Global Journal of Engineering and Technology Advances
eISSN: 2582-5003
Cross Ref DOI: 10.30574/gjeta
Journal homepage: https://gjeta.com/

(RESEARCH ARTICLE)

Check for updates

Effect of magnesium metal in the characteristics performance of a sucrose-based solid rocket propellant

Oyedeko K.F.K [1,*] and Egwenu S. O [2]

[1]*Department of Chemical Engineering, Lagos State University, Lagos, Nigeria*
[2]*Rocket propellant unit, Centre for Space Transport and Propulsion, Epe, Lagos Nigeria*

Global Journal of Engineering and Technology Advances, 2021, 06(02), 051–060

Publication history: Received on 10 January 2021; revised on 06 February 2021; accepted on 08 February 2021

Article DOI: https://doi.org/10.30574/gjeta.2021.6.2.0016

Abstract

This research work aimed at investigating the effects of magnesium metal (powder) and carbon on a potassium nitrate-sucrose (KNSU) solid propellant formulation. Characterization of propellant is very important to determine its performance before it can be suitable for use for a rocket flight or any mission. Ballistic loadcell method was used. The ballistic load cell instrumentation measured the thrust generated by the propellant, the propellant burn time and the exit temperature of the burning hot propellant gases. The carbon constituent which acts as an opacifier and coolant was kept constant at 2% in order to arrest some of the heat during the combustion process and helped to lower the combustion temperature, because high combustion temperature could lead to combustion chamber rupture or failure. Also, carbon was not increased beyond 2%, so as not to make the propellant excessively smoky because of presence of magnesium oxide and other solids in the combustion products that can cause air pollution, and could be harmful to human lives and the environment. The propellant specific impulse (117.9s), combustion temperature (1818K), heat ratio (1.1508), propellant molecular weight (38.88g/mole), propellant density (1874.6kg/m3), characteristics velocity (997.2m/s) and burn rate (0.00906m/s) were obtained. The effect of addition of magnesium which was optimized for 3% in the formulation contributed significantly in improving the overall performance of the propellant as parameters such as the specific impulse, chamber temperature, characteristics velocity and heat ratio were found to have higher values as compare to the KNSU propellant when magnesium was not present in the formulation. Basically, higher values of these parameters suggest better propellant performance. Also, in this case, when carbon was increased beyond 2%, the propellant was excessively smoky because of presence of magnesium oxide and other solids in the combustion products that can cause air pollution, and could be harmful to human lives and the environment.

Keywords: Magnesium; Propellant; Carbon; Potassium nitrate; Characterization; Rocket

1. Introduction

Most modern missile systems e.g Rocket motor make use of propellants to produce thrust force require to propel them [1]. Propellants consist mainly of fuel and oxidizer with some additives. An oxidizer produces oxygen for reaction with the fuel to give requires proportion [2]. Propellants are also of different types solid, liquid or hybrid based on the constituents of oxidizer and physical state of the fuel [3].

Solid propellants have been the commonly used due to reliability, cost effective and simple to design [4,5]. Many researchers have been working over the years to improve and develop new propellant formulations significantly to meet the need of the type reaching targets and missions with less weight [4,5]. and propellant mass fraction can be sometimes be up to 70-90% of the rocket total mass, then improving and developing new propellant formulations to meet this need is therefore very significant [6]. Potassium nitrate is so chemically stable that it is absolutely impossible

*Corresponding author: Oyedeko K.F.K
Lagos State University, Department of Chemical Engineering.

to make them burn or explode on their own. Potassium nitrate is well compatible with magnesium and sucrose fuel/binder. Ammonium nitrate on the other hand is very hygroscopic, and the absorption of moisture will degrade propellant made with ammonium nitrate. It is also more difficult to ignite compare to potassium nitrate [7,8].

Sucrose is used as the main fuel as well as the binder in this formulation. Sucrose is preferred to other sugar fuel such as sorbitol and dextrose because it is an excellent amateur propellant, providing good performance with high reproducible and reliable results. It has also showed more strength and toughness and provide eleven (11) number of oxygen atom for combustion with the oxidizer ($C_{12}H_{22}O_{11}$). It is available, cheap and safe to handle especially the table sugar, [8,9,10,11]

In this study we seek to improve the performer of KNSU propellant which was found to have a lot of exhaust smoke compared to fuel like NC-NG propellant, a relative low smoke fuel of high desired and military applications which is another very important reason that informs this study.

Also, KNSU propellant has some shortcomings of brittleness, hygroscopic nature and caramelization amongst others [11], but with the inclusion of carbon, the challenges of hygroscopicity and brittleness is tackled. The carbon acts as an opacifier absorbing some of the heat generated from the combustion thereby reducing the chamber temperature that could lead to rupture of the rocket motor. Very importantly, it also helps to prevent the propellant from absorbing moisture from the air, keeping it from becoming gummy and soft and helping to maintain the propellant dryness and ease of ignition Magnesium metal or powder is the fuel in the propellant mixture. A metalized fuel is needed to supply the intense heat required to sustain combustion with potassium nitrate. Other metals, such as aluminum, will work well, but aluminum can oxidize into aluminum oxide (Al_2O_3) in the process of combustion. These oxide particles tend to agglomerate, to form a condensed phase of liquid or solid larger particles and solidify in the nozzle as the gas temperature decreases. When in the liquid state, the oxide can form a molten slag which can accumulate in pockets and cause choking around the nozzle (mostly for nozzles that were not properly designed) and that can lead to erosive burning. These increase the problems of nozzle material erosion, heat transfer and nozzle material compatibility. It also complicates the prediction of theoretical specific impulse.[7,8,11,12]. However, magnesium was chosen because magnesium oxide (MgO) does not pose challenge to the nozzle, works well with sugar fuel/ binder and produces good propellant flames to reduce smoke.[4,5,8,11].

2. Material and methods

This method involves the selection of the propellant constituents, propellant preparation, performance parameters measurements and experimental testing for the model validation. Propellant fuel constituents or ingredients such as sucrose and magnesium were varied to see how their interactions improved the performance and their effects on the propellant formulation were studied. The propellant constituents used are oxidizer (potassium nitrate), fuel or binder (sucrose), fuel (magnesium) and opacifier (carbon). The propellant characteristics performance parameters of interest in this work are the specific impulse, density, combustion temperature, characteristic velocity, molecular weight and heat ratio of the propellant. The complete test procedure involves three stages: sample preparation, setting up the testing equipment and carrying out the ballistic test.

The propellant was produced using the re-crystallization method in order to obtain a more compact propellant fuel. The selected constituents of the propellant fuel were potassium nitrate, sucrose, magnesium and carbon and their proportion were in the ranges as follows 65%, 23–35%, 0–10%, and 0-2%, respectively. These chemical constituents or species were chosen for their performance, availability in the country, safety of handling the chemicals and ease of molding to the desired shape. For a 3.55kg weight of the propellant used in the experimental test, the appropriate quantity of each constituent was weighed as obtained from the formulation varying the percentage composition of sucrose and magnesium in order to have ten sample tests as shown in Table 2. The oxidizer (potassium nitrate) was crushed and grinded to a finer powder. The heating vessel is used to carry out the re-crystallization process, the thermostat was set to a lower temperature than that of the propellant slurry at 130 ^{0}C. Potassium nitrate with some water is first introduced. When it is thoroughly dissolved in the water, the sucrose is added, after both are well mixed, the magnesium is added, and lastly the carbon is introduced into the propellant slurry. Consistently and intermittently, uniform mixing is achieved by the aid of the stirrers. The carbon was last to be added because it immediately began to remove moisture from the propellant slurry thereby making stirring more difficult. The temperature was monitored and regulated in order to prevent caramelization of the propellant. Caramelization degrades the propellant and will not perform optimally as expected. The mixing was allowed to proceed until the propellant was ready for casting when it is observed that all constituents had been uniformly mixed together and a semi solid mixture was achieved. Maintaining uniformity and casting into desired shape was much easier. The presser is used for the removal of the air trapped inside

the propellant grains. This was to allow a higher burning rate with higher and uniform density. The propellant slurry (semi-solid) was then cast into a casting mould (Figure 1(a)) using the cast-in-place-core method to form the grain geometry and was then vibrated on a vibrator (Figure 1(b)) for about 10 s to ensure consistency and relatively flaw-free propellant grain. The casting mould was then removed after 2 hours, and the propellant was allowed to cure (solidify and dry) usually for 48 hours or more and the propellant is now ready for testing as shown in Figure 1(c).

A. 

B. 

C.

D. 

Figure 1 Propellant process stages

Table 1 Propellant production stages

Figure	**Process**
Figure A	Propellant mould/core
Figure B	Propellant Shape mould/inhibitor
Figure C	Propellant vibrator
Figure D	Propellant grain

Table 2 Percentage Composition of Propellant Samples Prepared.

Samples	**%KN**	**%SU**	**%Mg**	**%C**
Control	65	35	0	0
1	65	33	0	2
2	65	32	1	2
3	65	31	2	2
4 (optimized)	65	30	3	2
5	65	29	4	2
6	65	28	5	2
7	65	27	6	2
8	65	26	7	2
9	65	25	8	2
10	65	24	9	2
11	65	23	10	2

To setup a test, the propellant sample is fixed into the combustion chamber and a pyrogen igniter installed usually at the nozzle exit. The combustion chamber or rocket motor is then closely tightened up and placed in the load cell test stand with the pressure transducer inside to take readings to be converted to thrust. The load cell test-stand before carrying out a test, is calibrated with a dead weight pressure tester. The igniter is connected to a battery and a safety check is performed. After satisfactory pre-tests, the burn test takes place. The data acquisition system is initiated and the sample is burned. During a test large amounts of heat are produced, therefore the setup is allowed to cool until it returns to the initial testing temperature before another test is conducted repeating the cycle.

Experimental measurements of propellant characteristics performance parameters were carried out using the standard ballistic evaluation method at the premises of the Centre for Space Transport and Propulsion, Epe, Lagos state. The propellant specific impulse, propellant density, combustion temperature, characteristic velocity, the molecular weight and heat ratio were computed from the measured parameters as described in equations (1) – (8), respectively [13].

The specific impulse was derived using the thrust per unit weight flow rate of the propellant.

$$Isp(s) = \frac{F}{\dot{m}_p g} \tag{1}$$

Where Isp is the specific impulse, F is the measured average thrust (N) and $\dot{m}_p$ is propellant mass flow rate of combustion gases (kg/s).

$$\rho_p = \frac{\dot{m}_p}{A_e V_e} \quad \text{or} \quad \rho_p = \frac{m_p}{V_p} \tag{2}$$

Where ρ_p is the propellant density of the each of the components (kg/m^3), $\dot{m}_p$ is the propellant mass flow (kg/s), A_e is the nozzle exit area (m^2) and V_e is the exit velocity (m/s), m_p is propellant mass, A is exposed area of the propellant, Lp is the propellant length and V_p is the volume of propellant occupied in the rocket motor defined as;

$$V_p = AL_p = \frac{\pi\left(D^2 - d_c^2\right)}{4} L_p \tag{3}$$

$$T_c = \frac{T_e \left(P_c\right)^{\frac{k-1}{k}} \left(k+1\right)}{2\left(P_e\right)^{\frac{k-1}{k}}} \tag{4}$$

Where Tc is the adiabatic flame temperature (K), Te is the nozzle exit temperature (K), Pc is chamber pressure, Pe is nozzle exit pressure (assumed to be atmospheric pressure) (Pa) and k is the heat ratio.

$$C^* = \frac{\left(\sqrt{kR_g T_c}\right)\left(V_e\right)^2}{R_g T_c \left(C_f\right)^2} \tag{5}$$

Where C^* is the propellant characteristic velocity (m/s), Rg is the gas constant (J/kg K), k is the heat ratio, Ve is the exit velocity, C_f nozzle coefficient and A_t is the nozzle throat area in (m^2).

$$C_f = \frac{F}{P_c A_t 10^6} \tag{6}$$

$$A_t = \frac{\pi d_t^2}{4} \quad (7)$$

$$Mw = 2R^{'}K\left(\frac{T_c - T}{(k-1)(I_{sp}g)^2}\right) \quad (8)$$

R is the universal constant, while the propellant heat ratio was iterated from all the equations. The heat ratio was obtained from the value that equates the measure characteristic velocity.

3. Results and discussion

The experimental results from the load cell and mathematical calculations carried out are presented in Tables 3 – 5 and Figures 2 – 10.

Table 3 Measured Parameters from The Load Cell

S/N	Parameters	Symbol	Measured Result	Unit
1	Initial Thrust Reading	F_1	147.1	N
2	Maximum Thrust Reading	F_2	1174.8	N
3	Measured Thrust	F	$F_2 - F_1$ = 1027.7	N
4	Burn time	tb	4	S
5	Exit temperature	T_e	1301	K

Table 4 Ballistic Load Cell Results of Propellant Samples

Propellant Samples	%Mg Composition	Exit Temperature Te (K)	Thrust F(N)	Burn time tb(s)
Control	0	113	1017	3.93
1	0	1108	1013	3.9
2	1	1191	1018	4.0
3	2	1222	1022	3.88
4	3	1301	1027	4.0
5	4	1297	987	4.12
6	5	1278	934	4.26
7	6	1267	894	4.31
8	7	1214	887	4.33
9	8	1056	827	4.4
10	9	1034	805	4.51
11	10	1013	782	4.6

Table 5 Result of Propellant Characteristics Performance Parameters

Sample	Specific impulse Isp (s)	Heat ratio k	Chamber Temp. Tc (k)	Characteristic velocity c-star(m/s)	Propellant density P(kg/m³)	Molecular Weight (kg/mol)
Control	**114.8**	**1.1512**	**1576**	**973.2**	**1899**	**36.59**
1	113.4	1.1547	1568	963.3	1904	36.86
2	116.9	1.1524	1677	989.4	1906	37.09
3	113.9	1.1511	1716	966.5	1908	37.24
4	118.0	1.1508	1826	997.2	1910	37.33
5	116.8	1.1497	1816	988.3	1912	37.36
6	114.3	1.1487	1786	969.4	1914	37.34
7	110.6	1.1468	1764	942.3	1916	37.28
8	110.3	1.1456	1686	939.6	1918	37.18
9	104.5	1.1444	1463	896.1	1920	37.06
10	104.2	1.1432	1429	894.3	1922	36.9
11	103.3	1.1425	1398	887.1	1924	36.74

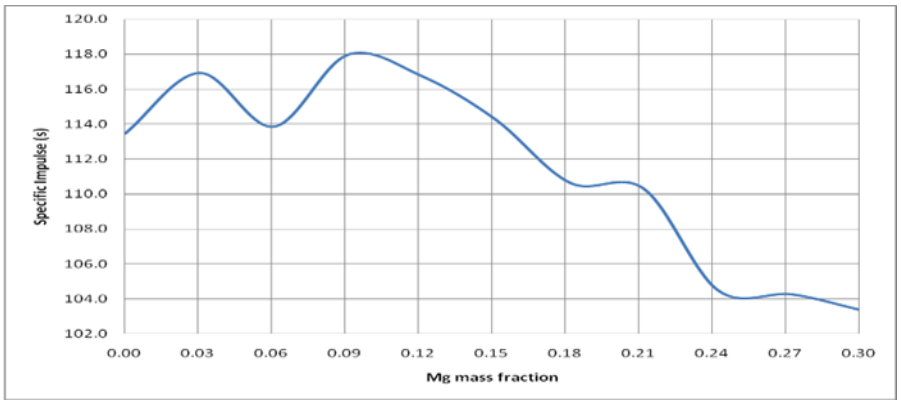

Figure 2 Specific Impulse vs. Magnesium Mass Fraction Curve

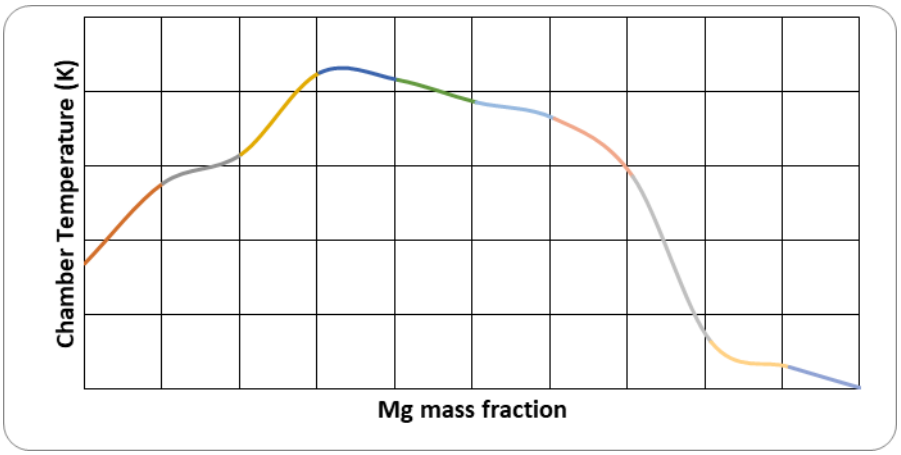

Figure 3 Chamber Temperature vs. Magnesium Mass Fraction Curve

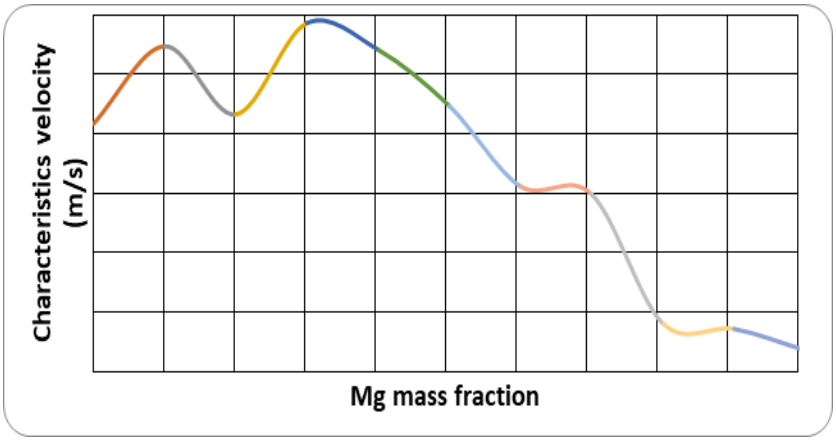

Figure 4 Characteristics vs. Magnesium Mass Fraction Curve

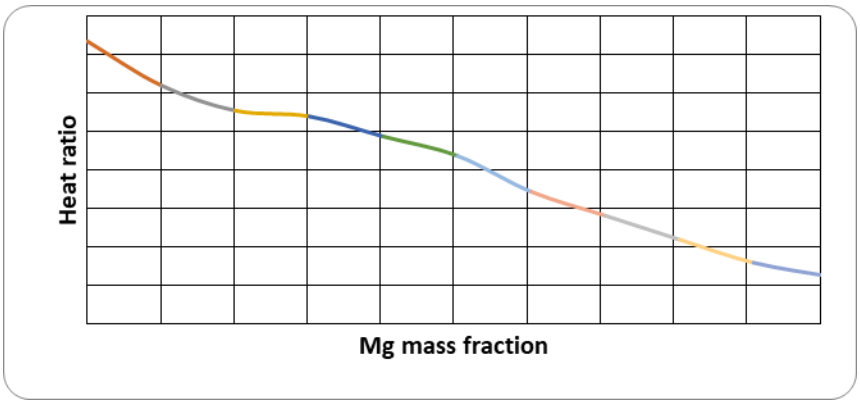

Figure 5 Propellant Density vs. Magnesium Mass Fraction Curve

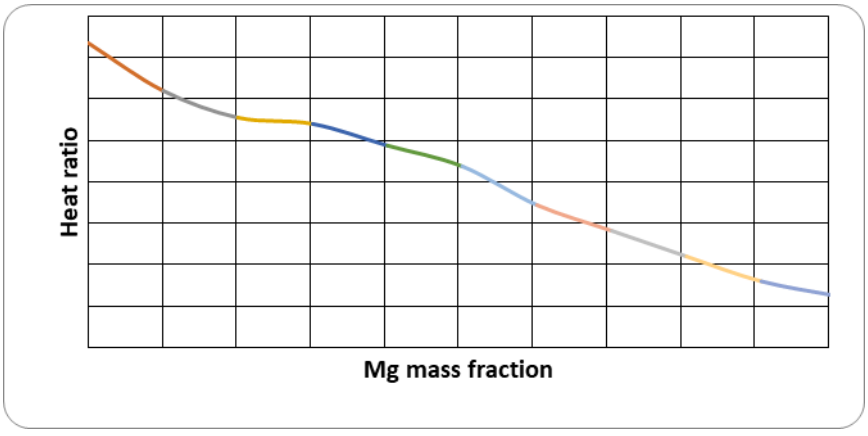

Figure 6 Heat Ratio vs. Magnesium Mass Fraction Curve

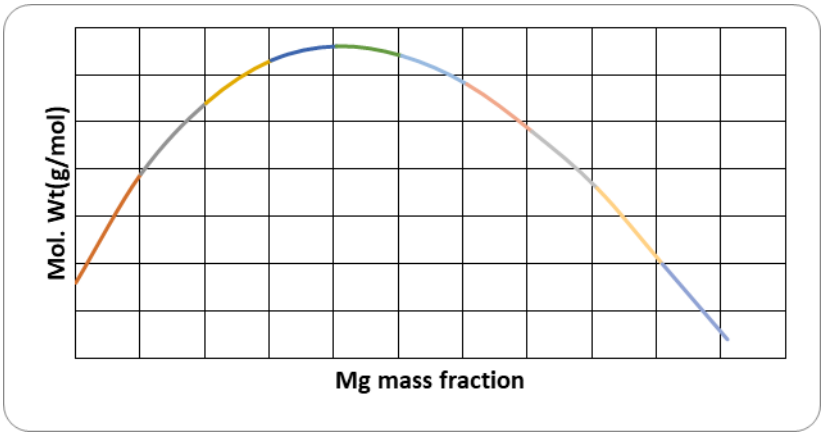

Figure 7 Molecular Weight vs. Magnesium Mass Fraction Curve

Table 6 Propellant Composition

Ingredients Used	%Composition	Mass Reqd.(kg)
Potassium Nitrate	65	32.5
Sucrose	30	15
Magnesium	3	1.5
Carbon	2	1
Total	100%	50kg

The load cell instrumentation was helpful in determining the thrust generated by the propellant and the duration of the burn, usually referred to as the burn time. Another very important parameter measured was the exit temperature of the propellant as presented in Table 4.

From the data obtained, computation analysis was employed using equations (1) – (8) to characterize the optimized propellant in order to determine the performance parameters. Table 5 is the result of the performance parameters obtained for the optimized formulation of KN(65%), SU(30%), Mg(3%) and C(2%) composition at a chamber pressure of 112Psi which gave result of propellant specific impulse (117.9s), combustion temperature (1818K), heat ratio (1.1508), molecular weight (38.88g/mole), propellant density (1874.6kg/m^3) and characteristics velocity (997.2m/s).

We observed that when the propellant formulation was ran on a propellant evaluation program (software), results were close to the result obtained from the ballistic test as well as the mathematical model problem solved. It proved that the solid rocket motor design had over 90 per cent efficiency and also, that basic assumptions made in the mathematical solutions were valid.

The test results according to Table 5 (samples 1-11) showed the effect of inclusion of magnesium in the propellant formulation as it significantly improved the overall performance parameters of the propellant. It was revealed that parameters such as specific impulse, chamber temperature, characteristics velocity, heat ratio were greatly improved upon as compare to when magnesium was not present in the formulation. Fundamentally, the higher the values of these parameters, the better the performance of the propellant.

Figure2 showed the behaviour of the specific impulse as magnesium is added to the formulation from 0-10 per cent. We saw that the specific impulse increased up to when magnesium was 3 per cent present in the formulation but started to fall when increased further.

Figure 3 also showed how the chamber temperature was behaving as magnesium is added to the formulation from 0-10 per cent. The chamber temperature increased upto when magnesium was 3 per cent present in the formulation but started to fall when increased further.

In Figure 4, the characteristic velocity graph also showed the same behaviour of rising characteristic velocity as magnesium is introduced into the formulation to about a little above 3 per cent before it began to decline.

Figure 5 showed a direct linear relationship between the propellant density and magnesium composition. The density of magnesium (1740kg/m^3) is higher than sucrose (1590kg/m^3) hence it was found to be increasing with increasing magnesium in the formulation. The implication of this is that, higher density propellant would require higher propellant mass which could make a rocket total lift-off mass higher to reach a specified altitude. This is usually avoided as efforts are to reach higher altitudes with lower propellant mass formulation. This was another cogent reason for the propellant to be optimized for 3 per cent to avoid a bulky propellant mass [6,7].

In Figure 6, the heat ratio had a direct relationship with the magnesium composition as the heat ratio kept reducing with increase in magnesium composition. Usually, the higher the chamber temperature measured, the higher the heat ratio value and vice-versa. This is the problem of workability, suitability and compatibility of the ingredients in the formulation. Magnesium workability and compatibility with sugar propellant is only minimal such that it should not exceed more than 3 per cent. Increasing the composition more than this led to lower heat ratio values that translated to lower chamber temperatures being recorded. For composite propellants, this may not be the case, as increase in magnesium composition could give higher heat ratio values and then higher chamber temperature as it can work well and more compatible with other binders/fuels such as Hydroxyl Terminated Poly Butadiene (HTPB) binder. This can be investigated to be researched upon later in the future.

Figure 7 showed a parabolic relationship between the molecular weight of the propellant and magnesium composition. Higher density usually gives higher molecular weight that results to higher propellant mass.

So, addition of magnesium to the formulation must really be justified by the amount of improvement brought upon the propellant performance. There will be no need to necessarily select and use a propellant of higher composition of magnesium when parameters such as specific impulse, thrust, characteristic velocity, chamber temperature, thrust coefficient and heat ratio are not raised to desired points when cost is excessively high.

The carbon constituent which acted as an opacifier and/or coolant was kept constant at 2 per cent in order to contain some of the heat during the combustion process from directly hitting the walls of the motor thereby limiting or reducing chamber temperature. The implication of this is that high combustion temperature could lead to the rocket motor rupture or failure. In this case, when increased beyond this percentage, the propellant was excessively smoky. Smoky fuel is the presence of high magnesium oxide (MgO) and other solids in the combustion products which result in air pollution, and can be harmful to human lives and the environment. A Smoky fuel is usually avoided especially in military application. This is the reason why fuel like NC-NG propellant which has very low soothes is employed. Also, carbon helped in aiding the curing of the propellant, improve the physical and mechanical properties such as plasticity (rubbery-like) texture [6, 7, 10].

4. Conclusion

It is important to characterize a propellant to pre-determine its performance parameters before being used for any particular mission. This was achieved by carrying out combustion of the propellant in a closed motor chamber (usually referred to as ballistic test).

The use of magnesium adequately reduces the challenge of a smoky propellant that the KNSU propellant produces, improved the specific impulse and thrust generated. The carbon was effective in addressing the challenge of hygroscopicity of the KNSU propellant and helped in uniformly distributing and containing the heat propagated from the propellant thereby lowering the combustion temperature. Also, improved curing quality of the propellant which makes it possible for longer period storage without absorbing moisture was achieved. This work has therefore solved the challenge of hygroscopicity of sugar propellants and improved the characteristics performance parameters of such propellants.

It is also very important to maintain the heating vessel during the re-crystallization process at a temperature not above 130 degree Celsius to prevent caramelization from occurring. Caramelization degrades the propellant and will not perform optimally as expected.

Compliance with ethical standards

Acknowledgments

We sincerely appreciate the contribution of Dr Adeniyi Gbadebo (head, rocket propellant unit), Engr Wole Onibon-oje, (head, combustion engines and rocket propellant division) and Dr Adetoro Lanre Moshood for the propulsion instrumentation used in this work (head, electronics and simulation division). All of Centre for Space Transport and Propulsion, Epe Lagos

Disclosure of conflict of interest

There are no conflicts of interest

References

[1] Gbadebo Omoniyi Adeniyi, Jamiu Adetayo Adeniran, Adewole Johnson Adesanmi, Funsho Alaba Akeredolu, Jacob Ademola Sonibare. Optimisation and performance evaluation of an environmentally friendly rocket composite propellant, Indian Chemical Engineer. 2018.

[2] Holzmann RT. Chemical Rockets and Flame Explosives Technology. Marcel Dekker, New York, NY, USA. 1969; 499.

[3] Jacqueline A. The Chemistry of Explosives, the Royal Society of Chemistry, Cambridge, United Kingdom. 1988; 9: 55 - 60.

[4] Liang Guozhu. Fundamentals of Rocket Propulsion, School of Astronautics, BUAA. September 2011; 440.

[5] Lyon M. (1991) Introduction to Rocket Propulsion, Propulsion DirectorateResearch, Development, and Engineering Center, U. S. Army Missile Command Redstone Arsenal. Alabama. 1991.

[6] Nakka R. Experimental Rocket Site. 2007.

[7] Nakka R. Solid Propellant Burn Rate, Experimental Rocketry Web 5, Nov. 2013.

[8] Oyedeko KF, Onyieagho A. Effect of Propellant Formulation onPropellant Properties. International Journal of Engineering Sciences & ResearchTechnology. 2018; 7(8): 305-313.

[9] Space Travel Guide. Web, 5 January 2014.

[10] Sutton GP, Biblarz O. Rocket Propulsion Elements, Wiley, New York, USA, 7th Edition. 1992.

[11] The MFC Propulsion Program, Application of Today's Propulsion Technology to Space Commercialization, Web. 10 December 2013.

[12] Yildirim C. Analysis of Grain Burn back and Internal Flow In Solid Propellant Rocket Motors in 3-Dimensions, Ph. D. Dissertation,Department of Mechanical Engineering, METU. 2007; 110-118.

[13] Yildirim C,Arksel M. Numericcal simulation of the grain burnback in solid propellant rocket motor. In proceedings of the 4th AIAA/ASME/SAE/SEE Joint proportion Conferencce and Exhibition. Tucson Arizona, USA.2005.