

**Modelling and Fabrication of Attitude Thrust Vectoring Control System
for Launch Vehicle in Model Scale Using Three Axis Gimbal Prototype.**

Submitted in partial fulfilment of the requirements for the award of Bachelor of
Engineering degree in Aeronautical

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(DEEMED TO BE UNIVERSITY)

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BONAFIDE CERTIFICATE

This is to certify that this Project Report is the bonafide work of MOULITHARAN. M (Reg. No. 39260019), LEKHA.T (Reg. No. 39260014), who carried out the Project entitled "MODELLING AND FABRICATION OF ATTITUDE THRUST VECTORING CONTROL SYSTEM FOR LAUNCH VEHICLE IN MODEL SCALE USING THREE AXIS GIMBAL PROTOTYPE." under our supervision from June 2022 to April 2023.

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ABSTRACT

The projects goal is to develop the three axis gimbal prototype and implement the thrust vector control algorithm that is suitable for the system for the derived mechanical concept for actuation of the model . the report explains in detail about design and development of the prototype for the small scale model rockets and its feasibility of implementation , The major topic covered in this report are construction of Solid rocket motor and its actuation mechanism with the given design principles and closed-loop control algorithm , the chosen concept of control algorithm can be executed over the system from the solely development flight avionics for stabilizing the rocket through out its mission trajectory and for the vertical take- off and landing model rockets , the concept which is being developed is based on the various research findings and a new approach of design and study towards enhancing the performance of the desired mechanism , whereas the system has undergone rigorous testing and validation in the phase of design process, the validation of the system is done through numerical simulation for its flow characteristics of solid rocket propellant and response characteristics of the avionics system. The main objective of the project is to attain deflection angle of $\pm 15^\circ$ for pitch , yaw and roll motion and have better performance than the existing model , meanwhile the developed system has to fulfill the requirement for making the model combustible capable and fully operational

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LIST OF SYMBOLS AND ABBREVIATIONS

SYMBOLS AND ABBREVIATIONS	DESCRIPTION
1. TVC	Thrust Vectoring Control
2. TVN	Thrust Vectoring Nozzle.
3. STOLEN	Short Take Off And Landing.
4. HARV	High Angle of Attack Research Vehicle.
5. SMLTD	Short Take Off And Landing Maneuverer Technology .
6. VTOL	Vertical Takeoff And Landing.
7. CFD	Computational fluid dynamics.
8. PCM	Pulse coded Modulation.
9. ACBB	Angular Contact Ball Bearing.
10. PVC	Poly Vinyl Chloride.
11. Dc	Combustion chamber diameter (mm). Dt. Throat diameter.
12. De.	Diameter at the exit (mm).
13. Lc	Length of the combustion chamber (mm).
14. Θ	Angle of inclination.
15. A	Area of circle (mm)
16. G	Area Ratio (mm).
17. Ae	Area at Exit (mm).
18. At	Area at Throat (mm).
19. Q	Mass Flow Rate (kg/s).
20. Pc	Pressure of the Air at Combustion Chamber (Bar)

CHAPTER 1

INTRODUCTION

1.1 PROJECT ORIENTATION

Recent advancements in the study of space have allowed us to better understand and tackle a wide variety of problems. Recent progress in the Thrust Vectoring Concept piques the user's interest and necessitates more effort on the part of the user in terms of study than was previously the case. The application of TVC is not limited to the space industry, but rather encompasses all areas of aeronautical technology. Thrust vector control, or TVC, is the ability of a vehicle to change its attitude or angular velocity by altering the direction of the push from its engine(s) or motor. It becomes very difficult to maneuver a system or a vehicle as its size and level of accuracy increase. It is important to build a thrust vector control nozzle that can withstand high temperatures, long periods of use, and severe erosion. After taking into account all of these factors, the final outcome should be a mechanism that is both lighter and more effective. This device exists so that the vehicle's pitch, yaw, and roll may be modulated by adjusting the propulsion. TVC may work with either one or several nozzles in a motor or engine. As a result, the idea of thrust vectoring has grown in significance throughout time.

The goal of this project is to provide a new testing environment in which a thrust vectoring control method may be evaluated. This method may be used to accomplish the appropriate level of rocket attitude control by adjusting the thrust vector. Creating a test facility involves creating a scale replica of a rocket with a detachable nozzle. In this case, the compressed air is constantly fed to the moveable nozzle. The nozzle must provide sufficient thrust to lift the test article, and the attitude of the rocket must be controlled by the algorithm.

All of the current thrust vectoring ideas are explained in this chapter's background, along with a short overview of nozzle basics and the many nozzle types that may be employed for each given project. The second section focuses on what should be considered while creating an effective nozzle.

Proper computation and a well designed mechanism are required for effective thrust vectoring. The following step, designing a nozzle, required the greatest attention to detail.

The amount of force generated by the nozzle is proportional to the mass flow rate, which is calculated in such a manner as to obtain the required mass flow rate. After all the math is done, a cad model of the nozzle is created. Software called COMSOL is used to examine the flow via the nozzle. If the theoretical and analytical findings are consistent with one another, then a perfect nozzle may be manufactured. Design the nozzle once you're done.

By altering the angle of the engine nozzle or other thrust-producing systems, the TVC system gives the vehicle the ability to steer in any direction. One such device that has been designed to manage the nozzle's alignment in all three dimensions is a prototype TVC mounted on a three-axis gimbal. The gimbal assembly of a three-axis gimbal TVC system may rotate around three transverse axes and is mounted to the engine nozzle. The engine nozzle may be adjusted in response to changes in the vehicle's orientation thanks to the gimbal assembly, which is controlled by a collection of actuators and sensors. To allow for efficient maneuvering and steady flight, the system is built to offer exact control of the thrust vector.

This report provides a thorough evaluation of a prototype TVC mounted on a three-axis gimbal for usage in spacecraft. The design of the gimbal assembly, the choice and integration of actuators and sensors, and the results of testing and validation are all included in this article. The prototype's success in testing shows that it can provide accurate and reliable control of the thrust vector in all three dimensions, making it an excellent candidate for use in spacecraft.

This Report will be structured as follows. The first section introduces the TVC system and describes the function of a TVC prototype mounted on a three-axis gimbal for use in spacecraft. The gimbal assembly's design, including the materials used, the layout of the parts, and the machining and assembly steps, are discussed Further .The control algorithm and software implementation, as well as the actuator and sensor choices, are discussed lastly , we detail the procedures, methods, and outcomes of the prototype's testing and validation. The primary findings and conclusions are summarized , and future research directions are dicussed

This Report's prototype of a three-axis gimbal TVC marks a major step forward in aerospace engineering and might be used in a wide range of rockets, satellites, and spacecraft. The prototype is a representation of the hard work and creativity of the engineers and scientists who created it, and it is hoped that it will lead to future developments in TVC system technology.

1.2 PROJECT OBJECTIVES

During flight, a rocket or spaceship might benefit from a thrust vector control system implemented on a three-axis gimbal prototype. In particular, a three-axis gimbal would be included to allow for precise control over the vector and magnitude of thrust generated by the rocket engines. Some of this project's most important aims may be Construction of a three-axis gimbal system for precise, on-the-fly control of the thrust vector. Using data from sensors (accelerometers, gyroscopes, and global positioning system), a thrust vector control algorithm is implemented to determine how to alter the gimbal angles. Prototype flight tests, including takeoff, landing, and a variety of maneuvers, in a simulated or actual environment. Minimizing fuel consumption and increasing efficiency by optimizing the design and control algorithm without sacrificing stability or controllability. Building a system that is both sturdy and trustworthy enough to function reliably for long periods of time in space. Rocket and spaceship control and stability may be enhanced by installing a thrust vector control system on a three-axis gimbal prototype, allowing for finer maneuvering and less chance of mission failure.

The idea is to give the rockets more control and make it easier for them to land safely. Aerodynamic fins and gimballed engines are used in conventional rocket designs for trajectory control during flight, but these approaches aren't ideal for vertical takeoff and landing. Small, movable nozzles will be used as part of the rocket's thrust vector control system, allowing for precise control over the rocket's propulsion. To provide perfect control during launch and landing, the system will be built to react rapidly and precisely to any changes in the rocket's orientation. The thrust vector control system for miniature rockets will be designed and tested as part of this project. The device will be tested to see whether it helps the rockets land more precisely and more precisely maneuver during vertical takeoff

and landing. The purpose of this research is to establish the viability of employing thrust vector control on small size rockets and to investigate its potential for future uses in space exploration and other disciplines.

1.2.1 Problem Statement:

- The system should be able to instantly change the direction of the thrust vector for any rocket engine, whether it be solid, liquid, or hybrid.
- Optimal performance requires that the thrust vector control system be made to work under a variety of weather conditions, including varying altitudes and weather.
- The control system should be flexible enough to handle bigger rockets or payloads, and it should be able to function with a variety of rocket configurations (such as single-stage and multi-stage rockets).
- The system should be built to function without human intervention, with sensors and algorithms on board to track the rocket's attitude and fine-tune the thrust vector as needed.
- Lightweight and compact design is essential so that the rocket's cargo capacity and performance are not compromised.
- The building materials utilised in the system must be able to survive the intense heat, vibrations, and pressures experienced during rocket launches and landings.
- Safe operation and the prevention of accidents or malfunctions during launch or landing need the inclusion of redundant components and fail-safes in the system's design.

- Reduced power consumption in the control system means more time and distance for your rocket in the sky.
- During the last phases of landing, the system should be able to provide precise control to guarantee a gentle and accurate touchdown.

1.2.2 Model Objective:

- Attain Deflection movement on 15° on developed gimbal prototype
- Make the developed model combustible capable to implement in the real time rocket
- Should be Developed for the purpose of vertical takeoff and landing model rockets.
- More efficient than other gimbaling schemes.
- Chosen mechanical concept of flex ball joint, where thrust vectoring is made easy.
- Study over the gimbal mechanism for flow properties
- Implementation of PID control algorithm over the developed model for landing model rocket and precise trajectory optimization
- Achieve zero steady state error from the control system

1.3 PROJECT STRUCTURE

In order to analyze flight mechanics, the initial phase of this research is to establish a modelling technique for thrust vectoring. Thrust vectoring modeling involves determining the effective deflection angle of vectoring thrust from the geometrical nozzle deflection angle and the amount of force lost owing to nozzle deflection. To instantly account for the impacts of thrust vectoring during flight mechanics calculations, the modeling of thrust vectoring must be connected with a flight mechanics analysis environment. Effective thrust deflection angle is calculated from geometrical nozzle deflection angle, and an analytical method is created for determining the loss of propulsion owing to nozzle deflection. This analytical solution is used in this project for conceptual modeling of thrust vectoring performance. The next step of the thesis project is to prepare all the inputs of the flight mechanics toolbox in order to run the flight mechanics analysis after the thrust vectoring modeling has been constructed and implemented. Steady-state trim calculation in Phalanx needs the rocket's mass and inertia in addition to the aerodynamic performance database for the specified aircraft configuration, control derivatives for all aerodynamic control surfaces, and an engine performance map. The necessary calculations are performed using the appropriate instruments for each group of inputs. Because this project's flight mechanics study of thrust vectoring control is conceptual in nature, the inputs preparation tools used are also conceptual in nature.



Fig 1.1 Workflow Chart

CHAPTER 2

BACKGROUND

The history of rockets spans long before the year 2000 beginning with invention, experimentation, discovery, and inspiration, which established the foundation for modern rocketry.

The Greek philosopher, mathematician, and astronomer Archytas is widely believed to have created and used an ephemeris.

Sometime between 428 and 347 B.C., a tiny bird-shaped contraption powered by a jet of steam or compressed air achieved controlled flight for the first time.

Polish Lithuanian "Kazimierz Siemienowicz" published a paper in which he proposed a rocket constructed of stages. Galileo constructed the principle of property that was later developed by Newton as inertia, which aided Newton in consolidating all of rocket science by developing the three well-known laws of motion, during the rapid technical advancements of the 16th and 17th centuries.

On March 16, 1926, the United States successfully launched the first liquid propellant rocket, however the results weren't all that stunning (the rocket only reached a height of 12.5 meters). The experiment was led by college professor Robert Goddard, who was the first to include gyroscopes system to control his rockets flight, parachute recovery system, and instrumentation payload compartment to isolate fuel and oxygen tanks from the chamber of combustor. He is commonly considered the progenitor of modern rocketry.

Dr. Werner von Braun, a figure in the German rocket program that resulted in the V2 "most powerful rocket at the time," ushers in the space age following World War II. He rose to prominence as an advocate for the American space program. He fled to the United States after the war and eventually became a citizen. His work centered on creating ICBM "intercontinental ballistic missiles" and America's first satellite "Explorer-1." As an engineering architect chief, he oversaw the production of Saturn V moon , which landed on the moon during the Apollo 12 mission. During his tenure, an era of astronauts emerged under the influence of American media.

Russia was and still is partially the leading space company, particularly in the production of cryogenic rocket engines, a type of engine that was built by only 7 major nations later, but Russia engines still led in 16 Efficiency by using a secret that was recently debunked consisting of fuel cycles and ratio of pumped fuel to oxygen ("the combustion chamber was reach in oxygen more than fuel").

SpaceX is one of the innovative companies that reduced the cost of engine production by incorporating the reusability factor and developing a potent rocket called "The Merlyn" to compete with Russian engines.

Recent rocket engines are more complex and diverse, ranging from liquid engines to hybrid and solid engines, not to mention ion and plasma turbines; the choice of engine largely depends on the mission's objectives, resources, and strategy. In the chapter devoted to experimentation, we will discuss our rocket engine type, why we chose it, and the entire process of its creation.

Control methods for rockets have also undergone numerous improvements, Optimal missile roll control, multi-variable guidance of gravity-turn rockets, two-point boundary value, and more are just some of the topics we've covered. as well as the use of simple methods ranging from tuning to disturbance observer.

The basic idea is to redirect the force produced by the average flow of an engine jet along the direction of the target. As a result, the intended system's attitude is altered as a result of an imbalance in its momentum. Rockets and aero planes benefit from increased maneuverability thanks to thrust vectoring.

When conventional aerodynamic control methods like as ailerons, flaps, elevator, and rudder become ineffective (P, 2014; Jeffrey D. Flamm*). There are two methods for doing thrust vectoring. There are two types of actuation systems: fluidic and mechanical. In comparison to mechanical actuating systems, fluidic actuating systems have their own benefits, such as being lighter in weight. To accomplish thrust vectoring with fluidic actuation systems, secondary fluid injection is used to deflect a portion of or the whole fluid jet away from the axis and in the desired direction. Systems like this are employed in rocket and missile nozzles with a large expanding area. Fluidic actuation systems are also the focus of

current research and development. A hydraulic or pneumatic actuator is used in mechanical actuating systems to execute the deflection of the jet. The vehicle's directional stability is maintained by TVC by rotating (gimbaling) the pushing chamber or re-directing the exhaust-gas stream.

Movable vanes or the addition of another liquid to the flow are two common methods for regulating the weakening gases. TVC is often used on boosters and higher stages (high-thrust systems) to regulate pitch and yaw attitude. Instead of using the tilt-rotate-control system, the shuttle uses attitude-control thrusters. For the pushed vector center to line up with the spacecraft's center of mass, it may be necessary to provide some means of coordinating the pushed vector. This structure must take into consideration changes in centre of mass as fuel burns to provide crucial fabricated resilience. If the influence torques caused by a misaligned pushed vector are modest enough, the spacecraft's reaction control framework (that is, beating thrusters) may be able to overcome them. Many engines start may involve deliberately spinning the whole vehicle.

In thrust vector control, the thrust generated by the engine or motor is modulated as the vector to be controlled, and the rocket, aircraft, or aero plane is guided along the desired trajectory by applying torque along one of three axes (pitch, yaw, or roll) with maximum angle shifting of 5 degrees.

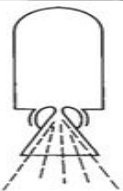
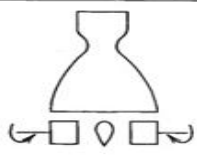
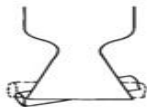
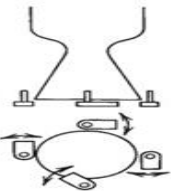
Gimbal or hinge  Universal joint suspension for thrust chamber L	Flexible laminated bearing  Nozzle is held by ring of alternate layers of molded elastomer and spherically formed sheet metal or plastic composite (fiberglass epoxy) S	Flexible nozzle joint  Sealed rotary joint S	Jet vanes  Four rotating heat resistant aerodynamic vanes in jet L/S
Jetavator  Rotating airfoil shaped collar, gimbaled near nozzle exit S	Jet tabs  Four paddles that rotate in and out of the hot gas flow S	Side injection  Secondary fluid injection on one side at a time S/L	Small control thrust chambers  Two or more gimbaled auxiliary thrust chambers L

Fig 2.1 Schematic diagrams for eight different TVC mechanisms

CHAPTER 3

LITERATURE SURVEY AND STATE OF ART

To ascertain the state of the art in thrust vectoring systems for airbreathing engines, a literature study was done. Several reports discussed the use of thrust vectoring to improve the mobility of combat planes. The one and only source discussing the use of thrust vectoring in the special case of orientation control is presented here. The published literature gives insight into several thrust vectoring systems and their relevance to different engine configurations and conditions, although being restricted in control applications. This lists the many vectoring technologies that were examined, who the paper's principal author was, and what the paper's key findings were.

Push vectoring is the technique of diverting some of the push produced by an airplane's engine in a direction other than the axial direction. An aircraft or missile's roll, pitch, and oscillation may all be managed by thrust vectoring. Thrust vector control (TVC) improves fighter aircraft's controllability and survivability. Maneuverability and control are made possible by TVC even at low airspeeds and extremely high angles of attack . Two operational aircraft that employ TVC to enhance mobility are the United States Air Force's F-22 and the experimental X-31.

There are a variety of TVC methods that may be used on a jet engine. Several techniques can be used to divert exhaust gases before they leave the nozzle, including rotating the engine in relation to the vehicle, rotating the engine nozzle to redirect the gases in a different direction, inserting movable vanes or paddles into the exhaust flow, secondary fluidic injection, and diversion. Generally speaking, these methods may be broken down into three groups: nozzle manipulation, secondary fluidic injection, and exhaust gas deflection. Any television commercial (TVC)

Because some of the engine's total thrust is being diverted away from the axial line, the procedure decreases the available axial thrust.

In order to improve the thrust of a jet engine, the velocity of the exhaust gases is often increased by use of a convergent or convergent-divergent nozzle. The latter kind uses an aerodynamic pharynx (or choke point) to achieve supersonic velocities in the exhaust gases. Thrust vectoring has traditionally been used by combat planes using a convergent-divergent nozzle configuration. For this reason, most TVC system publications focus on supersonic nozzles and the capabilities and flow qualities they provide.

3.1 MECHANICAL NOZZLE DEFLECTION

Thrust vectoring is often accomplished by altering the shape of the exhaust nozzle on a rocket motor or jet engine. The direction of the bullet's propulsion may be changed by physically turning the projectile. This is the standard procedure for rocket-powered launch vehicles and is often utilized for flying machines.

Convergent-divergent nozzles are sophisticated pieces of machinery that need precise calibration to operate properly under a wide range of pressure and flow rates. Changing the nozzle's shape, particularly the aerodynamic pharynx's design, to vector the flow might have unforeseen implications that diminish engine performance and thrust. It was postulated in 1990 by Berrier and Taylor that turning the flow using a gimbal mechanism upstream of the nozzle would have no influence on the vehicle's thrust. By rotating the flow at the low-velocity, subsonic area just before the pharynx, turning losses may be kept to a minimum. The results of their experiments support this claim. At geometric vector angles up to 25° and throughout a broad range of nozzle pressure ratios, the authors observe a peak thrust loss of just 2%. They also speculate that an internal accounting miscalculation is to blame for this setback.

There are instances of mechanical nozzle manipulation on two U.S. Air Force fighter aircraft: the F-35 and the F-22. The F-35 uses a gimbal, as seen in above figure

Primary Author	Technology Investigated	Date Published	Major Conclusions
Berrier	Gimbale d nozzle	1990	Turning the flow in the subsonic regime before the nozzle choke point is a highly efficient vectoring technique.
Carson	Variable geometry nozzle	1991	Meant to turn the flow in the sonic regime, this method accidentally manipulated the aerodynamic choke point, turning the flow in the subsonic regime. Also a very efficient technique.
Wing	Adjustable internal nozzle contouring	1997	Adjustable contouring did not produce satisfactory degrees of vectoring. This method also greatly reduced the overall thrust level. Not a viable method.
Deere	Separation Control	2003	Not effective at changing the location of the choke point, but controlling the flow separation in the nozzle did achieve large vectoring angles.
Flamm	Dual Throat Fluidic Vectoring	2007	Most effective when the two minimum areas are equal. This method also achieves large vectoring angles by controlling the flow separation.
Dungan	V-2 Post-exit vanes	2011	One of the earliest known uses of thrust vectoring. These vanes were functional, yet limited in effectiveness because they had not been fully characterized.
Berrier	Post-exit thrust vectoring vanes	1988	Applicable across a wide range of nozzle configurations. Capable of very large vectoring angles, but at the cost of overall thrust.
Bowers	F-18 Thrust Vectoring	1996	Used post-exit vanes despite thrust loss issues. Effectively improved the F-18's maneuvering abilities.
Alcorn	X-31 Thrust Vectoring	1996	Also used post-exit vanes. Tests demonstrated the ability of this method to stabilize a tailless plane.

Table 3.1 Evolution of Thrust vector control

Two U.S. Air Force fighter planes, the F-35 and the F-22, include mechanically manipulated nozzles. The F-35 uses a gimbal system for thrust vectoring, as shown in

Figure 2.1a, to achieve short takeoff and vertical landing times. The F-119 engine used by the F-22 is shown in Figure 2.1b, along with the amount of nozzle rotation in the pitch plane.

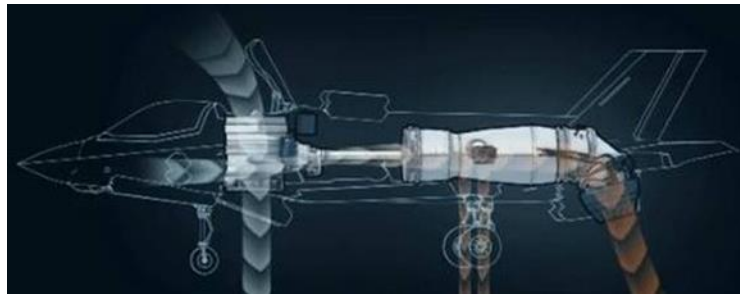


Fig 3.1 depicts the F-35B's use of TVC technology. The rear nozzle is visibly rotating..



Fig 3.2 This image depicts the F-35B's use of TVC technology. The rear nozzle is visibly rotating..

Both the F-22 and F-35 feature propulsion vectoring systems with a single axis. Researchers have also looked at what happens when various engine layouts are fitted with multi-axis thrust vectoring devices. In 1991, Carson and Capone suggested using a circumferentially split divergent nozzle flap that could direct thrust in any direction. A number of subscale nozzles with diverging parts of fixed geometry typical of the full-scale variable geometry were tried in order to investigate the impacts of this multi-axis system. Particularly interesting were the results of flow twisting in the divergent region when the flow is supersonic.

Carson and Capone determined the real aerodynamic throat moves because of the deflection of the divergent flaps and is largely independent of the throat's physical placement using pressure measurements along the divergent section. Orifice pressure ratios dictate the precise placement of the throat, which is significantly skewed with respect to the axial

centerline. Carson and Capone discovered that flow reversal is primarily accomplished at subsonic velocities as a consequence of the throat's inclination and positional alterations. This results in minimal or nonexistent spinning losses. According to their data, under vectored nozzle conditions, there is only a 1% reduction in thrust.

When the tested nozzles' pressure ratios were high enough to minimize flow separation on the nozzle's back side, Carson and Capone discovered that the nozzles were successful in vectoring thrust. Surprisingly, the nozzle was able to achieve resultant thrust vectoring angles larger than the geometric angles when the pressure ratio was high enough. At a nozzle pressure ratio of 3.0 and with the nozzle in an afterburning configuration, Carson and Capone reported an effective thrust vector angle of 37, whereas the geometric angle was only 30. Scientists found no difference in the recorded pitch and yaw vectoring angles. They concluded that their technique would be effective for multi-axis thrust vectoring, which was their original premise.

Accidental nozzle manipulation led to the observed translation and tilt of the aerodynamic pharynx in . Wing et al. presented a nozzle in which the internal contouring may be adjusted mechanically to control the translation and inclination of the throat. This was possible because the nozzle's four separate portions for adjusting the pharynx height. The internal flow routes of the exhaust gases may be modified by changing the shape of these portions. The use of these pieces resulted in a nozzle with a unique flat diamond or parallelogram form. The nozzle also had separate spaces for the upper and lower pharynx, allowing for directional movement in both directions. They aimed to accomplish flow twisting in the subsonic range and decrease thrust losses by adjusting the angle of the pharynx with reference to the centerline of To test the adaptability of their idea, Wing and his team constructed a large number of miniature nozzles of fixed forms. Even when not set up for vectored flow, the peculiar parallelogram nozzle proved ineffective. Once a particular threshold is reached, further extension of the divergent section is halted, and the upper and lower surfaces revert to being parallel. That's because increasing the orifice pressure ratio does nothing to increase the thrust provided by the underexpanded flow. Even at the highest possible thrust ratio, pressure readings reveal regions of flow separation, circulation, and under inflated flow sections. The peak thrust ratio produced by these components is 1-2% lower than that produced by conventional convergent-divergent nozzles.

Several combinations of pitch and yaw vectoring were evaluated by Wing et al. Pitch-direction thrust vectoring proved unsuccessful when the internal flow-path contours were changed by adjusting the four pharyngeal parts. Many times, and especially at lower pressure ratios, they were unable to significantly alter the direction of flow. The pharynx, as measured by pressure, was not developing along the predicted internal ridges and remained mostly perpendicular to the flow. They attempted to redirect the flow by placing strands of material on the ridges in an effort to magnify the impact of the physical pharynx. These ridges served their purpose of guiding the pharynx to the right place, but at the expense of severe propulsion losses. Limits on thrust vectoring and related performance loss resulted from nozzle exit flaps that stayed parallel to the original engine centerline, forcing the flow to turn in the supersonic zone.

The findings improved when the pharynx was moved about using the compartments, although this had a direct negative effect on the nozzle's output power. This technique only yielded 6 degrees of thrust vectoring at full displacement. In addition, lower nozzle pressure ratios resulted in less noticeable vectoring. Since most maneuvering applications that need thrust vectoring take place at low nozzle pressure ratios, this is undesirable.

Yaw vectoring was shown to have similar but distinct differences in thrust losses and flow difficulties compared to pitch vectoring. There was a key difference in that.

It was found that the maximum vectoring angle might reach about 11 degrees. The maximum angle flipped in certain nozzle types and declined with increasing pressure ratios. Even though 11 degrees is quite a little, it's not nearly enough for exceptionally huge and enormous combat planes to maneuver. Adjustable internal contouring is not a feasible solution for thrust vectoring due to the complicated flow interactions and accompanying thrust losses demonstrated in the pitch and yaw vectoring demonstrations. The nozzle.

3.2 SECONDARY FLUID INJECTION

Even though it does not change the nozzle's physical configuration, thrust vectoring by secondary fluidic injection follows the same idea as nozzle manipulation. Instead, a second fluid (usually exhaust air from the engine compressor or fan) is injected to regulate the

exhaust flow. The injection of fluid serves to create artificial orifice borders with varying sizes. Because they may be built as fixed-geometry nozzles without complex adjustable hardware, fluidic vectoring nozzles can have greater stealth properties and be less in weight.

The three most studied methods of fluidic thrust vectoring are counter flow, shock vector control, and pharyngeal shifting. Different approaches may be compared and contrasted in terms of their relative effectiveness. The effectiveness of these nozzles is measured by the amount of secondary flow achieved as a fraction of the main nozzle flow. Moreover, similar to previous thrust vectoring methods, propulsion losses are compared. The shock vector control technique works by erecting shock waves in the path of a supersonic flow. These shock barriers are effective in turning the flow around, but they reduce thrust overall. using thrust ratios between 86% and 93%, vectoring efficiencies of up to 3.3%/%-injection are possible using the shock vector control approach

The suction used in the counter-flow method is situated behind the orifice, between the collar and the trailing edge. When using the hoover, the collar's flow is reversed. A thrust differential is created when the pressure drops, and the speed goes up close to the region of reversed flow. Higher vectoring efficiency (between 92% and 97%) is achieved with this method's increased thrust ratios.

One drawback of this method is that it calls for the use of supplementary vacuum equipment.

Although fluidic pharyngeal shifting causes very small losses in thrust, it performs poorly. To change the subsonic flow, the nozzle throat is manipulated in a manner similar to the mechanical throat shifting approach described . The pharynx is reshaped in a way that is more successful than by traditional mechanical means. With this technology, you may keep your thrust ratio between 94% and 98% while achieving vectoring efficiencies of up to 2%-injection. . Similar to the throat-shifting approach, but with improved vectoring efficiency, Deere et al. suggested a novel method to modify the acoustic plane. The vectoring capabilities created by secondary flow injection in a 2-D dual-throat nozzle were analyzed using the computational fluid dynamics tool PAB3D. The dual-throat nozzle is designed with

two smaller throats that are separated by a larger chamber. In all trials, a secondary flow was injected at the upstream minimum region and maintained at a constant rate of 6% of the primary flow rate. The design factors for this set-up were as follows: upstream minimum height, fluidic injection angle, cavity angle, and cavity length. All configurations' vectoring efficiencies and thrust ratios were provided.

According to raising the cavity convergence angle from 9.55 degrees to 20 degrees improved thrust vectoring efficiency from 1.49 per cent of injection to 1.71 per cent of injection with no effects on thrust performance. Although vectoring performance was modestly improved, thrust ratios improved by 1.5% due to the shorter cavity length. Improving thrust vectoring efficiency at the expense of a 2.2% thrust penalty was possible by reducing the upstream minimum area to match the downstream minimum area.

The most effective increase in vectoring efficiency with the least impact on thrust ratios was achieved by adjusting the injection angle. Thrust vectoring efficiency was improved from 0.84 per cent of injection to 2.15 per cent of injection while the thrust ratio was only reduced by 0.6%. Although the researchers were trying to control the orientation and position of the acoustic line in the orifice, they point out that this is not necessary for the successful vectoring of the main flow.

Just plain manipulating. Instead, controlling the flow separation within the recess cavity was the key to achieving high vectoring angles. The standard throat-shifting approach of fluidic thrust vectoring was improved by adjusting the flow separation. Miniature thrust-vectoring dual-throat nozzle built and tested by Deere et al. To evaluate various design parameters, they built a small-scale axisymmetric dual-throat nozzle using data from , other sources, and their own work using PAB3D. They reevaluated the impacts of varying nozzle designs and secondary injection circumferential span, as well as the effects of cavity length and cavity convergence angle .

Because a nozzle with a single shape is inefficient across the board, research has focused on nozzles with adaptable geometries. The tested designs included nozzle expansion ratios tailored to certain flight phases. Secondary flows were injected at a constant rate of 150 in

all tests. While comparing 60 and 90 degree circumferential spans, the flow injection area remained constant.

Once again, Flamm et al. discovered that parity between the two minimum regions yields the best thrust vectoring performance. The equal geometry dual-throat nozzle outperformed the variable geometry nozzle in terms of vectoring angle and engine thrust performance across all pressure ratios evaluated. Thrust vectoring efficiency was reduced but thrust ratio was increased when the cavity length was shortened, as was the case in the prior computational analysis. The thrust ratio suffered when the angle of convergence of the cavity was increased, while the thrust vectoring efficiency improved somewhat. The experimental results showed that the vectoring angles produced by the 60-injection span were 1.5 to 2 degrees larger than those produced by the 90-injection span.

Eiders et al. recently investigated how secondary fluid injection may be used to regulate thrust vectoring on a cold-flow aerospike nozzle. By injecting a second fluid at an angle perpendicular to the orifice axis into the main flow field, vectoring forces might be produced. Their experimental aerospike nozzle was reduced in length by 57% of its whole theoretical length. Carbon dioxide compression was the working fluid.

Eiders et al. investigated how the placement of injection ports affected vectoring performance. Secondary injection test apertures were positioned at 20%, 80%, and 90% of the axial distance. The maximal force amplification factor of 1.4 was achieved when the injection point was situated at 90%. That is to say, compared to what the secondary injection might have accomplished on its own, the combined primary flow and secondary injection created 40% greater vectoring force. Specific shocks lasting up to 55 seconds were recorded at the 90% injection site. The average duration of the primary flow-specific impulse in the series of trials was around 38 seconds. The data taken during these studies showed that the nozzle pressure ratio had no effect on the amplification factor for the side force.

2.3 EXHAUST FLOW DEFLECTION

Direct flow manipulation employing post-nozzle exit vanes or paddles is an alternative to nozzle manipulation. Figures 2.3a and 2.3b display examples of aircraft employing paddles.

On German V-2 missiles from World War II, exhaust flow deflection, a precursor to modern thrust vectoring, was discovered. The four graphite vanes attached to the V-2's hot exhaust flow helped direct the missiles in the right direction. Four vanes, shown in Figure 2.2, were steered by a set of gyroscopes. While the V-2 often struggled to hit city-sized targets, its gyroscopic guidance system and use of thrust vectoring for maneuvering made it a cutting-edge vehicle for its time .

The effectiveness of a substantially more advanced paddle-based device was evaluated by Berrier and Mason in 1988 . They looked at the results of varying the paddles' curvature, placement in relation to the orifice, number of paddles, and degree of deflection, among other design and placement factors. Using a miniature compressed-air-powered single-engine simulator, they tried out many paddle layouts. Upon first glance, it was clear that paddles with more pronounced spoon shapes diverted the flow more than their radially curved counterparts. This is due to the fact that the terminal angle exceeds the geometric angle of the spoon by around 9.3 degrees. Researchers found that higher flow deflection always led to greater thrust losses.

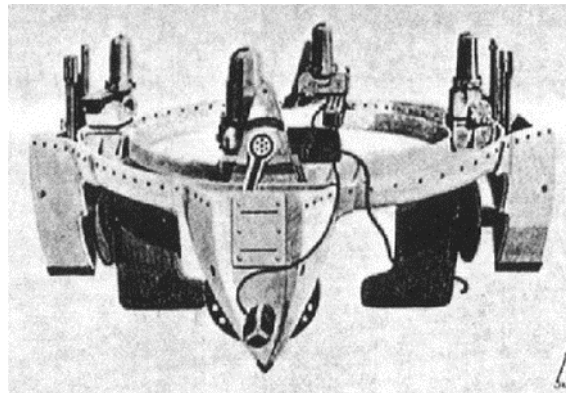


Fig 3.3 Illustration of the German V-2 rocket's graphite thrust vectoring vanes.

The resulting angle of the thrust vector was not significantly affected by where the paddle hinge was located (axially or radially), although both had an effect. Relocating the paddle's fulcrum further from the nozzle exhaust flow decreased the vector angles produced by the paddle's impingement on the flow. As predicted, the resulting angle of propulsion vector coincided with the deflection of one paddle. It was found that by deflecting a second paddle

by an equal amount, the direction of the vectored propulsion could be changed to practically any angle.

The effectiveness of paddles with three and four equally spaced blades was compared by Berrier and Mason. With four paddles, power could be easily vectored in either the pitch or yaw directions, but that was the only advantage. The three-paddle arrangement vectored well in the pitch direction but had trouble with yaw.

The resulting thrust vector angle was only mildly affected by the number of paddles deflected into the flow. As was previously mentioned, this sort of TVC has a direct correlation between the generated vector angle and the available propulsion, with the latter decreasing as the former increases. Researchers could not show a clear relationship between these numbers, but they did talk about one specific scenario with a three-paddle arrangement and a six-to-one orifice pressure ratio. When one paddle was fully deflected, the vector angle measured away from axial centering was 11.6, resulting in a 10% decrease in thrust ratio. After affixing a second paddle with its deflection all the way out, the

While the thrust ratio dropped by 21%, the resulting vector angle increased to 19 degrees. The system's efficiency would plummet as a result of these large losses. As mentioned in (2.1), these losses occur because of the rotation of the flow at supersonic speeds.

Despite the constraints of post-nozzle departure paddles, two full-scale flying demonstration projects—NASA's F-18 High Alpha Research Vehicle (HARV) and the international X-31 Enhanced Fighter Maneuverability demonstrator—utilized a three-paddle design. These two vehicles proved that, under certain conditions, fighter planes' mobility could be enhanced by using post-nozzle escape paddles. There is no noticeable loss of propulsion when the blades are not bent. While employing the paddles resulted in a large reduction in thrust, both vehicles spent some time in the post-stall flying envelope, greatly enhancing their responsiveness.

The experimental consequences of thrust loss revealed by Berrier and Mason were verified by the F-18 HARV team's observations. They also concluded that the distance between the

paddles was more important than the size of the paddles themselves. They found that by placing the oars closer together, less water would escape between them, making it easier to alter the current . The F-18 HARV is seen in Figure 2.3a doing ground-based vectoring trials. The major exhaust column is clearly directed upwards, although some of the flow can be seen escaping between the paddles.



Fig 3.4 Examples of thrust vectoring that used the exhaust flow deflection technique.

Figure 2.3b depicts the X-31's thrust vectoring system, which proved so successful that the research team undertook four further quasi-tailless flying trials. The tests were conducted to see whether the thrust vectoring technology might improve the vertical stabilizer's effectiveness. Significant weight, zero-lift drag, and radar observability savings might be possible if the vertical stabilizer could be shortened or deleted altogether. By combining the rudder with the computerized flight control system, the researchers were able to nullify the stabilizing effects of the vertical stabilizer. A system overlay was then installed, and the ailerons and TVC system were employed to stabilize the plane again. All four trials with nearly tailless flying were a huge success .

CHAPTER – 4

THEORETICAL BASE

4.1 DEFINITIONS AND FUNDAMENTALS

Solid propellant rocket motors are easier to use than liquid propellant rocket motors because they don't need separate tanks to hold the propellant, pipes and turbo-pumps to move the propellant to the combustion chamber, or injectors to get the propellant into the combustion chamber. Even starting SRM is a lot easier.

First, an internal (in professional motors) or exterior (in amateur and commercial motors) igniter gets the solid propellant burning. Gases produced by combustion are released into the combustion chamber when the grain's exposed surfaces catch fire. The result is an increase in pressure inside the chamber. The gases expand as they go through the nozzle and out of the device. The combustion chamber pressure may rapidly or slowly increase, depending on the grain structure. As a result, the rate at which mass is ejected increases and remains high until the process is stopped. The rocket engine is "choked" when the gas velocity approaches Mach 1 at the throat, the region with the smallest surface area. When the flow is obstructed, fluctuations in downstream pressure have no effect. Pressure variations in the chamber upstream do, however, have an effect. When this occurs, supersonic flow develops at the nozzle's bifurcation region. At the same time as this is occurring, the pressure of the hot gas on the motor's walls and the mass being ejected together produce thrust. This force acts counter to the flow's direction, but in the same direction. After the fuel runs out, the chamber's pressure declines exponentially until it equals that of ambient air. There is also a big difference between the SRMs used in professional aerospace applications and space exploration and the ones used by amateurs. The latter are much simpler and can be split into two types: single-use motors and motors that can be used more than once.

Since they are constructed from phenolic resins that erode during operation and cannot be deconstructed, single-use motors are designed to be used just once and often generate less total impulse than reusable motors.

Nevertheless, reusable motors may be used again since their components can be disassembled and discarded after each flight, making room for brand new parts and fresh grains of propellant. Metal casings are used in these motors so that they can resist repeated uses under different conditions. Nevertheless, the nozzles may also be made of metal, making them either reusable or disposable. Both phenolic resin and steel with graphite

inlays in the neck area are viable materials for the latter, although the latter is more susceptible to erosion.

Another system for defining solid rocket motors is extensively utilized in the hobbyist community. It classifies the motors from A to O according to the magnitude of the impulse they produce. Table 4.1 lays out this system of organization. As can be seen, the impulse increases by a factor of 2 for every letter range up to the maximum of 40960 N s, which is generally accepted as the upper limit for homemade rocket motors.

Next follows the letter, then the average thrust of the motor being identified. For example, the L995 (41% L) motor generates a typical thrust of 995 N and a total impulse of anything between 2560 N s to 5120 N s. The overall impulse of this 41% L motor is 3618 N s.

Class	Total Impulse (N · s)	Class	Total Impulse (N · s)
A	1.26 – 2.5	I	320 – 640
B	2.5 – 5	J	640 – 1280
C	5 – 10	K	1280 – 2560
D	10 – 20	L	2560 – 5120
E	20 – 40	M	5120 – 10240
F	40 – 80	N	10240 – 20480
G	80 – 160	O	20480 – 40960
H	160 – 320		

Table 4.1: Solid rocket motor letter classification by total impulse.

4.1.1 Motor Components

- Motor Casing is the most important component of the solid rocket motor where the combustion of SRM takes place, meanwhile it has to house the propellant and withstand the combustion temperature and pressure occurred in the whole system

during the flight operations, it is mainly responsible for transmission of all the thrust until it reaches bottom line.

- End Closure - The axial loads resulting from gas pressure are handled by the end closure utilized as the top-closing component.
- Nozzle – Boosts the motor's thrust by raising its speed and allowing the combustion products to expand to atmospheric pressure.
- Liner – Shields the container from the high heat produced inside the engine.

4.2 IDEAL NOZZLE THEORY

In order to produce the mathematical connections needed in the chapter on preliminary design to identify the primary design parameters, a brief explanation of the ideal nozzle theory will be provided here. The notion of an ideal propulsion system is often used early in the design process to simplify the gas dynamics principles and thermodynamic processes that occur inside the nozzle.

Assumptions that define a perfect nozzle include the following: There are no shock waves inside the flow, (1) the working fluid follows the ideal gas law, (2) adiabatic flow occurs, (3) no friction occurs at the walls, and (4) all exhaust gases are released in the direction of the nozzle axis, and (5) all inlet gases are recirculated. All cross-sections of the nozzle are assumed to have the same gas variables (temperature, pressure, velocity, and density) due to the assumption of quasi-one-dimensional flow.

The working fluid is considered to have a frozen composition, which means that the gas composition remains constant over the whole length of the nozzle (number 7). It is a steady state flow (9) and the flow is compressible and isentropic (see (8)).

Following the optimum nozzle analysis, a set of correction factors are always used to account for performance losses.

Energy conservation in an adiabatic and isentropic flow may be used to interpret the decrease in enthalpy between two axial cross-sections as a gain in kinetic energy, provided that potential energy changes are neglected.

$$h_x - h_y = c_p(T_x - T_y) = \frac{1}{2}(v_x^2 - v_y^2) \quad (4.1)$$

By using the conservation of mass principle of steady-state flows, the mass flow rate may be determined at a particular control volume input and outlet, and then expressed in terms of the cross-sectional area (A), the velocity (v), and the specific volume (Vs) (V).

$$\dot{m}_x = \dot{m}_y = \frac{Av}{V} \quad (4.2)$$

This is how the ideal gas law is put into words.

$$P_x V_x = R_g T_x \quad (4.3)$$

In a gas, the universal gas constant, R_u , multiplied by the molecular weight, MW, yields the gas constant, R_g .

The following equation is always true in all isentropic process

$$\frac{T_x}{T_y} = \left(\frac{P_x}{P_y}\right)^{\frac{\gamma-1}{\gamma}} \quad (4.4)$$

By adding the local static temperature to the kinetic energy contribution, we get an approximation of the adiabatic stagnation temperature.

$$T_0 = T + \frac{v^2}{2c_p} \quad (4.5)$$

The stagnation conditions will now be denoted by the subscript 0, while the subscripts 1, t, and 2 will continue to be used to describe the nozzle inlet, pharynx, and exit sections, respectively.

$$\frac{T_0}{T} = 1 + \frac{1}{2}(\gamma - 1)M^2 \quad (4.6)$$

and the ratio of static to stagnant pressure as

$$\frac{P_0}{P} = \left[1 + \frac{1}{2}(\gamma - 1)M^2 \right]^{\frac{\gamma}{\gamma-1}} \quad (4.7)$$

It is now possible to use the Mach number and equations (4.2), (4.3), (4.6), and (4.7) to calculate the area ratio between two nozzle cross-sections.

$$\frac{A_y}{A_x} = \frac{M_x}{M_y} \sqrt{\frac{\left[1 + \frac{(\gamma-1)}{2} M_y^2 \right]^{\frac{(\gamma+1)}{\gamma-1}}}{\left[1 + \frac{(\gamma-1)}{2} M_x^2 \right]}} \quad (4.8)$$

Another critical element that is closely connected to the produced thrust is the exhaust or exit velocity, which may be calculated using equation (4.1)

$$v_2 = \sqrt{2(h_1 - h_2) + v_1^2} \quad (4.9)$$

Nevertheless, when $A_t/A_1 \ll 1$, the intake velocity becomes so tiny that the term v_1^2 may be ignored, and the inlet characteristics (T_1 , P_1) change little from the stagnation circumstances (T_0 , P_0). As a result, using equations (4.1), (4.4), and the specific heat relation for isotropic gases ($c_p = \gamma R_g/(\gamma - 1)$), the latter formula may be expressed as

$$v_2 = \sqrt{\frac{2\gamma}{\gamma-1} \frac{R_u T_0}{MW} \left[1 - \left(\frac{P_2}{P_0} \right)^{\frac{(\gamma-1)}{\gamma}} \right]} \quad (4.10)$$

Cross-sectional areas of nozzles may be expressed in terms of local static pressure as an alternative to Mach numbers. This may be derived by using the equating procedure in equation (4.2) on the neck and on any other chosen part. For choked circumstances, $M_t =$

1, hence equations (4.3), (4.4), and (4.10) are replaced accordingly. In its final form, the statement reads as

$$\frac{A_t}{A} = \left(\frac{\gamma+1}{2}\right)^{\frac{1}{(\gamma-1)}} \left(\frac{P}{P_0}\right)^{\frac{1}{\gamma}} \sqrt{\frac{\gamma+1}{\gamma-1} \left[1 - \left(\frac{P}{P_0}\right)^{\frac{(\gamma-1)}{\gamma}}\right]} \quad (4.11)$$

It is possible to rewrite the formula for the mass flow rate under constant and clogged circumstances as

$$\dot{m} = A_t P_0 \gamma \frac{\sqrt{\left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{(\gamma-1)}}}}{\sqrt{\gamma R_0 T_0}} \quad (4.12)$$

The air pressure (P_3) around the motor's surface and the expulsion of a large volume of gas at high velocity combine to create thrust. Total thrust may be determined using the formula.

$$F = \dot{m} v_2 + (P_2 - P_3) A_2 \quad (4.13)$$

Keep in mind that P_3 disappears in space, and therefore the greatest force created at any given altitude occurs only when $P_2 = P_3$, which may seem counterintuitive. Even though it may seem that the thrust is greater if $P_2 > P_3$, the first term is actually reduced due to the incomplete expansion of the gas within the nozzle.

The integral of the thrust over time, or total impulse, is another crucial metric that must be determined.

$$I_t = \int_0^t F dt \quad (4.14)$$

including the ratio of thrust to weight flow, known as the specific impulse.

$$I_{sp} = \frac{I_t}{m_p g_0} = \frac{\Gamma}{\dot{m} g_0} = \frac{\Gamma}{\dot{w}} \quad (4.15)$$

The mass of the propellant, M_P , is multiplied by the acceleration of gravity, g_0 , at Earth's surface.

While being expressed as a time value, the particular impulse is a useful metric for comparing the efficiency of various motors and propulsion systems. A greater I_{sp} indicates that a motor is more efficient since more thrust is generated from the same quantity of fuel.

4.3 INTERNAL BALLISTICS

The study of the combustion properties of the solid propellant grain is what is meant by "internal ballistics." It details the conditions under which the propellant will burn and function.

The pace at which something is burned is the first of these variables. The rate of consumption of grain. The chamber pressure is directly proportional to the propellant thickness, or the regression of the web, which is roughly perpendicular to the web's surface. The APN model (see equation (4.16)) is the most used one for describing the relationship between burn rate and time. Burning rate may be defined by the chamber pressure (P_0), a proportional constant (a), and an exponential constant (n), according to an empirical generalization. The last two are experimental variables that change with the investigated propellant.

$$r_b = a P_0^n \quad (4.16)$$

It is necessary to collect extensive experimental data on the propellant through stringent laboratory characterization in order to implement alternative expressions of the burning-rate, such as those that take into account erosive burning, ambient temperature dependence, or increases in rocket acceleration.

Burning rate, burning area (A_b), and solid propellant density (p) all influence the mass flow rate at which gases are produced in the combustion chamber.

$$\dot{m} = A_b r_b \rho_p \quad (4.17)$$

Unlike the burning rate and burning surface, the propellant density does not vary over time in the preceding equation. These two variables may be employed throughout the design phase to get the desired results. The generated thrust profile will be progressive if, for instance, the grain is constructed such that A_b only grows with time. The thrust profile is progressive if A_b decreases over time, and neutral if A_b remains constant during the burn (see figure 4.

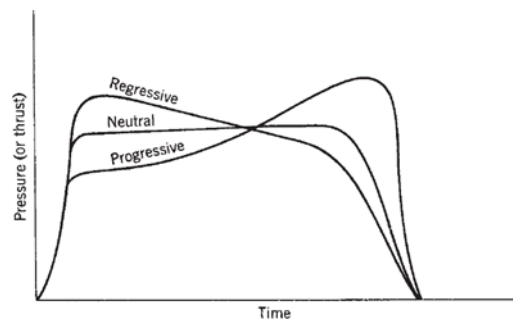


Figure 4.1: Types of thrust-time profiles.

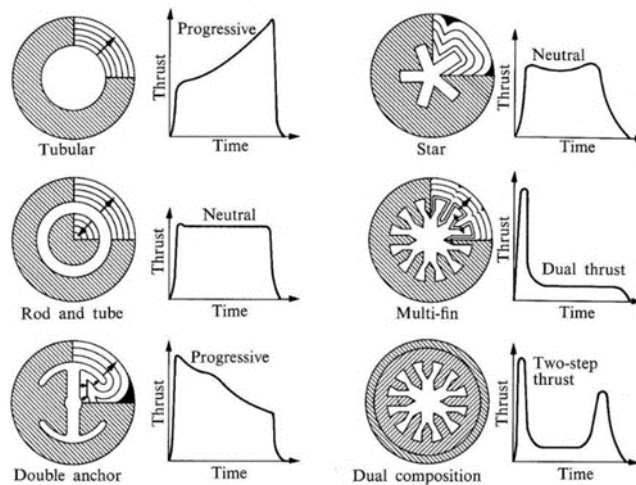


Fig 4.2 : Regression and thrust profiles of several grain configurations.

The burning to throat area ratio (A_b/A_t) is another crucial statistic utilized in the construction of solid propellant rocket motors. Using the mass conservation principle inside the chamber, we may describe this dimensionless coefficient, indicated by the symbol K_n , in terms of the pressure within the chamber.

$$K_n = \frac{A_b}{A_t} = \frac{P_0^{1-n}}{a \rho_b c^*} \quad (4.18)$$

Using the mass flow rate, the throat area, and the pressure within the chamber, we can get the characteristic velocity c^* .

$$c^* = \frac{P_0 A_t}{\dot{m}} = \frac{\sqrt{\gamma R_g T_0}}{\gamma \sqrt{\left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{(\gamma-1)}}}} \quad (4.19)$$

Only combustion temperature, gas constant, and specific heat ratio affect characteristic velocity. It is helpful for comparing propellant efficiency.

4.4 HEAT TRANSFER

High combustion chamber temperatures, which may reach up to 2500 K, are a major safety issue for rocket motors. Therefore, the greatest danger is the complete failure of the engine due to the melting or weakening of any of the structural components. Therefore, it is crucial to maintain the motor's integrity by monitoring and adjusting the temperature of its components.

4.4.1 Convective heat transfer at the nozzle

The nozzle walls are heated by convection and radiation from the hot gas. The first one affects the entire response because convection heat flux relies on gas velocity, and rocket motor nozzle velocities are supersonic. Radiation heat transfer may be crucial if the gas has many solid particles or liquid droplets, but we won't address it.

The convective heat flux that generates inside the nozzle is the following.

$$\dot{q} = h_{g_2} (T_{aw} - T_w) \quad (4.20)$$

where T_{aw} As a function of the local Mach number, the specific heat ratio, and the dimensionless Prandtl number, the adiabatic wall temperature is the value at which the flow is completely isentropic.

$$T_{aw} = T_0 \left(\frac{1 + \text{Pr}^{0.33} \left(\frac{\gamma-1}{2} \right) M^2}{1 + \frac{\gamma-1}{2} M^2} \right) \quad (4.21)$$

It is also possible to calculate the convective heat transfer coefficient, h_{g2} , using the semi-empirical Bartz formula.

$$h_{g2} = \left[\frac{b}{D_t^{0.2}} \left(\frac{\mu_0^{0.2} c_p}{\text{Pr}^{0.6}} \right) \left(\frac{P_0}{c^*} \right)^{0.8} \left(\frac{D_t}{r_c} \right)^{0.1} \right] \left(\frac{A_t}{A} \right)^{0.9} \sigma \quad (4.22)$$

Where,

$$\sigma = \frac{1}{\left[\frac{1}{2} \left(\frac{T_w}{T_0} \right) \left(1 + \frac{\gamma-1}{2} M^2 \right) + \frac{1}{2} \right]^{0.8-m/5} \left[1 + \frac{\gamma-1}{2} M^2 \right]^{m/5}} \quad (4.23)$$

$$\text{Pr} = \frac{\gamma}{1.77\gamma - 0.45} \quad (4.24)$$

D_t is the diameter of the throat, r_c is the radius of curvature, and μ_0 is the gas viscosity measured under stagnation conditions.

Prandtl's number may be written in terms of the specific heat ratio as, according to

Furthermore, the coefficients m and b in equations (4.22) and (4.23) are given to be the following

$$m = 0.75$$

$$b = 0.026 \text{ (for subsonic speed)}$$

$$b = 0.023 \text{ (for supersonic speed)}$$

From equation (4.22) there are some observations to be focused on:

- The heat flow is almost inversely proportional to the chamber pressure ($h_{g2} \sim P^{-0.8}$).
- Where A is least ($h_{g2} (A_t/A)^{0.9}$), the neck is where the greatest heat flow occurs.
- The heat flow increases with the molecular weight of the gas ($c_p \propto 1/MW$, $c \propto 1/MW$) $h_{g2} \propto MW^{0.5}$
- The heat flow increases with increasing radius of curvature ($h_{g2} (1/r_c)^{0.1}$)

4.4.2 Effect of Heat Losses

As the flow passes through a real nozzle, some of its heat is dissipated as convective heat flux via the nozzle walls. Since these losses are not transformed into kinetic energy, the motor's overall performance suffers. This is often included in when determining the effective exhaust velocity of the gases, which is lower than their ideal velocity.

Calculating the total heat flux lost at the nozzle walls requires integrating the heat flux per unit surface along the inner nozzle area.

$$\dot{Q}_w = \int_{S_{int}} \dot{q}_w dS \quad (4.25)$$

By dividing the losses by the total heat flux produced at the combustion chamber, the proportion of heat lost is then calculated.

$$\frac{\dot{Q}_w}{\dot{Q}_T} = \frac{\dot{Q}_w}{\dot{m}c_p T_0} \quad (4.26)$$

The loss of performance due to lower kinetic energy exiting the nozzle is often smaller than the heat flux losses. This means that the following connection is true

$$\dot{m} \left(\frac{v_2^2}{2} - \frac{v_2'^2}{2} \right) < \dot{Q}_w \quad (4.27)$$

where (as discussed in the preceding sections) v_2 is the ideal exhaust velocity and v_2' is the effective exhaust velocity.

Then, by separating v_2' from the prior relation, replacing equation (4.10), and rearranging certain parts, one may get the ratio of the effective exhaust velocity over the ideal exhaust velocity.

$$\frac{v_2'}{v_2} > \sqrt{1 - \left(\frac{\dot{Q}_w}{\dot{m}c_p T_0} \right) \left(\frac{1}{1 - \left(\frac{P_2}{P_0} \right)^{\frac{\gamma-1}{\gamma}}} \right)} \quad (4.28)$$

4.5 STRUCTURAL INTEGRITY

A solid-propellant rocket motor's casing serves as the combustion chamber in addition to housing the propellant. As a result, it must be able to endure all mechanical loads brought on by the gasses as well as stresses brought on by high temperatures and component thermal expansion.

Since the motor's construction and the loads placed on it are designed to be axysimmetric, all stresses generated by the motor are main stresses and are referred to by the terms listed

below in the cylindrical reference system: Hoop (or circumferential) stress (σ_H) and radial stress (σ_r) and axial stress.

For this reason, it is best to design the motor using a finite element analysis that takes into account the thermal and mechanical effects of the boundary conditions.

However, a preliminary design is still required to determine the size of some motor components and to set other design parameters, such as the maximum expected operating pressure. To this goal, the motor housing may be modeled as a cylindrical pressure vessel with closed ends.

If the thickness to internal radius ratio of the cylinder is smaller than 1/10, then the thin-walled cylinder approximation holds and a condensed membrane theory may be applied. Since the hoop and longitudinal stresses are assumed to be uniform throughout the wall's thickness and to be much bigger in size, the radial stress may be neglected.

However, if the thickness to internal radius ratio of the cylinder is more than 1/10, the thick-walled cylinder approximation is more accurate, and the Lam'e formulae must be applied.

Figure 4.4 compares the stress distributions for the thick-walled cylinder in both models.

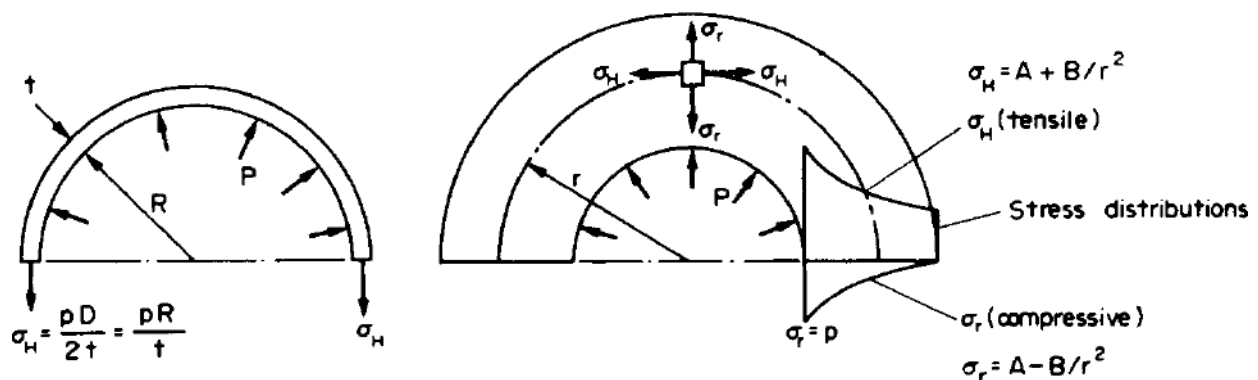


Figure 4.3: Stress distribution of thin (left) and thick (right) walled cylinders with internal pressure load.

4.5.1 Burst Pressure

Bursting, or exceeding the ultimate (or burst) pressure, is one of the most dangerous ways the structure might collapse. If it were to occur, the rocket would completely fail, the casing would collapse, and the engine would explode, posing a major threat to anybody around.

Designing the motor casing in such a way that the burst pressure is never achieved or exceeded is crucial.

The yield strength and ultimate strength of the case's materials, as well as its shape, affect this pressure. Applying Svensson's equations [13] and simulating the situation as a cylindrical pressure vessel with thin or thick walls, depending on its radius to thickness ratio, is an easy technique to determine this value.

$$P_u = \sigma_y \psi B_1 \text{ (thick cylinders)} \quad (4.92)$$

$$P_u = \frac{\sigma_y}{m} B_3 \text{ (thin cylinders)} \quad (4.92)$$

where m is the radius/thickness ratio, B_1 and B_3 are burst factors dependent on the strength ratio y/u , and is a geometrical factor dependent on m . Figure 4.5 shows the relationship between the burst factor and the strength ratio, while Figure 4.6 shows the relationship between the geometry factor and the radius to thickness ratio, both of which are provided by Huston and Josephs .

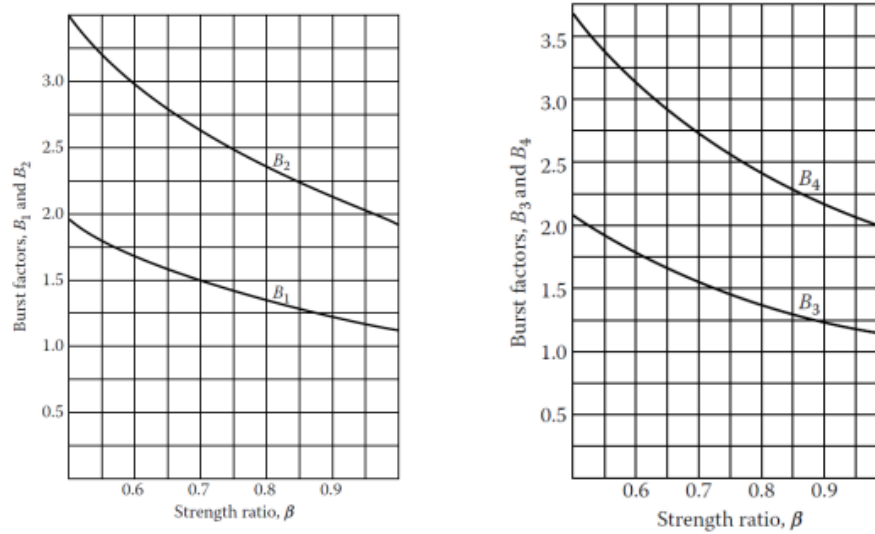


Fig 4.4 depicts a comparison between a thick cylinder (left) and a thin sphere (right), along with their relative strength ratios and burst factors. The numbers 1-4 denote spherical forms, whereas the numbers 1-3 denote cylindrical ones.

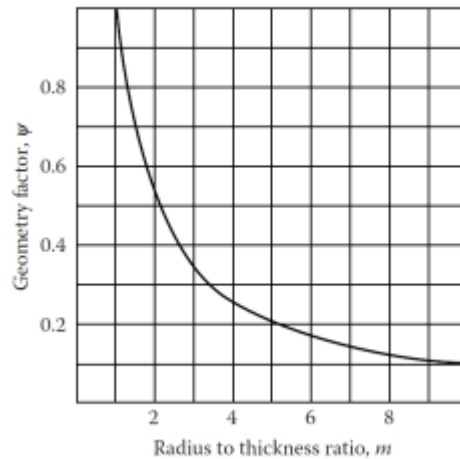


Fig 4.5 shows the geometry factor as a function of the strength ratio for thick cylinders and spheres.

4.5.2 MEOP pressure

When designing a reusable motor, bursting is obviously the most important failure mode to avoid. However, permanent deformation of the structure should also be considered. Something could happen if the yield strength of the casing is surpassed.

Maximum Expected Operating Pressure (MEOP) is a measure of how much pressure may be applied to a chamber before the casing fails. However, a buffer zone will be included to ensure the building holds up even if slight pressure variations near the limit cause the threshold to be crossed.

If the problem can be modeled as a thin-walled cylinder pressure vessel, then other equations will be used.

If the situation can be represented as a thin-walled cylinder, the stresses may be determined using the simplified membrane theory equations.

$$\sigma_{\theta\theta} = 2\sigma_{zz} = \frac{P_c R}{t} \quad (4.31)$$

where P_c represents the chamber pressure, R represents the average radius, and t represents the wall thickness.

Stresses can be computed using Lamé's equations if the case can be approximated as a thick-walled cylinder.

$$\sigma_{rr} = \frac{R_i^2 P_c - R_o^2 P_a}{R_o^2 - R_i^2} - \frac{(P_c - P_a) R_i^2 R_o^2}{r^2 (R_o^2 - R_i^2)} \quad (4.32)$$

$$\sigma_{\theta\theta} = \frac{R_i^2 P_c - R_o^2 P_a}{R_o^2 - R_i^2} + \frac{(P_c - P_a) R_i^2 R_o^2}{r^2 (R_o^2 - R_i^2)} \quad (4.33)$$

$$\sigma_{zz} = \frac{R_i^2 P_c - R_o^2 P_a}{R_o^2 - R_i^2} \quad (4.34)$$

where P_a is the surrounding pressure, R_o is the outer radius, and R_i is the inner radius.

The Von Mises stress, which has the following formula, is the yield criteria that will be utilized to calculate the design pressure.

$$\sigma_{vm} = \sqrt{\frac{1}{2} [(\sigma_{rr} - \sigma_{\theta\theta})^2 + (\sigma_{\theta\theta} - \sigma_{zz})^2 + (\sigma_{zz} - \sigma_{rr})^2]} \quad (4.35)$$

By introducing a safety factor and applying the yield criterion to the Von Mises stress expression on a material, the yield strength can be determined.

$$\sigma_y = SF_d \sqrt{\frac{1}{2}[(\sigma_{rr} - \sigma_{\theta\theta})^2 + (\sigma_{\theta\theta} - \sigma_{zz})^2 + (\sigma_{zz} - \sigma_{rr})^2]} \quad (4.36)$$

Chapter 5

Motor preliminary design

The design process begins at this stage in the thesis. The actions used in the design process to achieve the aims and adhere to the criteria are collected in the following sections.

All of the motor components will be shown and scaled in this chapter. Additionally, the choice of the propellants, the materials utilized, and the assembly process will all be discussed.

5.1 CASE SIZING

Since the motor case is the only part subject to these constraints, it must be sized first (see table 1.4). This part is vital because it prevents the chamber from collapsing, withstands the pressure within it, and can be used again and over again without wearing out or breaking.

The exterior diameter (D_{case}), thickness (t_{case}), and length (L_{case}) are the three factors that will determine the case.

Nevertheless, the first parameter is already specified by the GCR-01 standard, which specifies that the motor case's outside diameter must be 75 mm.

The maximum case length required by Motor is 500 mm, although the actual length may vary depending on how the design process plays out in the following phases. The length of the casing has an impact on the motor's performance since it directly influences the amount of thrust produced because it determines how much of the propellant is exposed (also referred to as the burning surface). Because of this, a bigger motor produces greater thrust for a given diameter. As a result, the case length will initially be 500 mm.

The thickness of the casing is the third design factor, and it has an impact on both the structural mass and the maximum pressure that the motor is permitted to work at. It will also specify whether the case may be handled as a thick-walled or thin-walled cylinder, as was shown in the previous chapter. It will initially be set at 2.5 mm, but depending on the findings of the study, it may need to be changed. It will start at 2.5 mm, but that might alter if the investigation shows anything different.

Parameter	Value
D_{case}	75 mm
L_{case}	500 mm
t_{case}	2.5 mm

Table 5.1: Summary of the design parameters of the motor case.

5.2 MATERIAL SELECTION

The performance of the motor and the gimbal prototype depends on the careful selection of materials for each component of the motor. The motor's maximum operating pressure is closely related to the material used for the casing, hence restricting the force it can produce. Furthermore, the material density plays an essential effect in the inert mass of the vehicle since the engine is carried within the rocket after the burnout.

The materials used then need to be both very sturdy and featherweight. When comparing different types of materials or alloys used in aerospace applications, specific stiffness (E/ρ), which is equal to the Young's modulus multiplied by the density of the material, is an important metric to consider.

It is preferable to use materials with a high stiffness, but brittle ones like polyvinyl chloride (PVC), ceramics, and certain metallic alloys (which also have significant stiffness values) must be avoided because to the possible safety issues connected with a motor failure.

The yield and ultimate tensile strengths, as well as the melting point, are other crucial characteristics to evaluate and contrast. Since the combustion gases reach their hottest point at the nozzle's throat, this latter factor is crucial there.

Metallic materials provide benefits in pricing and production. Metallic cases may be produced from cylinders of the required size that are currently available on the market, but composite cases need mandrels of the same diameter as the case to wrap the fibers. In addition to a dedicated oven for curing the matrix, making a composite case requires specialized equipment and software for winding the fibers around the mandrel.

After considering all of this information, it was decided that carbon fiber tube would be the best materials to employ in the engine design. In Table 5.2, we provide all of the relevant data for some of the most used alloys in amateur rocketry.

Carbon fiber tubes, owing to their great stiffness and strength, are capable of withstanding both forms of stress. The tubes' low thermal expansion coefficient is also vital to the rocket motor case's continued integrity when it's in use. The tube will not expand or contract excessively when temperatures change because of the tube's low thermal expansion coefficient.

Carbon fiber tubes also have the useful quality of being resistant to corrosion. When used as propellants, the harsh chemicals often found in solid rocket motors may

corrode and degrade the rocket motor casing. Carbon fiber tubes are very corrosion-resistant; thus, the casing won't suffer any structural damage from the rocket's harsh environment.

With their high strength-to-weight ratio, rigidity, low thermal expansion coefficient, and resistance to corrosion, carbon fiber tubes are a great material for solid rocket motor casings. To guarantee the carbon fiber tube can resist the high circumstances encountered during rocket operation, significant thought must be given to the manufacturing technique used to create the tube, as well as the design of the tube itself. Carbon fiber tubes' particular qualities and adaptability guarantee that they will maintain a significant role in aircraft applications.

Material	Density ρ (g/cm³)	Young's Modulus E (GPa)	Yield Strength σ_y (MPa)	Ultimate Strength σ_u (MPa)	Specific Stiffness E/ρ	Melting Temperature (°C)
Aluminum 2024-T6	3.0	71	370	480	23.7	500
Low-Carbon Steel SAE-AISI 1020	7.9	190	380	460	24.1	1420
Carbon fiber	1.65	340	500	800	206.6	2750
Stainless steel - 316	8.0	193	205	515	24.125	1387

Table 5.2: Comparison of structural material properties.

5.3 PRESSURE LIMITS

The equations for calculating the burst and MEOP pressures were introduced in the previous chapter and are now being used.

The table 5.1 numbers indicate that the motor case's radius-to-thickness ratio (m) is

$$m = \frac{r_{\text{case}}}{t_{\text{case}}} = \frac{37.5}{2.5} = 15 > 10$$

That's why it's OK to use a thick cylinder as a stand-in for the motor's housing.

Now that the casing material has been decided upon, the strength ratio (λ) may be calculated as

$$\beta = \frac{\sigma_y}{\sigma_u} = \frac{500}{800} = 0.625$$

Figure 4.6 and the left chart of Figure 4.5 both benefit from the introduction of these numbers, yielding geometrical factors of 1.265 and burst factors of 1.255, respectively.

Equation (4.29), now, may be used to determine the motor case's rupture pressure as

$$P_u = \sigma_y \psi B_1 = 500 \cdot 1.265 \cdot 1.255 = 793 \text{ MPa}$$

By substituting the Lam'e formulae (equations (4.32), (4.33), and (4.34)) into equation (4.36), and then isolating the pressure inside the chamber, the MEOP, also known as the design pressure (P_d), may be computed using the Von Mises yield criterion. The formula for determining the design pressure is as follows: a pressure of 101325 Pa, an inner and outer motor case radius of, respectively, and a safety factor of 1.5

$$P_d = 53.57 \text{ MPa}$$

Since $P_u/P_d = 14.80$, the burst pressure is more than or equal to 2 times the MEOP, as per requirement SR-01.

m	β	B_1	ψ
10	0.625	1.255	1.265
P_d	SF	P_u	P_u/P_d
53.57 MPa	1.5	793 MPa	14.80

Table 5.3: Summary of the pressure limit parameters of the motor case.

5.4 END CLOSURE AND SERVO MOTOR HUB

The end closure of a solid rocket motor is the piece that fits over the open end of the motor casing to keep the hot gases and burning fuel within. This is crucial for the motor's proper functioning and safety. The burning propellant creates extreme pressure and heat, and the end closure must be able to tolerate both while still providing a secure seal.

Several considerations, such as motor size and form, application, and manufacturing capabilities, will influence the design of the end closure. Metals including steel, aluminum, and titanium are often used in end closures, as are composites and ceramics.

The kind of propellant utilized in the motor will also influence the design of the end closure. The end closure, for instance, must be constructed to endure high temperatures and pressures if the engine employs a composite propellant.

The end cap must be strong enough to endure the heat of fire as well. Heat produced by the burning propellant may cause the motor casing and end closure to swell. A catastrophic motor failure may occur if the end closure is not built to withstand this expansion.

In most cases, the final closure undergoes extensive testing throughout the design and production phases to guarantee its durability and functionality. Physical tests of the closure's strength and endurance may be conducted alongside simulations of the firing circumstances.

The end closure serves the practical purpose of sealing the motor, but it may also affect the motor's efficiency. The total efficiency of a motor, for instance, may be affected by factors such as the form of the end closure, which controls the passage of exhaust gases. Therefore, it is essential to carefully optimize the design of the end closure in order to get the required performance characteristics.

In conclusion, the end closure plays a significant role in the development of solid rocket motors. It must be able to maintain its seal in spite of the extreme heat and pressure created by the burning propellant. The kind of propellant utilized and the intended performance characteristics are two of the many variables that will affect the design of the end closure. In most cases, the final closure will undergo extensive testing to guarantee its durability and efficiency.

End Closure	Value
Inner diameter	75 mm
Outer diameter	81.25 mm
Disc Diameter	150 mm

Servo hub	Value
Height	128 mm
Material used	SS-316
Disc Diameter	150 mm

Table 5.4 End Closure and Servo Hub parameters

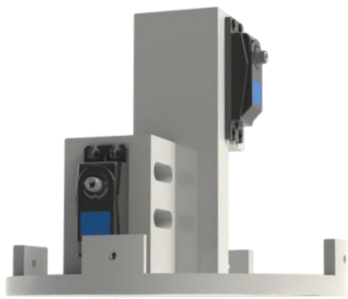


Fig 5.1: Servo motor Hub.



Fig 5.2 End Closure.

5.5 THERMAL INSULATION

High chamber temperatures are sought for in rocket science because they are indicative of peak performance and increased thrust. However, the motor construction must be shielded as much as possible from the combustion chamber's heat to prevent the walls from melting and to maintain the motor's yield and ultimate strength. Since the safe pressure of the shell is lower than the pressure within the chamber, the result would be catastrophic.

There are three types of cooling that may be used to avoid such catastrophic structural damage: First, a heat sink, then insulation, and finally absorption.

The first approach is seldom utilized since it is both inefficient and unreliable. The thick motor housing serves as a heat sink, removing the need for any further cooling measures within the chamber. This is because the chamber temperatures in these motors are very low, and the burning periods are brief (often 1 second or less). Since the motors in question for this thesis are projected to have combustion temperatures of at least 1500 K, and since a thicker case would increase the inert mass of the motor, this is not a viable alternative.

Second, the heat transmission between the casing and the propellant may be minimized by inserting an insulating liner (often made of cardboard or resin-coated cardboard) between the two. In order to reduce heat loss from the chamber to the building's walls, it is preferable

that these materials have a low thermal conductivity. They need to be sufficiently thick to continue isolating all the way through a fire, since they may still melt or burn beyond a certain temperature.

Parameter	Value
λ	0.064 W/(m · K)
t_{liner}	4.0 mm
t_{disk}	4.0 mm
t_{cast}	1.5 mm

low thermal conductivity. thick to continue isolating all they may still melt or burn

The third technique involves placing an ablative substance (such graphite or phenolic resins) between the flow and the structure, and this may be done at both the case and the nozzle. This substance breaks down as it is eroded by the flow, simultaneously forming a barrier layer. Since the throat area is diminished due to erosion when this technique is done at the nozzle, the motor's performance suffers as a result. Since the goal is to have a rocket engine that can be used more than once, this approach will be scrapped since it requires a new nozzle insert for each launch.

Therefore, it makes the most sense to use the insulating material. Since cardboard has a low heat conductivity (see table 5.5), is cheap and lightweight, and is often used by amateur rocketeers (meaning its dependability has previously been demonstrated), it was selected as the liner material. The liner will be placed between the grain and the case to prevent heat transfer, and the insulation disc will be placed between the grain and the front closure to prevent heat loss. The casting tube, in which the propellant will be created and to which it will be connected, will be made of the same material.

Thicknesses will initially be those listed in table 5.5, which are commonly used in COTS motors, however this may alter based on the outcomes of future temperature study.

Table 5.5: Summary of the thermal insulation design parameters of the motor case.

5.6 PROPELLANT SELECTION AND CHARACTERISATION

The rocket motor's solid propellant must be chosen after considering factors including toxicity, performance, cost, ease of production, and data availability.

A form of solid rocket propellant known as sugar-based KN propellants, often referred to as sucrose-based propellants, uses sugar as the primary fuel component. Due to their low toxicity, cheap price, and simplicity of manufacture, these propellants have become more and more common in recent years.

The potassium nitrate (KNO_3) oxidizer employed in these propellants is referred to as KN. Due to its stability and high oxygen content for burning, potassium nitrate is often used as an oxidizer in solid rocket propellants. On the other hand, the fuel employed in these propellants is a common sugar called sucrose.

Sucrose, potassium nitrate, and a minor quantity of additional ingredients including binders and plasticizers make up the foundation of sugar-based KN propellants. The potassium nitrate and sucrose are first heated and combined to create a slurry. This slurry is then cast into a solid grain and cured for a certain amount of time at a predetermined temperature.

The low toxicity of sugar-based KN propellants is one of its benefits. Sugar-based KN propellants don't include dangerous compounds like perchlorates or chlorates, unlike other solid rocket propellants such ammonium perchlorate composite propellants (APCP). They are now easy to handle and carry thanks to this.

The affordability of sugar-based KN propellants is another benefit. They are a more affordable alternative to conventional solid rocket propellants since the raw elements needed to create them are easily accessible and affordable.

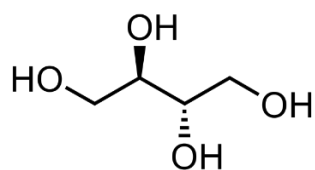
In addition to having a fast burn rate and specific impulse, sugar-based KN propellants may provide a lot of thrust per unit of propellant mass. They are thus perfect for miniature rocket applications including model rockets, high-power rockets, and sounding rockets.

However, there are several restrictions for sugar-based KN propellants. Their comparatively low density in comparison to other solid rocket propellants is one of the key drawbacks. This implies that in order to produce the same amount of thrust as other propellants, a larger volume of propellant is required. For space launch applications, where every kilogram of weight counts, this might be a problem.

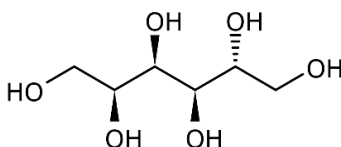
Finally, sugar-based KN propellants provide a secure, economical, and effective substitute for existing solid rocket propellants. Despite certain potential drawbacks, their special qualities make them perfect for small-scale rocket applications and research. Sugar-based KN propellants may be a feasible solution for larger-scale rocket applications with more study and development.

Properties	Sorbitol	Sucrose	Dextrose
Molecular Form	$C_6H_{14}O_6$	$C_{12}H_{22}O_{11}$	$C_6H_{12}O_6$
Density	1.489	1.587	1.54
Melting Point	101	186	148
Molecular weight	182.172	342.3	180.2
Enthalpy of formation	-1353.7	-564.7	-1274.6
Glass transition	-43.5	-65	NA

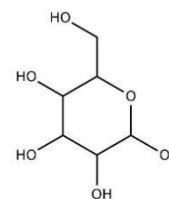
Table 5.6: propellant properties



Sucrose



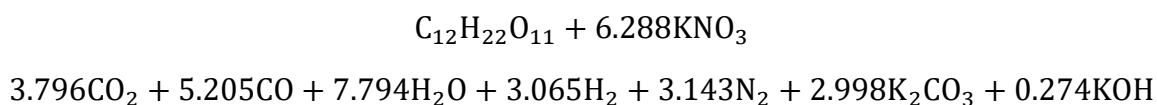
Sorbitol



Dextrose

Fig 5.3 Propellant Chemical Formula

For KNSU, KNSB and KNDX theoretical combustion equation for a propellant with an oxidizer-fuel (O/F) ratio of 65:35.



5.7 Grain Design

As was shown in the preceding chapter, propellant grains may be designed with holes of varying shapes. Depending on the form, the thrust curve profile and the amount of exposed propellant surface will change. This means that the unique characteristics of each application must be researched in order to develop the most suited grain while yet meeting the restrictions set.

However, there are requirements that must be followed in this instance. In accordance with EuRoC regulations [1], A minimum rail departure velocity of 30 m/s is required for the Phobos rocket's 3 m launch rail. The goal is to maintain as much control over the rocket's trajectory as possible. Since the rocket wouldn't be able to reach the required velocity with only the beginning push, this rules out a progressive thrust curve profile for the engine. On the other hand, a thrust profile that is entirely retrograde is undesirable because the rocket's tremendous acceleration might cause structural damage to the aircraft.

Therefore, a neutral profile thrust curve is best for the Phobos rocket. This may be accomplished by BATES grains, rod and tube perforations, or star perforations. The latter are preferable in this configuration for many reasons. The star and rod and tube perforations, on the one hand, are difficult to manufacture, tend to let unburned propellant into the chamber, and the first one eliminates the possibility of axisymmetric approximations for further motor analysis.

However, BATES grains are simpler to produce and, because of their axisymmetric shape, they leave behind less unburned propellant in the chamber walls.

The outside diameter and length of the grain are limited by the available storage capacity in the motor casing. The outside diameter of the grain is calculated to be 54 mm using the known parameters of the case diameter and thickness in addition to the insulator thickness. Since the nozzle has not yet been constructed, it is expected that it will penetrate into the case for the same amount of time as the front closure since it will be connected to the case and sealed in the same manner. With this information and the thickness of the insulating dick in mind, we can calculate that the overall length of the propellant is 850.4 mm.

A perforation size of 12 mm is being explored for the grain's inner diameter, however this might shift depending on whether or not the burn period is to be increased.

To conclude, a decent rule of thumb for the total number of grains is to make each grain 1.5 times longer than its outer diameter. To obtain the length closer to the target value, a total of 10 grains were determined to be optimal.

Parameter	Value
D_{grain}	54 mm
d_{grain}	12 mm
L_{grain}	250 mm
N_{grain}	1

Table 5.: Brief overview of the grain propellant's design specs.

5.8 NOZZLE SIZING

The motor's nozzle is responsible for transferring the heat generated in the combustion chamber into motion. In only a few centimeters, the flow within the nozzle accelerates to supersonic speeds, and it must be able to endure the high temperatures that are generated at the nozzle's walls and, in particular, its throat. Due to the fact that the geometry of this part is affected by factors such as grain design, propellant, and maximum chamber pressure, it is sized last.

This is how the design process works:

The thickness of the propellant consumed at a specific time interval is first represented by the regression variable x , and then the BATES grain's burning area is expressed in terms of this thickness.

$$\frac{dA_b}{dx} = \pi N_{grain} \cdot (2L_{grain} - 4d_{grain} - 12x) \quad (5.7)$$

$$\frac{dA_b}{dx} = \pi N_{grain} \cdot (2L_{grain} - 4d_{grain} - 12x) \quad (5.8)$$

Then, by substituting the $x|_{A_{b,max}}$ into A_b the maximum burning area is obtained

$$A_{b,max} = \frac{1}{2}\pi N \cdot \left[\frac{1}{3}(L_{\text{grain}} + d_{\text{grain}})^2 + D_{\text{grain}}^2 \right] = 951.10\text{cm}^2 \quad (5.9)$$

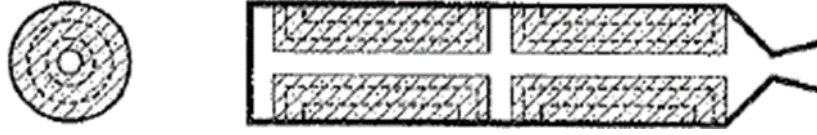


Figure 5.4: BATES grain regression.

Since the maximum chamber pressure is attained at $A_{b,max}$, knowing the maximum burning surface is crucial.

This enables us to get the nozzle throat cross-section using equation (4.18). Here, we can see that the Kn parameter is affected by a number of different variables, including the chamber pressure (P_c), the characteristic velocity (c), the burn-rate coefficients (a , n), and the propellant density (p). The latter three (a , n , and p) are assumed to be constant during the burn, as is c , which is dependent on T_c , and the MW of the combustion products. As a result, P_c is a necessary condition for Kn to take its maximum value (Kn_{max}), which occurs simultaneously with the maximum value of A_b .

This can be summarized in the following expression

$$Kn_{max} = \frac{A_{b,max}}{A_{throat}} \rightarrow A_{throat} = \frac{A_{b,max}}{Kn_{max}} = \frac{a\rho_p c^*}{P_d^{1-n}} A_{b,max} \quad (5.10)$$

Then

$$d_{throat} = 2\sqrt{\frac{A_{throat}}{\pi}} \quad (5.11)$$

Table 5.15 displays the calculated throat diameters for the two propellant types when used in the respective motors.

The nozzle's exit diameter will be determined by the size of the motor, which will be calibrated for operation at sea level by expanding the gases to an ambient pressure (P_a) of 101325 Pa. At the nozzle outlet, the cross-section may be calculated by using equation (5.12).

$$A_{\text{exit}} = \varepsilon \cdot A_{\text{throat}} \quad (5.13)$$

$$d_{\text{exit}} = 2 \sqrt{\frac{A_{\text{exit}}}{\pi}} \quad (5.14)$$

Note, that the throat area has been evaluated for the MEOP (P_d).

How long and what shape will the nozzle be? These are the last two outstanding questions.

Since only the axial exhaust velocity contributes to the formation of thrust, it is crucial that the nozzle be designed to achieve a leaving gas flow that is as axial as possible. Therefore, the nozzle's effectiveness increases as the exit angle (e) decreases.

Nozzles may be created in two basic shapes: bell-shaped and conical. Since bell-shaped nozzles may reach lower exit angles than conical nozzles while still operating Having the same expansion ratio () as a conical nozzle but being shorter in length, their primary use is in the commercial aviation sector (see Figure 5.3). The potential increase in price of bell-shaped nozzles, however, is a worry for the hobbyist sector. Raw material costs may be reduced, but they are not the major expense. Since the machining time of the component is what would ultimately end up fixing the cost of the nozzle in this circumstance, producing an ideal bell-shaped nozzle would need more attention, time, and money to create efficiently in order to keep efficiency.



Fig 5.5: Cad view of designed Nozzle

Because of its relative ease of implementation, a conical nozzle stands out as a promising design direction. However, due to the nozzles' fixed divergence angle of 2, not all gases will

be propelled in a straight line after exiting the nozzle. It is anticipated that efficiency would decline, and this is reflected in the following correction factor

$$\lambda = \frac{1 + \cos \alpha}{2} \quad (5.15)$$

While a smaller divergence angle increases efficiency, it also increases the overall length of the nozzle, therefore a compromise must be reached. In figure 5.4, the divergence angle is compared to the correction factor and the nozzle lengths of the two motors.

Parameter	Values
d_{throat}	14.99mm
d_{exit}	29.99 mm
L_{cone}	37.99 mm

Table 5.8 : Nozzle Parametr

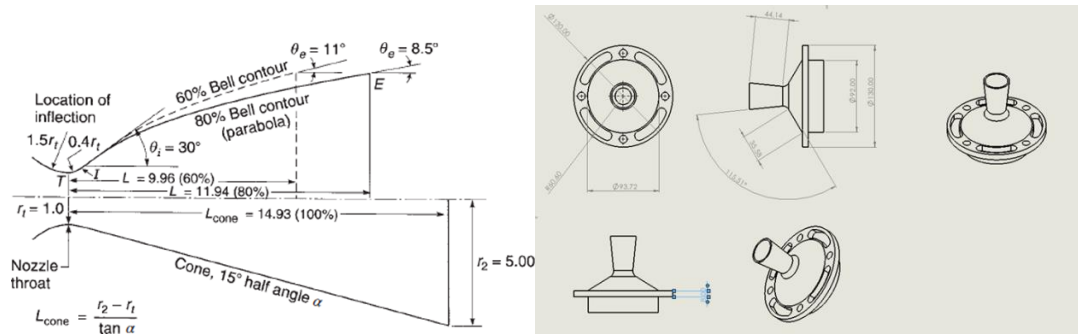


Fig 5.6 Nozzle Dimensions and Drawing

5.9 FLEX NOZZLE PROTOTYPE

Flexi nozzle A flexible, malleable nozzle that can be shaped-adjusted to alter the direction of the exhaust flow is how TVC operates. To do this, mechanical loads are applied to the nozzle, causing it to bend in various directions and change the thrust vector. The advantages of this technology include increased control over the movement of the rocket or aircraft as well as increased efficiency and weight reduction.

A renowned aerospace company's team of engineers and researchers has created a prototype for a flex nozzle TVC system. This prototype was made with the intention of being included into rocket engines, where it might enhance the rocket's overall performance and manoeuvrability.

The flexible nozzle used in the prototype is comprised of a lightweight, high-strength material, maybe a titanium alloy. The rocket engine's nozzle is connected to it, and a number of actuators are employed to apply mechanical loads to it, forcing it to bend in various directions. Input from sensors on the rocket is used by a computer system that controls the actuators to change the nozzle shape in real-time in response to the rocket's motions and trajectory.

The higher efficiency of this prototype over conventional TVC systems is one of its main benefits. This is due to the flex nozzle's ability to be optimised to deliver the best thrust vector control with the least amount of energy input, which lowers fuel consumption and extends the range of the rocket.

This prototype's adaptability and flexibility are further benefits. The system can be modified to meet the unique needs of various rocket and aircraft types, enhancing the accuracy and control of their motions.

Overall, the creation of this flexible nozzle TVC prototype marks a significant advancement in aerospace engineering. This technology has the potential to revolutionise the way we construct and handle rockets and aircraft, making them more competent and efficient than ever before thanks to its increased efficiency, flexibility, and precision.

5.9.1 Mechanism types for Thrust Vector Control:

TVC for a gimbaled nozzle:

A TVC system that has been around for a while is the gimbaled nozzle. To regulate the thrust vector, this device uses a pivoting nozzle that can be turned in various orientations. The simplicity and dependability of this system are its key benefits. Gimbaled nozzle TVC systems use more gasoline than flex nozzle TVC systems, however they may be hefty.

Flex Nozzle TVC:

A more recent advancement in TVC technology is the use of flex nozzle TVC systems. In these systems, the direction of the exhaust flow is controlled by adjusting the shape of a flexible, malleable nozzle. TVC systems with flex nozzles are more effective than those with

gimbaled nozzles because they use less fuel and provide the pilot more control over the direction of the rocket or aircraft. They are better for tiny aircraft and rockets since they are lighter than conventional gimbaled nozzle TVC systems.

Pulse Detonation Engine (PDE) TVC:

PDE TVC systems are a particular kind of TVC that generate thrust via the use of detonation waves. These systems provide fine thrust vector control and are very effective. PDE TVC systems, however, are complex and need sophisticated control systems to function. Additionally pricey, they are normally exclusively used in high-performance rockets and airplanes.

Aerodynamic TVC :

Aerodynamic TVC systems employ the airflow around the rocket or aircraft to regulate its motions. Although these devices are lightweight and simple, they are not as accurate or effective as other TVC systems. They are usually exclusively employed in tiny rockets and airplanes when weight is very important.

5.9.2 Developed Model and its Functioning

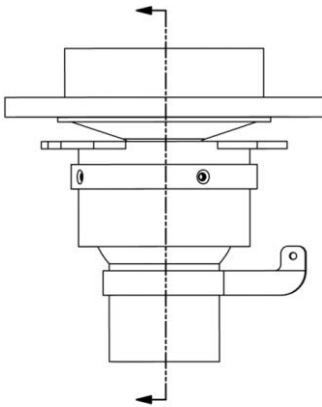


Fig 5.7 Flex Nozzle Prototype

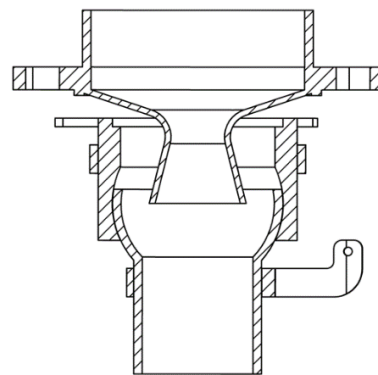


Fig 5.8 Section View of Prototype

The TVC system's flex nozzle is meant to function as a secondary nozzle, and it attaches directly to the engine's CD nozzle. The fundamental function of a rudder is to redirect the thrust that is released from the nozzle throat, giving the rocket more freedom of movement.

Because of its adaptability, the nozzle may be fine-tuned to provide optimal thrust vector control with little energy input, extending the flight time and range of a rocket or airplane.

A number of actuators impart mechanical loads to the nozzle in the flex nozzle TVC system, forcing it to bend in various directions. A computer system, fed data from sensors on the aircraft or rocket, manipulates the actuators to change the nozzle's shape in real time to optimize performance.

The system's versatility and adaptability are two of its main benefits over more conventional TVC setups. The flex nozzle TVC system may be adjusted to meet the needs of various aircraft and rockets, allowing for more precise and controlled flight.

The total weight of the plane or rocket may be kept low thanks to the flex nozzle TVC system's light construction. Engineers and designers in the aircraft industry like this method due to the positive effects it has on fuel economy and range.

Two actuator rods are being connected from flex nozzle to servo motor for its deflection movement

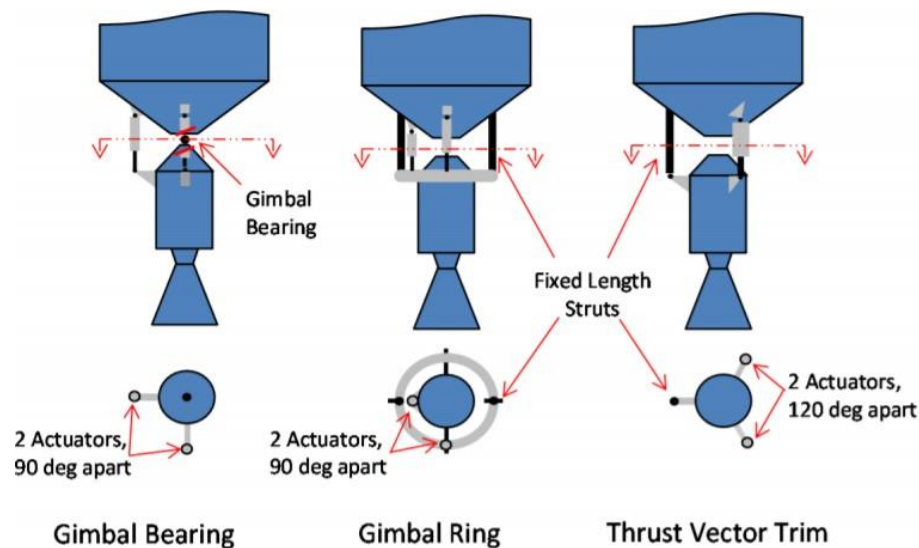


Fig 5.9 Schematic Representation of Nozzle Actation

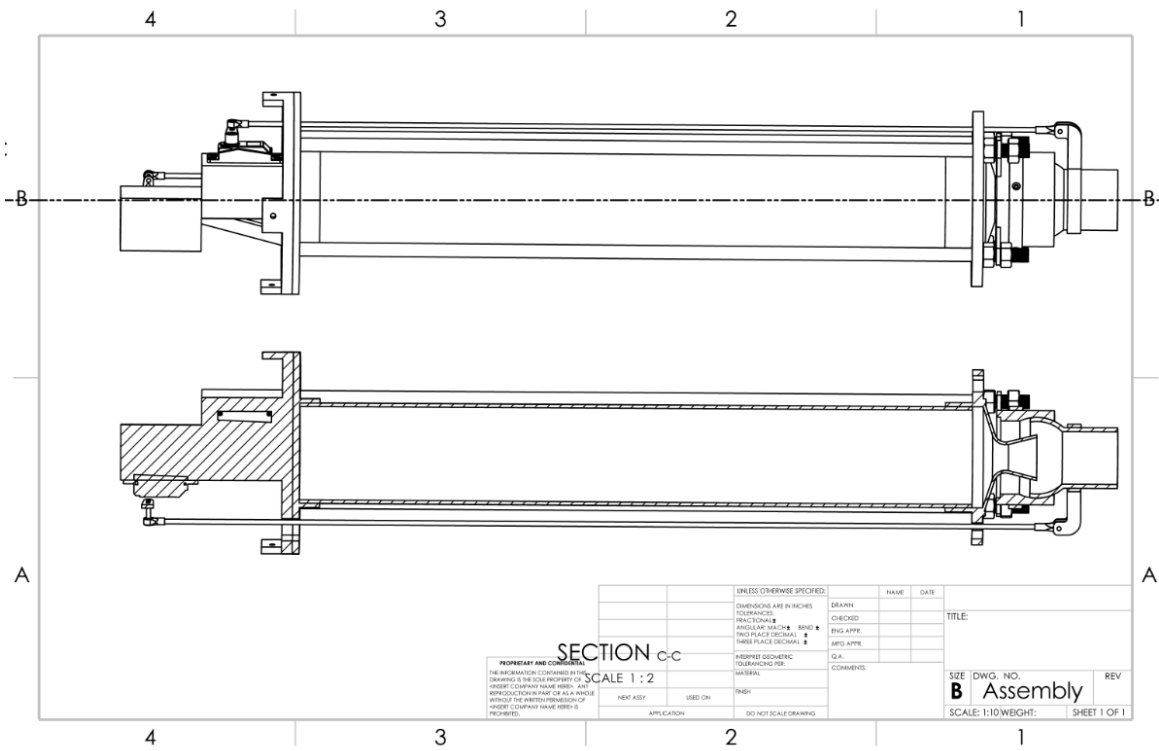


Fig 5.10 Assembly setup of the Model

CHAPTER 6

COMPUTATIONAL STUDY AND SIMULATION

6.1 MOTOR PERFORMANCE ANALYSIS

Motor Performance analysis has been carried out using Richard Nakka's Experimental sheet and open motor software and performance has been compared

The first step in the development process was to settle on a fuel type.

I opted for the "KNSB" propellant combination, which consists of 65% KNO₃ and 35% sorbitol.

This combination, sometimes referred to as "Candy" propellant, has been used successfully in amateur experimental rocketry due to its low risk, high reliability, and low cost.

It has an Isp of around 120 seconds and burns at about 1700 degrees Celsius.

The size of the combustion chamber came up next. The diameter of the chamber was determined by the size of the prefabricated components that might be used.

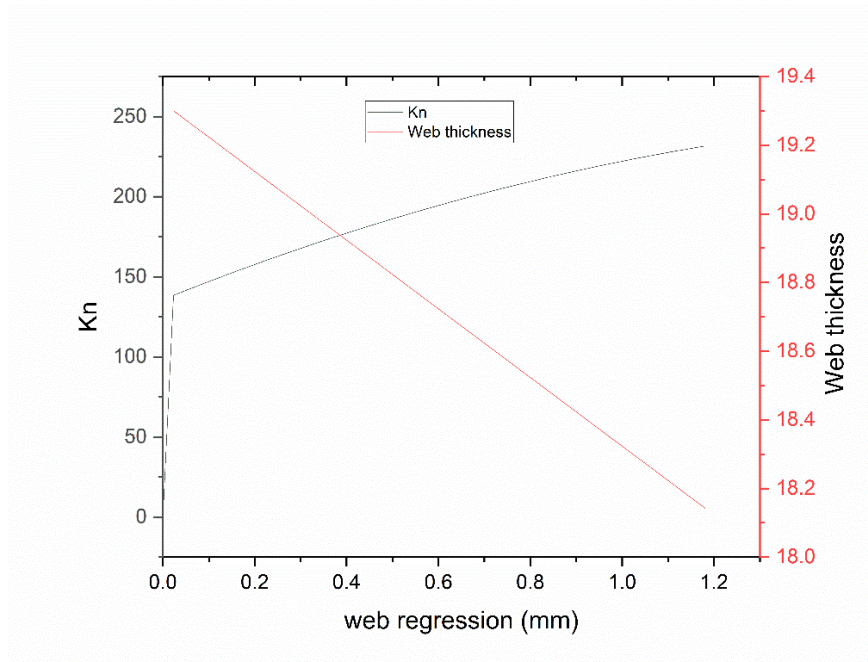


Fig 6.1 Kn vs Web Regression Graph

6.1.1 Richard Nakka 1D Analysis

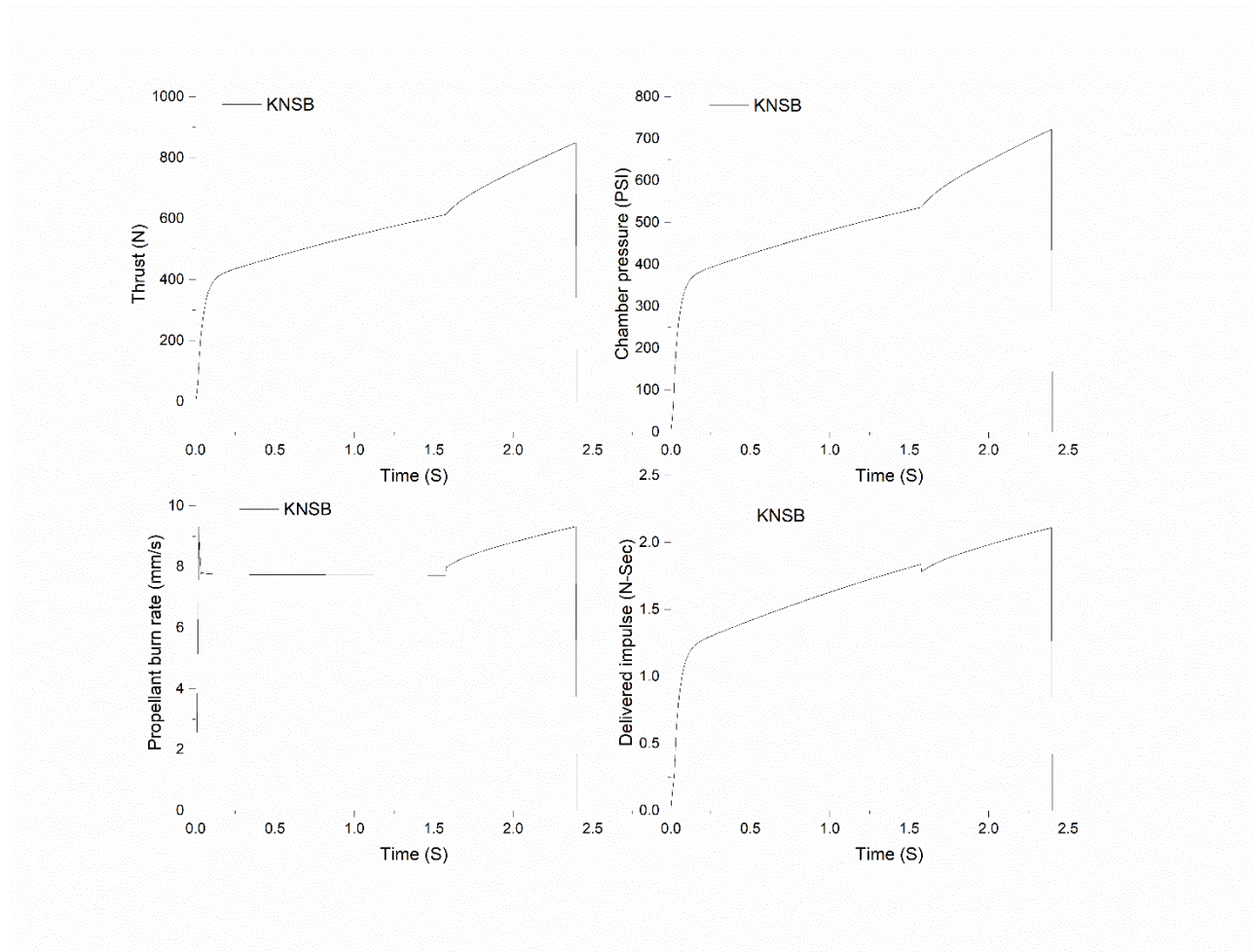


Fig 6.2 KNSB Results from Richard Nakka sheet

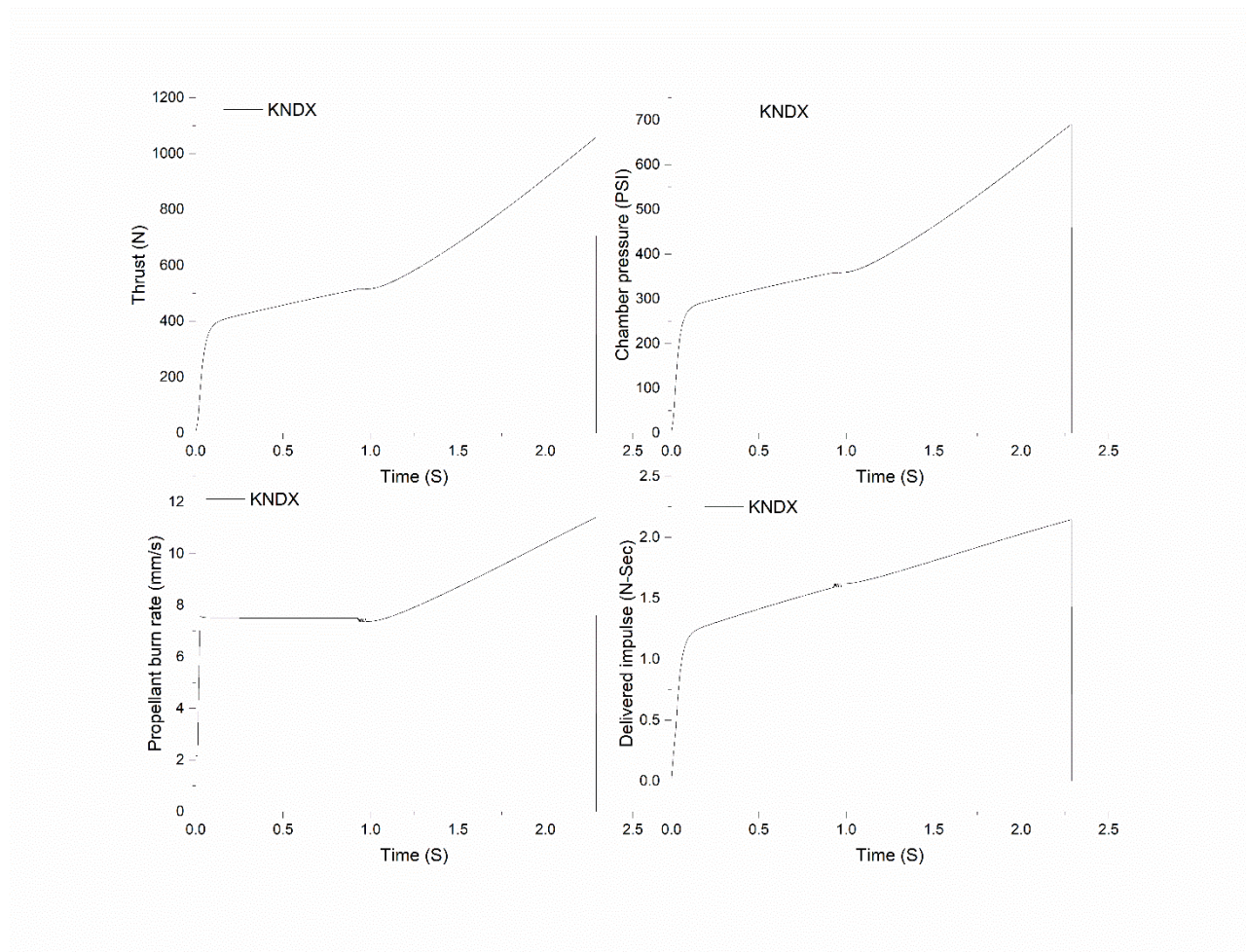


Fig 6.3 KNDX Results from Richard Nakka sheet

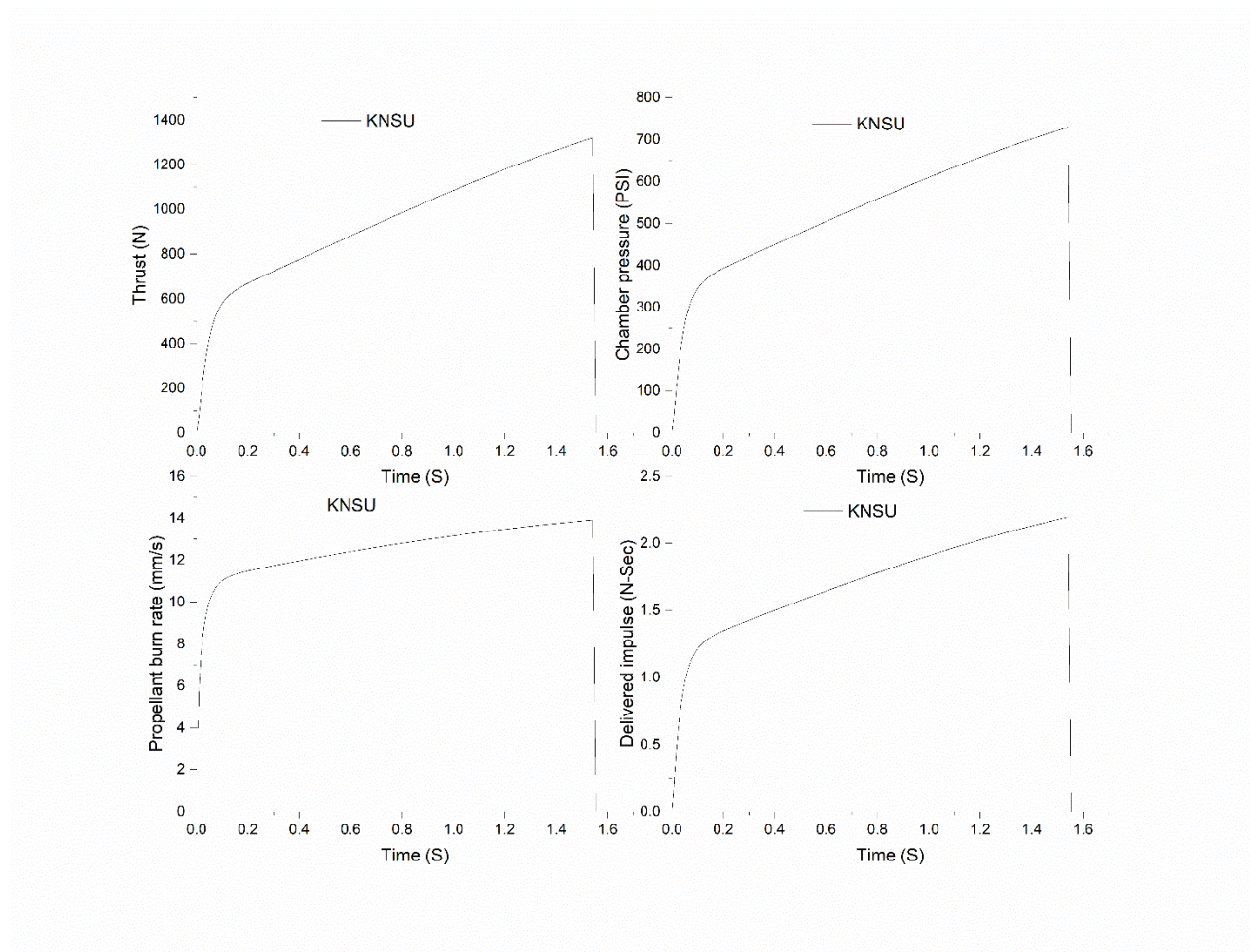


Fig 6.4 KNSU Results from Richard Nakka sheet

6.1.2 3D Flow Analysis on Nozzle and Chamber

- In a 3D axisymmetrical analysis, heat transfer in the circumferential direction is deemed negligible. It is anticipated that this will facilitate the analysis and reduce the computation time. The geometry and boundary conditions that will be applied are compatible with this symmetry, so it is acceptable.
- Transient evaluation. As previously observed in the selection of propellant, the combustion temperature
- The temperature of the combustion process reaches 1730 Kelvin, the motor burns out in 1.487 seconds. As a result, the total duration of the motor's operation is so brief that the steady-state is never attained; consequently, a transient thermal analysis describes the temperature distribution during operation more accurately.
- It is presumed that the chamber temperature remains constant throughout motor operation.
- It is presumed that the composition of the combustion products remains unaltered across the nozzle, i.e. the gas is evaluated at frigid conditions.
- The charring and ablation of insulators are not taken into account in this analysis.
- A forced convective heat transfer process is generated at the interior of the motor case, i.e. the combustion chamber, at boundary
- Regarding the heat flux generated by convection, the radiation emitted by the exhaust gases is deemed to be negligible.

The initial temperature for all motor components is set to 20degreeCelsius.

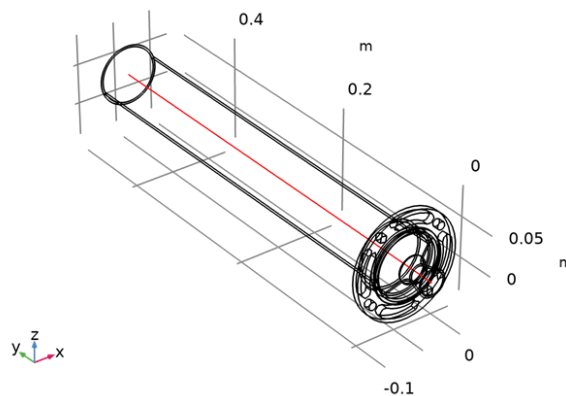


Fig 6.5 Flow sim representation along centerline

6.1.2.1 Geometry

The Above Figure depicts the geometry used in the thermal analysis. One limitation of COMSOL Multi-Physics software is that the geometry cannot be modified during the simulation, and neither can the temperature distribution of one geometry be used as input for a slightly different geometry. Therefore, throughout the thermal simulation, the propellant shape must remain fixed.

In this preliminary study, we assume the motor has the geometry it would have when the chamber pressure is at its highest. Because heat may diminish the motor case's yield and ultimate strengths, measuring its temperature distribution at the time of greatest loads is important. Therefore, the temperature distribution would be somewhat colder than it really would be if the original shape, with the propellant unburned, was employed. Using the geometry of that precise moment, the resulting temperatures will be somewhat greater than they otherwise would be, providing an extra margin of safety. The findings, however, will reveal that the altered geometry of the propellant caused by its consumption has a negligible impact on the thermal profile of the casing.

In contrast to the nozzle, where the gas flow is always in touch with the steel walls, the case is the sole part of the propellant that undergoes a change in shape as the burn advances.

6.1.2.2 Nozzle Temperature

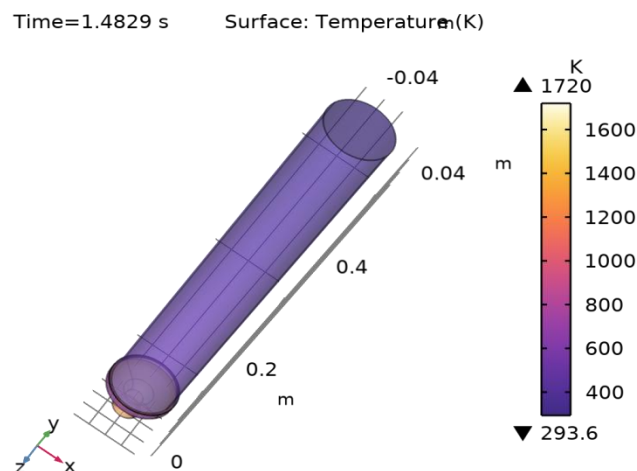


Fig 6.6 Temperature Distribution along the Model

At burnout, at $t = 1.4829$ s, the nozzle reaches its maximum temperature (1730 K = 1456.85 C). Stresses caused by thermal loads are at their highest at this precise moment, making it an important time for assessment. Figure shows the current temperature distribution for KNSU Propellant

Since the front closure is always at room temperature, this section focuses only on the nozzle. Therefore, as was previously observed, it must endure pressures throughout the full motor operation duration if it is to withstand the loads generated from the maximum chamber pressure moment.

6.1.2.3 Von Misses Stress Distribution on chamber and nozzle

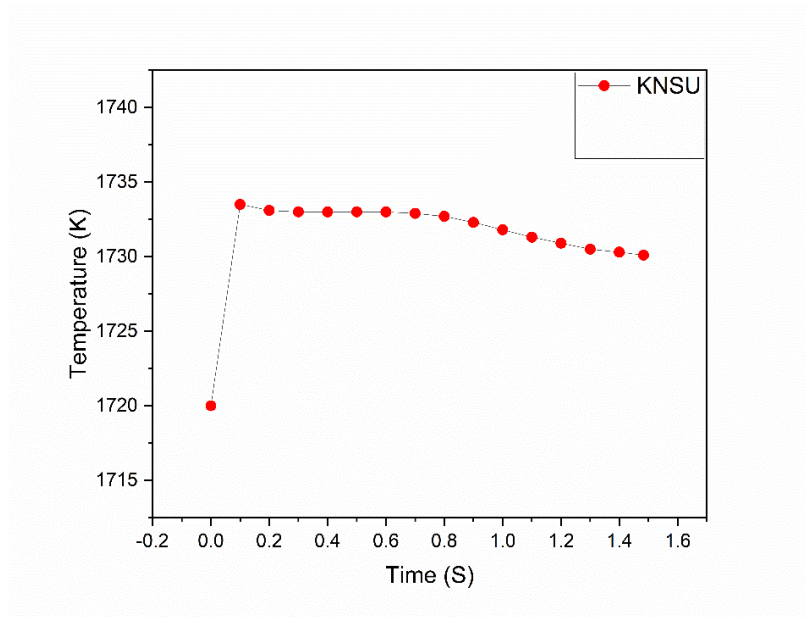


Fig 6.7 Temperature vs time graph For KNSU

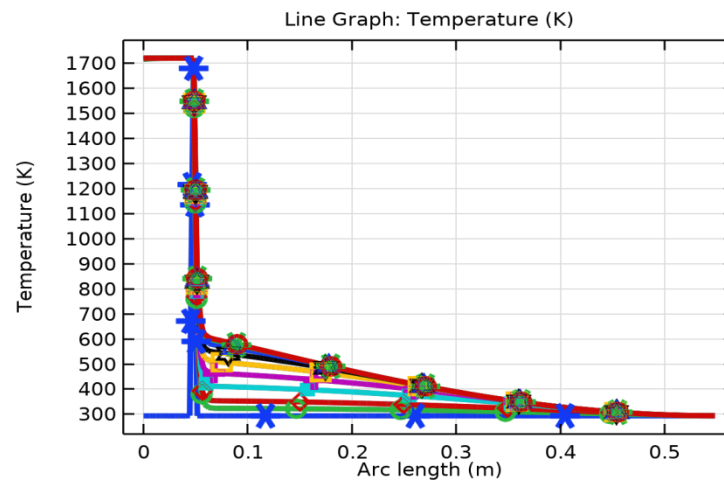


Fig 6.8 Temperature Vs Arc Length

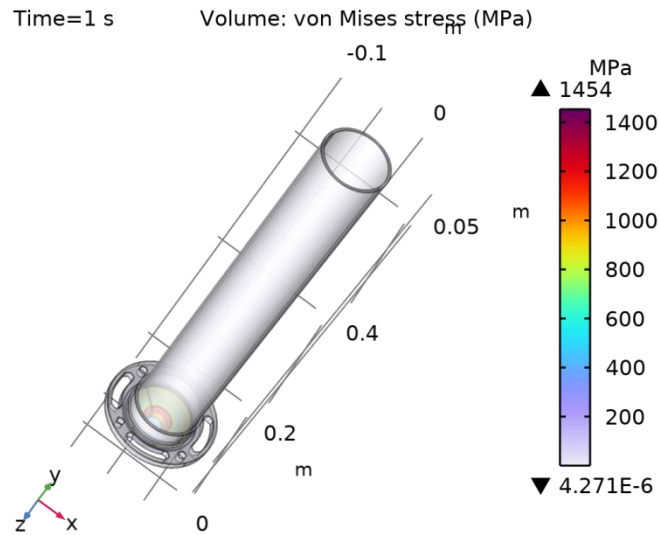


Fig 6.9 Von Misses Stress distribution for KNSU

The above Figure shows the Von Mises stress distribution for the whole nozzle during burnout; figure is included for elucidation. The Von Mises stress and temperature distribution within the nozzle are shown in the final graph.

These findings demonstrate, first, that stresses created at the nozzle are higher during burnout than during the moment of maximum chamber pressure (by more than 1000 MPa, when comparing maximum values).

Secondly, it is unclear whether or not the locations with greater stresses surpass the molybdenum yielding temperature, given the data acquired from literature paper only covers the range between 1400 and 2400 K and no further data was available.

Thirdly, the throat does not correspond with the greatest stress site (a point in the converging section close to the neck).

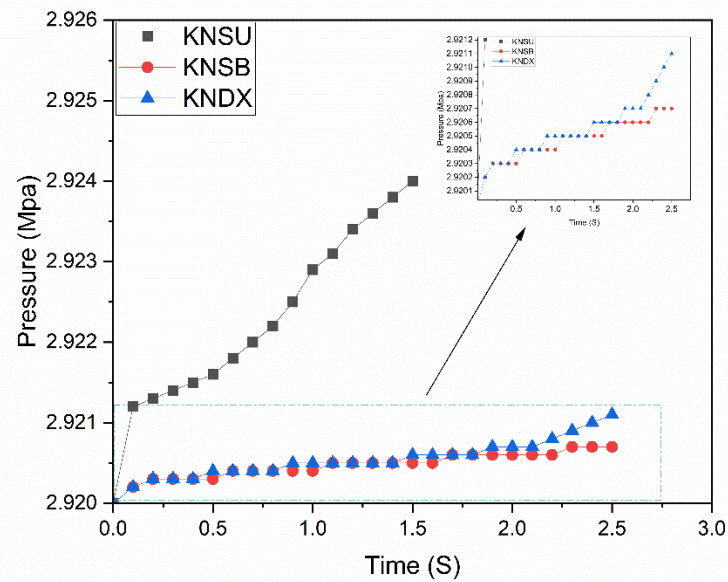


Fig 6.10 Pressure vs time

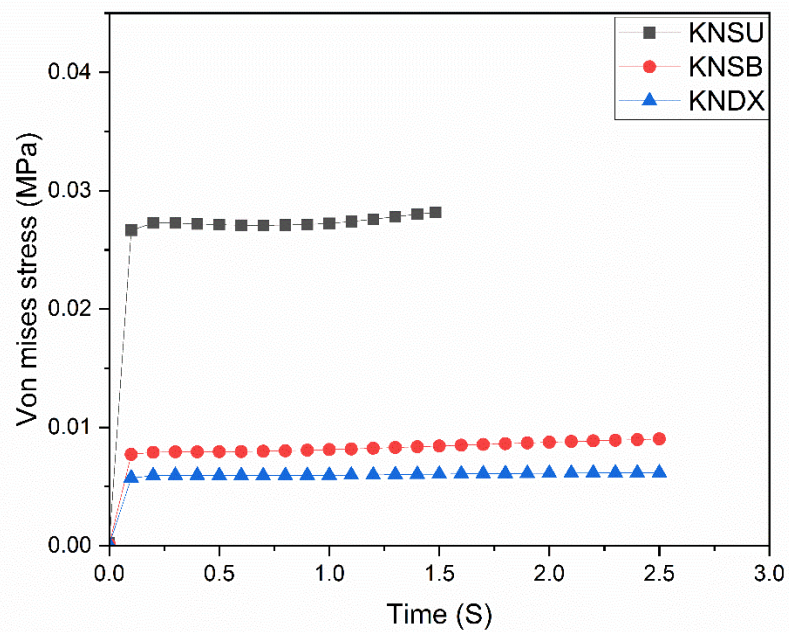


Fig 6.11 Von Misses Stress on chamber vs time

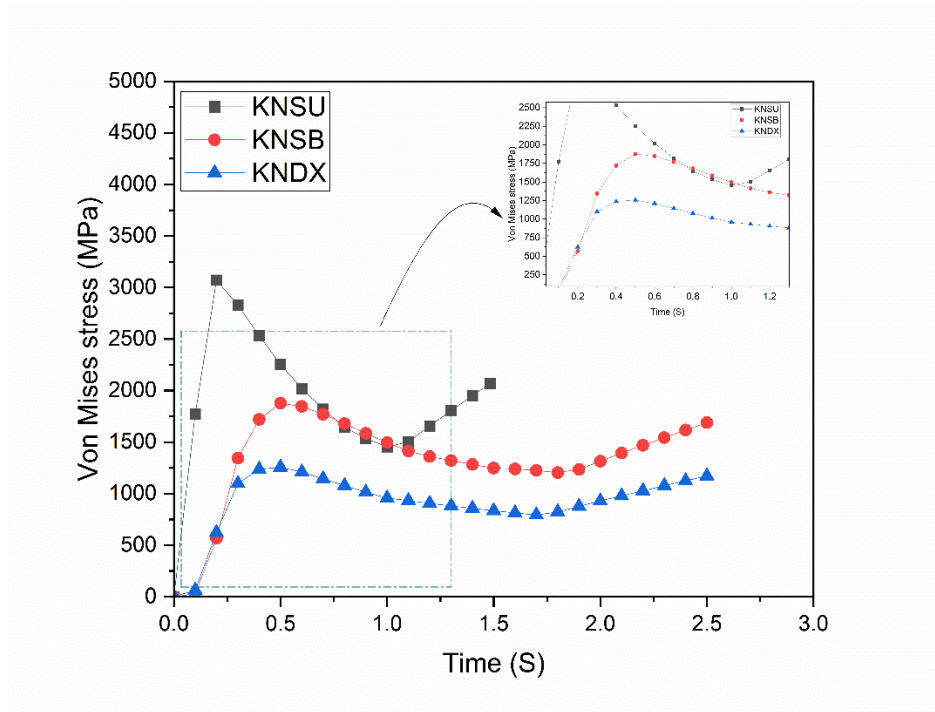


Fig 6.12 Von Misses stress on Nozzle vs time

The von Missses stress is analyzed for the three propellant charcteristics namely , KNSU KNSB and KNDX , where the average chamber pressure is around 1454 M_{pa}

6.1.2.4 Volumetric Strain Distribution on chamber and nozzle

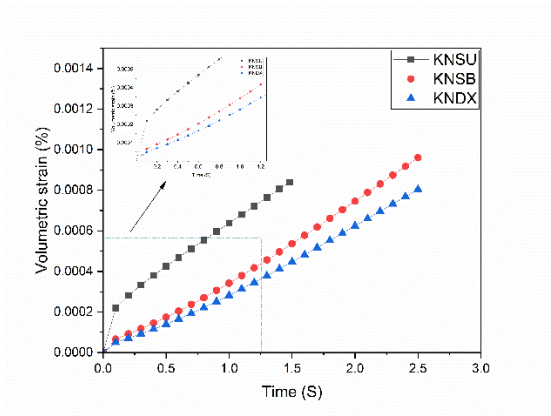


Fig 6.13 Volumetric Strain on Chamber

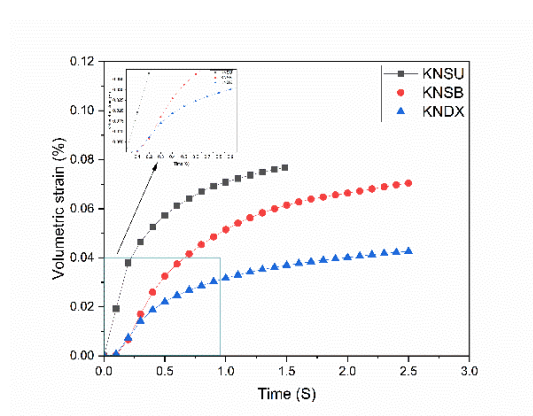


Fig 6.14 Volumetric Strain on Nozzle

6.1.2.5 Displacement Magnitude on chamber and nozzle

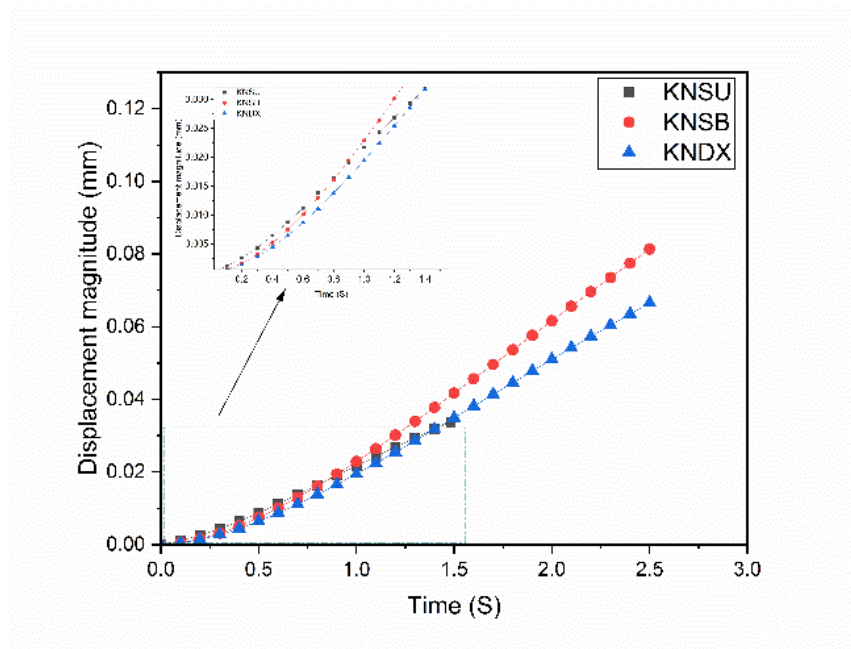


Fig 6.15 : Displacement Magnitude on chamber

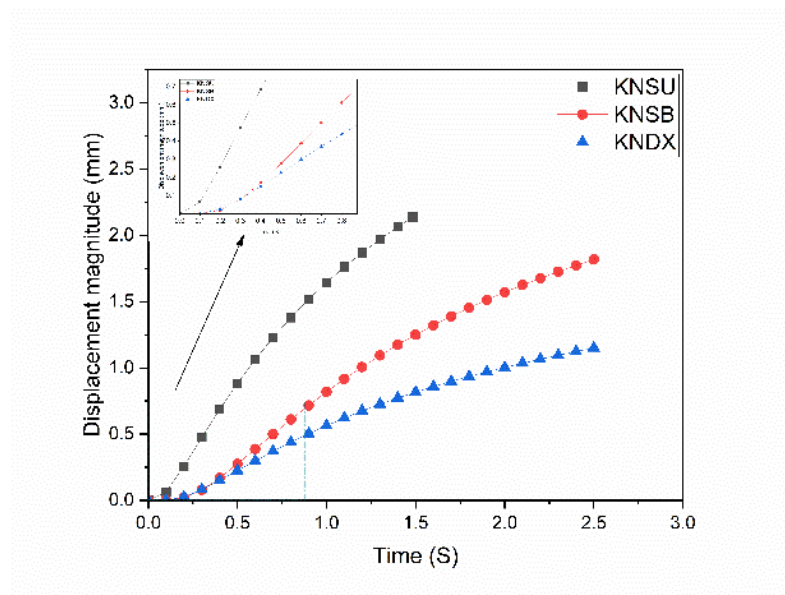


Fig 6.16 : Displacement Magnitude on Nozzle

6.2 Mach Simulation on TVC Prototype

2d- axis symmetry deflection analysis have been performed over the nozzle section with tvc prototype which acts as a extended secondary nozzle , the results have been accumulated for different angles of deflection for ie ($\theta = 15,10,5,0,-5,-10,-15$)

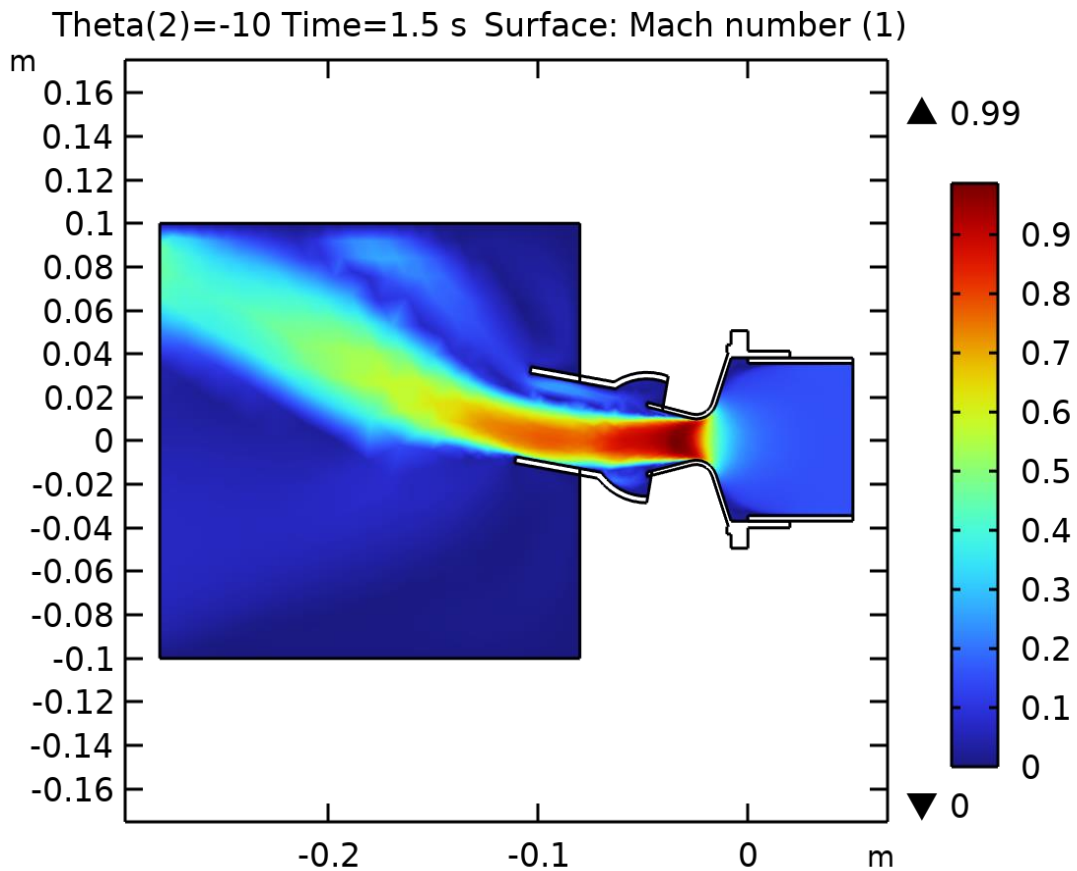


Fig 6.17 Mach Number at Exit Section Of TVC Nozzle

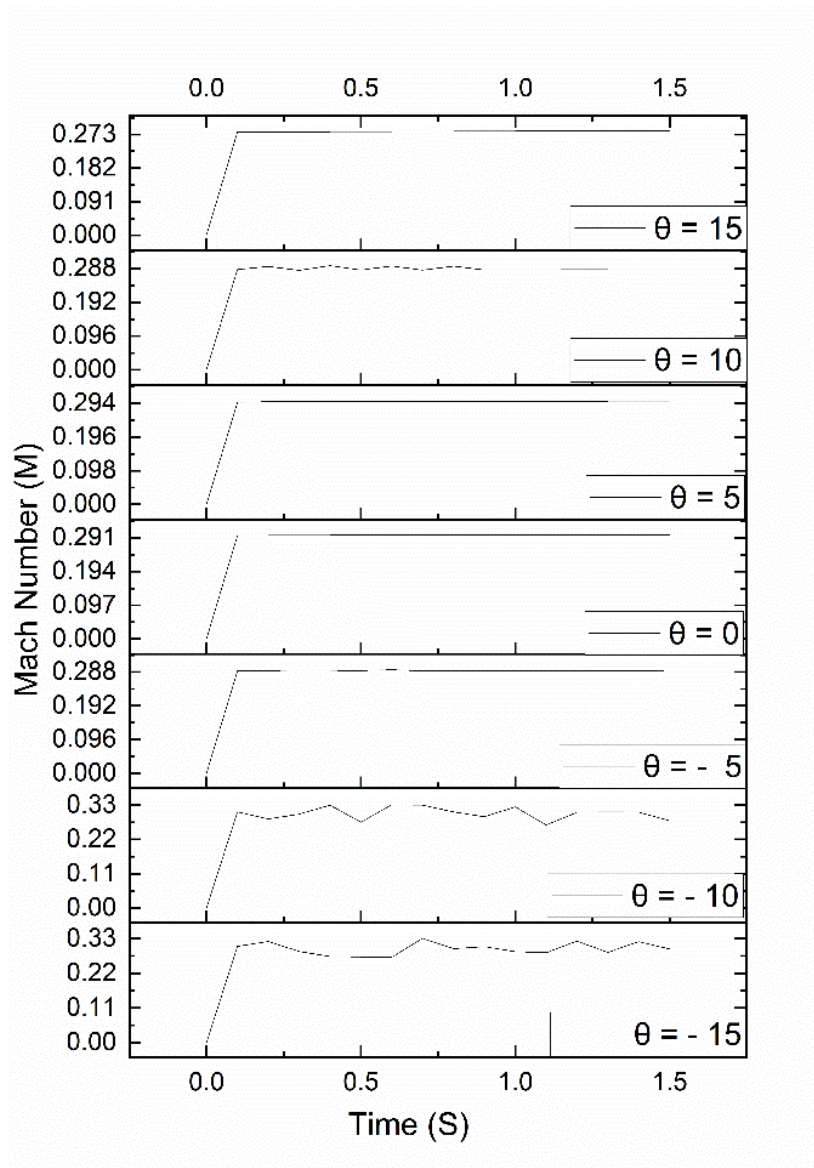


Fig 6.18 Mach vs Time for 6 different angle

CHAPTER 7

EXPERIMENTAL STUDY AND MODEL VALIDATION

7.1 TVC CODE GENERATION

in the first step, the following libraries are imported:

- `Arduino.h`
- `Wire.h`
- `Servo.h`
- `I2Cdev.h`
- `SPI.h`
- `SimpleKalmanFilter.h`
- `SD.h`
- `MPU6050_6Axis_MotionApps20.h`
- `helper_3dmath.h`
- `PID_v1.h`
- `Adafruit_BMP280.h`

In the next step the pin numbering is done for individual components, bearing in mind to use Pulse Width Modulation pins to RGB LEDs, buzzer and Servos. Since both MPU 6050 and BMP 280 work on I2C protocol, they can be connected to the same Serial Clock and Serial Data pins on the Teensy board. Because the SD card reader works on SPI bus, that is faster than I2C protocol, separate pins are defined for MOSI and MISO and Chip select pins.

A PID controller will be implemented to control 2 servos. A simple Kalman filter will be used to filter out the noise from the raw values of MPU and give stable reading. After this, the PID values along with quaternion vectors are defined. In the end other global variables required to execute the code are initialized.

Next step is to setup in which pins will be defined, sensors connection will be checked and initialized'.

In the loop section, a switch case is used to detect 4 states of the rocket namely:

'On Ramp: This state of the rocket indicates, that the rocket is placed on the launch pad and is ready for takeoff. The thrust vectoring is set to off in this mode.

It continuously calculates the acceleration in y direction and if this acceleration is beyond 2000, the launch is detected then the current state of the rocket is now In-Flight mode.

In Flight: In this mode, the flight computer continuously logs all the data from the available sensors. It then processes the data to calculate the

attitude of the rocket and passes it to the PID controller. The PID controller then outputs the data to servos and stabilizes the rocket.

This mode also aborts the flight if the rocket is tilted to 45 degree or more. If everything is well and good then code is written to calculate the maximum altitude that depends on the reading of the barometer. If the current altitude is lower than the previous altitude of the sample then the apogee is detected and this altitude is stored.

There are 3 types of recovery mode for the rocket. The first one is a predetermined time interval after the launch is detected. When this time is elapsed the current state changes to Recovery mode. The second one is an and function between maximum altitude is greater than minimum altitude specified in the code and if the current altitude is lesser than (maximum altitude minus 3m). The third one is also an and function of (if current altitude is greater than or equal to minimum altitude) and velocity is negative. This is one of the safest modes to activate Recovery mode.

Abort Flight: The flight is aborted and if the height is lesser than 15m, pyros are deactivated. Else if the altitude is greater than 15m, recovery mode is activated.

Recovery: If this mode is activated then the pyro channels are activated and the parachutes are deployed. Thrust vectoring is stopped and the servos are position to 90 degrees.

Land detection is also programmed in which the computer checks acceleration in x and y to be close to zero and waits are 5 count of these values. If the count is 5 then the land is detected and data logging is stopped

7.2 PCB Design and Fabrication

PCB design is done in AUTODESK's EAGLE software. First the schematic design is done where each connection is made for the components. Then the board is designed according to the schematic. The board is sized 115 * 65 cm. The schematics and board design are shown below.

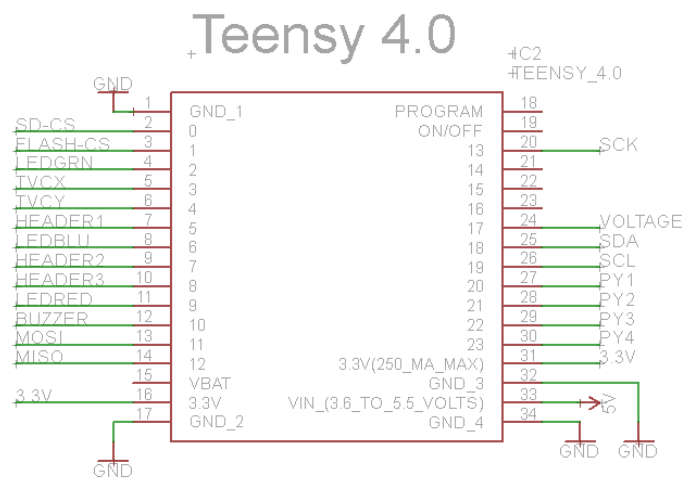


Fig 7.1 Teensy 4.0

the buzzer along with two servos ServoX and ServoY requires pulse width modulation (PWM) pins. Hence Buzzer is connected to Digital pin 9 and The two servos are connected to pin 3 and 4 respectively of the Teensy micro-controller. Pin 5,6,20 and 21 are connected to 4 pyro channels. Chip-select (CS), Master out Slave in (MOSI), Master in Slave out (MISO) and Serial clock (SCK) of SD card reader, are connected to Digital pins 10, 11, 12 and 14 respectively. LED terminals are connected to Analog pins A1,A2 and A3. Serial data and clock are connected to pin 18 and 19 respectively. Voltage measure pin is connected to Analog pin A9. The header pins are connected to 7,8 and 22 for the requirement of any Serial communication in future projects

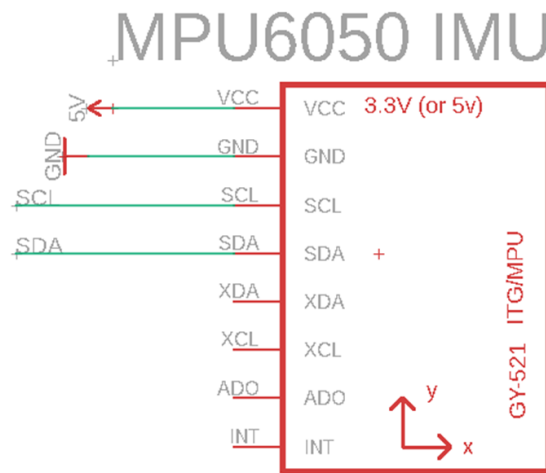


Fig 7.2 MPU 6050

MPU 6050 and BMP 280, work on I2C protocol and requires only SCL and SDA pins that are connected to Teensy

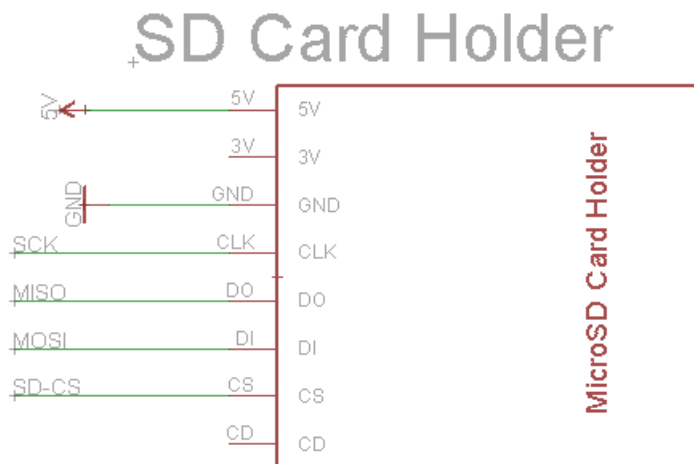


Fig 7.3 Micro SD Card Holder

SD card reader works on SPI (Serial Protocol Interface), hence requires 4 pins these are Chip-select (CS), Master out Slave in (MOSI), Master in Slave out (MISO) and Serial clock (SCK) as shown in Fig 10.20. The pyro signals consisting of Terminal block, N channel

MOSFET, pull-up and pull-down resistors are shown. The PY is the pyro signal that connects Gate terminal to Teensy

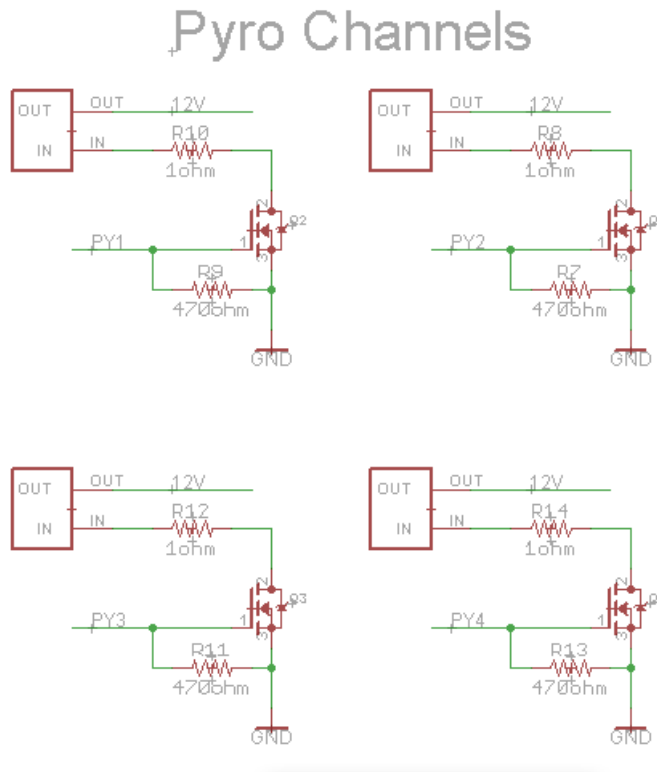


Fig 7.4 Pyro Channels

Capacitors are used to filter out noise during voltage stepdown in Voltage regulators and their schematic is as shown in Fig 10.22, Schematic for servo is shown in the above image , where SERVOX or SERVOY is the signal connection to Teensy.

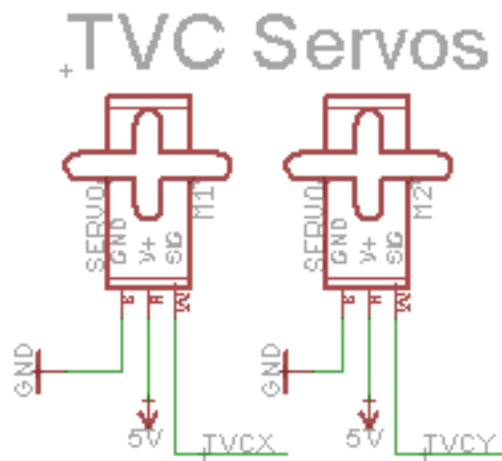


Fig 7.5 TVC Servos

shows the Buzzer and RGB LED schematic, LED requires current limiting resistors to prevent and damage to LEDs hence, three 330-ohm resistors are used for each terminal whereas buzzer does not require any. Fig 10.25 Header pin block J5 and JP2 are extra terminals that can be used to connect any peripherals that work on SPI or I2C protocol.

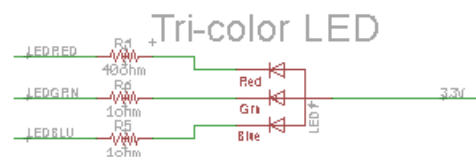


Fig 7.6 Tri- Colour RGB LED

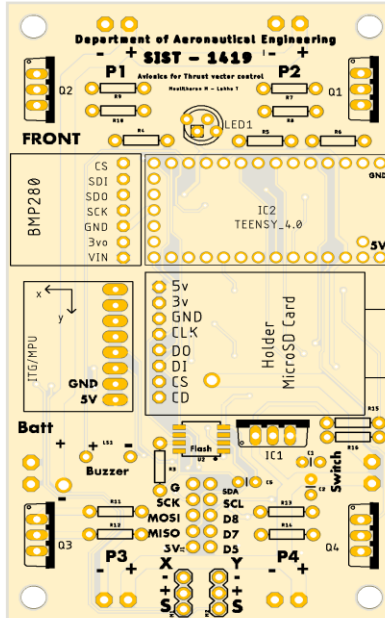


Fig 7.7 PCB Layout – Front View

The board contains 2 main layers of copper. Top layer – 5V and Bottom layer – Ground. The placement of components is a result of several iterations that is convenient for the sensors. From the schematic design, the components are already connected to their respective pins through what is known as air-wires in Eagle. In the board design, Copper traces of suitable width has to be drawn following these wires. Holes known as ‘Via’ can be used to jump from bottom of the board to the top and vice versa. For the battery terminal and to power all the pyros, the width is 2mm which is calculated from the current and temperature rise through this path. For all the rest of the signal connections, the width is 0.25mm. The board design is as shown in Fig 10.27, the red and blue paths represent top and bottom layer traces respectively.

The PID response for the Arduino code is recorded and is shown in Fig blue line signifies launch detection signal, red line shows detection signal, orange and green line shows the PID response signal for two servos for the applied orientation.

7.3 TVC Prototype Model Fabrication and Validation

The Tvc Prototype has been successfully manufactured using CNC technique and assembled in university lab facility and model orientation and dimensions are verified as per the drawings Given for the Manufacturer , The Flex nozzle gimbal prototype which is designed has given an excellent cut of sheer angle for 15 degrees of deflection movement for three degree of freedom , the servo motor which acts a actuator were able to lift the weight of cnc machined parts with its capacity of 45 kg torque per cm



Fig 7.8 Top View Of Assembly



Fig 7.9 Nozzle view of the assembly

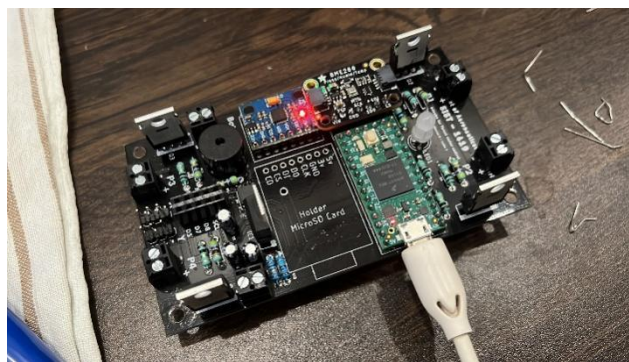


Fig 7.10 Fabricated Avionics For TVC

SIST-1419 Avionics Board has been manufactured and assembled and made it available for general Flight of Landing model rockets , most convenient for any sort of flight missions.

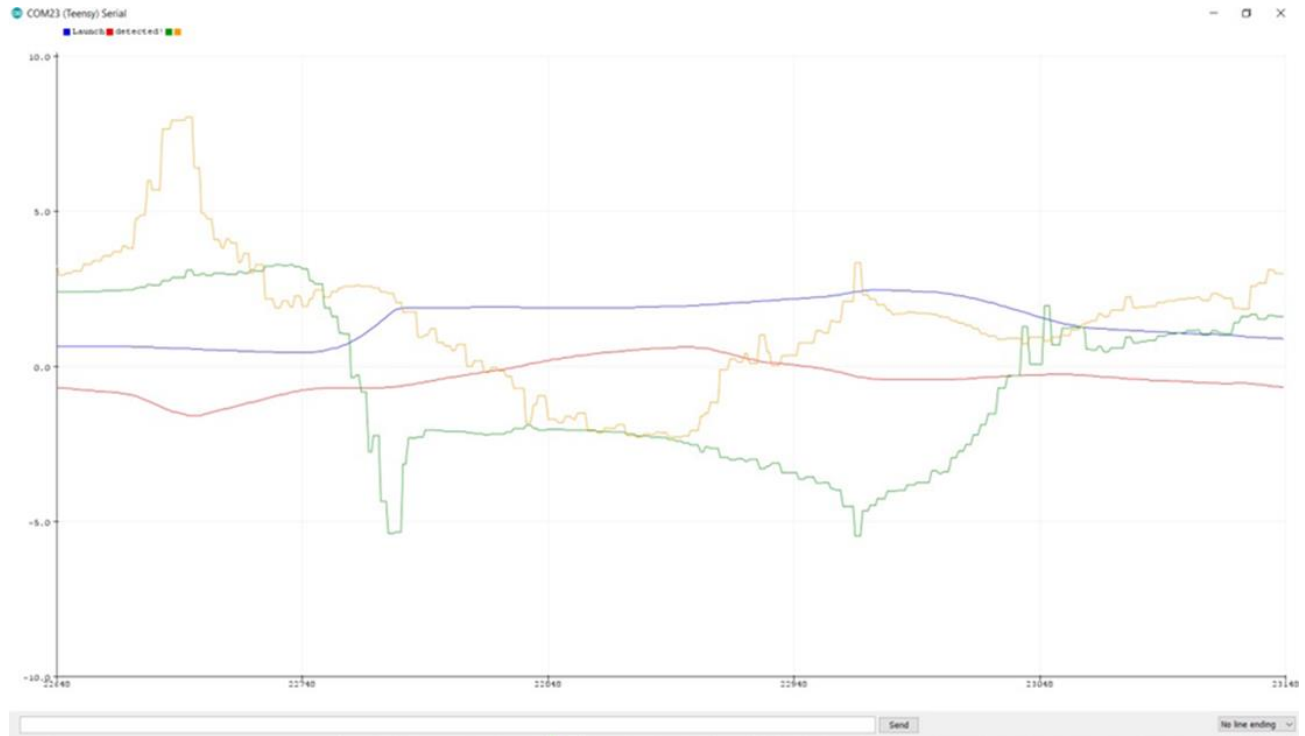


Fig 7.11 Response Characteristics from Mpu 6050 for three axis of motion

The Reponse of the servo motor actuation with respect to its co-ordinates has been observed and validated with zero-characteristic error from the system feedback closed loop control and integration of mechanical and avionics system made successful after verifying the result .

CHAPTER 8

RESULTS AND DISCUSSION

building rocket boosters and engines that operate in vacuum or at sea level, creating the ideal navigation and control systems in ascent or descent modes, and building a reusable rocket engine are just a few of the research and testing that go into rocket propulsion and control.

Exceptional landing and re-launch techniques aid in reducing the yearly cost of the rocket business.

We modelled the control of Three axis gimbal prototype as a linear model using a linear quadratic regulator and PID tuning approach in order to demonstrate a thrust vector controlling system for a small size rocket vehicle employing a pitch and yaw flex nozzle gimbaled design. The results showed that both controllers performed well, although the PID performed better in terms of precision.

A nonlinear approach with quaternion presentation will be the most reliable way to modulate the attitude of rockets and to avoid any issues like the gimbal lock issue accord when dealing with Euler representation. Adding a roll control will help to fully control the rocket attitude instead of only pitch-yaw control, an optimal and feedforward control should be conducted because they are the most reliable and recent control ways used in the rocket industry.

Although , all the propellant characteristics were studied by flow simulation in comsol Multiphysics application and for all six different deflection angles and mechanism has achieved 15 degrees of deflection angle .

We gimballed the whole motor in our TVC design (chamber and nozzle), however a superior design would simply control the nozzle, forcing us to switch the motor's structural material from aluminium to stainless steel for attaining the combustible capable

In order to get better results on how to obtain angle data directly and more precisely from an MPU6050 module, a thorough study of this filter theory should be undertaken since

CHAPTER 9

FUTURE WORK

Future development may include enhancing the prototype's performance via more testing and optimisation. In order to assess the prototype's resilience and suitability for actual usage, this may include testing it in more demanding conditions or with bigger payloads.

The thrust vector control algorithm should also be further honed and optimised to increase its responsiveness and precision. This can include using more sophisticated control methods, such model-based or adaptive control.

The developed TvC Gimbal prototype Can be tested and implemented in liquid based bi-propellant engines .

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