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Effects of Propellant Fabrication on the Design for N-Class KNSB Solid Rocket Motor

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Abstract

In the age of the Internet of Things (IoT), satellite launching has become more common than in the past decades. To date, the rocket system is the only means of bringing satellites into Earth's orbit. However, the technology is very localized, and Malaysia is still in an embryonic stage. The main motivation is to produce a reliable rocket motor using relatively cheap and easily available resources. In this thesis, the best manufacturing casting method that yields more than 95% density ratio was studied for KNSB sugar-based propellant. The N-Class SRM was fabricated, tested and discussed. Each propellant manufacturing method was repeated for three samples and the sample density was calculated. The fabricated SRM was static test fired and the results were discussed with root cause analysis. The results were compared with the theoretical and discuss. The findings from the test static test show that the ejection of the nozzle during the static firing test is most probably case by the external and internal voids caused by the manufacturing method. The suggested method is to change from cartridge form to case-bonded propellant and use compression during casting to remove the internal air bubble that produced internal voids.

Discipline: Aerospace, Mechanical Engineering

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1 Introduction

Solid rocket motor (SRM) components can be as simple as three parts, which are the propellant, the casing, and the nozzle, as shown in Figure 1 (Tom Benson, 2021). More parts were included due to the evolution of design and limitations, such as the addition of an insulation layer in the motor casing to minimize heat transfer from the high combustion temperature in the chamber, which causes the casing to be weaker than its original properties (Haymes & Gal, 2017). Additionally, there is an O-ring seal to ensure that components between the body and nozzle are securely airtight. Various problems that arose during the development of rocket technology played a role in the technological revolution that occurred in other related fields. It was developed under a NASA Small Business Innovation Research (SBIR) program contract at NASA's Kennedy Space Center in Florida to extend the technology developed for NASA to other applications. Examples include the development of aerogel in 1930, which was first used as a heat insulation to keep rocket fuels cool in liquid form, before being used in a simpler techniques such as water bottles to keep the water cool for a long period of time for cyclists, and the research into rocket engine plumbing resulted in the development of a computer cooler (Andrew Wagner, 2020).

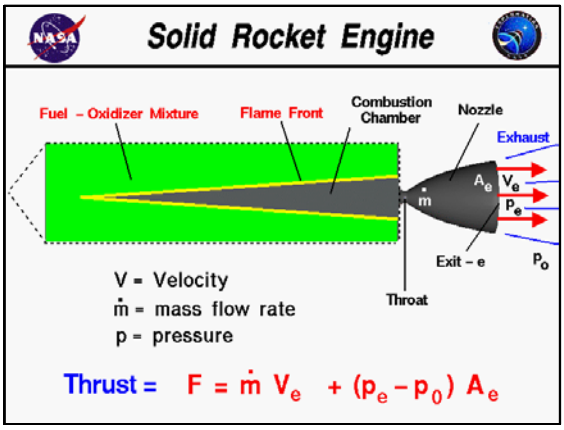

Figure 1: Solid rocket motor parts (Tom Benson, 2021).

The SRM casing acts as a combustion chamber and pressure vessel for solid propellant. It is crucial to have a robust SRM case to protect and store propellant grain. Moreover, the case must endure the highly pressurised and heated combustion chamber for grain burning activities (Jaafar et al., 2004; Sariyam Teja & Subramanyam, 2019). Materials such as titanium alloys and aluminium alloys are used in smaller rockets, while nickel alloy steel was commonly used in larger rockets (Hartfield et al., 2003). Material strength, enhanced temperature properties, rigidity or deformation characteristics, corrosion resistance, and ease of fabrication must be considered during material selection (Sariyam Teja & Subramanyam, 2019). The selection of materials for SRM components is done based on high specific strength, high specific modulus, fabrication easiness, easy availability, critical requirements and service conditions (Hartfield et al., 2003).

Hot gases from SRM propellant combustion will exhaust through the nozzle. It has been reported that nozzle design has a major influence on SRM efficiency (Manski & Hagemann, 1996).

Fixed, submerged, external, blast tube, and thrust vector control are commonly SRM nozzles used in tactical missiles (Jaafar et al., 2004). The design depends on the purpose or mission of the rocket. The nozzle is a very important element in applications involving the expansion of hot gases such as ramjet, turbojet and rocket engines. An increase in specific impulse obtained with an increase in combustion chamber pressure is almost entirely caused by the increase in expansion ratio through the nozzle (Morrell, 1950). The smallest cross-section area of the nozzle is called the throat of the nozzle, where the combustion gases flow through it at a maximum flow rate (known as a chocked condition) and the gas is further expanded in the diverging section.

The combustion pressure and temperature depend on variables such as burn rate, specific impulse and characteristic velocity, which depends on the propellant composition. For a high specific impulse and sturdy application such as in the military, AP-based propellant is commonly used due to its high reliability and relatively simple manufacturing process (Hartfield et al., 2003). However, the cost of propellant is relatively high and it is considered a controlled material. Alternatively, one can opt for cheaper and more readily available compositions, such as potassium nitrate-sorbitol, KNSB propellant, where the process of SRM development is almost the same as the AP-based propellant.

Thus, the main objective of this study is to evaluate propellant fabrication methods that yield more than a 95% density ratio, which is to be implemented in the manufacturing of KNSB SRM propellant. Moreover, The N-Class SRM was designed, fabricated, tested and discussed. The design of SRM included the SRM propellant, SRM nozzle, SRM bulkhead and SRM case.

2 Method

2.1 Solid Rocket Motor Design

The modular design concept was used in this approach, where each component of the SRM is customizable, as shown in Figure 2. This type of design approach was used to allow the customization of parts at a lower cost compared to integral design. The design of each part becomes more simplified with this approach where each of the parts can be fabricated and modified separately. The disadvantage of the modular design is that the parts need to be assembled using a fastener. However, this disadvantage can be used to our advantage since the fastener can act as the safety valve whereby if the pressure exceeds the threshold value, the fastener will fail.

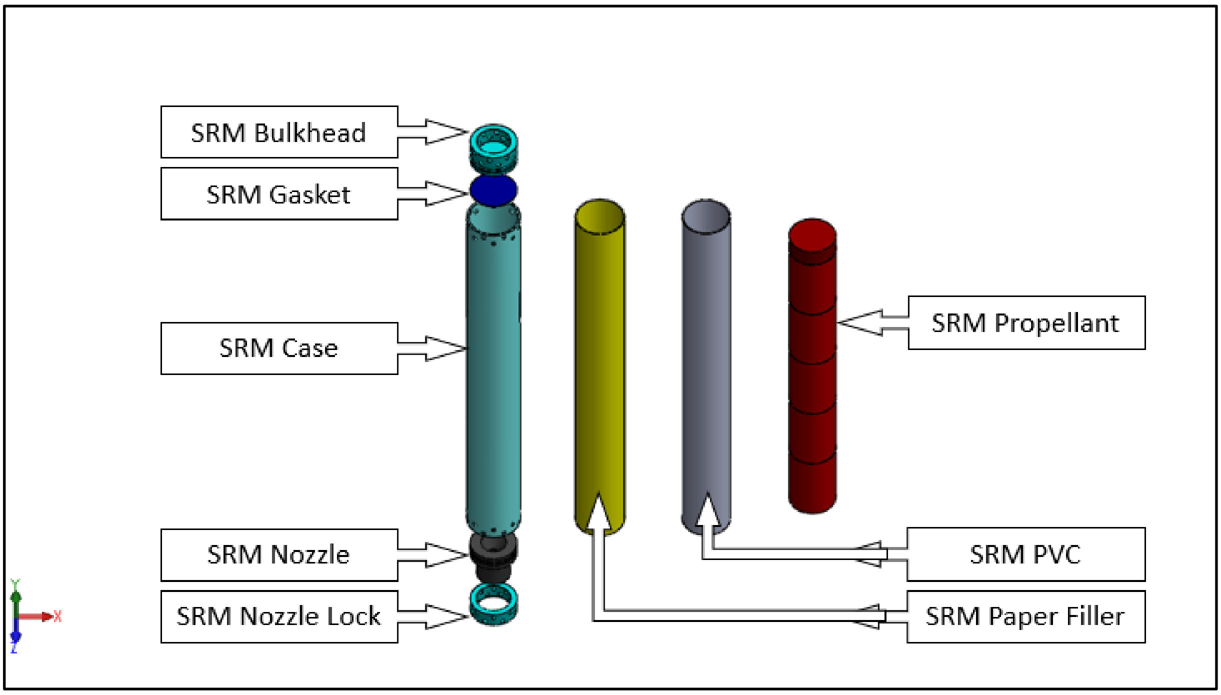

Figure 2: Solid rocket motor art.

2.1.1 Solid Rocket Motor Propellant

The most crucial part of the SRM is the propellant. The design of the SRM starts with the dimension of the propellant. Targeting the total impulse of N-Class SRM, which is 16,000 Ns, the propellant design profile was calculated. Prior to that, the thermodynamic properties of the propellant were estimated using ProPEP 3 software. The software has been tested and validated by comparing the theoretical calculation and experimentation by the master student of Universitari en Enginyeria Aeronàutica, France (Ruiz, 2019).

The SRM propellant was designed by using sugar-based propellant considering the cost-effectiveness and availability of the raw materials. Sorbitol powder was used as fuel in this study. This is due to the ease of manufacturing process where the sorbitol powder has a lower melting point compared to others such as sucrose and dextrose. Moreover, the study has proved that sorbitol will not easily caramelize even in high temperatures up to 200 °C (Nakka, 2018). The composition data and burning rate have been studied and recorded (Olde, 2019).

The SRM propellant grain was produced in a cartridge-loaded or freestanding grain. The SRM propellant grain was manufactured separately from the case by casting into a cylindrical PVC mould and then loaded into or assembled into the case (Nigar et al., 2021). This method used less cost and less misspend if there is any faulty on the SRM propellant grain produced since only one cartridge needs to be replaced compared to the whole SRM propellant if using case-bonded grain.

The production of propellant can be divided into two main processes, which are raw material preparation and the casting process. There are four types of casting processes as mentioned in Table 1. The casting process was tested to evaluate the best method to produce KNSB SRM propellant with a density of at least 95% of the theoretical density (Nakka, 2018). The preparation of the raw material process will be similar to all casting processes. The production takes place in a controlled environment in the laboratory where the temperature and humidity are consistently monitored to ensure no environmental factors will affect the production of the propellant.

Table 1: Propellant moulding methods.

No	Methods
1	Pour only during casting temperature without doing anything else until the sample case is full
2	Pour during casting temperature is reached then the case is shaken
3	Pour during the casting temperature is reached, then the mixture is left to be a bit cold and hard before compressing the propellant
4	Add the mixture with Sodium Laureth Sulfate (SLS) which is a surfactant to increase the viscosity of the propellant mixture.

Each of the methods was repeated three times to obtain three samples for each method. The propellant samples were cast into cylinder cases for weight. The densities of each sample were calculated, recorded and compared to the theoretical density of the SRM propellant.

2.1.2 Solid Rocket Motor Case and Nozzle

The SRM case used in this study is the standardly available material in the market. The case is made up of aluminium T6-6061 with physical properties listed in Table 2. The sizing SRM case has a standard nominal size of 5 inches with a 0.127 m outer diameter and thickness of 4.75 m. Based on these properties, the safety factor of the material was calculated based on the maximum hoop stress exerted by the SRM case.

Table 2: Solid rocket motor case properties in analysis system (ANSYS) software (Ashby, 2021).

Properties	Value
Material	Aluminium T6-6061
Density (kg/m3)	2713
Tensile Ultimate Strength (Pa)	3.131x108
Tensile Yield Strength (Pa)	2.592x108

The design of the SRM nozzle was made up of graphite material. Graphite is one of the most used materials for a nozzle throat because of its high-temperature resistance, high thermal conductivity and increasing toughness at elevated temperatures (Nigar et al., 2021). The SRM nozzle profile was calculated theoretically based on 500 Psi (3.45 MPa) chamber pressure. The nozzle profile is a very important part of the SRM since its function is to optimize the performance of the rocket and the stability of the propellant combustion.

A factor that needs to consider between the SRM propellant and SRM nozzle profile is its port-to-throat ratio. The ratio studied needs to have more than 2 (Ropia et al., 2020). This is to avoid erosive burning on the combustion chamber where the Mach number at the combustion chamber is too high. The relationship between the nozzle throat, chamber pressure and force is shown in (1) and (2).

$$P_c = \frac{A_b}{A_{n_t}} \times \rho_p \times C^* \times r_b \quad (1),$$

$$F = C_f \times A_{n_t} \times P_c \quad (2),$$

For greater efficiency, the gas flow needs to expand completely in the divergence section where in that process the pressure will decrease almost to ambient pressure and the maximum velocity produced from the nozzle is achieved. To obtain the best performance the expansion ratio of the nozzle was determined from ProPEP 3 software and the optimum exit nozzle area was determined.

2.2 Solid Rocket Motor Static Test

The expected performance of the SRM was calculated theoretically by calculation. Figure 3 shows the expected performance in terms of force and chamber pressure produced by the SRM presented in graphical form. The initial force of the SRM is expected to increase from around 1,500 N up to 4,300 N. The direct increase profile trend is influenced by the tubular SRM propellant shape.

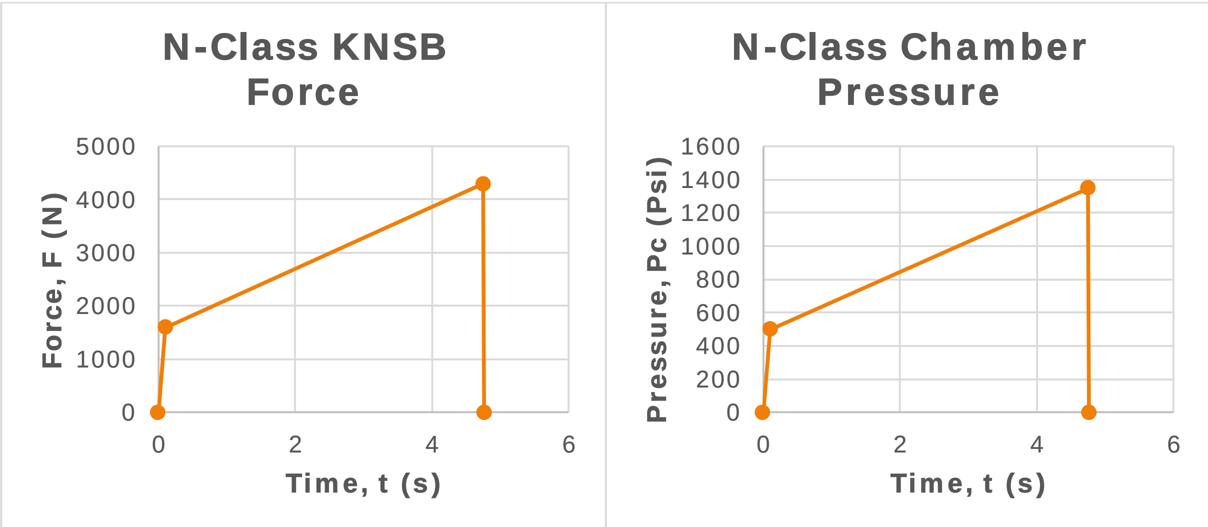

Figure 3: Performance of SRM in form of force and chamber pressure produced.

The SRM static test was done according to the safety regulation of NAR (Steve Humphrey et al., 2014). The NAR Standards & Testing Committee Motor Testing Manual states that during static motor testing, the separation distance shall be equal to or greater than the launch separation distance in the NAR Safety Code for the class of motor that is being fired.

The operation was conducted at a large space area where the facilities and support equipment are sufficient for this test. The technical preparations were conducted in a few phases. The first phase is by visiting and making sketch preparation of the testing site equipment and minor modification that need to be implemented for the testing. The second phase is the integration phase, where the manufactured load cell mount and a dummy SRM are prepared to be in the actual test. After final adjustments are made, the third phase is the actual day of the testing.

The 100,000 N beam-type load cell was mounted perpendicular to the wall and was connected to the Arduino DAQ and tested the system three times to obtain the readings and data storage before the actual test. The Arduino DAQ is capable of recording 11 data per second which is sufficient to plot the thrust time graph curve for the SRM.

Lastly, the igniter wiring connection was tested and inserted into the SRM through the nozzle throat. The place needs to be clear as in the NAR rules before ignition.

3 Result and Discussion

The theoretical properties computed from the ProPEP 3 software were the exact solutions obtained through the chemical formulations and equations. Table 3 provides a summary of the most crucial parameters for the continuation design of the SRM parts.

Table 3: Summary of ProPEP 3 results.

	Symbol	Empirical		SI	
Density	ρ_p	0.0663750	lb/in^3	1,837.2600	kg/m^3
Chamber Temperature	T_c	1,591.1380	F	1,139.3378	K
Chamber Pressure	P_c	500.00	Psia	3,447,500.00	Pa
Specific Heat Ratio	k	1.1373		1.1373	
Molecular Weight	MW	35.3389		35.3389	
Ideal Specific Impulse	I_{sp}	114.3722	s	114.3722	s
Characteristic Velocity	c^*	,986.0070	ft/s	910.0905	m/s
Expansion Ratio	ε	5.8500		5.8500	

A suitable manufacturing process for the SRM propellant has been studied prior to its manufacturing. The SRM propellant casting technique has been studied to identify the best method to produce propellant with a good density ratio of greater than 95% of the theoretical density. The casting method and its density produced are presented in Table 4.

Table 4: Density ratio of different casting method.

Casting Method	Sample 1 (g/cm3)	Sample 2 (g/cm3)	Sample 3 (g/cm3)	Average (g/cm3)	Density Ratio (%)	CV
Pour	1.6725	1.67	1.6851	1.6759	89.44	3.94E-03
Pour with shake	1.7721	1.79	1.7931	1.7851	95.27	5.19E-03
Pour, cool and compress	1.8001	1.8221	1.81	1.8107	96.64	4.97E-03
Mix with SLS	1.799	1.8021	1.81	1.8034	96.25	2.57E-03

The ideal density was computed using ProPEP 3 software yielding a density of 1.8737 g/cm3 A minimum density ratio of 95% was targeted for a good propellant mixture Density (Nakka, 2018). This mean that the minimum density ratio of the propellant mixture should be more than 1.78 g/cm3. From the results the best average density for the propellant mixture is by pour, cool and compress with a density of 1.81 g/cm3 and a density ratio of 96.64%. Additionally SLS shows as the second-best method while pour and shake is the third-best method that exceeds the threshold density ratio. The sample coefficient of variation is low (<1) showing that the number of samples tested was enough and the propellant fabrication was homogenous. However, the fabrication of propellant in this study continued with the pour and shake method due to the manufacturing difficulties.

The design of SRM parts continued using the exact solution given by ProPEP 3 software. Through mathematical calculation, the SRM profiles of all parts are summaries as in Table 5.

Table 5: Solid Rocket Motor profile.

Profile	Value
Target Total Impulse, I_{total} (s)	16,000
Propellant Weight, w_p (N)	113.64
Propellant Mass, m_p (kg)	11.58
Propellant Volume, V_p (m^3)	0.0063
Propellant Length, l_p (m)	0.8711
Initial Burning Surface, A_{b_i} (m^2)	0.1095
Maximum Burning Surface, A_{b_m} (m^2)	0.2951
Burning Rate, r_b (mm/s)	6.7355
Area Nozzle Throat, A_{n_t} (m^2)	0.0004
Area Nozzle Exit, A_{n_e} (m^2)	0.0021
Thrust Coefficient, c_f (unitless)	1.2958
Initial Pressure, P_{c_i} (Pa)	3,447,500
Maximum Pressure, P_{c_m} (Pa)	9,292,792
Initial Force, F_i (N)	1,593.9441
Maximum Force, F_m (N)	4.296.5025

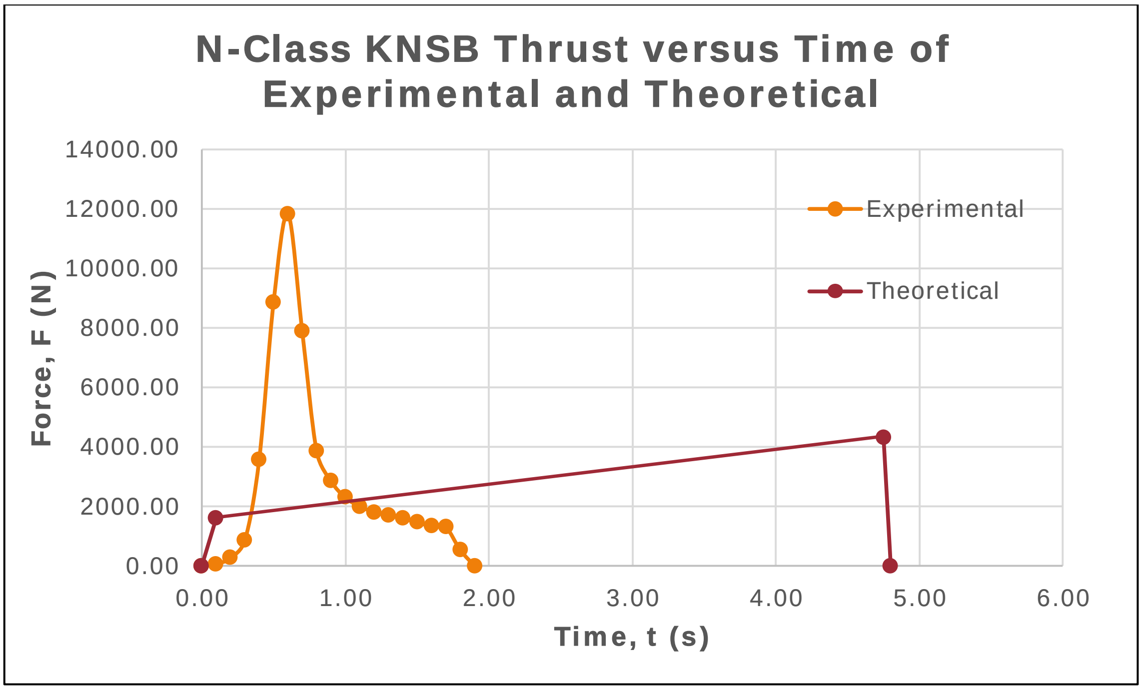

Figure 4: Experimental and theoretical thrust-time curve of the solid rocket motor.

It is observed that the predicted initial thrust occurred at $t \approx 0.3$ seconds, which is 0.2 seconds delayed from the theoretical prediction. This is likely due to the initial temperature of the propellant produced by the igniter, thus delaying the pressure build-up in the combustion chamber (Yaman et al., 2014). The burning of the SRM shows stable combustion until $t \approx 0.6$ seconds as shown in Figure 5 and Figure 6.

Figure 5: Initial combustion of solid rocket motor at $t \approx 0.1$ second.

Figure 6: Combustion of solid rocket motor at $t \approx 0.3$ second.

Furthermore, it is also observed that the thrust increased abruptly to a maximum of almost threefold the predicted value. This observation is attributed to the extreme pressure in the combustion pressure, which caused the ejection of the SRM nozzle (as shown in Figure 7 – the nozzle was ejected at approximately 0.6 seconds after the ignition). This is followed by a sudden drop in the thrust within about 0.3 seconds before gradually decreasing until the end of burn time.

Figure 7: Combustion of solid rocket motor at $t \approx 0.6$ second.

The possible root causes of the solid rocket motor nozzle ejection were identified and presented in the form of a fishbone diagram, as shown in Figure 8. The main factors can be caused by either the design itself or problems during manufacturing. The sub-causes can also be seen in the diagram.

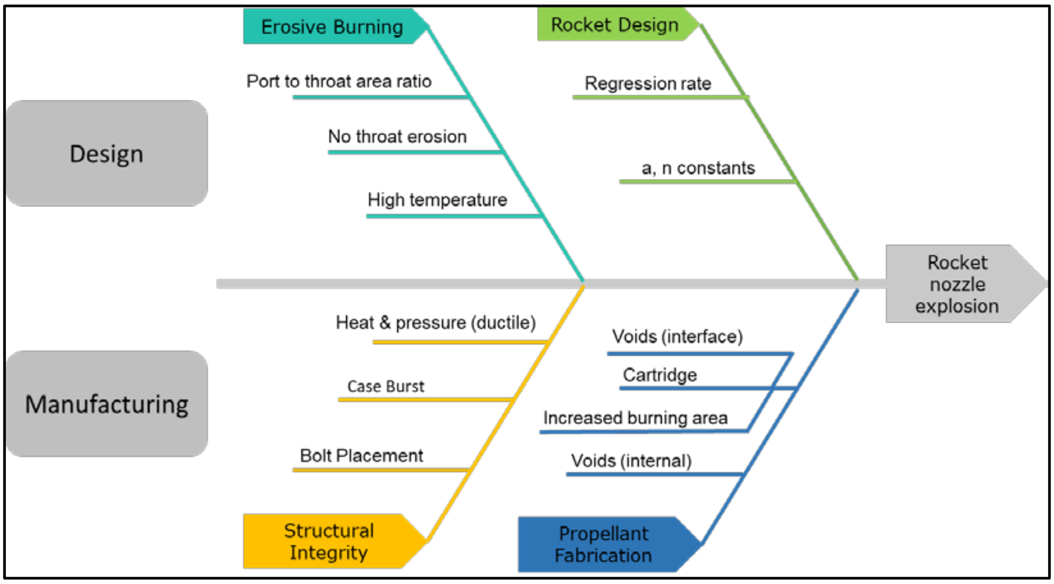

Figure 8: Root cause analysis using fishbone diagram.

The degree of likelihood of each factor is presented in Table 6. The likelihood ranges from 1 to 10, where higher values represent a higher probability of the cause contributing to the nozzle ejection.

Table 6: Degree of likelihood for every factors.

<table>
<tr><th colspan="2">Factors</th><th>Degree of Likelihood</th></tr>
<tr><td rowspan="6">Manufacturin
g</td><td colspan="2">Propellant fabrication</td></tr>
<tr><td>Increase burning area</td><td>8</td></tr>
<tr><td>Internal voids</td><td>5</td></tr>
<tr><td colspan="2">Structural integrity</td></tr>
<tr><td>Heat and pressure (ductility)</td><td>8 (secondary)</td></tr>
<tr><td>Bolt placement</td><td>6 (secondary)</td></tr>
<tr><td rowspan="7">Design</td><td colspan="2">Rocket design</td></tr>
<tr><td>Burning or regression rate</td><td>5</td></tr>
<tr><td>a, n constant</td><td>5</td></tr>
<tr><td colspan="2">Erosive burning</td></tr>
<tr><td>Port-to-throat ratio</td><td>4</td></tr>
<tr><td>No throat erosion</td><td>4</td></tr>
<tr><td>Temperature</td><td>6</td></tr>
</table>

From Table 6, it is expected that the cause of the nozzle ejection is more likely due to manufacturing factors. There is a high possibility that the actual performance of the rocket motor is different from the theoretical design due to the manufacturing process, particularly with propellant manufacturing. The formation of external and internal voids is possible during the casting of the propellant, which results in an increased burning surface area and causes an increased burn rate. This consequently resulted in an increase in combustion pressure and thrust produced (Yaman et al., 2014).

The main concern during the manufacturing of the propellant is the quality of the cured propellant. Although many factors have been considered during the manufacturing proses, it is deduced that the main contributing factor for the nozzle ejection is due to the over-pressure. The magnitude of chamber pressure during the combustion is estimated based on the thrust data obtained, which is approximately fivefold the designed pressure. The figure shows the SRM case post-firing. Based on the observation of the SRM case condition, a few deductions were made on what happens inside the SRM case during the static firing.

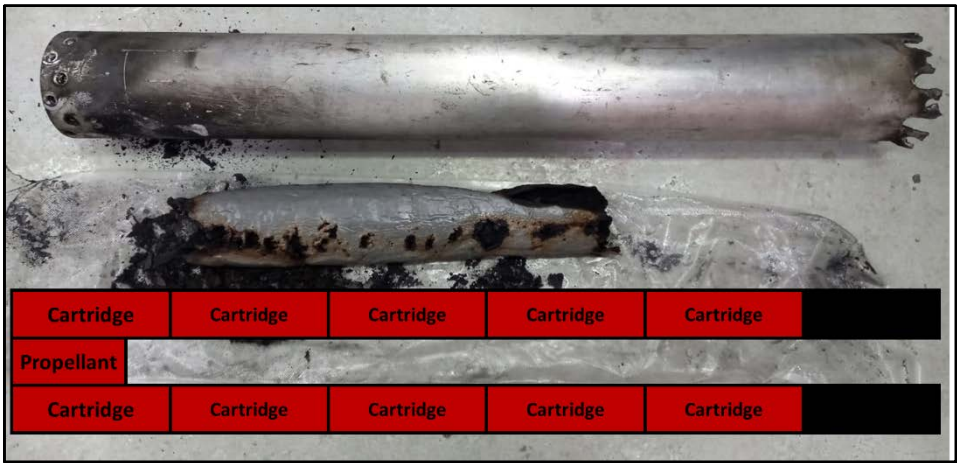

Figure 9: Solid rocket motor case after static firing test.

The manufacturing of the SRM propellant is using cartridge loads. The cartridge is attached together by melting one end of the propellant cartridge and then pressed together to the other cartridge end under light pressure. However, due to the propellant cartridge configuration, there might be some gaps formed between the cartridge interfaces during the assembly. The gaps will increase the initial burning surface area, which will result in an enhanced burning rate and consequently the chamber pressure. Figure 9 illustrates how the gap between the propellant cartridges supported by the condition of the PVC cylinder after firing will increase the burning surface area. It can be seen in this figure that there are possibilities for the combustion gases to leak through these gaps, initiating burning at the outer surface of the propellant. It is observed that the top and bottom parts of the PVC inside the case are completely burnt. This will further enhance the pressure peak in the combustion chamber and consequently resulted in high force produced causing the nozzle to detach from the solid rocket motor body.

To confirm this assumption, the second set of propellant cartridges was fabricated following the exact same method as discussed earlier. This cartridge set was sent for an x-ray to see if there are any voids forming. Figure 10 shows the x-ray results of the attached propellant cartridge.

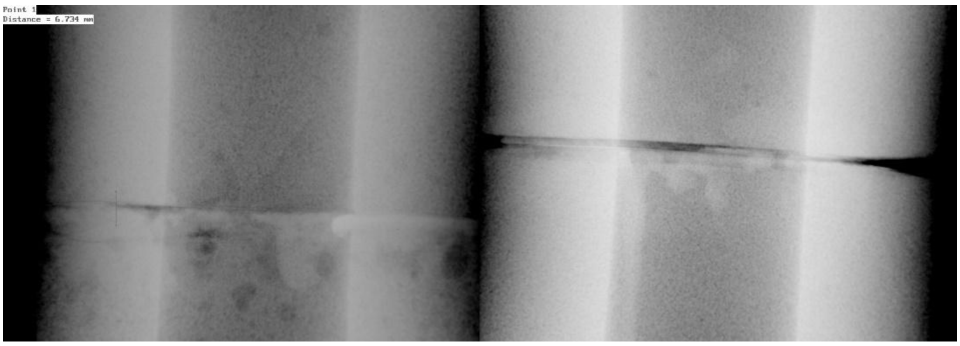

Figure 10: SRM propellant cartridge attachment.

The interface line is clearly seen in the x-ray images, indicating gaps/voids in the interface. This result confirms the assumption. There are a few suggested solutions to this problem. One of the direct solutions is to change the fabrication process from using cartridge-loaded to case-bonded propellant. This solution will eliminate the gap that could be formed from attaching two or more separate cartridges.

Apart from the cartridge interface gaps, internal voids were identified as a possible factor that increases the burning surface area. The formation of the internal voids is confirmed through the X-ray image, as shown in Figure 11.

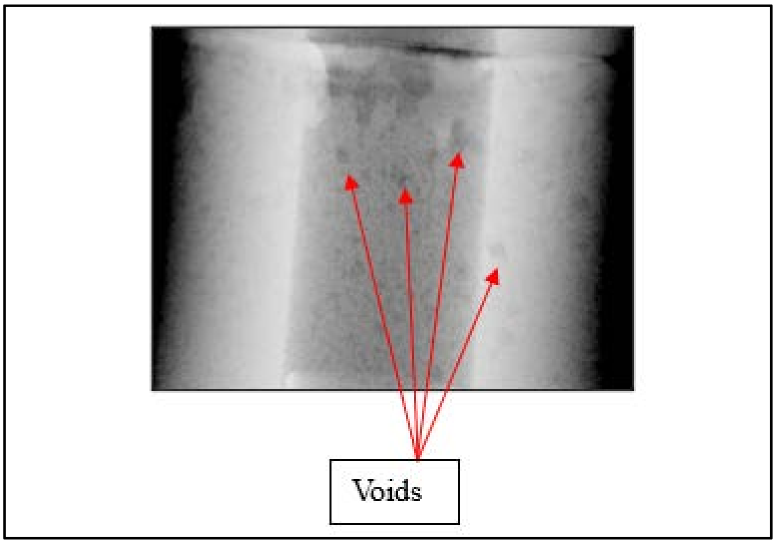

Figure 11: X-Ray of the propellant shows the formation of internal voids.

The darker regions on the X-ray image represent the internal voids formed within the propellant. There are multiple spots of dark black regions within a single cartridge, which will increase the burning surface area. Unlike the increased burning surface area due to the interface gaps, these voids increase the local burning surface area during the combustion, which may result in pressure oscillation inside the combustion chamber and abruption of a larger enough piece of the propellant. This piece can stick to a narrow place of the burning propellant or choke the minimum cross-section of the nozzle. This can cause a sharp catastrophic jump of the booster trust and overpressure in the chamber head (Biggs, 2002). Furthermore, these voids could cause propellant structural failure which is a critical defect that is caused by the cracks and voids in solid propellant and slots of booster joint segments. Thus, it is important to ensure that trapped air bubbles or voids are not formed during the casting of the propellant.

4 Conclusion

In this study, it is best to use the pour and compression method to have a high propellant density ratio. Although the static firing test did not provide the expected quantitative performance data, the after-test analysis gives an overview of the flaw in propellant manufacturing that can be improved. The most critical point is the external and internal voids will affect largely the SRM performance and need to be reduced. The three approaches can be applied to minimize/eliminate the formation of voids. The first approach is by lifting the casting mould cylinder, followed by tapping it repeatedly against a solid surface prior to the insertion of the coring rod. These steps help dispel trapped air. The second approach is by curing the propellant under pressure. In this method, an axial force is applied to the top surface of the propellant by using hydraulic compression while it is curing. This method reduces the possibility of trapped air formation by forcing them to be removed by the pressure applied to the propellant during the curing process. This results in a homogeneous propellant grain that has a near-ideal density. This approach is supported by the findings reported in Table 4, where the propellant cast under pressure yields the highest density and lowest coefficient of variation. The third approach is deaeration by using high-frequency vibration. In this approach, the casting mould cylinder is vibrated while pouring the propellant slurry. This will help to dispel trapped air inside the propellant and thus reducing the internal voids.

5 Availability of Data And Material

Data can be made available by contacting the corresponding author.

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