the size of solid micro-thrusters is very small, and this kind of thrusters use heating resistance wire or semiconductor bridge ignition [13], their charge requires the propellant to have a high thermal characteristics, short ignition delay time, excellent loading performance and other conditions. The traditional AP/HTPB propellant has high specific impulse, but due to its excellent thermal stability, it is difficult to ignition in normal solid micro-thruster. It is suggested to choose LS with high thermal characteristics as the main components of propellant. Due to the addition of appropriate amount of binder, such as NC and SQ-2 which contains the NC can improve propellant energy, and improve the processing technology of the explosive LS，so different ratio of the LS/NC system and the LS/SQ-2 system was chosen for further analysis of the thermal decomposition.

3. Thermal decomposition characteristic analysis

3.1 Test instruments

Tests in this paper were performed using the High Pressure Differential Scanning Calorimeter (DSC) thermal analyzer produced by Swiss Mettler Company. The model is HP DSC827e, its temperature range is 0 ~ 700 ℃, and pressure range is 0.1 ~ 10Mpa. Thermogravimetry (TG) thermal analyzer produced by Swiss Mettler Company with temperature range of 0 ~ 1600 ℃ is also used. Crucibles use imported Al_2O_3 (stamped). Thrusters are measured by the balance produced by the Swiss Mettler Company, the model is Toledo AB135-s.

3.2 Test conditions

The experiment was performed at atmosphere pressure, 0.1Mpa, with temperature range 50 ~ 500℃ for the DSC sets, and 50~1000℃ for the TG set of measurements; The heating rate is kept at 15℃/min for both DSC and TG sets of measurements. The sample size is smaller than 1mg, and the sample amount is between 0.5~0.7mg. The atmosphere: is pure N_2.

3.3 Test program

In order to compare the influences of NC and SQ-2 on the thermal decomposition characteristics of LS, three propellants with different ratios of LS/NC and four different ratios of LS/SQ-2 were designed, as shown in Table 3. By comparing the three LS/NC formula, adding different proportions of NC to LS, the changes in the thermal stability of LS was investigated. At the same time by comparing the four LS/SQ-2 propellant formula, adding different proportions of SQ-2 to LS, the changes to the thermal stability of LS were investigated. Finally, the same proportion of LS/NC and LS/SQ-2 formula was compared. Those comparisons will help to identify the best components to reduce the thermal stability of LS. The lowest thermal stability formula will be chosen as the solid propellant type for micro solid propellant rocket motors.

Table 3. Tested Propellant Formulation of LS/NC and LS/SQ-2.

Propellant Formulation	Ratio			
LS/NC	—	5:5	6:4	7:3
LS/SQ-2	4:6	5:5	6:4	7:3

3.4 Discussion of test results

3.4.1 The thermal decomposition curve of LS/NC propellant and the results analysis

The peak temperature Tp when propellant was decomposing is used to evaluate the thermal stability of propellants. The energy characteristics of propellant was showed by the exothermic quantity ΔH of a unit mass of the sample. DSC curves of pure NC, pure LS and three mixtures of LS/NC with different ratios are shown in Fig. 1, Fig. 2 and Fig. 3 respectively. Table 4 gives the corresponding characteristic temperature and exothermic quantity of the three curves in Fig. 3.

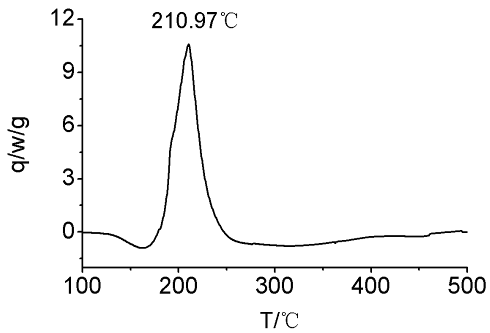


Fig. 1. DSC curve of pure NC.

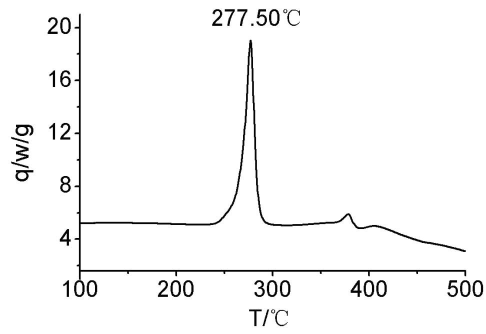


Fig. 2. DSC curve of pure LS.

Table 4. Characteristic temperature and exothermic quantity of the three ratios of LS/NC (The average value of many tests in the table).

Propellant	Ratio	T_{f1}/°C	ΔH_{f1}/J/g	T_{f2}/°C	ΔH_{f2}/J/g	$\Delta H_{f1}+\Delta H_{f2}$/ J/g
	5:5	210.33	895.83	259.87	782.06	1677.89
LS/NC	6:4	209.12	719.61	258.15	676.97	1396.58
	7:3	210.20	409.58	257.45	527.74	937.32

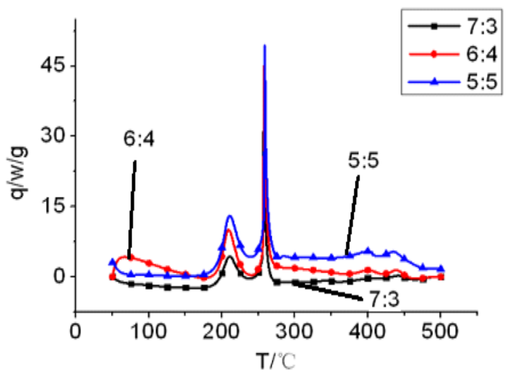


Fig. 3. DSC curve of LS/NC(7:3,6:4,5:5).

Fig. 1 and 2, give the LS and the NC exothermic peak, which shows that the decomposition of LS can release large amount of heat in a very short time, the decomposition peak temperature is 277.50°C. In contrast, the decomposition of NC is relatively slow. Its decomposition peak temperature is 210.97°C, which is lower than that for LS. The LS/NC system has two exothermic peaks, denoted by f1 and f2, corresponding peak temperatures as T_{f1},T_{f2}, and heat recorded as ΔH_{f1}, ΔH_{f2}.

It can be seen that the peak temperature f1 of the three LS/NC propellants is the same as the peak temperature of NC, which shows that f1 in the LS/NC system is the decomposition peak of NC; The peak temperature f2 of three of propellants are 17°C lower than the LS peak temperature, which shows that joining NC can reduce the LS decomposition temperature. This is due to the exothermic reaction between the breakdown products of NC such as NO, CO, NO_2 and other active materials with LS and its decomposition products, which reduces the LS decomposition temperature peak. It is quite conceivable that NC can reduce the LS thermal stability.

3.4.2 The thermal decomposition curve of LS/SQ-2 propellant and the results analysis

DSC/TG curves of SQ-2 and the four mixtures of different ratios of LS/NC are shown in Fig. 4 and 5. Table 5 presents the characteristic temperature and heat release of the four curves corresponding to Fig. 5.

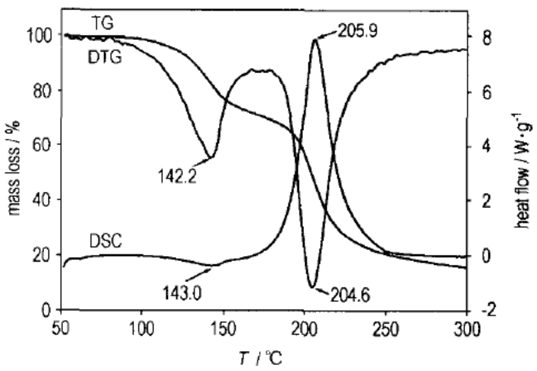


Fig. 4. DSC and TG-DTG curves of SQ-2.

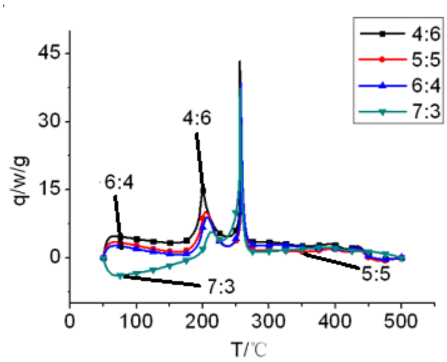


Fig. 5. DSC curves of LS/ SQ-2 at ratios of 7:3,6:4,5:5,and 4:6.

Table 5. The characteristic temperature and heat release of four kinds of LS / SQ-2 ratio.

Propellant	Ratio	T_{f3}/°C	ΔH_{f3}/J/g	T_{f4}/°C	ΔH_{f4}/w/g	$\Delta H_{f3}+\Delta H_{f4}$/ J/g
LS/SQ-2	4:6	200.85	832.81	257.79	494.65	1384.53
	5:5	204.00	743.20	260.44	525.71	1268.91
	6:4	208.73	594.78	261.76	644.74	1239.52
	7:3	211.52	255.96	263.58	748.81	1004.53

The SQ-2 DTG curve from Fig. 4 shows two mass loss processes: the first peak, whose peak temperature is 142.2℃, corresponds to the main NG (Nitroglycerin) volatilization or decomposition, because in the 140℃, low loading density conditions, nitroglycerin thermal decomposition curve is deceleration. Accelerating trend is not obvious in the initial stage of thermal decomposition of nitroglycerine. A second peak appears at temperature 204.5℃, and is believed mainly correspond to the decomposition of NC which can be known in Fig. 1; From the DSC curve a more steady endothermic process and a large exothermic peak can also be seen, the former peak temperature is 143.0℃ corresponds to the DTG curve of the nitroglycerin volatile peak, the latter peak temperature is 204.6℃, which is the same as DTG peak temperature of the mass loss of NC thermal decomposition. Fig. 5 shows that there are two decomposition exothermic peak in LS/SQ-2 propellant series, denoted by f3 and f4, the peak temperature recorded T_{f3}, T_{f4}, heat release are denoted as ΔH_{f3}, ΔH_{f4}.

It can be seen from Figs. 4 and 5 and Table 5 that the first peak temperatures (200.85~211.52℃) of the four LS/SQ-2 are close to the decomposition peak temperature of SQ-2 (205.9℃), therefore it can be concluded that the first peak temperature f3 corresponds to the decomposition peak temperature of SQ-2. The second peak temperature f4 is the decomposition peak temperature of SQ-2 under the action of LS. The second peak temperature of LS/SQ-2 is about 20℃ lower than the peak temperature of pure LS. The reasons are as follows: first, the decomposition of SQ-2 is the main decomposition of nitroglycerin and nitration cotton. Thermal decomposition of nitroglycerin are made up of two steps: the first step is the breaking of O-NO_2, to release NO_2, and to generate substances such as RCHO; The second step is the process that autocatalytic of nitric acid ester was accelerated by NO_2. The initial stage of the decomposition process of nitrification cotton is heavy denitration of O-NO_2 to escape NO_2, and followed by breaking of the oxygen bridge between the rings, the last is breaking of the oxygen bridge between the ring and carbon skeleton. Second, SQ-2 also contains lead oxide, the lead oxide act as a catalyst in the sub-surface area or surface area, to change the decomposition process of nitrate and LS. So the decomposition products of SQ-2 and lead oxide can make a small amount of the LS first decompose to reduce the peak temperature of f3, thus SQ-2 also can reduce the thermal stability of LS.

Compare different ratios of LS/SQ-2, it can be found that increasing the proportion of SQ-2, the peak temperature f4 tends to decrease, indicating that a small amount of SQ-2 decomposition products and LS react, lowering the LS exothermic peak temperature, which reflected SQ-2 plays a catalytic acceleration on the decomposition of the LS.

3.4.3 Compare thermal stability between LS/NC and LS/SQ-2

In order to compare the influence degree of NC and SQ-2 on the thermal stability and the energy characteristic of LS, thermal decomposition curves of the same ratio of the LS/NC and LS/SQ-2 are compared and analyzed. Fig. 6, 7 and 8 are the thermal decomposition curves of LS/NC and LS/SQ-2 at ratios for 5:5 6:4 7:3 respectively

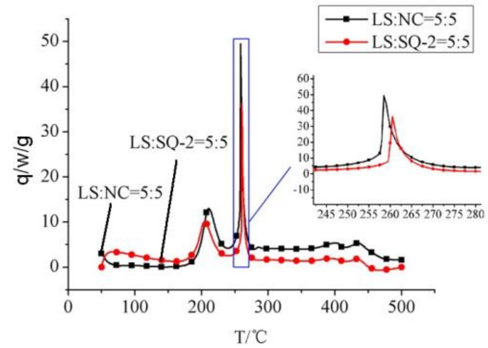


Fig. 6. Thermal decomposition curve of the ratio for 5:5 of LS/NC and LS/SQ-2.

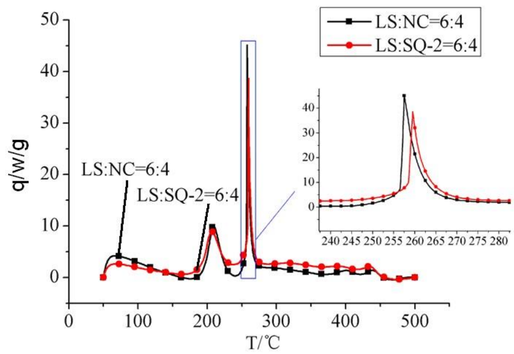


Fig. 7. Thermal decomposition curve of the ratio for 6:4 of LS/NC and LS/SQ-2.

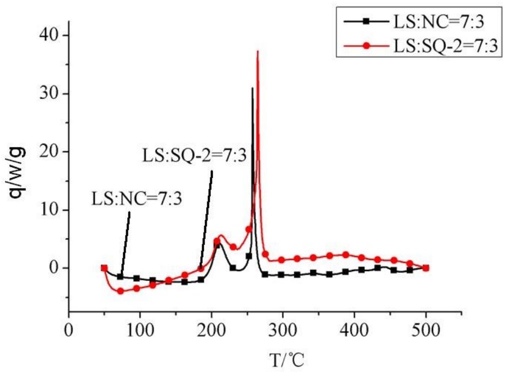


Fig. 8. Thermal decomposition curve of the ratio for 7:3 of LS/NC and LS/SQ-2.

The second peak temperatures of the three LS/NC mixtures are lower than that of the LS/SQ-2, For example, the second peak temperature of the propellant with a ratio of 7:3 of LS/NC is 6℃ lower than that of LS/SQ-2. This shows that the ability of NC in reducing the thermal decomposition temperature of LS is higher than that of SQ-2. Generally LS/NC propellant has low thermal stability, meets better the requirements for thermal stability of micro propellant. With the ratio of two components gradually reduces (i.e. ratio for 7:3 down to 5:5), the second peak temperatures of LS/NC and LS/SQ-2 get closer, which shows that as the content of NC and SQ-2 is increased, the difference between them in improving the thermal stability of LS is reducing. NC and SQ-2 both could reduce the thermal decomposition temperature of LS, NC has more significant effect. The ratio for 5:5 of LS/NC has the lowest thermal stability。

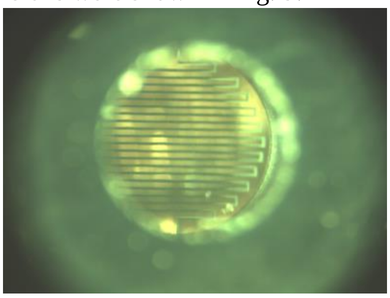
4. Micro propellant ignition experiments

4.1 Experimental design

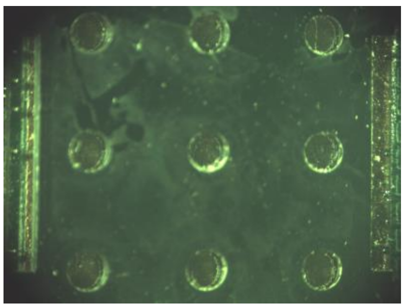
The principle of micro solid rocket propulsion is that the ignition circuit ignited the solid propellant, then the propellant burned in the combustion chamber produces high-temperature and high-pressure gas, the gas is ejected from the nozzle to generate counter-thrust. The ignition circuit is the most important and most complicated preparation process, most difficult part in the micro solid engine. The current common ignition is mainly the resistance wire heating fire.

The circuit mainly consists of the substrate, the ignition resistance, wires and a pad. The ignition circuit substrate commonly uses a silicon (surface to be oxidized) or a heat resistant glass through coating, photoetching, dry etching to make the ignition resistance. Wires and pads form on the substrate. Whether or not a micro-thruster can work properly depends mainly on the reliability of the ignition circuit and the ignition power, so the most crucial part of the circuit is the ignition resistance.

In the experiments propellants with different ratios of LS / NC and LS/SQ-2 as shown in Table 6 are first produced by an extrusion grain molding device, the grain is cylindrical and the diameter is about 0.5 mm. Ignition experiments used semi-finished MEMS solid micro-thruster. The structure is micro machined ignition wire array: the aluminum ignition wires placed on the 0.5 mm thick silicon wafer. The developed ignition resistance wires dimensions were shown in Fig. 9.



(a) Ignition wire distribution



(b) Micro-thruster array for ignition

Fig. 9. The ignition resistance wires.

Table 6. Propellant formulation of LS/NC and LS/SQ-2.

Number	LS (%)	SQ-2 (%)	NC (%)
#1-1	50	50	0
#1-2	60	40	0
#1-3	70	30	0
#1-4	80	20	0
#2-1	50	0	50
#2-2	60	0	40
#2-3	70	0	30
#2-4	80	0	20

4.2 Experimental results

When the micro-thruster is at work, constant voltage electric power is supplied, which can save energy and is easy to control. The constant voltage power supplied to energize the tested propulsions. At the same time, the high-speed camera system records work processes of the ignition device at a photography frequency of 8000 frames / sec. The ignition experimental results were shown in Table 7, each result is the average of three experiments, the difference of each result is not more than 5%. Fig. 10 is an ignition experiment picture.

Table 7. The ignition experimental results.

Ignition Power（W）	2.16	2.43	2.61	2.80	3.00	3.20	3.41
#1-1	No	No	Yes	-	-	-	-
#1-2	No	No	No	Yes	-	-	-
#1-3	No	No	No	No	Yes	-	-
#1-4	No	No	No	No	No	Yes	-
#2-1	Yes	Yes	Yes	-	-	-	-
#2-2	No	Yes	Yes	-	-	-	-
#2-3	No	No	No	Yes	-	-	-
#2-4	No	No	No	No	Yes	-	-

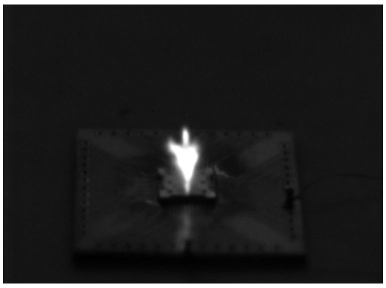


Fig. 10. A typical picture from the ignition experiments.

From Table 7 we can conclude that

(1) Comparing #1-1 and #2-1, they both contain the same number and variety of oxidant LS，the minimum ignition power of #1-1 is 2.61W while #2-1 is 2.16W. It can be seen that ignition performance of NC propellant is better than SQ-2 propellant. The same conclusion can be draw by comparing respectively #1-2 and #2-2, #1-3 and #2-3, #1-4 and #2-4.

(2) Comparing #1-1,#1-2,#1-3 and #1-4, represents a decrease of the content of SQ-2 in the propellant from 50% to 20%, ignition powers are increasing. This shows that decreasing the content of SQ-2 will reduce propellant ignition performance. This is also true when the content of NC is decreased, as for the cases #2-1, #2-2, #2-3 and #2-4. This shows that decreasing the content of NC will reduce propellant ignition performance.

The above analysis shows that the LS / NC mixture has a better ignition performance than the LS/SQ-2 combination at the same ratio. Equal proportion of LS to NC at 5:5 produces an ignition power of 2.16W, which is less than other matching propellant ignition power. Possible reasons for this result are: the thermal decomposition figures from the previous show that the initial decomposition peak temperature of LS is higher, instead of the initial decomposition peak temperature of NC and SQ-2 is low, when the content of LS in the propellant increases, the required initial ignition temperature of the propellant increases, therefore the corresponding ignition power will increase. The decomposition peak temperature of the LS/NC mixtures are lower than that of the LS/SQ-2 with the same ratio, so less power is required to ignition. However, after the LS starting to decompose, the reaction is very fast and intense, the initial decomposition peak temperature of NC and SQ-2 is low, but the subsequent reaction is more gentle, so the content of LS in the propellant is higher, the reaction after ignition is more intense. After summing up energy characteristics, thermal decomposition characteristics and ignition performances, LS / NC is more suitable to be used as a micro-motor propellant.

5. Numerical simulations on performance of Micro-Thruster

In order to predict the thrust and impulse which the solid micro-thruster produces after the charge burning, the three-dimensional numerical simulation was carried out for the thruster.

5.1 Physical model

Simulations were performed for the solid micro-thruster used in the above experiments. A schematic diagram and the dimensions of the model are shown in Fig. 11 and 12. The combustion chamber pressure is $1MPa$, the nozzle exit pressure is 1000Pa; The inlet gas mass flow rate is $0.15g/s$. The thruster shell material is silicon, the thermal conductivity is $157w/(m \cdot k)$, the density is $2400kg/m^3$, the specific heat is $700J \cdot kg^{-1} \cdot K^{-1}$. The combustion products were assumed to complete gas, the specific heat is a function of temperature.

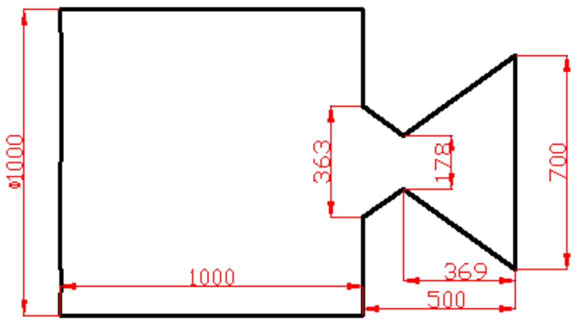


Fig. 11. Micro-thruster structure dimension (The unit is μm).

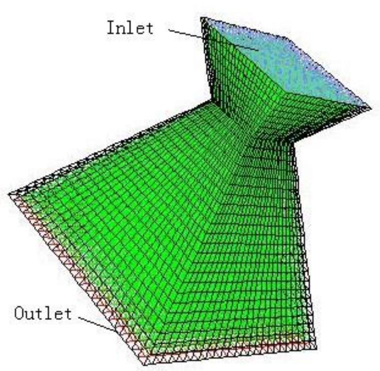


Fig. 12. The schematic diagram of micro-thruster nozzle.

5.2 The initial and boundary conditions of calculation

In the numerical simulation, the following types of boundary conditions were applied: inlet boundary conditions: the inlet boundary was used in the mass flow rate. The mass flow rate is $0.15\,g/s$ in the equalizing section (Fig. 13). Exit conditions: The pressure at the nozzle exit is 1,000Pa, considering the case to beadiabatic, no slip wall was used at the walls. In consideration of heat loss, the wall where gas and solid contact with was used the thermal coupling interface, the velocity took no slip condition.

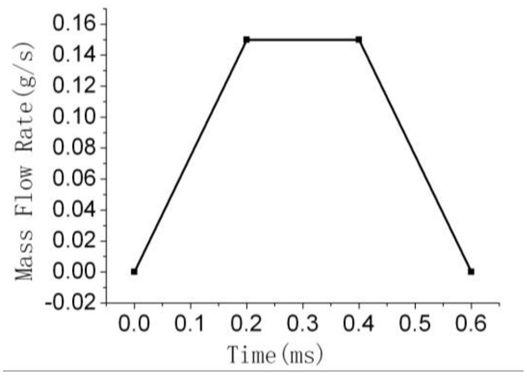


Fig. 13. The inlet mass flow rate.

5.3 Analysis of results

The micro-thrusters with the different charge weights which were 0.08mg and 0.1mg were simulated by using a turbulence model, the working time is 0.6ms. The thrust curve from calculation is shown in Fig. 14.

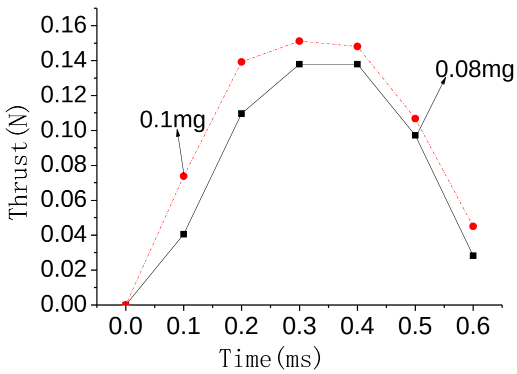


Fig. 14. The thrust curve of the different charge weights.

When the charge weight is 0.08mg, the micro-thrusters pulse peak was 0.1379N, total impulse is $5.373 \times 10^{-5} N \cdot s$ from calculations. Increasing the charge weight to 0.1mg with other conditions kept the same, the micro-thrusters pulse peak was 0.1512N, the total impulse is $6.415 \times 10^{-5} N \cdot s$, the thrust peak increased 9.64%, the total impulse is increased by 19.4%. From the calculation results, higher the charge weight of the micro-thruster, greater the thrust. The charge weight has strong great effect on thruster performance, so in the follow-up study, the control on the charge weight of micro-thrusters is particularly important.

6. Thrust test experiments of micro-thrusters

6.1 Test program

In the experiments, the impulse of the solid micro-thrusters was tested by a pendulum system. The impulse which is the accumulation of the force over time is equal to the momentum increment of force vectors. Pendulum is an indirect impulse measuring device which makes the tested impulse transform into the swing angle. Fig. 15 shows the impulse test system used in the experiments, which consists of pendulum, laser interferometer, damper and damping platform etc.

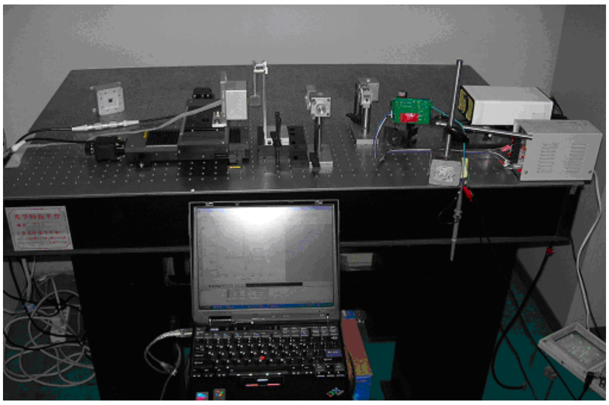
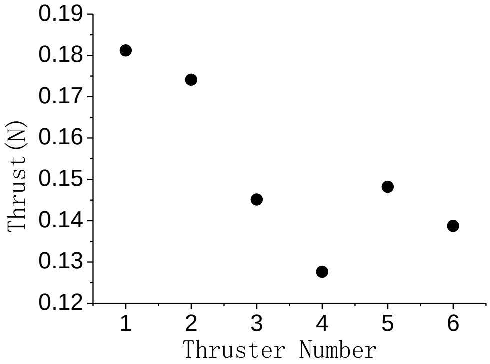


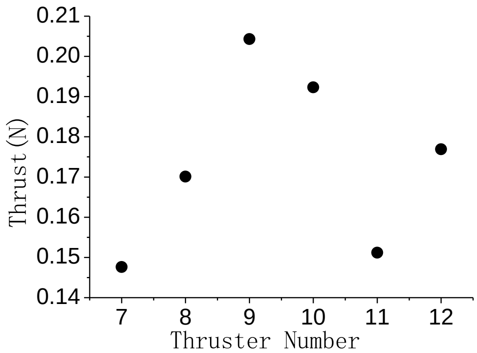
Fig. 15. The Impulse Test System.

6.2 Test results

The simple propellant is difficult to ignition, but the sensitivity of simple dynamites or primary explosive are too high, they are difficult to control and outbreak in a very short time, also have a short duration of action. In this experiment the propellant formulation of 6:4 LS / NC was chosen, the ignition tests were carried out on a plurality of thrusters. The measured thrusts are shown in Fig. 16.



(a) The thrust data of 0.08mg



(b) The thrust data of 0.1mg

Fig. 16. Thrust Test Data.

If the charge weight of thrusters is 0.08mg, according the tests, the average thrust peak is 0.1543N, the average thrust impulse is about $6.34 \times 10^{-5} N \cdot s$. The thrust peak is 11.89% higher, the total impulse is 17.99% higher than the calculated value from the simulations. When the charge weight of thrusters is increased to 0.1mg, the average thrust peak is 0.1789N, the average thrust impulse is about $7.48 \times 10^{-5} N \cdot s$, the measured thrust peak is 18.3% higher than the calculated value, the total impulse increases by 16.6%. The experimental data are shown in Table 8. Experimental and theoretical values have a certain gap, which may have a number of factors: first, the theoretical calculation based on the macro-thruster model which may not be applicable for the millimeter and micrometer thruster; second, the thrust also has relation to propellant charge density, the combustion performance and many other factors, the theoretical calculation has many approximate parts, which seriously affecting the accuracy of the calculation; Third, the working process of micro-thruster charge may not be ordinary combustion process and entirely the detonation process, but it is closer to deflagration state in the combustion process.

Table 8. Test data.

Number（0.08mg）	Thrust Peak（N）	Number（0.1mg）	Thrust Peak（N）
1	0.1812	7	0.1476
2	0.1741	8	0.1701
3	0.1451	9	0.2043
4	0.1276	10	0.1923
5	0.1482	11	0.1512
6	0.1387	12	0.1769
Calculation Value	0.1379		0.1512

It can be seen from Fig. 16 that there exists large variation in the measured thrusts , which is caused by two main reasons. First, the difference of charge weight and density, lead to the great different on thrust and duration of action; Second, the heat resistance and the ignition power are different, which caused the different on the intensity of propellant combustion, thus affected the burning time and energy of propellants. Two improvements were proposed, firstly the charge method should be standardized, the charge weight and density are unified; Secondly because of the different resistance values of resistors is largely affected by the processing technology, the next step, the process parameters should be further improved, to ensure the approximative resistance value, the same heating power, the unity burning intensity. Although some differences exist between the theoretical results of thrust and impulse and the measured, the trend is consistent, with the charge weight of micro-thrusters increasing, thrust and impulse is greater.

7. Conclusions

(1) The thermal decomposition peak temperature of LS is 277.50°C, adding NC can reduce the decomposition temperature of LS, the magnitude of the drop is about 20°C. Increased propellant energy is found after adding NC.

(2) SQ-2 can reduce the thermal stability of LS. The decomposition temperature of LS is reduced by 14°C after adding SQ-2, the propellant energy can also be improved.

(3) It is found that NC can significantly reduce the thermal stability of LS propellant system and improve the propellant energy, and is better than SQ-2. The LS/NC performs better and is more suitable as energy sources for micro solid rocket motors than LS/SQ-2 at the same ratio.

(4) For LSNC, equal proportion at 5:5 produces an ignition power of 2.16W, which is less than other matching propellant ignition power.

(5) If the charge weight is 0.08mg, the micro-thruster produces a pulse peak of 0.1379N, and the total impulse is $5.373\times10^{-5} N\cdot s$ from calculations. Increasing the charge weight to0.1mg, the pulse peak is 0.1512N, and the total impulse is $6.415\times10^{-5} N\cdot s$, giving a thrust peak increase of 9.64%, and the total impulse increase of 19.4% .

(6) The measured thrust peak is 11.89% higher than the calculated value. The measured total impulse is 17.99% higher. When the charge weight of thrusters is increased to 0.1mg, the average thrust peak is 0.1789N, the average thrust impulse is about $7.48\times10^{-5} N\cdot s$, the thrust peak increases 18.3% than the calculation value, the total impulse increases 16.6%.

(7) Although some differences exist between the theoretical results and the measured, the trend is consistent, with the charge weight of micro-thrusters increasing, thrust and impulse is greater.

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