Dexter : Design, Fabrication, Testing & Validation of a Portable Tabletop Solid Rocket Motor - Static Performance Thrust Stand with a Wirelessly Controlled Data Acquisition System.

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Abstract - *Scientific experimental activities on rocket propulsion being accessible to student researchers is one of the key factors for research and development in the field of aerospace engineering. However, even under the regulatory limits of amateur rocketry, the required rocket motors and their test facilities are unavailable to the majority of institutes in India and other developing countries. Concerning this issue, this paper details the research and development of Dexter, which is a portable tabletop solid rocket motor static performance thrust stand with a wirelessly controlled data acquisition system, built to provide research and development opportunities for students to gain in-depth knowledge and practical experience about rocket propulsion systems and advance in the field of aerospace engineering. The development of this project is explained in two phases described as 1. Design & Fabrication & 2. Testing & Validation. The design and fabrication phase presents the technical drawings, CAD models, schematic diagrams, PCB layouts, fabrication materials, techniques, prototypes of the portable tabletop thrust stand, solid rocket motor, wirelessly controlled data acquisition system, and its components. The testing & validation phase presents an experimental investigation of G class solid rocket motor's static performance, computing the thrust, mass flow rate, and supersonic exhaust velocities through a convergent-divergent nozzle over varying propellant mixtures ratios using ($C_6H_{14}O_6$) D-Sorbitol as fuel and (KNO_3) Potassium Nitrate as an oxidizer, performing data visualization and graphical representation through MATLAB Software with a comparative study between the analytical solutions and the experimental results over selected solid rocket motor performance parameters from the testing phase and thus validates the entire project with acceptable error margins. These errors and difficulties faced during the project are measured and discussed with the future scope. Conclusions of this project and its importance to the aerospace research community are outlined.*

Key Words **:** Solid Rocket Motor, Thrust Stand, (DAQ) Data Acquisition, Design, Fabrication, Static Performance, Analytical Study, Experimental Testing.

Nomenclature **:** Dc – Core Diameter ; MR – Mixture Ratio ; Ve – Exhaust Velocity ; Ae/A* – Area Ratio ; R – Reynolds Number ; M –Molecular Weight ; k –Specific Heat Ratio ; ρ – Density ; f0 - Oxidizer Mass Fraction ; Do – Outer Diameter ; W – Dry Weight ; Cp – Specific Heat Coefficient at constant pressure ; Cv – Specific Heat Coefficient at constant volume ; Mesup – Supersonic Mach Number ; Mesub – Subsonic Mach Number ; T0 – Stagnation Temperature ; Te – Exhaust Temperature ; ff - Fuel Mass Fraction ; A – Area ; C* – Characteristic Velocity ; Ct – Thrust Coefficient ; T – Thrust ; It – Total Impulse ; m – Products Mass ; mp – Propellant Mass ; ṁ – Mass Flow Rate ; Isp – Specific Impulse.

1. INTRODUCTION

In a broad sense, in the solid rocket, the fuel and oxidizer are mixed into a solid propellant using appropriate binders and additives packed into a solid motor casing with various grains controlling the burn rate. When the mixture is ignited, combustion takes place on the surface of the propellant producing great amounts of exhaust gases at high temperature and pressure. The amount of thrust that is produced depends on the motor design i.e classification based on its (D) diameter, nozzle design, (Ae/A*) area ratio which accelerates the flow, (Ve) exhaust velocity, fuel, oxidizer used, oxidizer to fuel ratio i.e (MR) mixture ratio, and propellant (M) Molecular weight, etc. After completion of design, horizontal or vertical static tests of rocket motors are conducted to evaluate and determine the performance of propulsion systems required for aerospace applications. These tests are fully experimental and instrumented. Physical parameters recorded and calculated are (T) Thrust, (Isp) Specific impulse, (ṁ) Mass flow rate, and (Ve) Exhaust velocity. Temperature, pressure, strain, vibration, shock, and acoustic levels can also be measured. The instrumentation sensors used are load cells. Thermocouples, pressure transducers, flow meters, accelerometers, strain gauges can also be used. The rocket motor is test-fired and the data is recorded on data acquisition systems. Detailed analyses are carried out for performance evaluation.

1.1 Research Objectives

a. Design & development of Dexter : portable tabletop solid rocket motor static thrust stand with wireless DAQ system.

b. Study and research of solid rocket motor propulsion systems & preparation of various propellant mixtures.

c. Investigation on propellant performance over variable oxidizer to fuel ratios i.e (MR) mixture ratio.

d. Demonstration, visualization, and analyses of overall solid rocket motor static performance parameters.

e. Innovation in the field of aerospace engineering to provide research and development opportunities for students to gain in-depth knowledge and practical experience about rocket propulsion systems.

2. DESIGN & FABRICATION

2.1 Portable Tabletop Solid Rocket Motor Static Performance Thrust Stand

The experimental setup is a self-developed portable tabletop thrust stand capable of demonstrating and carrying out static performance tests of solid rocket motors. The setup uses linear guided horizontal force computing methodology to determine the rocket thrust output and a vertical force computing methodology to determine the propellant weight flow rate which simply leads to the determination of the mass flow rate considering the earth's gravitational acceleration. Both methodologies utilize a single-point shear beam load cell. Technical drawings and CAD models are generated on SolidWorks 2011 for the assembled portable tabletop solid rocket motor thrust stand as well as its components. Most of the components are fabricated using combinations of lathe, vertical milling, and drilling machine operations and are assembled using component-specific nuts and bolts. The entire structure is manufactured using high grade light weight aluminium & anti-corrosive stainless steel. [Fig : 1,2,3,4,5,6,7].

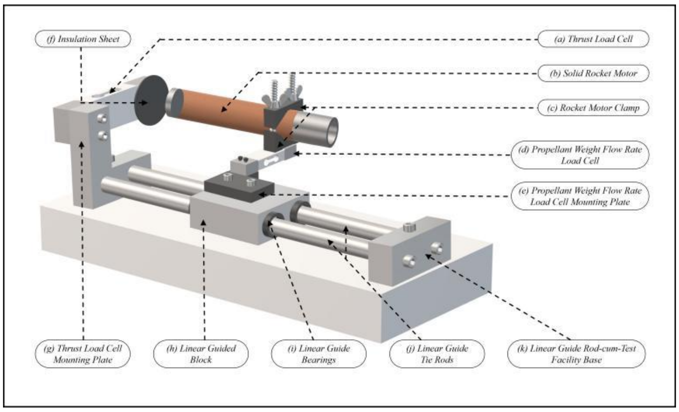

Fig - 1: Portable Tabletop Static Performance Thrust Stand

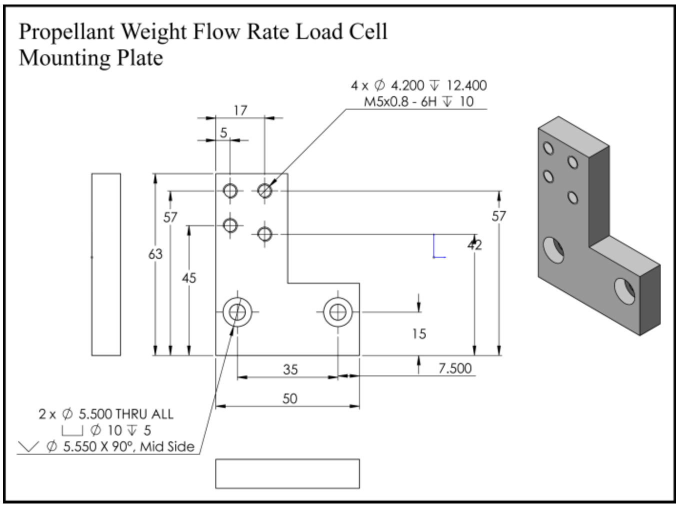

Fig - 2: Weight Flow Rate Load Cell - Mounting Plate

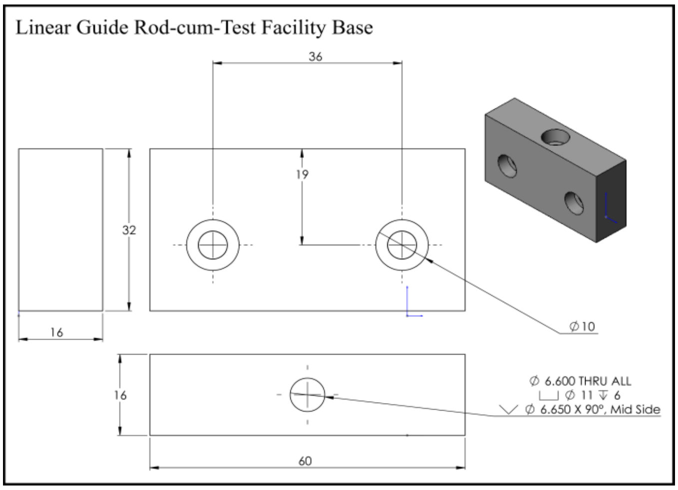

Fig - 3: Linear Guide Rod-cum Thrust Stand Base

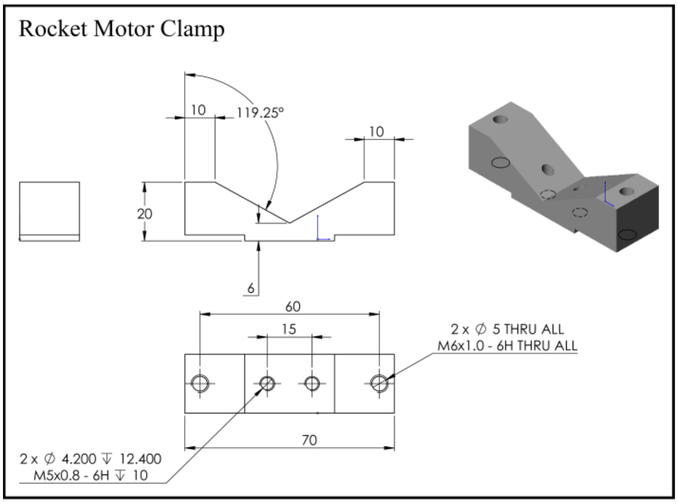

Fig - 4: Solid Rocket Motor Clamp - Symmetrical Half

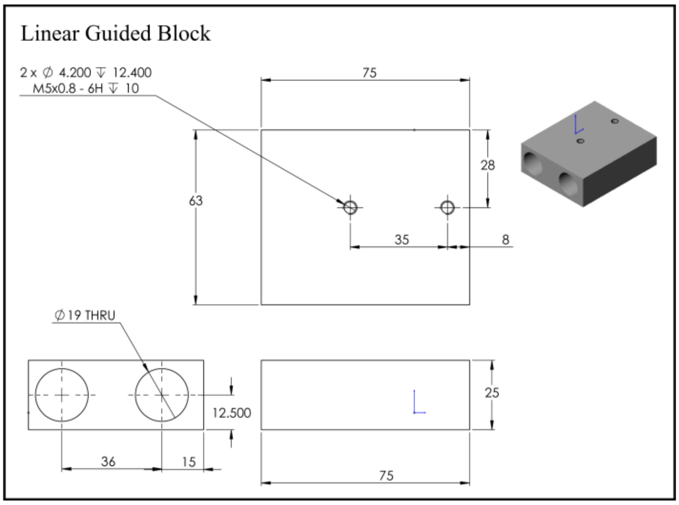

Fig - 5: Linear Guided Block-cum Mounting Plate Base

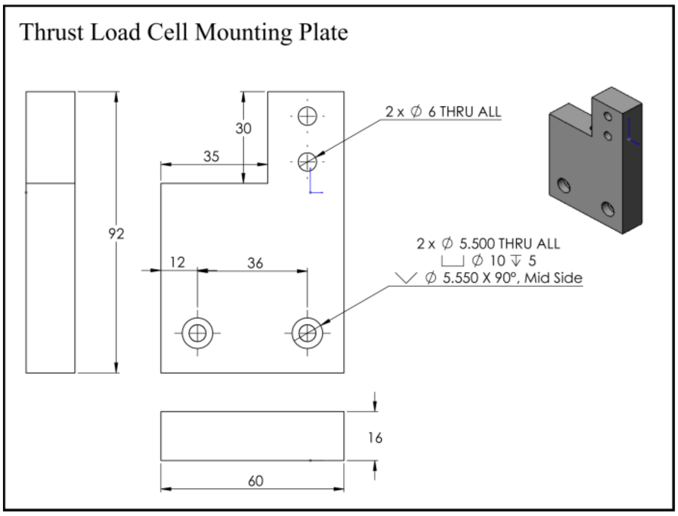

Fig - 6: Thrust Load Cell - Mounting Plate

Fig - 7: Shear Beam Load Cell - Thrust & Weight Flow

2.2 Solid Rocket Motor (G - Class)

Solid rocket motor comprises of rocket motor casing made up of stainless steel which acts as a combustion chamber and a pressurized vessel, this chamber is coated with readily available portland cement for insulation pertaining to its fire and heat resistant properties then filled with solid propellant during the testing phase. One end of the rocket motor casing is fitted with a mild steel end cap while the other end houses a mild steel convergent-divergent nozzle capable of accelerating pressurized exhausts to higher supersonic speeds. [Fig : 8,9,10,11].

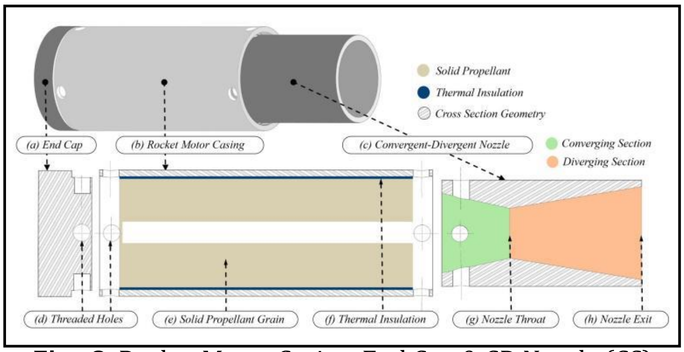

Fig - 8: Rocket Motor Casing, End Cap & CD Nozzle (CS)

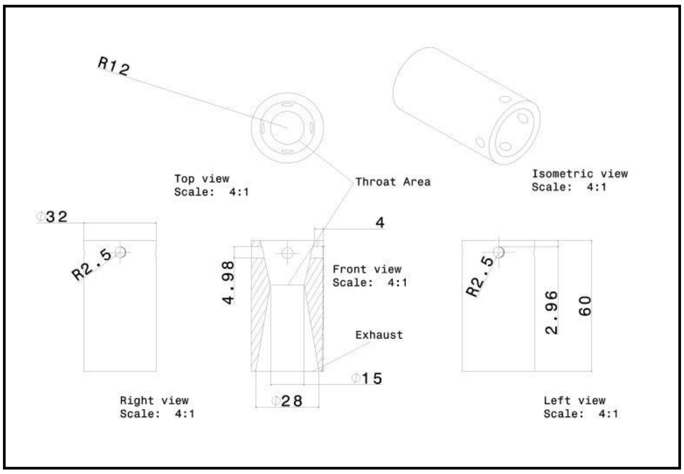

Fig - 9: Converging Diverging (CD) Nozzle

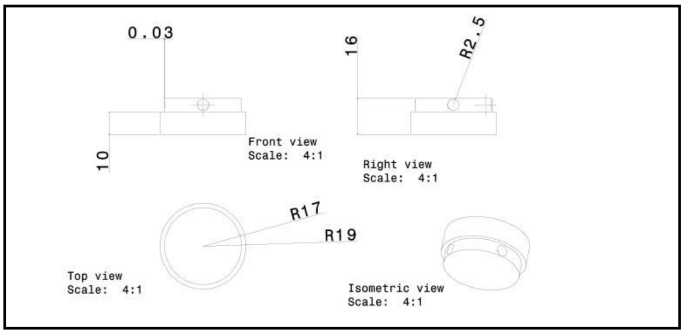

Fig - 10: End Cap

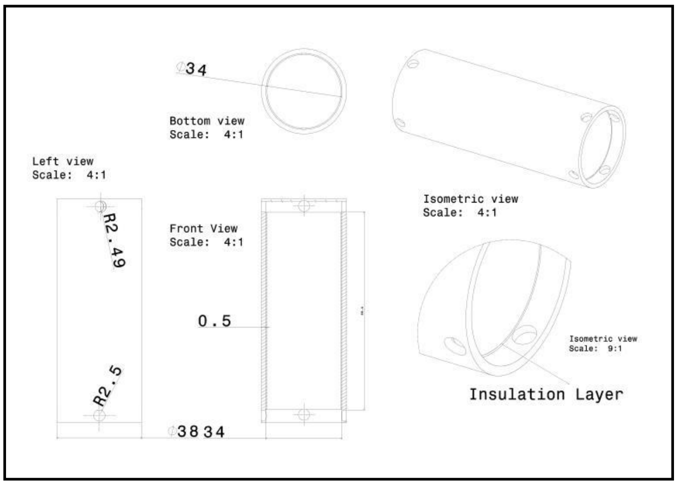

Fig - 11: Rocket Motor Casing

2.3 Wireless Data Acquisition System (DAQ)

DAQ system accumulates raw data from the load cells & various sensors and processes on it to indicate the final output of experimental data. It takes the analog information through sensors and by using electronic components like micro-controller and amplifiers converts it to digital output. Data is acquired wirelessly with a Wi-fi module i.e. ESP32 which is connected to a Node-MCU micro-controller. The schematics and the PCB design of the system were made on an online PCB design software named EasyEda. The schematic and the PCB include the entire DAQ system along with other subsystems such as Ignition System, Wireless data transmission, and machine control system. [Fig : 12,13]. The data obtained from the experiment is presented in tabulated form in [Table 4].

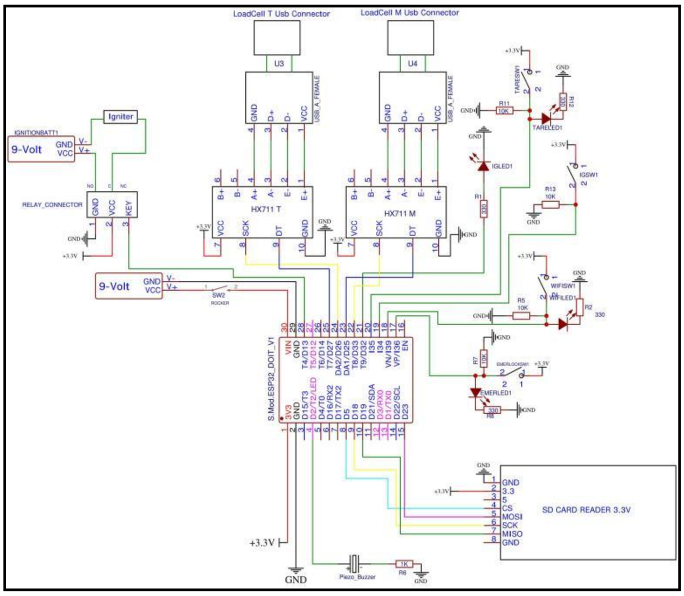

Fig 12: Schematic - Data Acquisition (DAQ) system

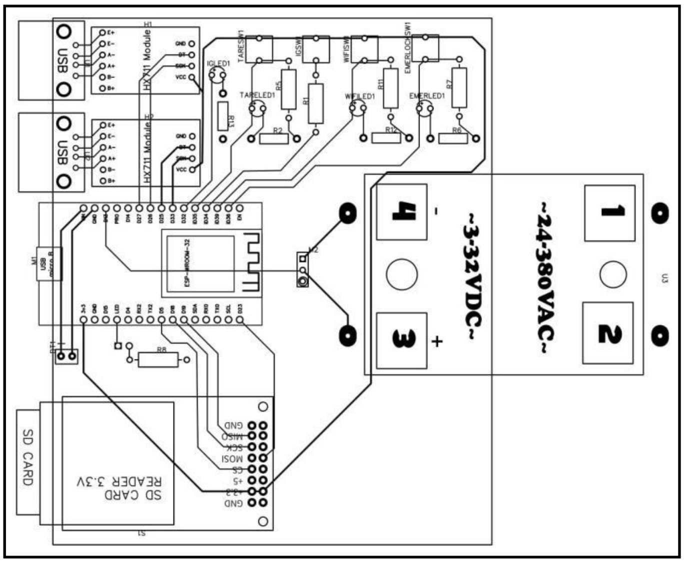

Fig 13: PCB Layout - Data Acquisition (DAQ) system

2.4 Propellant (Solid)

KNSB is mostly used in experimental rocketry. KNSB Propellant has 35% of D-Sorbitol ($C_6H_{14}O_6$) as fuel & 65% of Potassium Nitrate (KNO_3) as an Oxidizer. The compounds are easily available, it does not contain any toxic element which makes it a safe propellant to do experiments with. Cost is relatively less. The preparation of fuel is simple & can be done using household appliances. Generally, Sorbitol is a white fine powder and potassium nitrate appears in white crystalline form. Sorbitol and Potassium Nitrate are collected in one vessel. The mixture is heated until caramel is formed, the fuel should not be overcooked and is then filled in the rocket motor casing. Catalysts and Additives such as Ferrous Oxide, Magnesium and Aluminium powder, etc can be added in calculated proportions up to Max = 5% of the Propellant. [Fig : 14]. The propellant component & mixture chemical data is presented in tabulated form in [Table : 1,2,3].

Fig 14: KNSB - Propellant Preparation

Table - 1: Propellant Component - Chemical Properties

Chemical Parameter	Sorbitol	Potassium Nitrate	Magnesium
Molecular form	$C_6H_{14}O_6$	KNO_3	Mg
Density	1.489 g/cm^3	2.109 g/cm^3	1.738 g/cm^3
Melting Point	94-96 °C	334 °C	650 °C
Molecular Weight	182.172 g/mol	101.103 g/mol	24.305 g/mol
Enthalphy of Formation	-1353.7 kJ/mol	-494.00 kJ/mol	NA
Boiling Point	296 °C	400 °C	1091 °C

Chemical Parameter	KNSB
Specific impulse	164 s
Characteristic exhaust velocity	909 m/s
Combustion temperature	1327 °C
Density	1.841 g/cm^3
Mass fraction of condensed phase	0.436
Ratio of specific heats	1.1361
Effective molecular weight of exhaust product	39.869 g/mol
Burn rate at 1 atm	2.6 mm/s

Table - 2: KNSB Propellant - Chemical Properties

2.5 Ignition & Insulation Systems

Fig 15: ZPP Pyrotechnic Igniters

Igniter system - Pyrotechnic initiator, composition (ZPP) zirconium – potassium perchlorate. The following igniter is placed at the end of the combustion chamber near the end cap. Internal Heat Resistant Insulation is made up of Gypsum, Portland based Cementitious plasters. This prevents the excessive heating & meltdown of the rocket motor. [Fig : 15].

3. TESTING & VALIDATION

3.1 Combustion Process

The chemical equation for the combustion of Potassium Nitrate and Sorbitol is $5O_2 + 4KNO_3 + 2C_6H_{14}O_6 \rightarrow 12CO_2 + 4H_2O + 2N_2 + 2K_2CO_3$. Potassium carbonate is released by the combustion reaction and is neither flammable nor explosive. It has a rating of two for health but since it is hygroscopic, it absorbs water soon after it is created and is therefore diluted and harmless. Similarly, all of the other chemicals are safe with health hazard ratings of zero.

3.2 Input Parameters

Parameters of Study : G Class Solid Rocket Motor

1. Length of total Rocket motor (with Nozzle) = 148 mm
2. Diameter of Combustion chamber = 34 mm
3. Length of Combustion chamber = 88 mm
4. Diameter of Nozzle throat = 15 mm
5. Diameter of Nozzle exit = 28 mm
6. Length of Nozzle = 60 mm
7. Area of Throat = 176.625 mm^2
8. Exit area of Nozzle = 615.44 mm^2

Table -3: Various Propellant Mixtures Used

Mix No	Mass of Oxidizer	Mass of Fuel	Mass of Additive	Mixture Ratio
1	78.12 g	38.47 g	3.7 g	2.03
2	72.29 g	44.30 g	3.7 g	1.63
3	66.47 g	50.13 g	3.7 g	1.32

Oxidizer (KNO_3) , Fuel ($C_6H_{14}O_6$) & Additive (Mg). Mix 1 is the best propellent mixture with MR = 2.03 [Table : 3].

3.4 Analytical Study & Calculations.

Universal gas constant "R" is having value at STP conditions

$$R = 8.314 \ \frac{J}{Kmol}$$

Molecular weight of KNSB propellant is

$$M = 39.869 \ \frac{g}{mol}$$

Value of specific gas constant is

$$R' = \frac{R}{M} = 208.4 \ \frac{J}{KgK}$$

$$\gamma = \frac{C_p}{C_v} = 1.1361, \text{ ratio of specific heat capacity.}$$

$T_0 = 1600\ K$ (Temperature - combustion products form completely for KNSB propellant)

Applying the area mach relation for de-Laval nozzle with $\frac{D_e^2}{D_t^2} = 3.4844$

$$\frac{A_e}{A_t} = \frac{1}{M_e^2}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M_e^2\right)\right]^{\frac{\gamma-1}{\gamma+1}}$$

$$3.4844^2 = \frac{1}{M_e^2}[0.9362(1+0.06805 M_e^2)]^{15.6950}$$

After solving we get $M_{e,supersonic} = 2.4314$ and $M_{e,subsonic} = 0.1739$

$$M_e = \frac{V_e}{a_e} \quad \text{(Relation between velocity and mach No. at exit)}$$

$$V_e = 2.4314 * \sqrt{1.1361 * 208.4 * 1600} \quad as \quad a_e = \sqrt{\gamma * R * T_0}$$

$$V_e = 1496.48 \ \frac{m}{s}$$

$$I_{sp} = \frac{V_e}{g_0} = 152.54 \text{ s}$$

$$\frac{T_0}{T_e} = 1 + \frac{\gamma-1}{2}M_e^2 \quad \text{(Relation between temperature and mach no.)}$$

Exhaust temperature : $T_e = 1140.98\ K$

Characteristic velocity C^* is given by,

$$C^* = \sqrt{\frac{RT_0}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}}}$$

$$C^* = 908.189 \ \frac{m}{s}$$

$I_{sp} = C^* * C_t$ (Specific impulse is product of characteristic velocity and throat coefficient.) $C^t = 0.16796$

*Analytical Calculations (Theoretical) for Mix 1

Supersonic Exhaust Velocity (Ve) = 1496.48m/s

Specific Impulse (Isp) = 152.54 s

3.5 Experimental Observations

Table - 4: Thrust vs Time vs Propellant Remaining

Time	Thrust (N)	Propellant Mass (g)
0 (start)	0.29	120.30 (start)
0.05	0.31	119.51
0.14	3.36	118.44
0.24	12.34	117.23
0.34	19.20	116.08
0.44	23.61	114.12
0.53	26.12	111.15
0.63	31.62	106.72
0.73	36.13	104.53
0.82	40.79	102.87
0.92	45.12	100.12
1.02	51.89	99.01
1.12	54.77	97.37
1.21	57.72	95.93
1.31	60.86	93.56
1.41	62.87	86.51
1.51	63.41 (max)	73.21
1.6	60.56	60.05
1.7	49.33	49.32
1.8	34.1	32.68
1.9	19.2	22.43
1.99	8.36	17.95
2.09	3.57	15.42
2.19	1.92	13.69
2.29	1.39	12.56
2.38	0.49	9.67
2.48	0.31	7.12
2.58	0.33	4.24
2.67 (end)	0.06	0 (end)

*Experimental Observations (Measured) for Mix 1

Solid Rocket Static Thrust (T) max = 63.41 N

Average Propellant Mass Flow Rate ≈ 45 g/s

Burn Time = 2.67 seconds. Fuel Left = 0 grams

3.6 Experiment Setup

The entire experimental setup is self-developed. The tests were conducted at MIT ADT University. 3 Tests were performed on 8th March 2020. The objective of the experimental setup was to measure the thrust generated, propellant mass flow rate, and calculate the supersonic exhaust velocity along with specific impulse. [Fig : 16,17].

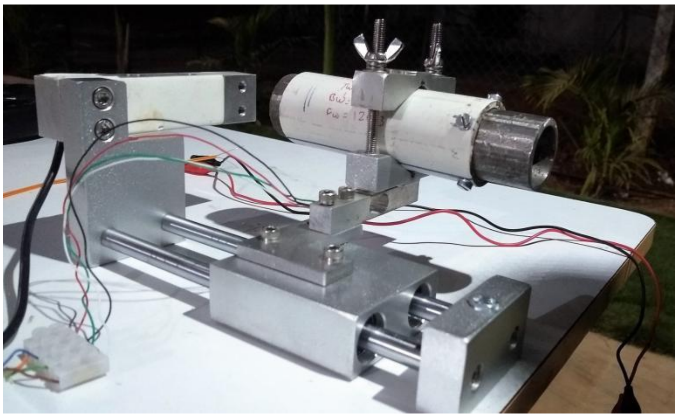

Fig -16: Thrust Stand & Solid Rocket Motor - Final Setup

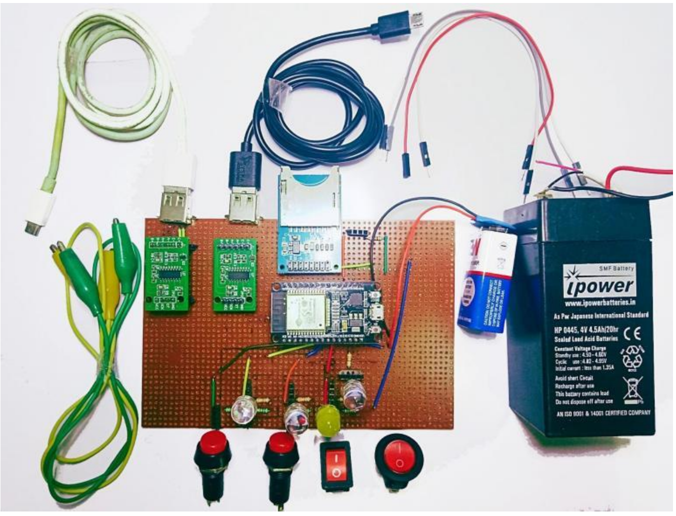

Fig -17: Wireless DAQ System - Final Setup

3.7 Final Results & Calculations

1. Thrust (T) max = ṁ × Ve max ; [Table 4] [Fig : 18,19,20].

Peak thrust = 63.41 N through experiments, average mass flow rate calculated (ṁ) = 45.00 g/s through experiments Ve max = 63.41 / 45.00 / 1000 ; Ve max = 1409.11 m/s ;

2. Specific Impulse (Isp) = Ve / g0 ; g0 = 9.81 m/s2

Ve max =1409.11; Isp = 1409.11 / 9.81 ; Isp = 143.64 s

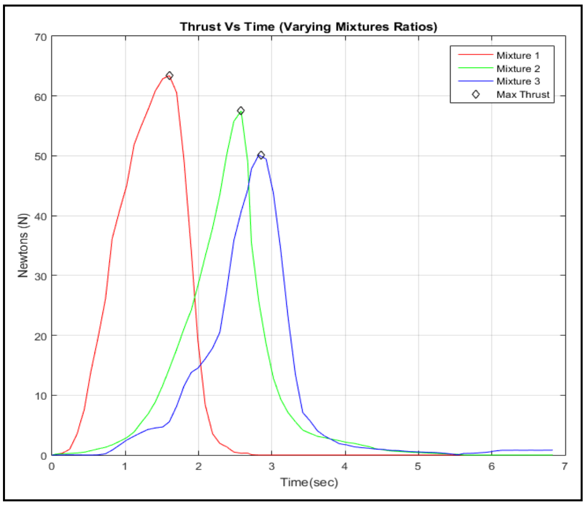

Fig -18: Thrust vs Time ; Mix : 1 ; 2 ; 3 ;

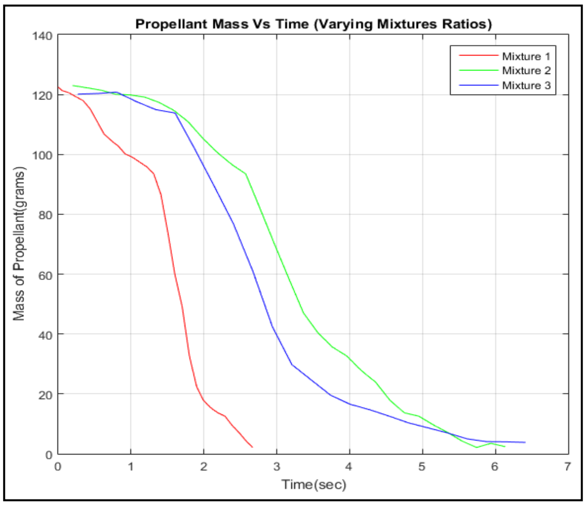

Fig -19: Propellant Mass vs Time ; Mix : 1 ; 2 ; 3 ;

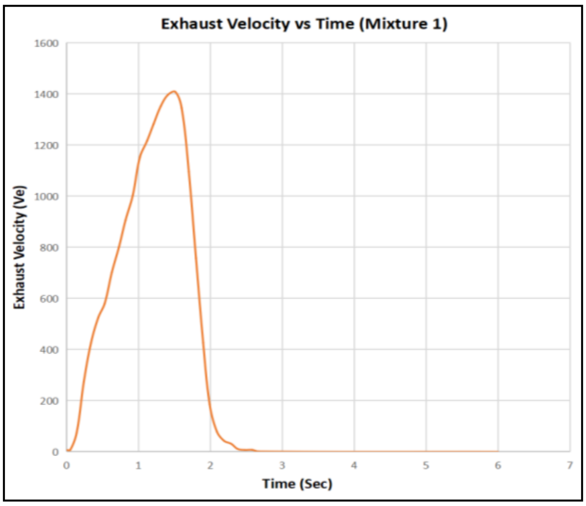

Fig -20: Exhaust Velocity vs Time ; Mix : 1 Only

3.8 Comparison between Analytical (Calculated) and Experimental (Observed) Results

The experimental results are within the considerable range of analytical results (-5.83 % error) & thus proves the authenticity of Dexter & the experiment. [Table : 5]

3.9 Comments & Discussion on the Rocket's Static Performance over Variable Mixture Ratio (MR)

As the ratio of oxidizer to fuel ratio increases the thrust obtained also increases until the optimum Mixture Ratio is achieved. The propellant consumption increases, the mass flow rate increases, the Burn time decreases, and Specific Impulse increases. Hence Mixture 1, MR of 2.03 i.e Oxidizer (potassium nitrate) = 66% and Fuel (Sorbitol) = 33% and Additive (Magnesium) serves as the optimum propellant ratio. The testing & validation phase use Mix 1 as standard.

4. CONCLUSIONS

a. As demonstrated in this paper, Dexter, which is a portable tabletop solid rocket motor static performance thrust stand with a wirelessly controlled (DAQ) data acquisition system has been self-developed to study, visualize and analyze the solid rocket motors performance characteristics at a scientific level. [Fig : 21].

b. KNSB propellant used is of low cost and an be made with readily available ingredients. The study of various Mixture Ratios and their effect on the Solid Rocket Motor performance was successfully demonstrated.

c. As the Oxidizer to Fuel ratio reaches optimum value, we see an increase in thrust produced, decrease in propellant consumption with higher supersonic exhaust velocities. Errors and accuracy of the self-developed system limit the experimental results to be exactly equal to the analytical solutions but are very close.

e. A futuristic & innovative approach was used to design, fabricate, test & validate a rocket test facility capable of demonstrating propulsion based research and experiments while promoting fellow undergraduate students from multidisciplinary branches to take part in the field of aerospace engineering.

Performance Parameter	Analytical Data	Experimental Data	Percent Deviation
Isp (Specific Impulse)	152.54 s	143.64 s	- 5.83 %
Ve (Exhaust Velocity)	1496.48 m/s	1409.11 m/s	- 5.83 %
Supersonic Mach No	2.4310	2.289	- 5.83 %

Table - 5: Analytical vs Experimental Data

Fig -21: Project Dexter - Successful Prototype Test

4.2 Error Solutions & Future Scope

To develop a vibration oscillation damping mechanism to overcome the error present in the current mechanism to achieve higher accuracy. Isolate the collected data from the instrument sensor errors and noise generated. Take the losses due to friction into consideration and add the losses accordingly. Conduct further experiments in a vacuum chamber to remove the environmental oxygen from affecting the experimental results.

5. FUNDING

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6. COMPETING INTERESTS

The authors declare that they have no competing interests.

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